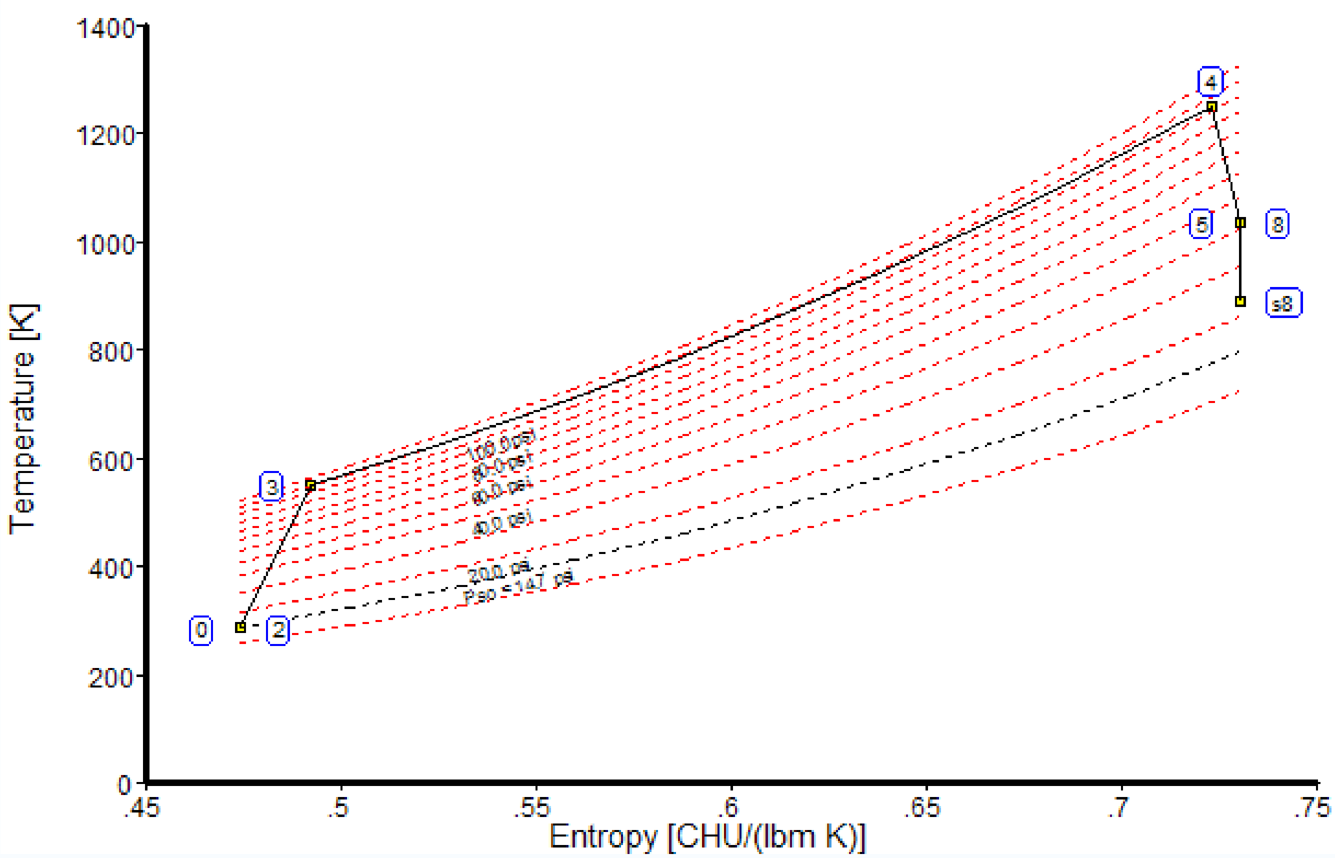
Jet engine performance

Design Point

TS Diagram



Typical Temperature vs. Entropy (TS) Diagram for a single spool turbojet. Note that 1 CHU/(lbm K) = 1 Btu/(lb °R) = 1 Btu/(lb °F) = 1 kcal/(kg °C) = 4.184 kJ/(kg·K).

Temperature vs. entropy (TS) diagrams (see example RHS) are usually used to illustrate the cycle of gas turbine engines. All the reader really needs to know about entropy is that it represents the degree of disorder of the molecules in the fluid and that it tends to increase!

Apart from stations 0 and 8s, stagnation pressure and stagnation temperature are used. Station 0 is ambient.

The processes depicted are:

Freestream (stations 0 to 1)

In the example, the aircraft is stationary, so stations 0 and 1 are coincident. Station 1 is not depicted on the diagram.

Intake (stations 1 to 2)

In the example, a 100% intake pressure recovery is assumed, so stations 1 and 2 are coincident.

Compression (stations 2 to 3)

The ideal process would appear vertical on a TS diagram. In the real process there is friction, turbulence and, possibly, shock losses, making the exit temperature, for a given pressure ratio, higher than ideal. The shallower the positive slope on the TS diagram, the less efficient the compression process.

Combustion (stations 3 to 4)

Heat (usually by burning fuel) is added, raising the temperature of the fluid. There is an associated pressure loss, some of which is unavoidable

Turbine (stations 4 to 5)

The temperature rise in the compressor dictates that there will be an associated temperature drop across the turbine. Ideally the process would be vertical on a TS diagram. However, in the real process, friction and turbulence cause the pressure drop to be greater than ideal. The shallower the negative slope on the TS diagram, the less efficient the expansion process.

Jetpipe (stations 5 to 8)

In the example the jetpipe is very short, so there is no pressure loss. Consequently, stations 5 and 8 are coincident on the TS diagram.

Nozzle (stations 8 to 8s)

These two stations are both at the throat of the (convergent) nozzle. Station 8s represents static conditions. Not shown on the TS diagram is the expansion process, external to the nozzle, down to ambient pressure.

Design Point Performance Equations

In theory, any combination of flight condition/throttle setting can be nominated as the engine performance Design Point. Usually, however, the Design Point corresponds to the highest corrected flow at inlet to the compression system (e.g. Top-of-Climb, Mach 0.85, 35000ft, ISA) , The design point net thrust of any jet engine can be estimated by working through the engine cycle, step by step. Below are the equations for a single spool turbojet.

Freestream

$$T_1 = t_0 \cdot (1 + (\gamma_c - 1) \cdot M^2/2)$$

$$P_1 = p_0 \cdot (T_1/t_0)^{\gamma_c/(\gamma_c - 1)}$$

Intake

$$T_2 = T_1$$

$$P_2 = P_1 \cdot \mathrm{prf}$$

Compressor

$$T_3 = T_2 \cdot ((P_3/P_2)^{(\gamma_c - 1)/(\gamma_c \cdot \eta pc)}$$

$$P_3 = P_2 \cdot (P_3/P_2)$$

Combustor

$$T_4 = \mathrm{RIT}$$

$$P_4 = P_3 \cdot (P_4/P_3)$$

Turbine

Equating the turbine and compressor powers, we have:

$$w_4 \cdot C_{\mathrm{pt}}(T_4 - T_5) = w_2 \cdot C_{\mathrm{pc}}(T_3 - T_2)$$

A simplyfying assumption sometimes made is for the addition of fuel flow to be exactly offset by an overboard compressor bleed, so mass flow remains constant throughout the cycle.

$$P4/P5 = (T4/T5)^{\gamma_t/((\gamma_t - 1) \cdot \eta_{\mathrm{pt}})}$$

Jetpipe

$$T_8 = T_5$$

$$P_8 = P_5(P_8/P_5)$$

Nozzle

$$t_{8\mathrm{s}} = T_8/((\gamma_t + 1)/2)$$

$$p_{8\mathrm{s}} = P_8/((T_8/t_{8\mathrm{s}})^{\gamma_t/(\gamma_t - 1)})$$

$$V_8^2 = 2gJC_{pt}(T_8 - t_{8\mathrm{s}})$$

$$\rho_{8\mathrm{s}} = p_{8\mathrm{s}}/(R \cdot t_{8\mathrm{s}})$$

$$A_8 = w_8/(\rho_{8\mathrm{s}} \cdot V_8)$$

Gross Thrust

$$F_g = C_{\mathrm{x}}((w_8 \cdot V_8/g) + A_8(p_{8\mathrm{s}} - p_0))$$

Ram Drag

$$F_r = w_0 \cdot V_0/g$$

Net Thrust

$$F_n = F_g - F_r$$

The calculation of the combustor fuel flow is beyond the scope of this text, but is basically proportional to the combustor entry airflow and a function of the combustor temperature rise.

Note that mass flow is the sizing parameter: doubling the airflow, doubles the thrust and the fuel flow. However, the specific fuel consumption (fuel flow/net thrust) is unaffected, assuming scale effects are neglected.

Similar design point calculations can be done for other types of jet engine e.g. turbofan, turboprop, ramjet, etc.

Cycle improvements

Increasing the overall pressure ratio of the compression system raises the combustor entry temperature. Therefore, at a fixed fuel flow and airflow, there is an increase in turbine inlet temperature. Although the higher temperature rise across the compression system implies a larger temperature drop over the turbine system, the nozzle temperature is unaffected, because the same amount of heat is being added to the total system. There is, however, a rise in nozzle pressure, because turbine expansion ratio increases more slowly than the overall pressure ratio. Consequently, net thrust increases, implying a specific fuel consumption (fuel flow/net thrust) decrease.

So turbojets can be made more fuel efficient by raising overall pressure ratio and turbine inlet temperature in unison. However, better turbine materials and/or improved vane/blade cooling are required to cope with increases in both turbine inlet temperature and compressor delivery temperature. Increasing the latter may also require better compressor materials.

Off-design

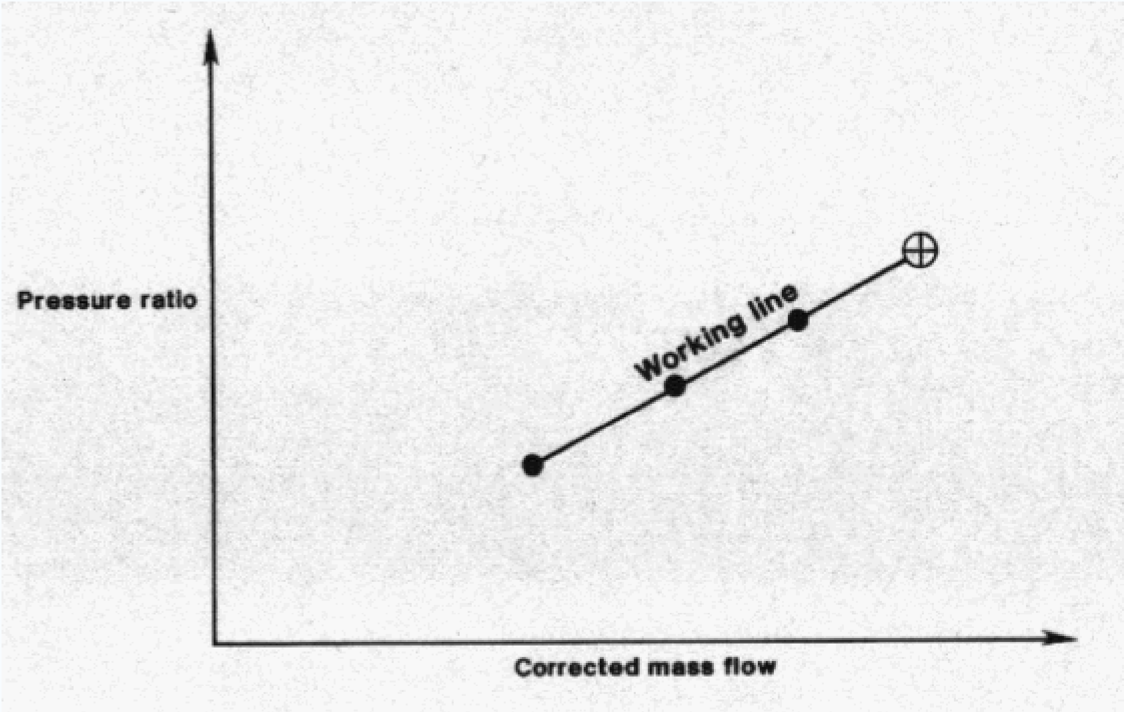
General

An engine is said to be running off-design if any of the following apply:

a) change of throttle setting
b) change of altitude
c) change of flight speed
d) change of climate
e) change of installation (e.g. customer bleed or power off-take)
f) change in geometry

Although each off-design point is effectively a design point calculation, the resulting cycle (normally) has the same turbine and nozzle geometry as that at the engine design point. Obviously the final nozzle cannot be over or underfilled with flow. This rule also applies to the turbine nozzle guide vanes, which act like small nozzles.

Simple Off-design Calculation



Typical compressor working line generated using Simple Off-design Calculation

Design point calculations are normally done by a computer program. By the addition of an iterative loop, such a program can also be used to create a simple off-design model.

In an iteration, a calculation is undertaken using guessed values for the variables. At the end of the calculation, the constraint values are analyzed and an attempt is made to improve the guessed values of the variables. The calculation is then repeated using the new guesses. This procedure is repeated until the constraints are within the desired tolerance (e.g. 0.1%).

Iteration Variables

The three variables required for a single spool turbojet iteration are the key design variables:

1) some function of combustor fuel flow e.g. RIT

2) corrected engine mass flow i.e. $w_{2\mathrm{cor}}$

3) compressor pressure ratio i.e. P_3/P_2

Iteration Constraints (or Matching Quantities)

The three constraints imposed would typically be:

1) engine match e.g. Fn or w_{fe}or T_3, etc

2) nozzle area e.g. A_{8calc}vs A_{8despt}

3) turbine flow capacity e.g. $w_{4corcalc}$vs $w_{4cordespt}$

The latter two are the physical constraints that must be met, whilst the former is some measure of throttle setting.

Note Corrected flow is the flow that would pass through a device, if the entry pressure and temperature corresponded to ambient conditions at sea level on a Standard Day.

Results

Plotted above are the results of several off-design calculations, showing the effect of throttling a jet engine from its design point condition. This line is known as the compressor steady state (as opposed to transient) working line. Over most of the throttle range, the turbine system on a turbojet operates between choked planes. All the turbine throats are choked, as well as the final nozzle. Consequently the turbine pressure ratio stays essentially constant. This implies a fixed $\triangle T_{\mathrm{turb}}/RIT$. Since turbine rotor entry temperature, RIT, usually falls with throttling, the temperature drop across the turbine system, $\triangle T_{\mathrm{turb}}$, must also decrease. However, the temperature rise across the compression system, $\triangle T_{\mathrm{comp}}$, is proportional to $\triangle T_{\mathrm{turb}}$. Consequently, the ratio $\triangle T_{\mathrm{comp}}/T_1$must also fall, implying a decrease in the compression system pressure ratio. The non-dimensional (or corrected flow) at compressor exit tends to stay constant, because it 'sees', beyond the combustor, the constant corrected flow of the choked turbine. Consequently, there must be a decrease in compressor entry corrected flow, as compressor pressure ratio falls. Therefore, the compressor steady state working line has a positive slope, as shown above, on the RHS.

Ratio $RIT/T_1$is the quantity that determines the throttle setting of the engine. So, for instance, raising intake stagnation temperature by increasing flight speed, at a constant RIT, will cause the engine to throttle back to a lower corrected flow/pressure ratio.

The Simple Off-design Calculation outlined above is somewhat crude, since it assumes:

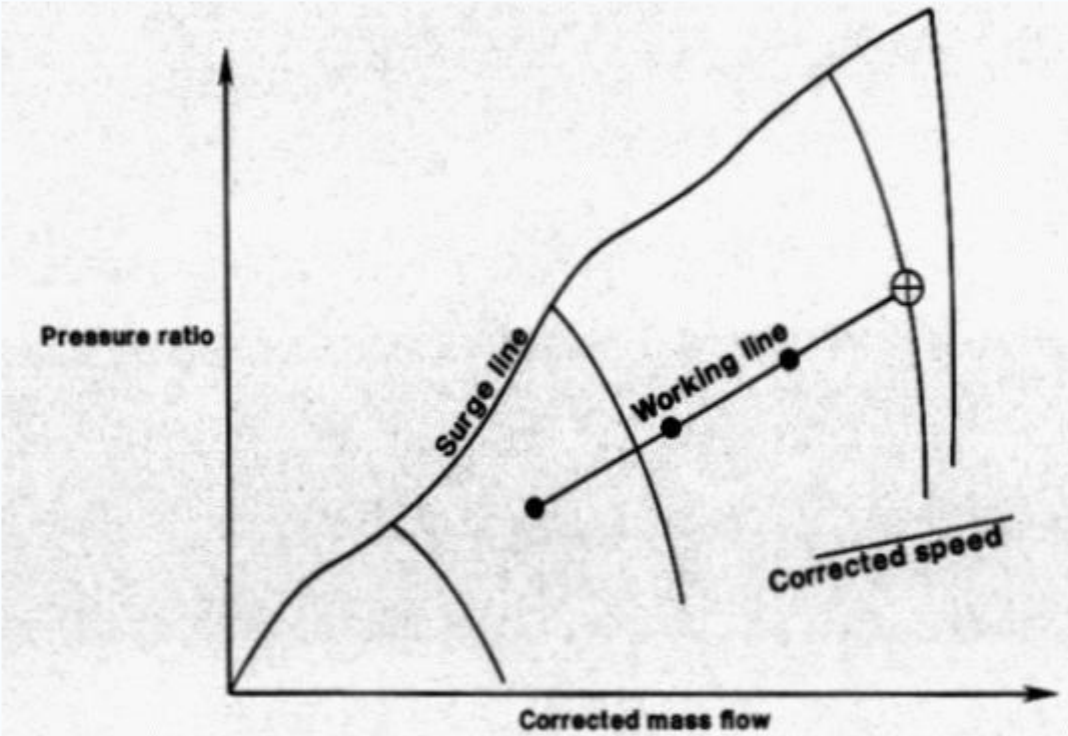
1) no variation in compressor and turbine efficiency with throttle setting

2) no change in pressure losses with component entry flow

3) no variation in turbine flow capacity or nozzle discharge coefficient with throttle setting

Furthermore, there is no indication of relative shaft speed or compressor surge margin.

Complex Off-design Calculation



Typical compressor working line generated using Complex Off-design Calculation

A more refined off-design model can be created using compressor maps and turbine maps to predict off-design efficiencies, relative shaft speeds, etc.

The iteration scheme is similar to that of the Simple Off-design Calculation.

Iteration Variables

Again three variables are required for a single spool turbojet iteration, typically:

1) some function of combustor fuel flow e.g. RIT

2) compressor corrected speed e.g. N_{cor}

3) an independent variable indicative of the compressor operating point up a speed line e.g. β.

So compressor corrected speed replaces corrected engine mass flow and Beta replaces compressor pressure ratio.

Iteration Constraints (or Matching Quantities)

The three constraints imposed would typically be similar to before:

1) engine match e.g. F_n or w_{fe}or T_3, etc

2) nozzle area e.g. $A_{8\mathrm{geometric design}}$ vs $A_{8\mathrm{calc}}/C_{\mathrm{dcalc}}$

3) turbine flow capacity e.g. $w_{4\mathrm{corcalc}}$ vs $w_{4\mathrm{corturbchar}}$

Plotted on the LHS are the results of several off-design calculations, showing the effect of throttling a jet engine from its design point condition. The line produced is similar to the working line shown above, but is now plotted on the compressor characteristic and gives an indication of corrected shaft speed and compressor surge margin.

Performance Software

Over the years a number of software packages have been developed to estimate the design and off-design performance of various types of gas turbine engine. Most are used in-house by the various aero-engine manufacturers, but several are available to the general public (e.g. **GasTurb** http://www.gasturb.de, **EngineSim** http://www.grc.nasa.gov/WWW/K-12//airplane/ngnsim.html).

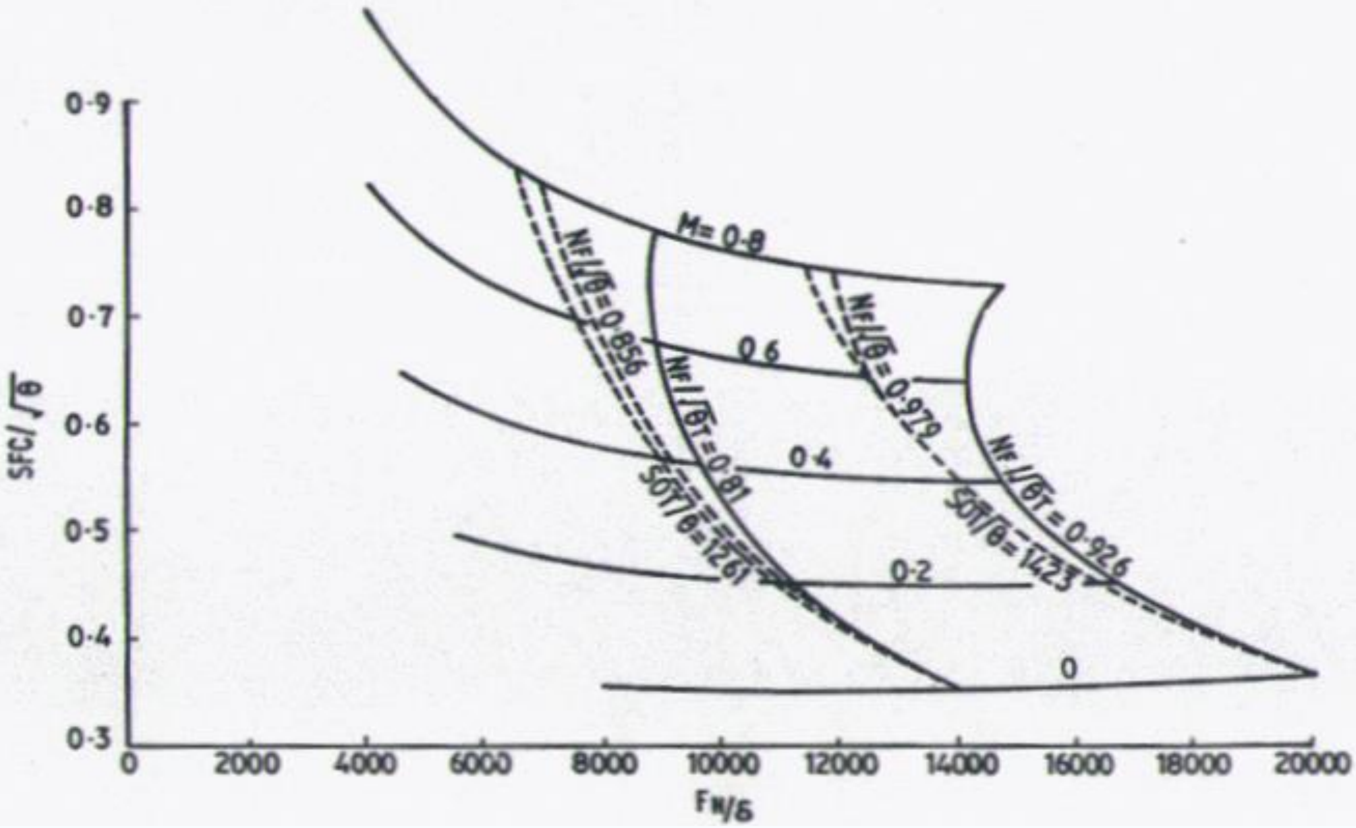
Husk Plot

A Husk Plot is a concise way of summarizing the performance of a jet engine. The following sections describe how the plot is generated and can be used.

Thrust/SFC Loops

Specific Fuel Consumption (i.e. SFC), defined as fuel flow/net thrust, is an important parameter reflecting the overall thermal (or fuel) efficiency of an engine.

As an engine is throttled back there will be a variation of SFC with net thrust, because of changes in the engine cycle (e.g. lower overall pressure ratio) and variations in component performance (e.g. compressor efficiency). When plotted, the resultant curve is known as a thrust/SFC loop. A family of these curves can be generated at Sea Level, Standard Day, conditions over a range of flight speeds. A Husk Plot (RHS) can be developed using this family of curves. The net thrust scale is simply relabeled Fn/δ, where δ is relative ambient pressure , whilst the SFC scale is relabeled $SFC/\sqrt{\theta}$, where θ is relative ambient temperature. The resulting plot can be used to estimate engine net thrust and SFC at any altitude, flight speed and climate for a range of throttle setting.



Typical Husk Plot

Selecting a point on the plot, net thrust is calculated as follows:

$$Fn = (Fn/\delta) \cdot \delta$$

Clearly, net thrust falls with altitude, because of the decrease in ambient pressure.

The corresponding SFC is calculated as follows:

$$SFC = (SFC/\sqrt{\theta}) \cdot \sqrt{\theta}$$

At a given point on the Husk Plot, SFC falls with decreasing ambient temperature (e.g. increasing altitude or colder climate).The basic reason why SFC increases with flight speed is the implied increase in ram drag.

Although a Husk Plot is a concise way of summarizing the performance of a jet engine, the predictions obtained at altitude will be slightly optimistic. For instance, because ambient temperature remains constant above 11000m (36089ft) altitude, the Husk Plot would yield no change in SFC with increasing altitude. In reality, there would be a small, steady, increase in SFC, owing to the falling Reynolds Number and Specific Heat effects.

Thrust Lapse

The nominal net thrust quoted for a jet engine usually refers to the Sea Level Static (SLS) condition, either for the International Standard Atmosphere (ISA) or a hot day condition (e.g. ISA+10 °C). As an example, the GE90-76B has a take-off static thrust of 76,000 lbf (360 kN) at SLS, ISA+15 °C.

Naturally, net thrust will decrease with altitude, because of the lower air density. There is also, however, a flight speed effect.

Initially as the aircraft gains speed down the runway, there will be little increase in nozzle pressure and temperature, because the ram rise in the intake is very small. There will also be little change in mass flow. Consequently, nozzle gross thrust initially only increases marginally with flight speed. However, being an air breathing engine (unlike a conventional rocket) there is a penalty for taking on-board air from the atmosphere. This is known as **ram drag**. Although the penalty is zero at static conditions, it rapidly increases with flight speed causing the net thrust to be eroded.

As flight speed builds up after take-off, the ram rise in the intake starts to have a significant effect upon nozzle pressure/temperature and intake airflow, causing nozzle gross thrust to climb more rapidly. This term now starts to offset the still increasing ram drag, eventually causing net thrust to start to increase. In some engines, the net thrust at say Mach 1.0, sea level can even be slightly greater than the static thrust. Above Mach 1.0, with a subsonic inlet design, shock losses tend to decrease net thrust, however a suitably designed supersonic inlet can give a lower reduction in intake pressure recovery, allowing net thrust to continue to climb in the supersonic regime.

The thrust lapse described above depends on the design specific thrust and, to a certain extent, on how the engine is rated with intake temperature. Three possible ways of rating an engine are depicted on the above Husk Plot. The engine could be rated at constant turbine entry temperature, shown on the plot as SOT/θ. Alternatively, a constant mechanical shaft speed could be assumed, depicted as $N_F/\sqrt{\theta}$. A further alternative is a constant compressor corrected speed, shown as $N_F/\sqrt{\theta_T}$. The variation of net thrust with flight Mach number can be clearly seen on the Husk Plot.

Other Trends

The Husk Plot can also be used to indicate trends in the following parameters:

1) turbine entry temperature

$$SOT = (SOT/\theta) \cdot \theta$$

So as ambient temperature falls (through altitude or climate), turbine entry temperature must also fall to stay at the same non-dimensional point on the Husk Plot. All the other non-dimensional groups (e.g. corrected flow, axial and peripheral Mach numbers, pressure ratios, efficiencies, etc will also stay constant.

2) mechanical shaft speed

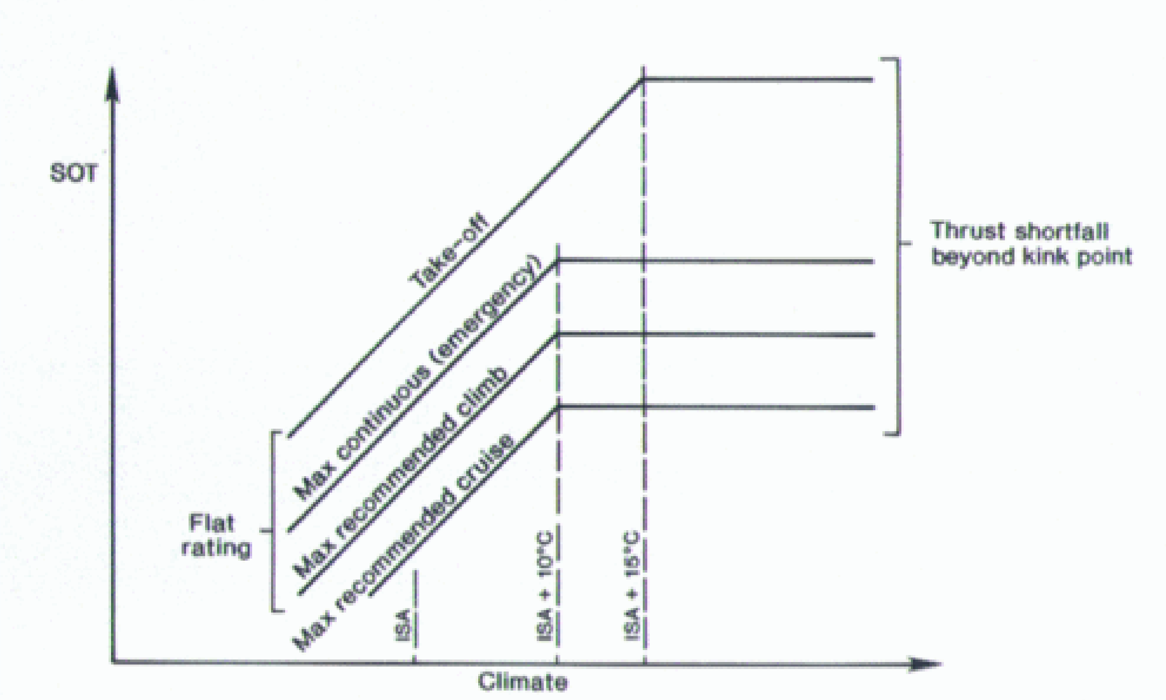
$$N_F = (N_F/\sqrt{\theta}) \cdot \sqrt{\theta}$$

Again as ambient temperature falls (through altitude or climate), mechanical shaft speed must also decrease to remain at the same non-dimemsional point.

By definition, compressor corrected speed must remain constant at a given non-dimensional point.

Rated Performance

Civil



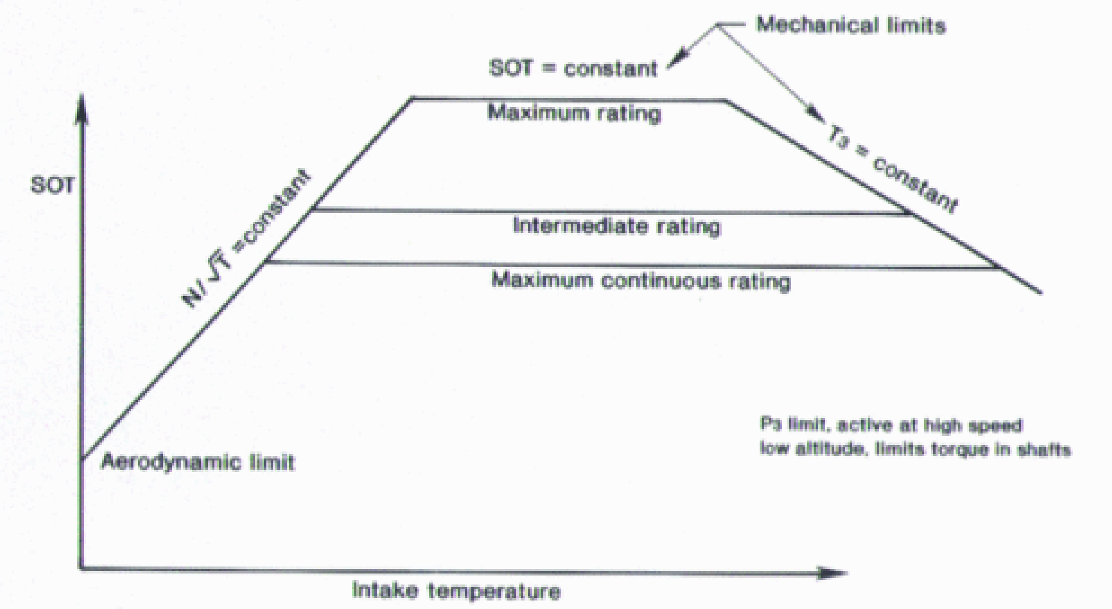
Typical civil rating system

Nowadays, civil engines are usually flat-rated on net thrust up to a 'kink-point' climate. So at a given flight condition, net thrust is held approximately constant over a very wide range of ambient temperature, by increasing (HP) turbine rotor inlet temperature (RIT or SOT). However, beyond the kink-point, SOT is held constant and net thrust starts to fall for further increases in ambient temperature. Consequently, aircraft fuel load and/or payload must be decreased.

Usually, for a given rating, the kink-point SOT is held constant, regardless of altitude or flight speed.

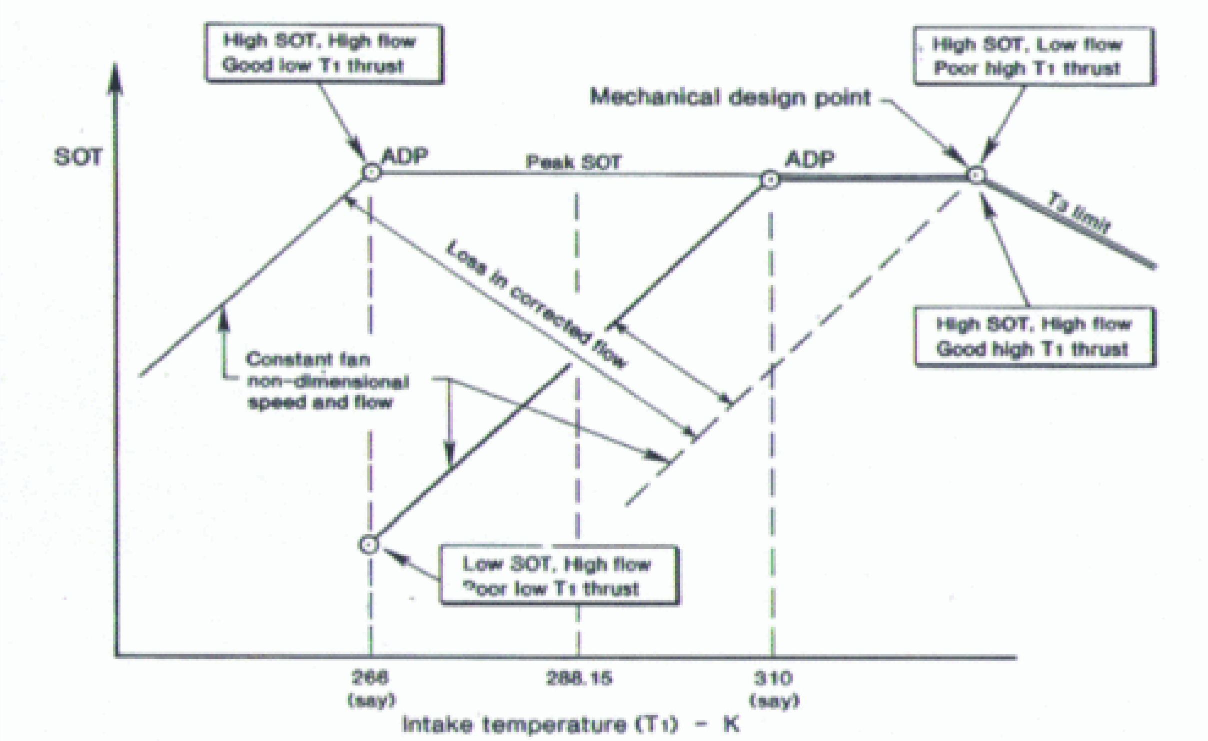
Some engines have a special rating, known as the 'Denver Bump'. This invokes a higher RIT than normal, to enable fully laden aircraft to Take-off safely from Denver, CO in the summer months. Denver Airport is extremely hot in the summer and the runways are over a mile above sea level. Both of these factors affect engine thrust.

Military



Typical military rating system

The rating systems used on military engines vary from engine to engine. A typical military rating structure is shown on the left. At low intake temperatures, the engine tends to operate at maximum [corrected speed]() or [corrected flow](). As intake temperature rises, a limit on (HP) turbine rotor inlet temperature (SOT) takes effect, progressively reducing corrected flow. At even higher intake temperatures, a limit on compressor delivery temperature (T_3) is invoked, which decreases both SOT and corrected flow.



Impact of design intake temperature

The impact of design intake temperature is shown on the right hand side.

An engine with a low design T_1 combines high corrected flow with high rotor turbine temperature (SOT), maximizing net thrust at low T_1 conditions (e.g. Mach 0.9, 30000 ft, ISA). However, although turbine rotor inlet temperature stays constant as T_1 increases, there is a steady decrease in corrected flow, resulting in poor net thrust at high T_1 conditions (e.g. Mach 0.9, sea level, ISA).

Although an engine with a high design T_1 has a high corrected flow at low T_1 conditions, the SOT is low, resulting in a poor net thrust. Only at high T_1 conditions is there the combination of a high corrected flow and a high SOT, to give good thrust characteristics.

A compromise between these two extremes would be to design for a medium intake temperature (say 290 K).

As T_1 increases along the SOT plateau, the engines will throttle back, causing both a decrease in corrected airflow and overall pressure ratio. As shown, the chart implies a common T_3 limit for both the low and high design T_1 cycles. Roughly speaking, the T_3 limit will correspond to a common overall pressure ratio at the T_3 breakpoint. Although both cycles will increase throttle setting as T_1 decreases, the low design T_1 cycle has a greater 'spool-up' before hitting the corrected speed limit. Consequently, the low design T_1 cycle has a higher design overall pressure ratio.

Nomenclature

- A flow area
- $A_{8\mathrm{calc}}$ calculated nozzle effective throat area
- $A_{8\mathrm{despt}}$ design point nozzle effective throat area
- $A_{8\mathrm{geometricdesign}}$ nozzle geometric throat area
- C_{pc} specific heat at constant pressure for air
- C_{pt} specific heat at constant pressure for combustion products
- C_{dcalc} calculated nozzle discharge coefficient
- C_x thrust coefficient
- g acceleration of gravity
- F_g gross thrust
- F_n net thrust
- F_r ram drag
- J mechanical equivalent of heat
- M flight Mach number
- N_{cor} compressor corrected shaft speed
- t static pressure
- P stagnation (or total) pressure
- P_3/P_2 compressor pressure ratio
- prf intake pressure recovery factor
- R gas constant
- SFC specific fuel consumption
- RIT (turbine) rotor inlet temperature

- t static temperature
- T stagnation (or total) temperature
- T_1 intake stagnation temperature
- T_3 compressor delivery total temperature
- V velocity
- w mass flow
- $w_{4\mathrm{corcalc}}$ calculated turbine entry corrected flow
- $w_{2\mathrm{cor}}$ compressor corrected inlet flow
- $w_{4\mathrm{cordespt}}$ design point turbine entry corrected flow
- $w_{4\mathrm{corturbchar}}$ corrected entry flow from turbine characteristic (or map)
- w_{fe} combustor fuel flow
- β arbitrary lines which disect the corrected speed lines on a compressor characteristic
- θ ambient temperature/Sea Level,Standard Day,ambient temperature
- θ_T total temperature/Sea Level,Standard Day,ambient temperature
- δ ambient pressure/Sea Level ambient pressure
- ρ density
- γ_c ratio of specific heats for air
- γ_t ratio of specific heats for combustion products
- η_{pc} compressor polytropic efficiency
- η_{pt} turbine polytropic efficiency

Corrected flow

Corrected Flow is the mass flow that would pass through a device (e.g. compressor, bypass duct, etc) if the inlet pressure and temperature corresponded to ambient conditions at Sea Level, on a Standard Day (i.e. 14.696lbf/in², 518.7R).

Corrected Flow , $w\sqrt{\theta}/\delta$, can be calculated as follows, assuming Imperial Units:

$$w\sqrt{\theta}/\delta = w(\sqrt{T}/\sqrt{518.7})/(P/14.696)$$

Corrected Flow is often given the symbol ***wc*** or ***wr*** (for referred flow).

So-called Non-Dimensional Flow $(w\sqrt{T}/P$**)** is proportional to Corrected Flow:

$$w\sqrt{T}/P = w\sqrt{\theta}/\delta * \sqrt{518.7}/14.696$$

The equivalent equations for Preferred SI Units are: (101.325kPa, 288.15K)

$$w\sqrt{\theta}/\delta = w(\sqrt{T}/\sqrt{288.15})/(P/101.325)$$

$$w\sqrt{T}/P = w\sqrt{\theta}/\delta * \sqrt{288.15}/101.325$$

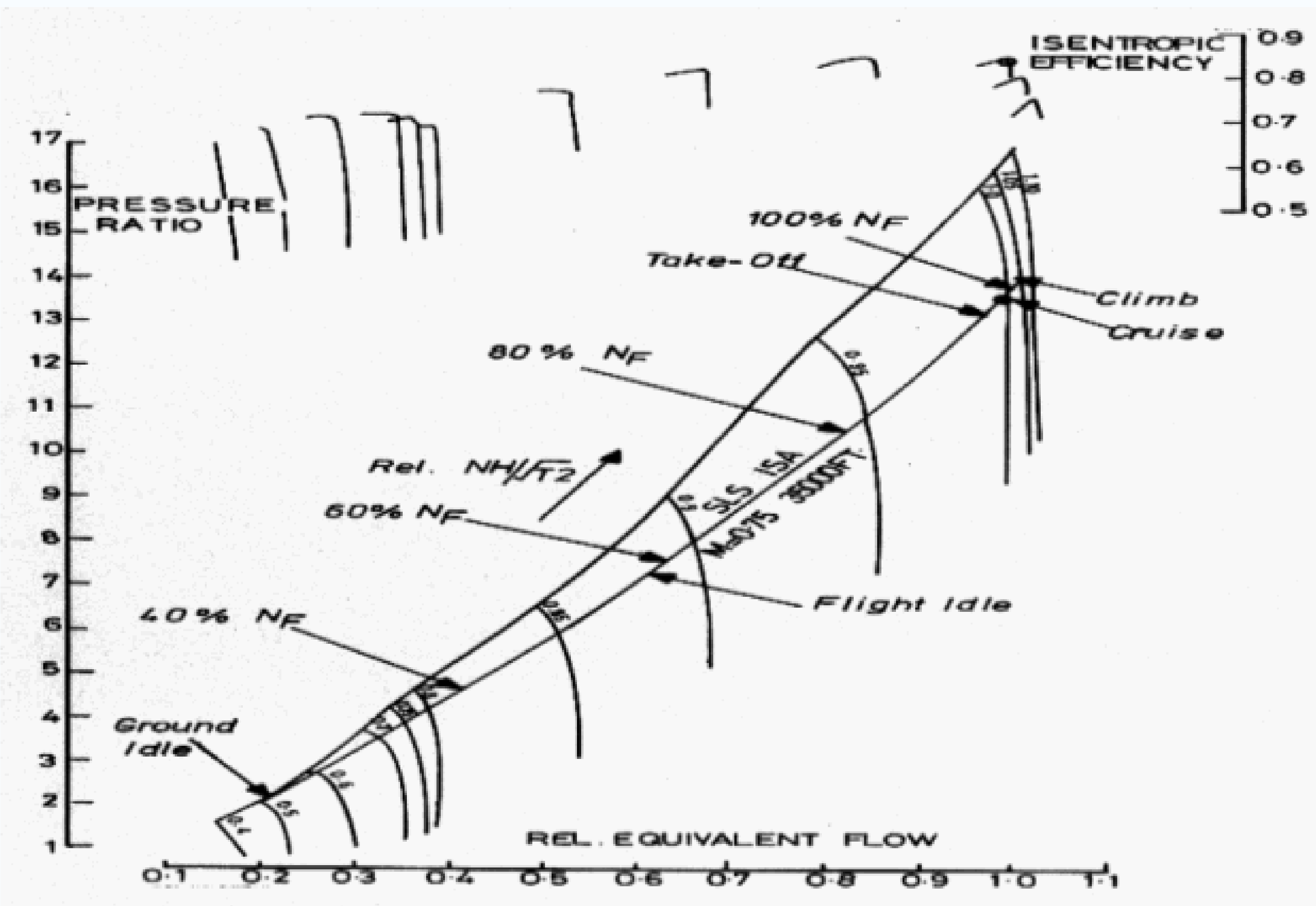
Nomenclature:

- ***P*** Stagnation (or Total) Pressure
- ***T*** Stagnation (or Total) Temperature
- ***w*** Real Mass Flow
- **δ** Referred Pressure
- **θ** Referred Temperature

In relative form, Corrected Flow, Referred Flow and Non-Dimensional Flow are all measures of axial Mach number.

Compressor map

Each compressor (or fan) in a gas turbine engine has an operating map. Complete maps are either based on compressor rig test results or are predicted by a special computer program. Alternatively the map of a similar compressor can be suitably scaled.



Typical high pressure compressor map

Compressor maps are important, since they are an integral part of predicting the performance of a gas turbine engine, both at design and off-design conditions. Fans and turbines also have operating maps, although the latter are significantly different in appearance to that of fans and compressors.

High pressure compressor map

A typical compressor map (or characteristic) for a high pressure (HP) compressor is shown above. Let us examine the various features of this map.

Flow axis

The x-axis is usually some function of compressor entry mass flow, usually corrected flow or non-dimensional flow, as opposed to real flow. This axis can be considered a rough measure of the axial Mach number of the flow through the device.

Pressure ratio axis

Normally the y-axis is pressure ratio (Pexit/Pinlet), where P is stagnation (or total head) pressure.

DeltaT/T (or similar), where T is stagnation (or total head) temperature, is also used.

Surge line

The slightly kinked diagonal line on the main (i.e. lower) part of the map is known as the surge (or stall) line. Above this line is a region of unstable flow, which is an area best avoided.

A compressor surge, typically, causes an abrupt reversal of the airflow through the unit, as the pumping action of the aerofoils stalls (akin to an aircraft wing stalling).

Surge margin

As the name suggests, surge margin provides a measure of how close an operating point is to surge. Unfortunately, there are a number of different definitions of surge margin. A popular one in use is defined as follows:

SM = 100% * (Rs -Rw) /Rw

where:

Rw ~ Pressure ratio at the working line, be it steady state or transient

Rs ~ Pressure ratio at surge, at same flow as Rw

Speed lines

The slightly curved, near vertical, lines on the main part of the map are the (constant rotational) corrected speed lines. They are a measure of rotor blade tip Mach number.

Note on the illustration that the speed lines are not distributed linearly with flow. This is because this particular compressor is fitted with variable stators, which open progressively as speed increases, causing an exaggerated increase in flow in the medium to high speed region. At low speed, the variable stators are locked, causing a more linear relationship between speed and flow.

Also note that beyond 100% flow, the speed lines close up rapidly, due to choking. Beyond choke, any further increase in speed will generate no further increase in airflow.

Efficiency axis

A sub-plot shows the variation of isentropic (i.e. adiabatic) efficiency with flow, at constant speed. Some maps use polytropic efficiency. Alternatively, for illustrative purposes, efficiency contours are sometimes cross-plotted onto the main map.

Note that the locus of peak efficiency exhibits a slight kink in its upward trend. This due to the choking-up of the compressor as speed increases, with the variable stators closed-off. The trend line resumes, once the variables start to move open.

Working line

Also shown on the map, is a typical steady state working (or operating/running) line. This is a locus of the operating points of the engine, as it is throttled.

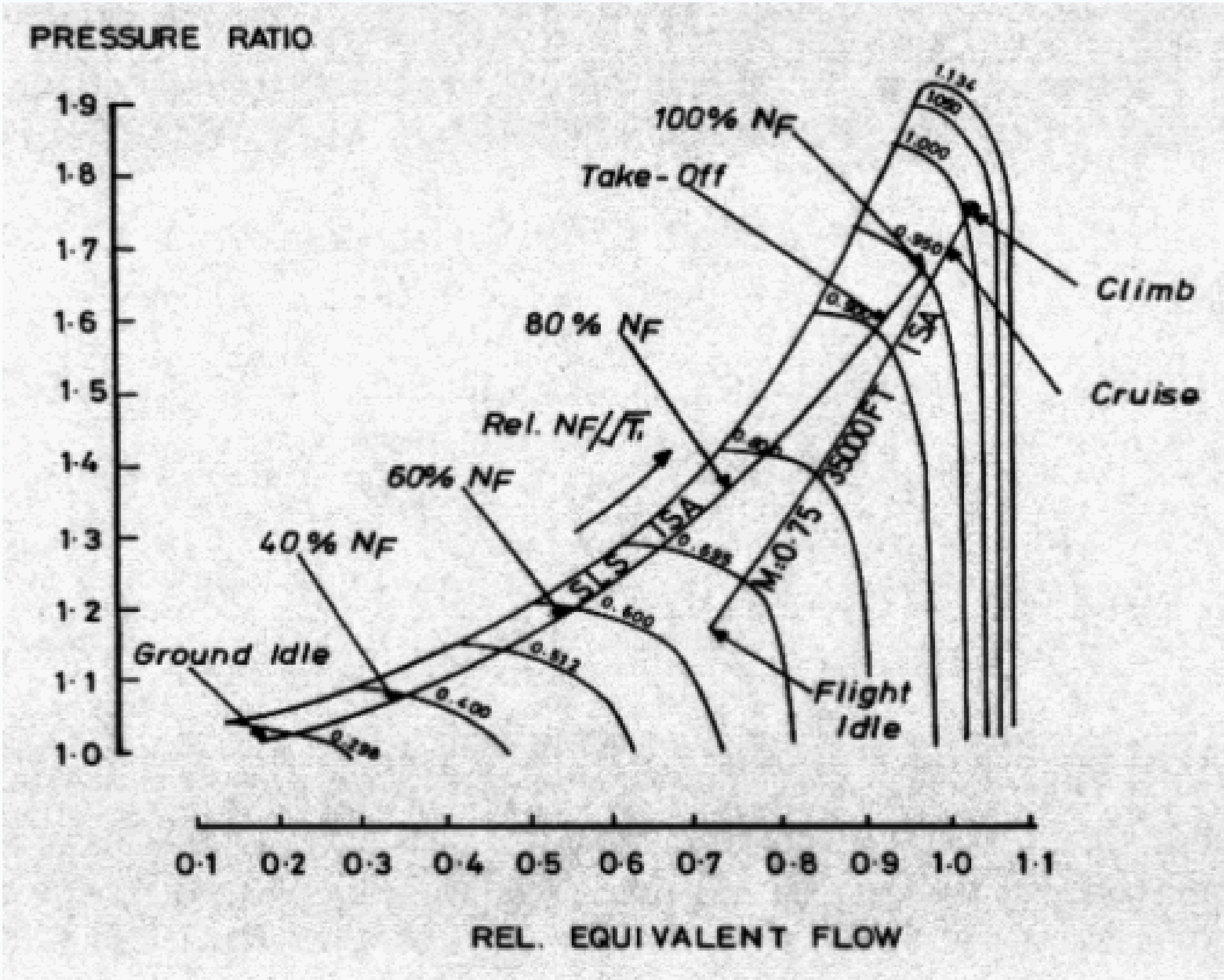
Being a high pressure ratio device, the working line is relatively shallow. If the unit had no variable geometry, there would be handling problems, because the surge line would be very steep and cross the working line at part-flow..

During a slam-acceleration from, say, a mid-throttle setting, the compressor working line will move rapidly towards surge and then slowly approach the steady state operating point, further up the map. The reverse effect occurs during a slam-deceleration. These effects are caused by the sluggish response of the spool (i.e. inertia effects) to rapid changes in engine fuel flow. Compressor surge is a particular problem during slam-accelerations and can be overcome by suitable adjustments to the fuelling schedule and/or use of blow-off (bleeding air off the compressor, for handling purposes).

In the particular example shown, a slam-accel from ground idle would cause an HP compressor surge. Opening the blow-off would help, but some changes to the variable stator schedule might also be required.

Because an HP compressor 'sees' the choked flow capacity of the HP turbine, the compressor working line is hardly affected by flight conditions. The slope of the working line approximates to a constant corrected outlet flow.

Fan map



Typical high bypass ratio fan map

As the second illustration shows, a low pressure ratio fan (such as that used on a high bypass ratio turbofan) has a range of working lines. At high flight speeds, the ram pressure ratio factors up the cold nozzle pressure ratio, causing the nozzle to choke. Above the choking condition, the working lines tend to coalesce into a unique steep straight line. When the nozzle unchokes, the working line starts to become more curved, reflecting the curvature of the nozzle characteristic. With falling flight Mach number, the cold nozzle pressure ratio decreases. Initially this has no effect upon the position of the working line, apart from the curved (unchoked) tail, which becomes longer. Eventually, the cold nozzle will become unchoked at lower flight Mach numbers, even at full throttle. The working lines will now become curved, gradually migrating towards surge as flight Mach number decreases. The lowest surge margin working line occurs at static conditions.

Owing to the nature of the constraints involved, the fan working lines of and mixed turbofan are somewhat steeper than those of the equivalent unmixed engine.

The fan map shown is for the bypass (i.e. outer) section of the unit. The corresponding inner section map typically has longer, flatter, speed lines.

Military turbofans tend to have a much higher design fan pressure ratio than civil engines. Consequently the final (mixed) nozzle is choked at all flight speeds, over most of the throttle range. However, at low throttle settings the nozzle will unchoke, causing the lower end of the working lines to have a short curved tail, particularly at low flight speeds.

However, ultra-high bypass ratio turbofans have a very low design fan pressure ratio (e.g. 1.2, on the bypass section). Consequently, even at cruise flight speeds, the cold (or mixed final) propelling nozzle is unchoked. The fan working lines become more spread-out with flight Mach number, because they move bodily up the map, towards the top right hand corner. As a result, the static working line can be well into surge.

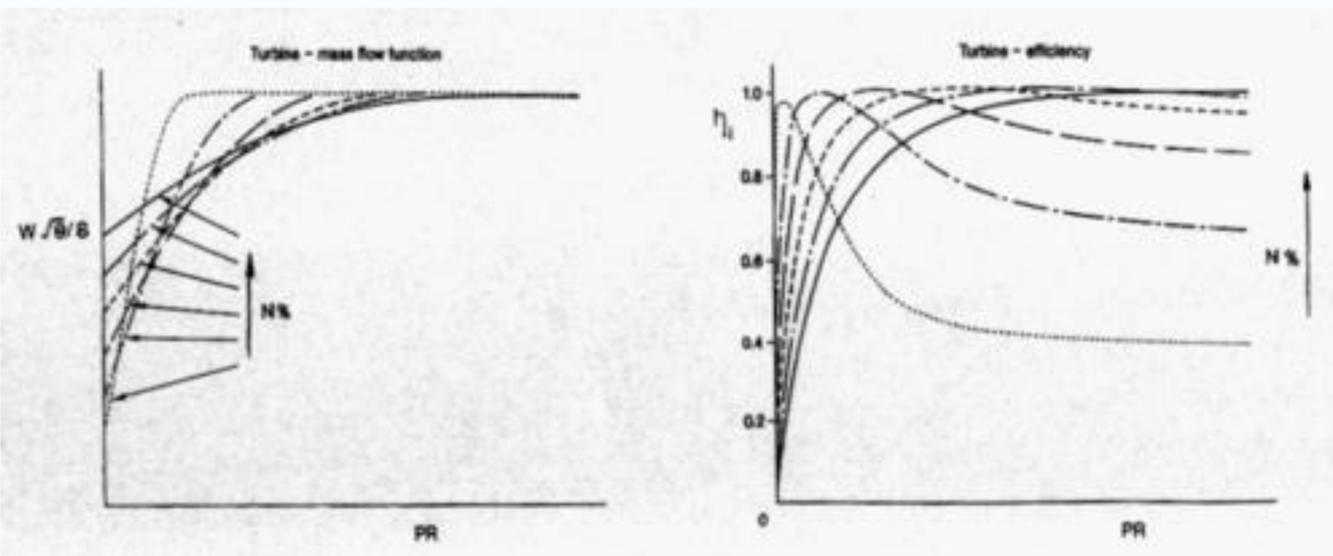
One solution is to have a variable area cold (or mixed) nozzle. Increasing the nozzle area at low flight speeds brings the fan working line away from surge.

An alternative solution is to fit a variable pitch fan. Scheduling the pitch of the fan blades has no impact upon the position of the fan working lines, but can be used to move the surge line upwards, to improve fan surge margin.

Turbine map

Introduction

Each [turbine]() in a [gas turbine]() engine has an operating map. Complete maps are either based on turbine rig test results or are predicted by a special computer program. Alternatively the map of a similar turbine can be suitably scaled.



Typical turbine map

Description

A typical turbine map is shown on the right. In this particular case, the x-axis is pressure ratio, but deltaH/T (roughly proportional to temperature drop across the unit/component entry temperature) is also often used. The other axis is some measure of flow, usually non-dimensional flow or, as in this case, [corrected flow](), but not real flow. Sometimes the axes of a turbine map are transposed, to be consistent with those of a compressor map. As in this case, a companion plot, showing the variation of isentropic (i.e. adiabatic) or polytropic efficiency, is often also included.

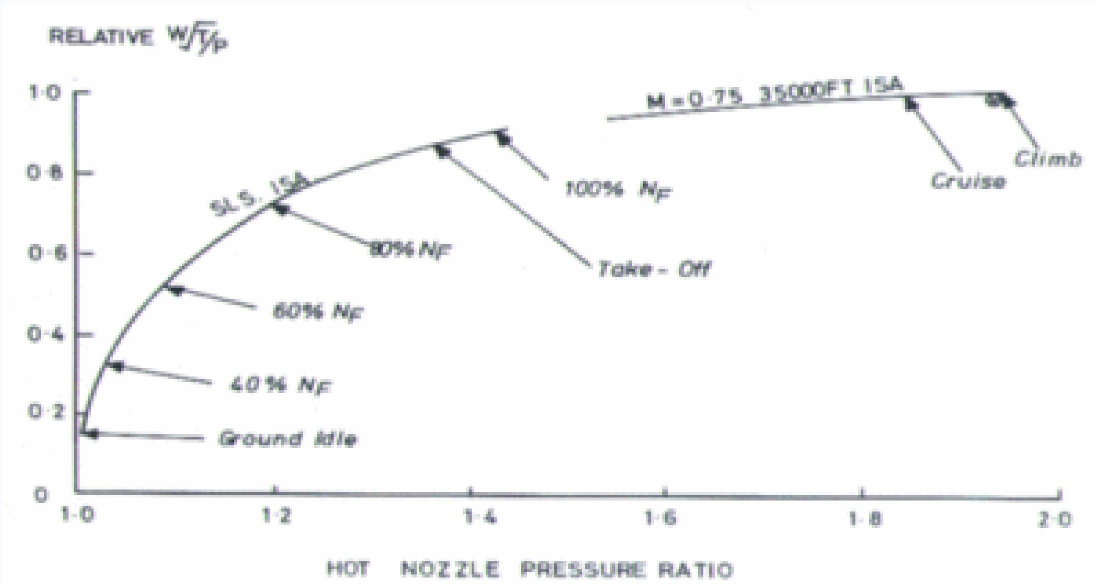
In this example the turbine is a transonic unit, where the throat Mach number reaches sonic conditions and the turbine becomes truly choked. Consequently, there is virtually no variation in flow between the corrected speed lines at high pressure ratios.

Most turbines however, are subsonic devices, the highest Mach number at the NGV throat being about 0.85. Under these conditions, there is a slight scatter in flow between the speed lines in the 'choked' region of the map, where the flow for a given speed reaches a plateau.

Unlike a compressor (or fan), surge (or stall) does not occur in a turbine. This is because the flow through the unit is all 'downhill', from high to low pressure. Consequently there is no surge line marked on a turbine map.

Working lines are difficult to see on a conventional turbine map, because the speed lines bunch-up. One trick is to replot the map, with the y-axis being the multiple of flow and speed. This separates the speed lines, enabling working lines (and efficiency contours) to be cross-plotted and clearly seen.

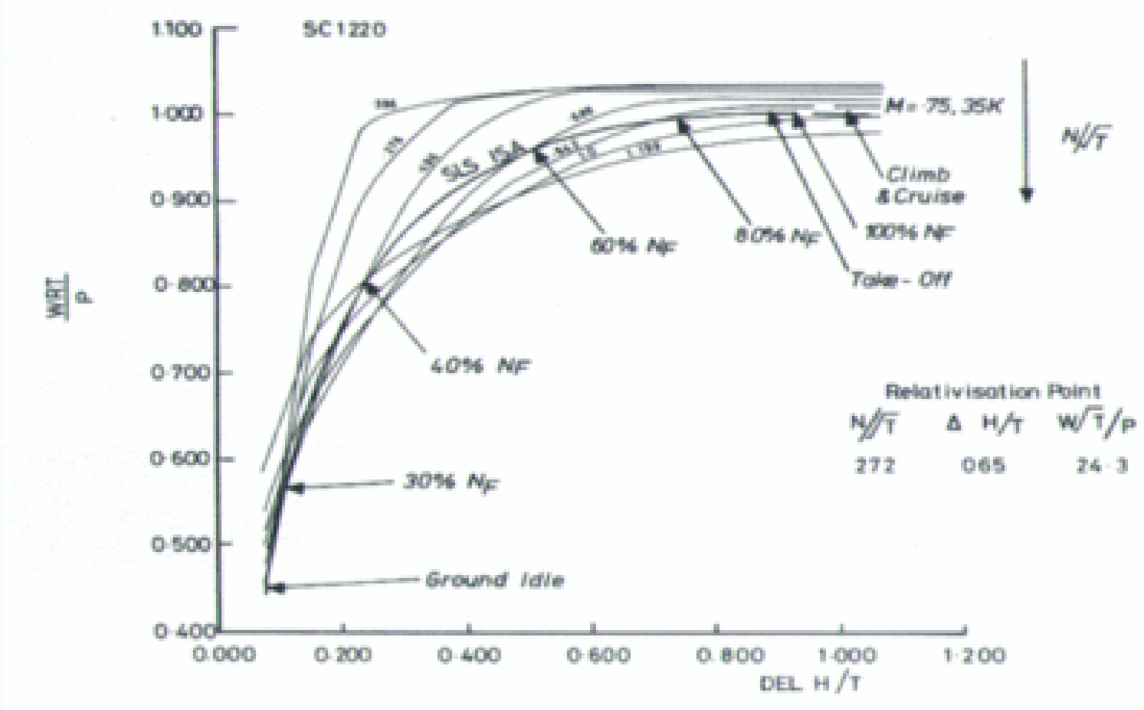
Progressive unchoking of the expansion system



Typical primary nozzle map

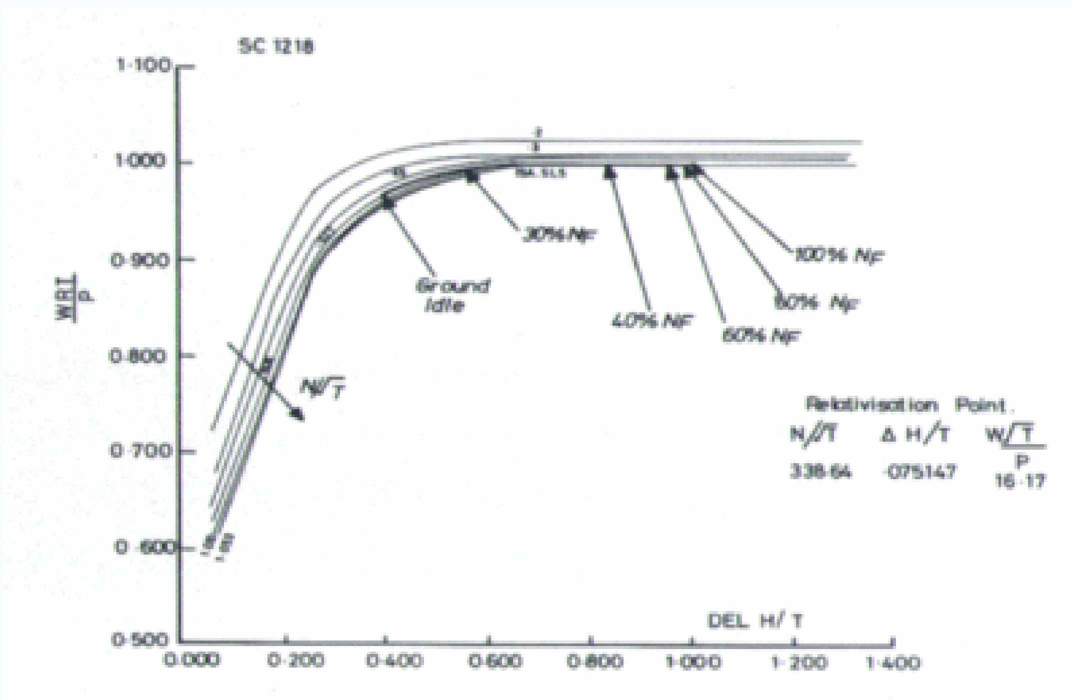
The following discussion relates to the expansion system of a 2 spool, high bypass ratio, unmixed, turbofan.

On the RHS is a typical primary (i.e. hot) nozzle map (or characteristic). Its appearance is similar to that of a turbine map, but it lacks any (rotational) speed lines. Note that at high flight speeds (ignoring the change in altitude), the hot nozzle is usually in, or close to, a choking condition. This is because the ram rise in the air intake factors-up the nozzle pressure ratio. At static (e.g. SLS) conditions there is no ram rise, so the nozzle tends to operate unchoked (LHS of plot).



Typical low pressure turbine map

The low pressure turbine 'sees' the variation in flow capacity of the primary nozzle. A falling nozzle flow capacity tends to reduce the LP turbine pressure ratio (and deltaH/T). As the left hand map shows, initially the reduction in LP turbine deltaH/T has little effect upon the entry flow of the unit. Eventually, however, the LP turbine unchokes, causing the flow capacity of the LP turbine to start to decrease.



Typical high pressure turbine map

As long as the LP turbine remains choked, there is no significant change in HP turbine pressure ratio (or deltaH/T) and flow. Once, however, the LP turbine unchokes, the HP turbine deltaH/T starts to decrease. Eventually the HP turbine unchokes, causing its flow capacity to start to fall. Ground Idle is often reached shortly after HPT unchoke.

Jet Propulsion/Mechanics

Engine ratings

Engines are certified to deliver standard thrusts depending upon atmospheric conditions. Thrust is typically measured in kN or lbs.

Maximum Takeoff thrust

This is the maximum thrust that the engine can deliver for 5 minutes at standard sea level atmosphere. Peak thrust is usually achieved when the engine is static.

Maximum Climb thrust

This is the maximum thrust that the engine can deliver for 5 minutes at standard sea level atmosphere.

Maximum Cruise thrust

The thrust allowable for unlimited flight duration at the design altitude.

Maximum Continuous thrust

Also called the maximum maneuver thrust. Sometimes is same as maximum cruise.

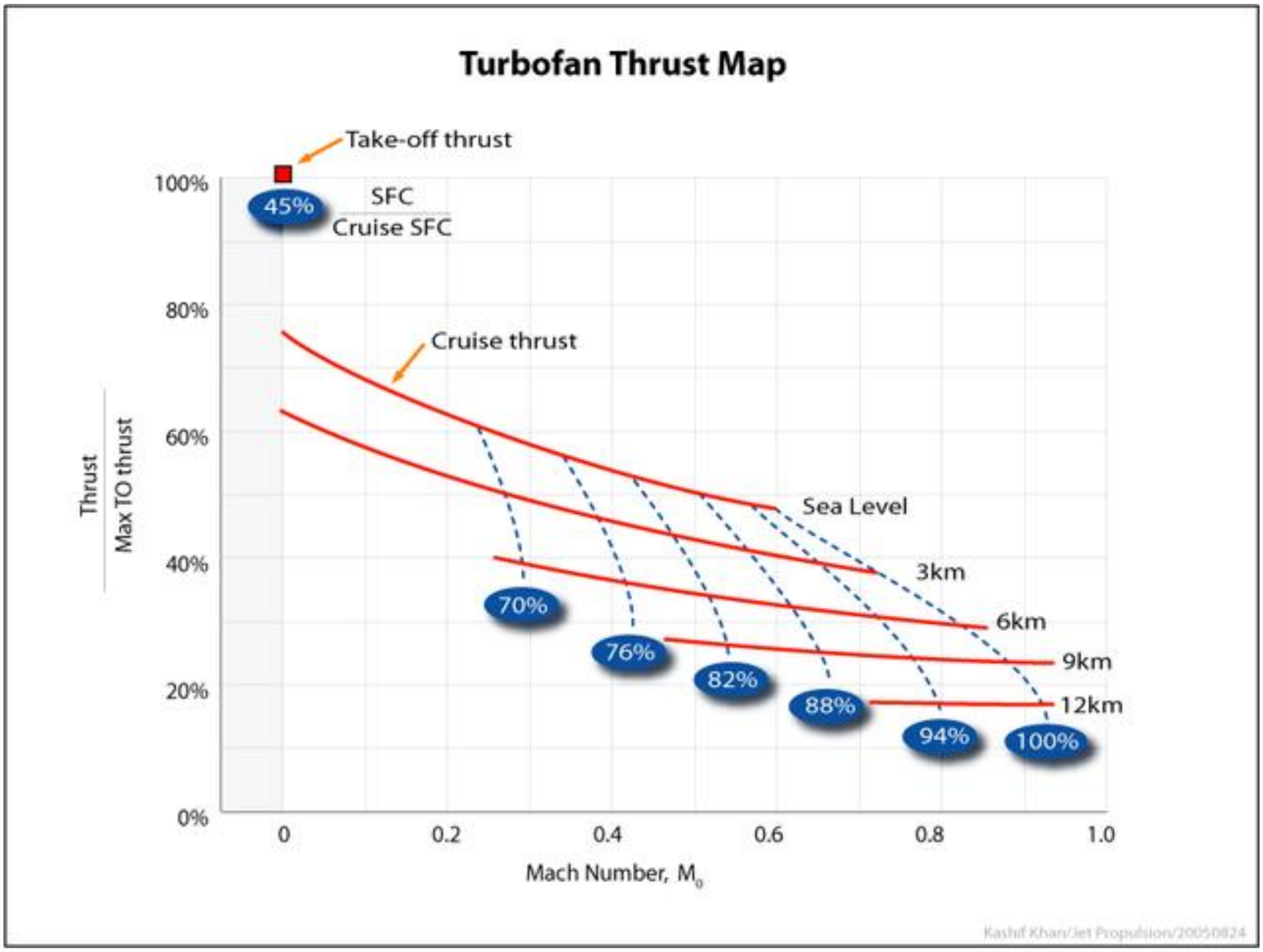
Specific fuel consumption

Typically quoted in mg/Ns for engines. Is usually much higher at cruise than at static.

A typical high bypass engine will consume about 8mg/Ns at maximum takeoff and 15mg/Ns at maximum cruise thrust.

Example ratings

The figure below shows the typical behaviour of a modern turbofan. The orange curves show maximum cruise thrust at altitude. The TO thrust is significantly higher than the cruise thrust at sea level since it is permitted for short durations only..



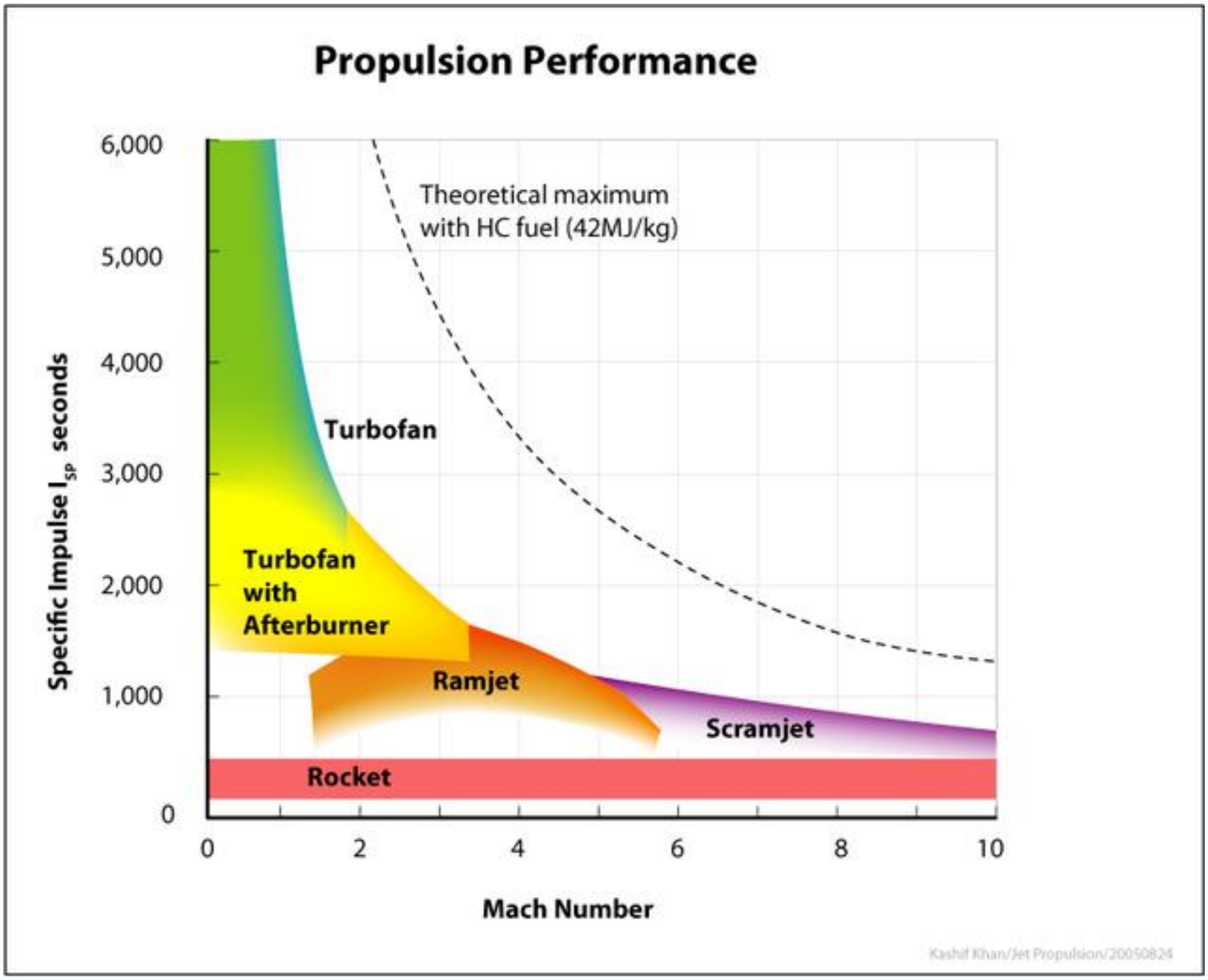
Typical large engines today have TO thrusts approaching 480kN. Typicalcruise SFC is around 15mg/Ns. A low bypass military engine will have TO sfc of about 18mg/Ns which rises to 50mg/Ns if afterburner is used.

Specific impulse

Specific impulse is defined as the thrust (N) divided by the fuel weight flow rate (N/s). The resulting measure is usually quoted in seconds and defines the weight fraction that is necessary to give a particular delta V for a rocket or range for an aircraft with a given lift to drag ratio.

For a jet engine the specific impulse can be determined from the specific fuel consumption. For example an engine that has an SFC of 15mg/N-s will have a specific impulse of 6800s. Current turbofan engines have cruise SFC below 15.

For a chemical rocket the specific impulse ranges to about 500s.



Thrust

The thrust of a jet engine is determined by the difference in momentum of the fluids flowing in and out of the engine. If the mass of fuel added is negligible then the thrust is:

$$T = \dot{m}(u_{exit} - u_{inlet})$$

Range

The Breguet Range Equation gives the range achieved by a vehicle. For a constant L/D ratio of the aircraft

$$Range = uI_{sp}\left(\frac{L}{D}\right)\ln\left(\frac{W_{initial}}{W_{final}}\right)$$

where

R = distance flown (m)

u = velocity (m/s)

I_{sp} = specific impulse (s)

L/D = lift-to-drag ratio (dimensionless)

$W_{initial}$ = gross aircraft weight at the start of cruise (kg)

W_{final} = gross weight at the end of cruise (kg)

Example

For an aircraft with 50% fuel, velocity of 600 m/s, an L/d of 10 and engines with a specific impulse of 3000 the range is:

$R = (600 m / s)(3000 s)(10)\ln(2) = 12{,}477 km$

Jet Propulsion/Thrust

Jet Propulsion/Drag

Jet Propulsion/Thermodynamic Cycles

Gas turbines are based on the Brayton cycle.

All jet engines and gas turbines are heat engines that convert thermal energy into useful work. The useful work may be in the form of mechanical power, as from a shaft which may be used to drive a propeller, a vehicle, a pump, an electric generator, or any other mechanical device. In Jet engine applications the work is in producing compressed air and combustion products which are then accelerated to provide reaction propulsion.

The thermal efficiency for a shaft application is calculated using the ratio of output mechanical energy divided by input thermal energy. For propulsive applications the definition of efficiency involves the velocity of the jet through the air since a stationary engine anchored to the ground produces no propulsive work. In such a case a propulsive efficiency is defined which relates the propulsive work delivered to the total mechanical energy produced by the engine.

Gas turbine energy exchange

Brayton Cycle

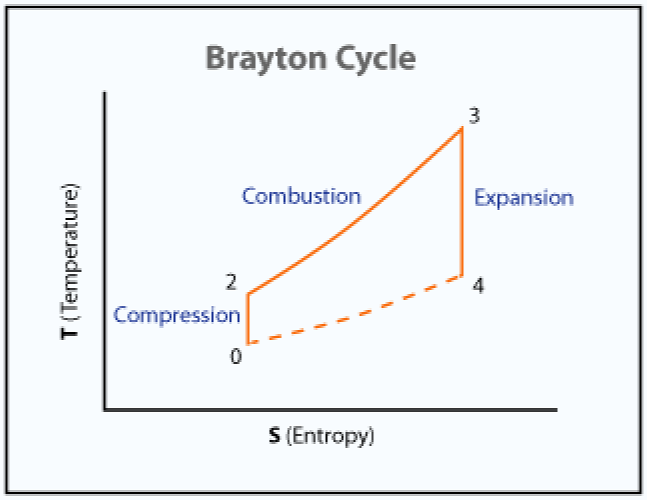
The brayton cycle consists of

0-2: isentropic compression

2-3: constant pressure heating

3-4: isentropic expansion

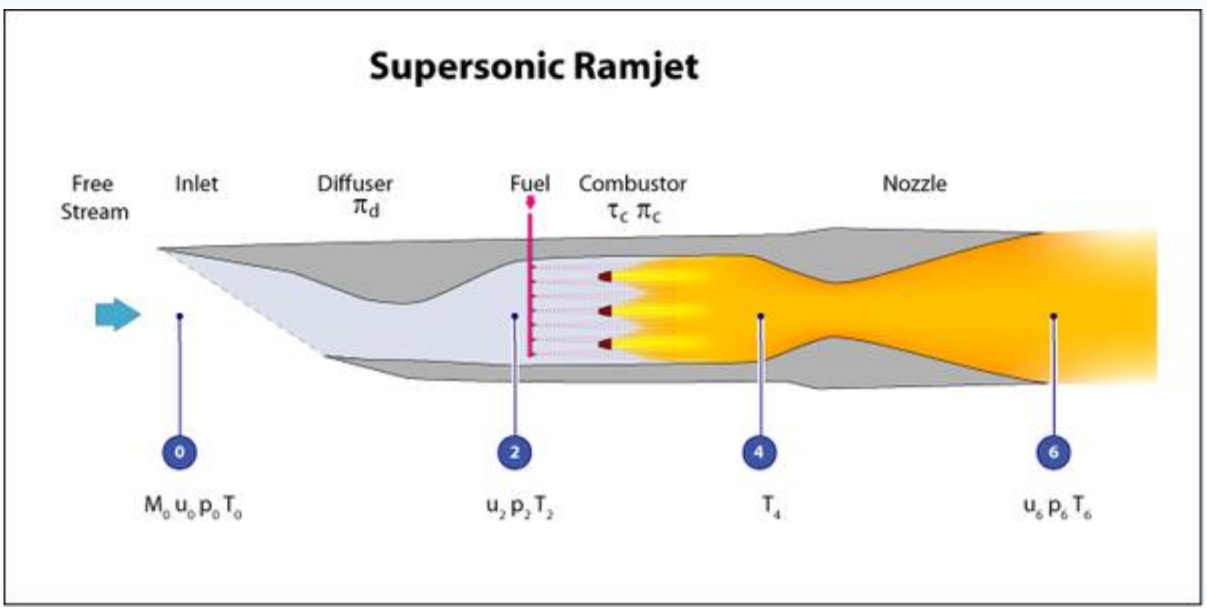
4-0: constant pressure cooling (absent in open cycle gas turbines)



Ramjet cycle

A ramjet uses the open Brayton cycle. In the diagram below a 2D supersonic intake is shown downstream of which is a divergent subsonic diffuser. Fuel is then injected into the compressed air and evaporates producing a mixture that is ignited when it reaches the flame front. The flameholders provide the turbulent circulation necessary to stabilize the flame, since deflagration velocities are usually much smaller (<10m/s) than the average velocity of air in the combustor. The combustion products are then exhausted through the nozzle.

To understand how thrust is produced if we assume that the flow of fuel is negligible compared to the air mass then the exhaust flow will be at approximately the same Mach number as the input flow. However the total temperature of the exhaust is much higher and the exit velocity will be correspondingly higher than the input velocity. This difference in velocity (and momentum) produces thrust.



The temperature rise in the intake-diffuser is related to the freestream Mach number M_0:

$$\frac{T_2}{T_0} = 1 + \frac{\gamma - 1}{2} M_0^2$$

Maximum efficiency is reached if temperature rise in combustor is small.

$$\eta = \frac{[(\gamma - 1)/2] M_0^2}{1 + [(\gamma - 1)/2] M_0^2}$$

where $\gamma = c_p/c_v \approx 1.4$ is the ratio of specific heats of air.

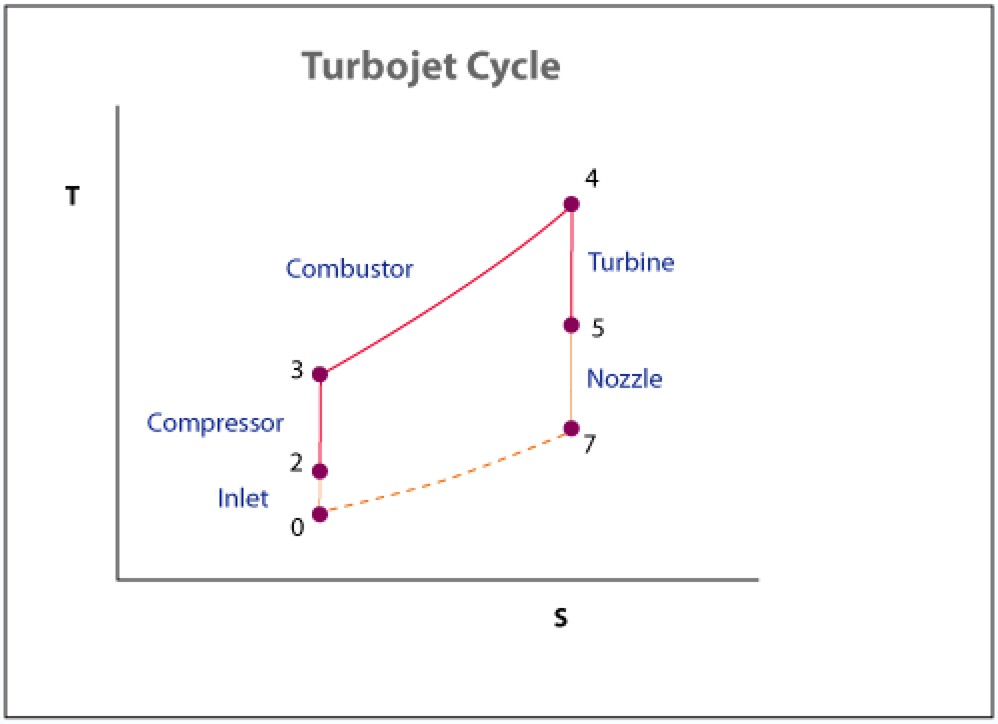
Ramjets are inefficient at subsonic speeds and their efficiency improves at supersonic speeds.

At hypersonic speeds the compression and dissociation processes make full diffusion unattractive and supersonic combustion is being researched. Scramjet slow the air down to low supersonic speeds and then burn high flame velocity fuels such as hydrogen or methane to try to get net thrust.

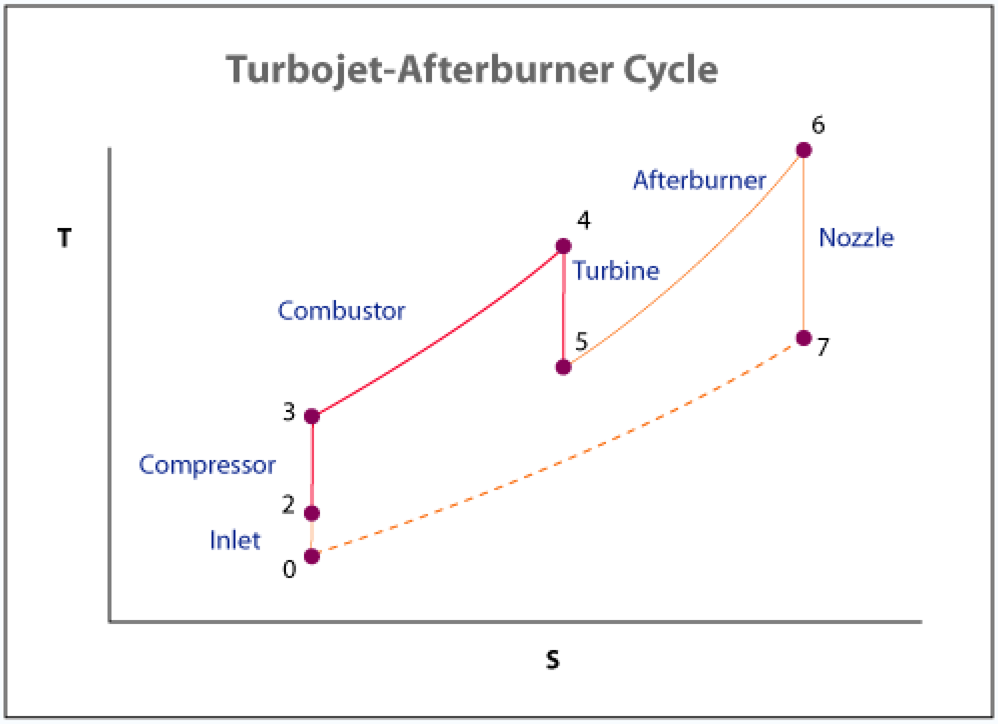
Turbojet cycle

Adding a compressor to a ramjet powered by a turbine in the exhaust allows increased combustor inlet temperature, and a consequent increase in possible thermal efficiency. The turbine however is limited in the temperature it can handle, so maximum power is also limited.

In the T-S diagram below the presence of the compressor allows us to raise the combustor inlet temperature (3). The raising of the combustor segment increases the cycle area and the thermal efficiency.



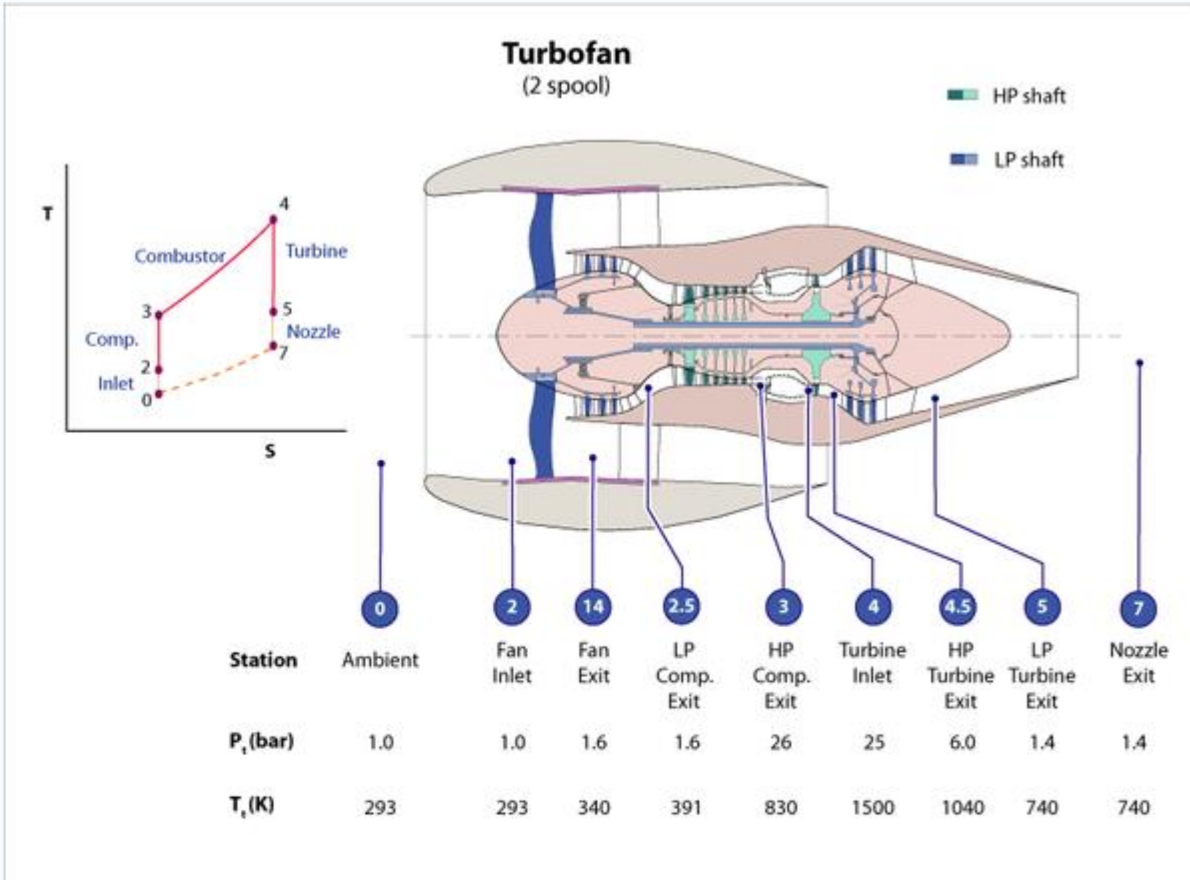
Addition of an afterburner (5-6) allows thrust augmentation as can be seen from the increased area of the diagram shown below. The afterburner operates in the higher entropy range and has lower efficiency than the base turbojet.



Turbofan cycle

A turbofan diverts some of the pressure energy of the core flow to power a fan which moves a larger mass flow, providing an increase in thrust and propulsive efficiency.

Turbofans normally have two or three shafts. Since the diameter of the fan is larger the same tip speed can be achieved at a lower rpm than the smaller diameter compressor and two shafts become necessary. The alternate method is to employ a gearbox to step down the shaft speed which is used in some smaller turbofans. In most turbofans however a multistage LP turbine is used to extract the same energy with smaller stage loadings and lower tangential velocity. The smaller diameter HP compressor is run with one or two turbine stages with higher tangential velocity than the LP turbine.



Jet Propulsion/1D Analysis

Ramjet

The thrust produced by a ramjet is

$$T = \dot{m}u_{inlet}(\sqrt{\tau_b} - 1)$$

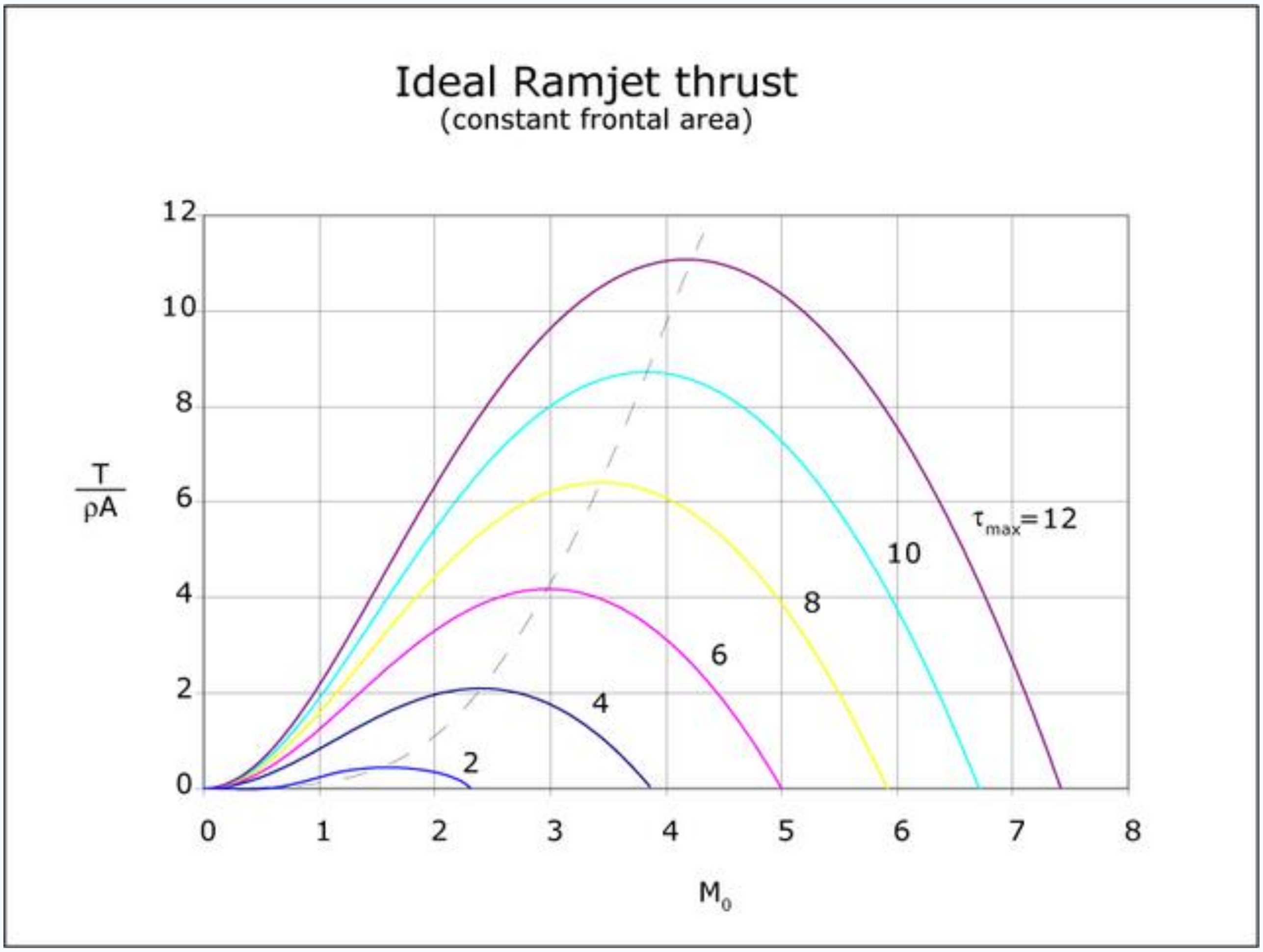
where τ_b is the the total temperature ratio produced by the combustor. Metallurgy and the availability of cooling will limit the maximum temperature that can be sustained in the combustor. We can define τ_{max} as the maximum total temperature ratio compared to the inlet conditions and τ_c as the temperature ratio relating the static and total temperatures. Then

$$\tau_{max} = \tau_b\tau_c = \tau_b\left(1 + \frac{\gamma - 1}{2}M_0^2\right)$$

Thus as τ_c increases with speed for a fixed maximum temperature τ_{max} stays constant and τ_b must reduce. If the theoretical frontal area of the ramjet is constant then the mass flow through the ramjet will increase linearly with the Mach number. At the same time the heat added diminishes. The thrust then is:

$$T = A\rho(M_0 a_0)^2 \left[\sqrt{\frac{\tau_{max}}{1+\frac{\gamma-1}{2}M_0^2}} - 1\right]$$

When $\tau_{max}=\tau_c$ the term in brackets goes to zero and the thrust vanishes. The thrust for a given τ_{max} is shown in the following figure (the dashed line is the peak thrust):



Jet Propulsion/Aerodynamics

The stagnation or total temperatures and pressures are needed to measure the energy additions in high speed gas flows that occur in gas turbines. Using the Mach number allows us to factor in the compressibility of the gas.

Basic principles

Stagnation temperature

Stagnation temperature is the temperature of the gas if it is brought to rest adiabatically. Adding the kinetic energy to the internal energy of the gas we get the relation

$$c_p T_t = c_p T + \frac{u^2}{2}$$
$$T_t = T + \frac{u^2}{2c_p}$$

where T_t is the total(stagnation temperature of the flow. The Mach number of the flow is

$$M = \frac{u}{a} = \frac{u}{\sqrt{\gamma RT}}$$

Substituting

$$T_t = T + \frac{\gamma RTM^2}{2c_p}$$

since $R = c_p - c_v$ and $\gamma = c_p / c_v$

$$T_t = T(1 + \frac{\gamma M^2}{2}(c_p - c_v)/c_p)$$
$$\frac{T_t}{T} = \left(1 + \frac{\gamma - 1}{2} M^2\right)$$

This is the temperature if the gas is brought to rest adiabatically.

Stagnation pressure

If the deceleration is isentropic then the stagnation pressure is:

$$\frac{p_t}{p} = \left(\frac{T_t}{T}\right)^{\gamma/(\gamma-1)}$$

Duct flow

A steady inviscid adiabatic quasi-one dimensional flow obeys the following equations:

Differential continuity equation

d (ρ u A) =0

Differential momentum equation

dp=-ρ u du

Differential energy equation

dh + u du=0

Rearranging continuity

$$\frac{d\rho}{\rho} + \frac{du}{u} + \frac{dA}{A} = 0$$

Rewrite momentum equation

$$\frac{dp}{\rho} = \frac{dp}{d\rho}\frac{d\rho}{\rho} = -udu$$
$$\frac{dp}{d\rho} \equiv \frac{\partial p}{\partial \rho}$$

The velocity of sound is:

$a = (dp / d\rho)^{1/2}$

Rearranging and substituting:

$a^2 = (dp / d\rho)$
$a^2\, d\rho / \rho = -u\, du$
$$\frac{d\rho}{\rho} = -\frac{udu}{a^2} = -\frac{u^2}{a^2}\frac{du}{u} = -M^2\frac{du}{u}$$

Substituting into continuity equation

$$M^2\frac{du}{u} - \frac{du}{u} - \frac{dA}{A} = 0$$

We get the area velocity equation:

$$\frac{dA}{A} = (M^2 - 1)\frac{du}{u}$$

Thus for acceleration (positive *du/u*) the area must decrease for Mach numbers below 1 and increase for Mach numbers above 1.

The relationship between Mach number and duct area related to the throat area A* is:

$$\left(\frac{A}{A^*}\right)^2 = \frac{1}{M^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{\frac{\gamma+1}{\gamma-1}}$$

The temperature relation is

$$\frac{T}{T_t} = \left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{-1}$$

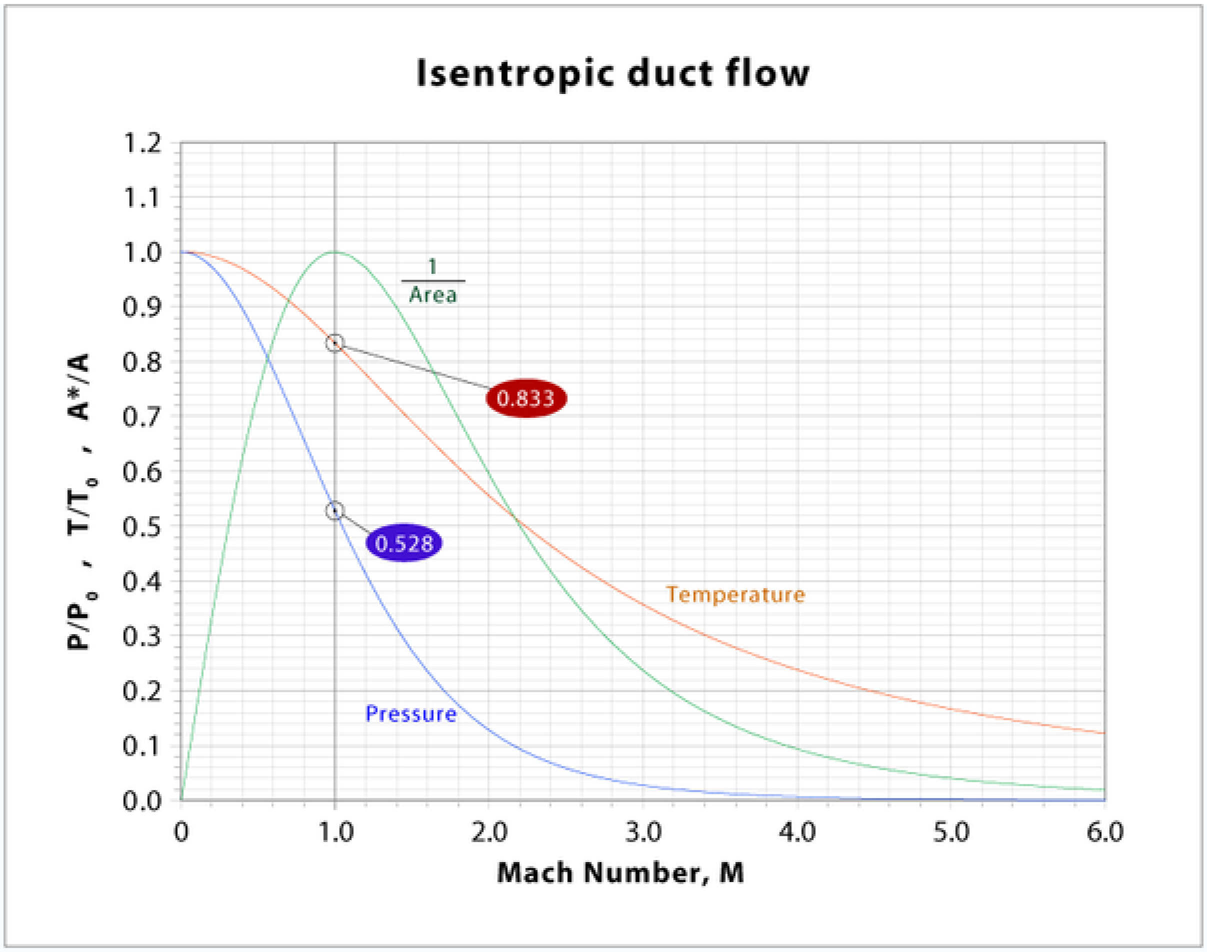
the pressure relation

$$\left(\frac{p}{p_t}\right)^2 = \left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{-\frac{\gamma}{\gamma-1}}$$

and the density relation

$$\left(\frac{\rho}{\rho_0}\right)^2 = \left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{-\frac{1}{\gamma-1}}$$

The figure below shows the realtionships for γ of 1.4.



[[1]]

Mass flow

The figure above shows this exchange for a fluid with γ=1.4 undergoing an adiabatic expansion. Sonic velocity (Mach 1) is achieved when the pressure drops to 0.528 and the area for a particular mass flow is minimum at this Mach number. The flow at this condition is said to be choked and any further reductions in duct area will not produce acceleration of the stream. The mass flow per unit area is

$$\dot{m} = A\rho v$$

$$\dot{m} = A\rho_0 \left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{-\frac{1}{\gamma-1}} V$$

Diffusers

Diffuser pressure recovery

An ideal diffuser would recover the stagnation pressure, but practical diffusers cannot bring the fluid velocity to zero and have losses. The pressure recovered by such a diffuser is:

$\pi_d = p_{t2} / p_{t0}$

Shocks

The total temperature across a shock remains constant but the total pressure is lost.

Normal shock

The Mach number M_2 after the shock is:

$$M_2^2 = \frac{1 + [(\gamma - 1)/2]M_1^2}{\gamma M_1^2 - (\gamma - 1)/2}$$

The density & velocity relation

$$\frac{\rho_2}{\rho_1} = \frac{u_1}{u_2} = \frac{(\gamma + 1)M_1^2}{2 + (\gamma - 1)M_1^2}$$

the pressure relation

$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma + 1}(M_1^2 - 1)$$

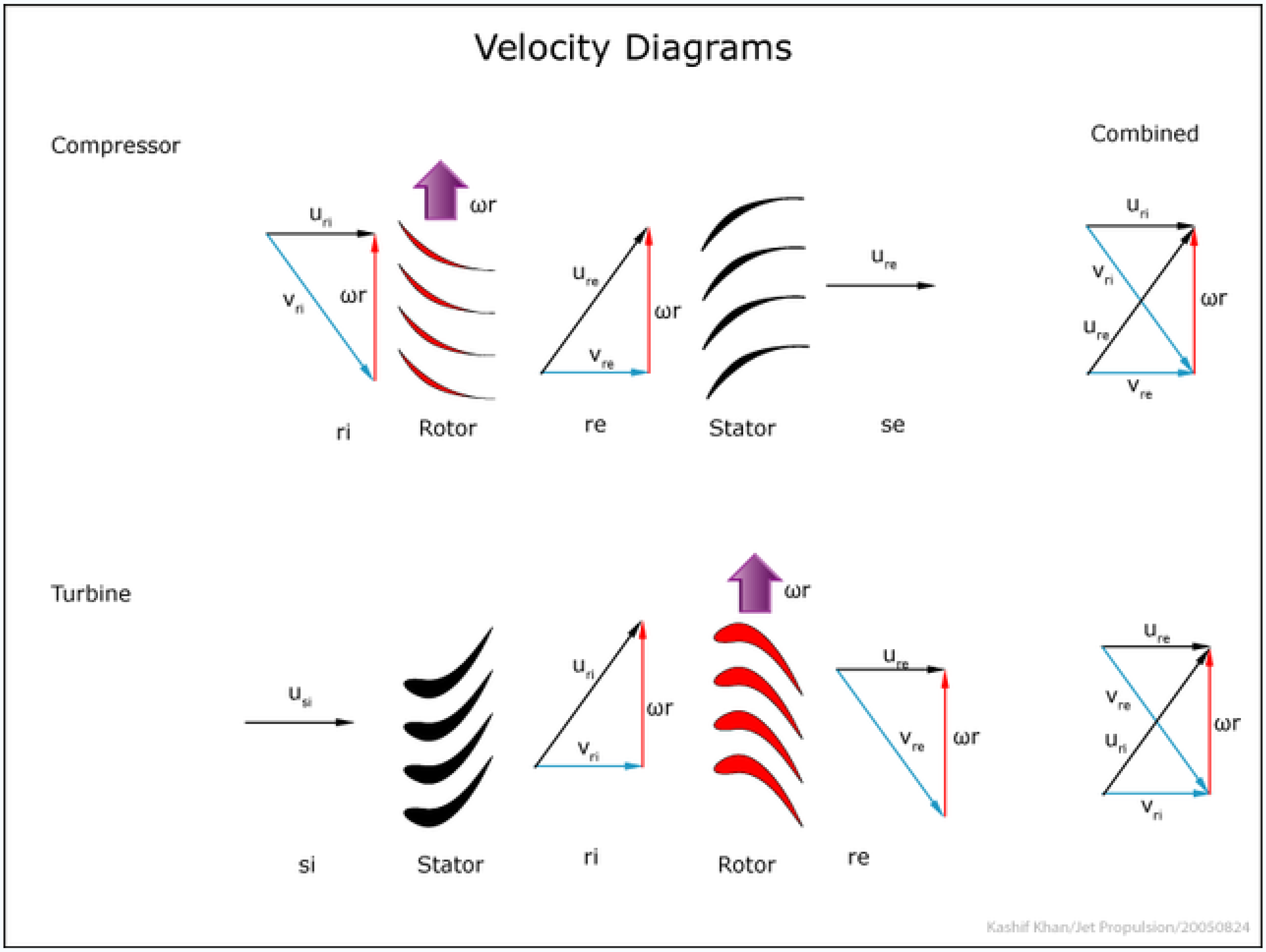
and the temperature relation

$$\frac{T_2}{T_1} = \frac{h_2}{h_1} = \left[1 + \frac{2\gamma}{\gamma + 1}(M_1^2 - 1)\right] \frac{2 + (\gamma - 1)M_1^2}{(\gamma + 1)M_1^2}$$

Inclined shock

Velocity triangles

Basic analysis of the effect of the blade rows on the airflow can be done through velocity triangles. The figure below shows a basic set of velocity triangles for compressor and turbine rows.



In the compressor the airflow is decelerated (diffused) in the velocity frame of the blade cascades. If the velocity of the inflow is u in the static frame then the rotor blades see the vector sum of u and ωr which is the velocity of the blades whirling at angular speed ω at that particular radius r. The rotor blades turn the flow incoming in the rotor frame of reference at velocity v_{ri} and diffuse it down to velocity v_{re}. The stator sees the incoming flow at u_{re} and diffuses it down to u_{se} and turns it back to the axial direction. The typical axial Mach number is around 0.6 and the rotor angular Mach number (ωr/a) is kept as high as possible to maximize the compression per stage. Since the fundamental gas dynamic process in the compressor is subsonic and supersonic diffusion, the limitations are imposed by the boundary layers and adverse pressure gradients which amplify them. The ultimate limit of compression is when the boundary layer diverges on the suction side of the blades and the blade row stalls.

The velocity triangles for compactness can be combined as shown on the right side since the angular velocity of the blade row is equal at inlet and exit.

In the turbine the airflow is accelerated (nozzled) in the velocity frame of the blade cascades. The incoming flow is accelerated and turned by the stators (nozzle vanes) to velocity u_{ri} and directed towards the turbine rotor. In the rotor frame of reference the flow comes in axially (or nearly so) at velocity v_{ri} and is accelerated and turned to velocity v_{re}. The exit velocity u_{re} then is nearly axial once the angular velocity ωr of the turbine blades is subtracted.

The turbine's degree of reaction is the kinetic energy that occurs in the turbine rotor compared to the total kinetic change. The triangles above describe an approximately 50% reaction turbine. Impulse turbines have the rotor frame velocities v_{ri} and v_{re} changing only in direction indicating that inlet and exit incidence angles of the blades are equal in magnitude if the axial velocity does not change.

Axial Compressors

Axial compressors are typically designed numerically since the flows in them are highly complex and three dimensional. The compression produced by the stage is determined by the tangential Mach number. The flow through the compressor depends on the axial Mach number and the area of the annulus. Improvements in design of the blades have allowed relative Mach numbers of 1.5 being achieved at the tip of the fan. The hub may be half the tip radius of the blades and the tangential velocity can vary by a factor of 2. Blades which operate supersonically for part of their span are called transonic.

Solidity is the ratio of the chord of the blades to the tangential distance subtended by the blade. Aspect ratio relates the blade chord to the blade length. The modern trend is towards lower aspect ratios. Higher aspect ratio blades tend to be lighter and blade loss is slightly less catastrophic an event. They often have part span shrouds to prevent flutter. Wide chord blades have recently been engineered to provide better performance, since they allow higher pressure ratios to be achieved. The width of the blade allows for a better shock structure in the supersonic regions of the blade, and a lower pressure gradient that delays separation. They can also avoid part span shrouds since their torsional rigidity is higher.

The blades act like staggered airfoils and they can tolerate a few degrees of incidence before the loss factor diverges. The loss factor is defined as the loss in total pressure divided by the dynamic pressure of the incident flow (p_t-p). The minimum loss factor ranges from about 0.02 increasing with the inlet Mach number.

The compressor blade rows perform diffusion in reducing the velocity difference while increasing pressure.

Blade profiles

The earliest compressors employed circular arc blades. Double circular arcs have also been used, while modern compressors use more sophisticated 3D CFD designed blades.

Subsonic blades

For subsonic blades the passage widens as the air goes through it and simultaneously turns. The blades have higher inclination on the leading edge relative to the axial direction which reduces at the trailing edge resulting in a widening channel through which air must flow. The convex (surface) surface presents a large adverse pressure gradient which tends to enlarge the boundary layer. If the adverse gradient exceeds a critical level then flow separation and blade stalling occurs.

Aspirated compressor blades evacuate the suction side boundary layer and allow for larger diffusion.

Transonic blades

Fan blades are typically transonic. The incident flow approached the blade at supersonic velocity. The initial diffusion happens through a converging wedge shaped passage that creates multiple inclined shocks terminated by a stronger normal shock in the passage that makes the flow subsonic. The subsonic flow is then further diffused by a diverging passage as in subsonic blades.

Supersonic blading is easy to see in the outer periphery of the fans of commercial airliners. The leading edges are sharp and appear to be curved slightly in the opposite direction to create the supersonic wedge. The incident flow while highly 3 dimensional is qualitatively comparable to the flow into a 2D intake of a supersonic aircraft such as the F-15.

For compressor stages it is advantageous to bring the flow subsonic by the use of variable stators. Most large modern engines have variable stators that allow subsonic blading to be used while providing good performance throughout the operational envelope. The variable stator adds swirl to the flow so that the Mach number variation between root and tip is reduced preventing stall at the root of the blades.

Multistage compressors

For multiple compressor stages on a shaft the inlet Mach number progressively drops as the air is compressed and heats up.

Mass flow

The mass flow in a duct is maximum if the Mach number is close to unity. The axial Mach number through the engine is kept close to one to reduce the blade heights. The blockage introduced by the hub and casing boundary layers, as well as the cross sectional area of the blades reduces the mass flow below the theoretical value. Actual axial Mach numbers range up to 0.6.

Loss mechanisms

Real compressors suffer from various loss mechanisms.

Tip leakage

Hub and casing boundary layers

Seal leakages

TBD

Stage performance

The corrected speed of the engine is defined as

Corrected speed $= \dfrac{N}{(T_{t2}/T_t}$

Off design behavior

A multistage compressor operating at speeds lower than designed or with lower pressure ratio than designed, will load the front stages more than the rear stages. This can result in stalling of the front stages. Variable stators and multiple shafts can be sued to solve this. Most modern turbofans have multiple shafts as well as variable stators in the front compressor stages. The variable stators balance the compression between the front and rear stages at off design conditions.

Matching

The compressor and turbine flows are “matched” to provide sufficient flow through the turbine, as well as sufficient power at the right rpm for the compressor. The temperature increase in the combustor allows us to calculate the relative areas required

Transients

Surge

To accelerate and engine the fuel added in the combustor is increased. This increases the temperature and the pressure in the combustor which now has to be generated by the compressor. If the compressor is too close to stall a surge may happen where the compressor stalls. In extreme cases the flow is reversed through the compressor and the hot combustor gases exit the front of the compressor. The cycle then continues at the Helmholtz frequency of the system till the disturbance is damped out. The engine controller is tasked with ensuring that the compressor doesn’t reach the surge line during acceleration.

In deceleration the fuel quantity is decreased and if the flame becomes too lean a flameout may occur.

Rotating stall

Unsteady flows in the compressor may cause some sections to stall (stall cells). These rotate with the blades and propagate from blade to blade, possibly exciting vibrations that can cause damage. The rotating stall precedes a full scale surge in which the compressor stalls in the entire circumference.

Jet Propulsion/Engine ratings

Engines are certified to deliver standard thrusts depending upon flight conditions. Thrust is typically measured in kN or lbs. A 'rating' is a predefined power setting that the pilot can select which may be appropriate for particular flight conditions. Rating terminology differs between civil and military aircraft, reflecting the different requirements of these types of aviation.

Civil Aircraft Ratings

The following ratings are typical of commercial airliners. The aircraft/engine manufacturer will have to declare two principal ratings to the certifying authorities, since these define the safe limits of operation of the engine/aircraft - these are the Maximum Take-Off (MTO) rating, and the Maximum Continuous Thrust (MCT or MCN) rating.

Maximum Takeoff thrust(MTO)

This is the maximum thrust that the engine can deliver for 5 minutes in the take-off envelope of the aircraft. Peak thrust is usually achieved when the engine is static, however the most demanding condition for a modern turbofan engine is end-of-runway or lift-off conditions, typically at about 0.25Mn. This condition usually generates the highest stresses and temperatures in the engine, hence use of this rating is only permitted for up to 5 minutes of operation.

It is used, as the name suggests, for take-off when the aircraft is at its heaviest and has to be accelerated to take-off speed in a finite runway distance. The higher the thrust available from the engine, the shorter the runway can be, or the greater the aircraft payload can be. This affects which airports an aircraft can be operated from, and the economics of operation. As an alternative to payload, a higher thrust rating allows more fuel load to be carried into the air, so extending range of operation. These trade-offs between available thrust, runway length, aircraft weight and range may need to be assessed for each flight, and is part of a commercial pilots preparation prior to take-off. An aircraft may take-off with less than maximum take-off thrust to reduce wear on the engine and extend its life. This is usually termed a 'derated' take-off, and is used to reduce engine maintenance costs.

It is a condition of certification that an aircraft should be able to take-off if one engine fails at the most critical point in the take-off run, when it is going too fast to be able to come to a safe stop in the remaining runway. In the case of twin engine aircraft, they have to be capable of taking off on one engine, so that in normal operation 'de-rate' is usually applied as an excess of thrust is available.

If an engine exceeds its 'redline' speeds or temperatures when running at MTO thrust, it is no longer considered airworthy.

Sometimes referred to as 'TOGA' thrust, short for take-off/go-around.

Maximum Continuous thrust (MCT)

Outside the MTO flight envelope, the MCT rating defines the maximum thrust that can be demanded by the pilot from the engine. As such, it has particular significance with respect to engine failure in flight, as the aircraft will have to proceed to its destination or nearest diversion airport at max continuous thrust. If the engine cannot achieve this thrust level whilst staying within it operating limits for engine speed and temperature, (them 'amber line'), it is no longer considered airworthy.

Maximum Climb thrust (MCL)

This is the thrust rating the manufacturer recommends be used during the climb phase of a typical flight. It may be the same as max continuous thrust, and usually is for a three or four engined aircraft. The top of the climb phase is typically the most challenging condition for a turbofan engine outside the take-off regime, and is a critical design requirement. De-rate can be applied to MCL thrust to extend engine life, but at the cost of a slower time to climb and slightly increased trip fuel consumption.

Maximum Cruise thrust (MCR)

Sometimes defined, but not a particularly useful rating since in cruise the pilot/autopilot will use the thrust required to maintain constant altitude and air speed to meet with air traffic control requirements.

Flight Idle

The idle rating is the minimum thrust that can be used whilst the aircraft is in flight. It is largely defined by the requirement to keep the engine running, possibly supplying secondary services to the aircraft such as hydraulic and electrical power, and, especially at high altitude, to supply passenger air at a minimum pressure. The flight idle rating is important in that the lower it is, the quicker the aircraft can descend (without going into a dive).

High or Approach idle

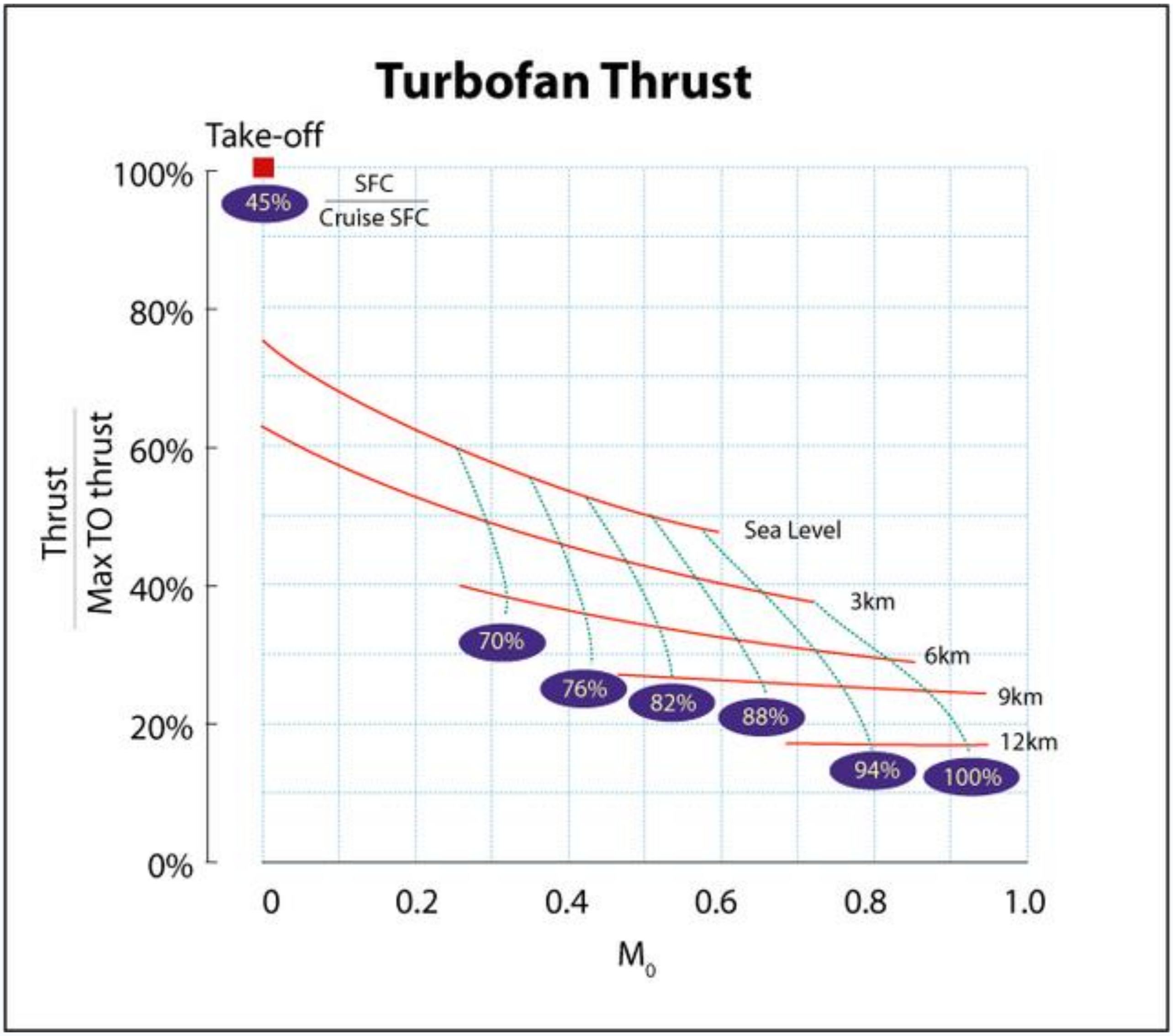
In the final phases of approach to landing it is important to be able to provide rapid response to throttle movements, this may require the engine to be running at a higher speed than ideal to be able to provide rapid acceleration if required. There may be a maximum response time requirement to achieve 'TOGA' thrust if a landing is aborted.

Ground Idle

Used for maneuvering on the ground. Typically defined by the need to keep the engine running and supplying power and services to the aircraft. Generally, the lower this value the better, since brake wear is a significant factor in aircraft running/maintenance costs.

Example ratings

The figure below shows the typical behaviour of a modern turbofan. The orange curves show maximum cruise thrust at altitude. The TO thrust is about 25% higher than the cruise thrust at sea level since it is permitted for short durations only.



Military Ratings

Combat arcraft have very different requirements to civil aircraft, and different rating terminology is used, especially for aircraft using reheat or afterburning for thrust augmentation.

Military Thrust

Typically used to define the maximum available thrust without use of reheat. Sometime referred to as max 'dry' thrust.

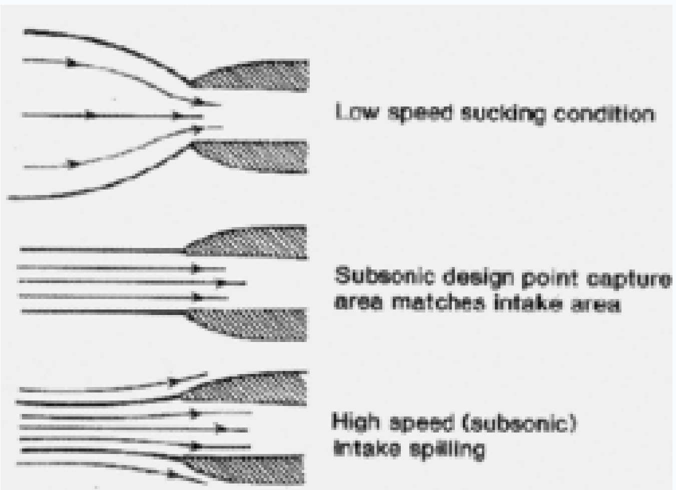
Design considerations

The various components named above have constraints on how they are put together to generate the most efficiency or performance. However the performance and efficiency of an engine can never be taken in isolation; for example fuel/distance efficiency of a supersonic jet engine maximises at about mach 2, whereas the drag for the vehicle carrying it is increasing as a square law and has much extra drag in the transonic region. The highest fuel efficiency for the overall vehicle is thus typically at Mach ~0.85.

For the engine optimisation for its intended use, important here is air intake design, overall size, number of compressor stages (sets of blades), fuel type, number of exhaust stages, metallurgy of components, amount of bypass air used, where the bypass air is introduced, and many other factors. For instance, let us consider design of the air intake.

Air intakes

Subsonic inlets



Pitot intake operating modes

Pitot intakes are the dominant type for subsonic applications. A subsonic pitot inlet is little more than a tube with an aerodynamic fairing around it.

At zero airspeed (i.e., rest), air approaches the intake from a multitude of directions: from directly ahead, radially, or even from behind the plane of the intake lip.

At low airspeeds, the streamtube approaching the lip is larger in cross-section than the lip flow area, whereas at the intake design flight Mach number the two flow areas are equal. At high flight speeds the streamtube is smaller, with excess air spilling over the lip.

Beginning around 0.85 Mach, shock waves can occur as the air accelerates through the intake throat.

Careful radiusing of the lip region is required to optimize intake pressure recovery (and distortion) throughout the flight envelope.

Supersonic inlets

Supersonic intakes exploit shock waves to decelerate the airflow to a subsonic condition at compressor entry.

There are basically two forms of shock waves:

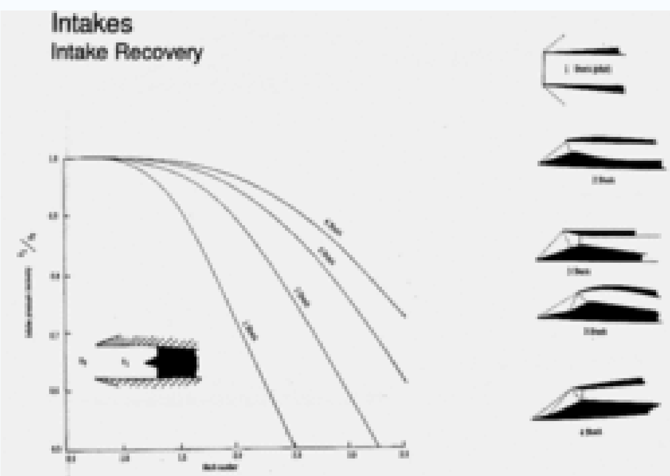
1) Normal shock waves, which are perpendicular to the direction of the flow.

2) Conical, or oblique, shock waves, which are angled rearwards, like the bow wave on a ship or boat.

Note: Comments made regarding 3 dimensional conical shock waves, generally apply to 2D oblique shock waves

Normal shock waves tend to cause a larger drop in stagnation pressure, than the weaker conical shock waves. Basically, the higher the supersonic entry Mach number to a normal shock wave, the lower the subsonic exit Mach number and the stronger the shock. Although conical shock waves also reduce Mach number, the outlet flow remains supersonic.

A sharp-lipped version of the pitot intake described above for subsonic applications performs quite well at moderate supersonic flight speeds. A detached normal shock wave forms just ahead of the intake lip and 'shocks' the flow down to a subsonic velocity. However, as flight speed increases, the shock wave becomes stronger, causing a larger percentage decrease in stagnation pressure (i.e. poorer pressure recovery). An early US supersonic fighter, the F-100 Super Sabre, used such an intake.



pressure recovery improvements resulting from the use of complex shock wave systems

More advanced supersonic intakes exploit a combination of conical shock wave/s and a normal shock wave to improve pressure recovery at high supersonic flight speeds. Conical shock wave/s are used to reduce the supersonic Mach number at entry to the normal shock wave, thereby reducing the resultant shock losses. An example of this was found on the SR-71's Pratt & Whitney J58s that could move a conical spike fore and aft within the engine nacelle, preventing the shockwave formed on the spike from entering the engine and stalling the engine, whilst keeping it close enough to give good compression.

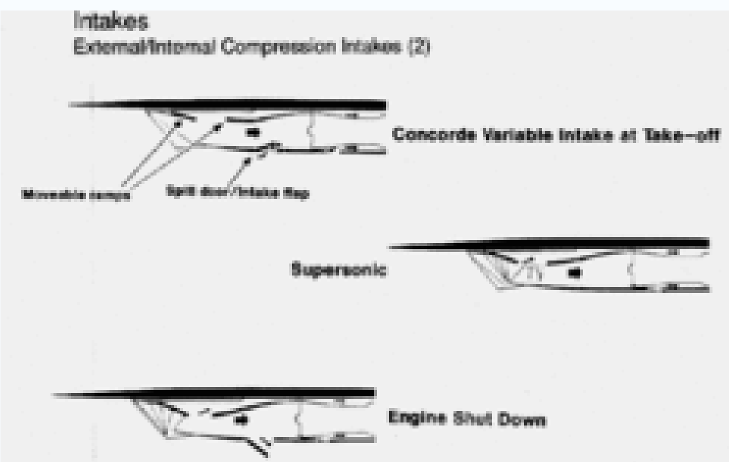
Many second generation supersonic fighter aircraft featured an inlet cone, which was used to form the conical shock wave. This type of inlet cone is clearly seen on the English Electric Lightning and MiG-21 aircraft, for example.

The same approach can be used for air intakes mounted at the side of the fuselage, where a half cone serves the same purpose with a semicircular air intake, as seen on the F-104 Starfighter and BAC TSR-2.

A more sophisticated approach is to angle the intake so that one of its edges forms a ramp. An oblique shockwave will form at start of this ramp. The Century series of US jets featured a number of variations on this approach, usually with the ramp at the outer vertical edge of the intake which was then angled back inwards towards the fuselage. Typical examples include the Republic F-105 Thunderchief and F-4 Phantom.

Later this evolved so that the ramp was at the top horizontal edge rather than the outer vertical edge, with a pronounced angle downwards and rearwards. This approach simplified the construction of the intakes and permitted the use of variable ramps to control the airflow into the engine. Most designs since the early 1960s now feature this style of intake, for example the F-14 Tomcat, Panavia Tornado and Concorde.

One of the problems with supersonic intakes is that they can deliver more corrected (or non-dimensional) flow than the engine itself can handle, particularly if the engine is throttled back. Some of the difference can be absorbed by the normal shock wave moving forward to a smaller flow area/lower entry Mach number, which weakens the shock, thereby reducing the outlet corrected flow. However, steps must be taken to prevent the normal shock from going forward of the intake lip, as this will disrupt the flow entering the intake. More extreme excesses in corrected flow can be accommodated by spilling air overboard through a trapdoor or supplementing the secondary flow of an ejector type final nozzle.



Concorde intake operating modes

Inlet cone

Inlet cones (sometimes called shock cones) are a component of some supersonic aircraft. They are primarily used on ramjets, such as the **turboramjets** of the SR-71 or the pure ramjets of the D-21 Tagboard and Lockheed X-7. More examples of inlet cones can be found on the Su-7 Fitter and the MiG-21 Fishbed, both of which use conventional jet engines.

Purpose

The main purpose of an inlet cone is to slow down the flow of air from supersonic flight speed to a **subsonic** speed, before it enters the engine. Most jet engines need subsonic airflow to operate properly, and require a **diffuser** to prevent supersonic airflow inside the engine. At supersonic flight speeds a conical shock wave, sloping rearwards, forms at the apex of the cone. Air passing through the conical shock wave (and subsequent reflections) slows down to a low supersonic speed. The air then passes through a strong normal shock wave, within the diffuser passage, and exits at a subsonic velocity. The resulting intake system is more efficient (in terms of pressure recovery) than the much simpler pitot intake.

Shape

The inlet cone is shaped so that the shock wave that forms on its apex is directed to the lip of the intake; this allows the engine to operate properly in supersonic flight. As speed increases, the shock wave becomes increasingly more oblique. As a result, some inlet cones are designed to move axially to maintain the shock-on-lip and allow efficient operation over a wider range of flight speeds.

Operation

At subsonic flight speeds, the conical inlet operates much like pitot intake. However, as the vehicle goes supersonic a conical shock wave appears, emanating from the cone apex. Conical (and oblique) shock waves are akin to the bow wave on a ship. As the flight Mach number increases, the conical shock wave becomes more oblique and eventually impinges on the intake lip.

Care must be taken to prevent the normal shock wave, which forms in the diffuser, coming forward too far and upsetting the flow field external to the intake lip. With a ramjet, this occurs if excessive fuel is injected into the combustor, raising internal pressure too far. However, with a turbojet or turbofan, the problem arises when the engine is throttled back, causing a mismatch between intake airflow and engine mass flow. A trapdoor is needed to dump excess flow overboard.

Alternative Shapes

Some air inlets feature a biconic centrebody to form two conic shock waves, both focused on the lip of the intake. This improves pressure recovery. Concorde used so-called 2D inlets, where the nacelle is rectangular and flat ramps replace the dual cones just described. Some aircraft use a semi-conic centrebody.

Many supersonic aircraft dispense with the conical centrebody and employ a simple pitot intake. A detached, strong, normal shock appears directly in front of inlet at supersonic flight speeds, which leads to a poor pressure recovery.

SABRE

SABRE (Synergic Air BReathing Engine) is a proposal for a hydrogen-fuelled air breathing rocket engine/jet engine for propelling the Skylon launch vehicle into low earth orbit.

SABRE is the ultimate design of Alan Bond's series of liquid air cycle engine (LACE) and LACE-like designs that started in the early/mid-1980s for the HOTOL project.

History

The precooler-based LACE idea was originally explored by Marquardt and General Dynamics in the 1960s as part of the US Air Force's aerospaceplane efforts. This work eventually culminated in medium-thrust engines that ran for several minutes at a time. Although the program was generally successful, changing priorities and USAF funding led to the idea being abandoned.

In an operational setting the system would be placed behind a supersonic air intake which would compress the air through ram compression, then rapidly cool it in a heat exchanger using some of the liquid hydrogen fuel stored onboard. The resulting liquid air was then processed to separate out the liquid oxygen for burning in the engine. The small amount of now-warmed hydrogen used up in this process was too difficult to feed back into the engine, so it was simply dumped overboard.

HOTOL's engine, the RB545, was very similar to the original LACE system, but much simpler in detail. Like the LACE system it used an air intake and heat exchanger system, but was able to do away with much of the complexity of the USAF design, which included pumps and storage tanks along each step of the separation process. RB545 was a "straight-through" design that used careful arrangement of the components to dump the unwanted liquified gasses directly overboard, instead of pumping it around from tank to tank. From that point on the RB545, like LACE before it, consisted of a fairly conventional rocket engine.

A new design

In 1989, after funding for HOTOL ceased, Bond and several others formed Reaction Engines Limited to continue research. The RB545's liquid hydrogen precooler had issues with embrittlement, patents and The Official Secrets Act, so Bond went on to develop SABRE in its place.

Like the RB545, the SABRE design is not a conventional rocket engine, but a precooled turborocket that burns hydrogen fuel and air.

At the front of the engine a single moving shock cone inlet slows the air to subsonic speeds and then passes the air through a precooler. Behind the precooler, the SABRE system consists of a number of different engine components, each tuned to a different portion of the flight. SABRE uses two "pure" rocket engines surrounded by a ring of smaller combustors similar to ramjets.

The precooler

As the air enters the engine at high speeds, it becomes very hot due to compression effects. The high temperatures are traditionally dealt with in jet engines by using heavy copper or nickel based materials, and by throttling back the engine at the higher airspeeds to avoid melting. Instead, using a gaseous helium coolant loop, SABRE dramatically cools the air in a heat exchanger but avoids liquification.

Previous versions of SABRE such as HOTOL put the hydrogen fuel directly through the precooler, but inserting a helium cooling loop between the air and the cold fuel avoids problems with hydrogen embrittlement in the air precooler.

Avoiding liquifaction improves the efficiency of the engine since less liquid hydrogen is boiled off; even simply cooling the air needs more liquid hydrogen than can be burnt in the engine core, the excess is dumped overboard (presumably through a suitable nozzle in afterburner mode.)

However, the dramatic cooling of the air raised a potential problem: it is necessary to prevent blocking the precooler from frozen water vapour. A suitable precooler, which rejects condensed water before it freezes has now been experimentally demonstrated.

The compressor

The cooled air is then passed into a reasonably conventional turbocompressor, similar in design to those used on a jet engine, but in this case powered by a gas turbine running on the helium loop, rather than off combustion gases as in a conventional jet engine.

The engines

After being launched and brought to speed by a short burst of the rockets, the jets are started, fed by air bled from the shock cone. At this point the precooler/turbocompressor is not being used. As the craft ascends and the outside air pressure drops, more and more air is passed into the compressor as the effectiveness of the ram compression alone drops. In this fashion the jets are able to operate to a much higher altitude than would normally be possible.

At Mach 5.5 the jets become inefficient and are powered down, and stored liquid oxygen/liquid hydrogen is used for the rest of the ascent in the separate rocket engines; the turbopumps are powered off of the helium loop from the heat produced by cooling the engine.

The helium loop

The 'hot' helium from the air precooler, and cooling the combustion chambers is recycled by cooling it in a heat exchanger with the liquid hydrogen fuel.

The loop forms a self starting Brayton cycle engine, and is used to both cool critical parts of the engine, but also to power turbines and numerous miscellaneous parts of the engine.

The heat passes from the air into the helium. This heat energy is not entirely wasted, it is in fact used to power the various parts of the engine, and the remainder is used to vapourise hydrogen, which is burnt in a Ramjet.

Losses

The losses from carrying around a number of engines that will be turned off for some portion of the flight would appear to be heavy, yet the gains in overall efficiency more than make up for this. These losses are greatly offset by the different flight plan. Conventional launch vehicles such as the Space Shuttle usually start a launch by spending around a minute climbing almost vertically at relatively low speeds; this is inefficient, but the optimum for pure-rocket vehicles. In contrast, the SABRE engine permits a much slower, shallower climb, airbreathing and using wings to support the vehicle, giving far lower fuel usage before lighting the rockets to do the orbital insertion.

Performance

The designed thrust/weight ratio of SABRE ends up several times higher—up to 14, compared to about 5 for conventional jet engines, and 2 for scramjets. This high performance is a combination of the cooled air being denser and hence requiring less compression, but more importantly, of the low air temperatures permitting lighter alloy to be used in much of the engines. Overall performance is much better than the RB545 engine.

The engine results in very good fuel efficiency—about 2800 seconds. Typical all-rocket systems are around 450 at best, and even "typical" nuclear powered engines only about double this.

The combination of high fuel efficiency and low mass engines means that a single stage to orbit approach for Skylon can be employed, with airbreathing to mach 5.5+, and with the vehicle reaching orbit with more payload mass per take-off mass than just about any non-nuclear launch vehicle ever proposed.

Like the RB545, the precooler idea adds mass and complexity to the system, normally the antithesis of rocket design. In addition, the precooler is also the most aggressive and difficult part of the whole SABRE design. The mass of this heat exchanger is an order of magnitude better than has been achieved previously; experimental work has proved that this can be achieved. The experimental heat exchanger has achieved heat exchange of almost 1 GW/m^3, believed to be a world record. Small sections of a real precooler now exist.

The **ATREX** engine developed in Japan is an experimental precooled jet engine that works as a turbojet at low speeds and a ramjet up to mach 6.0.

ATREX uses liquid hydrogen fuel in a fairly exotic single-fan arrangement. The liquid hydrogen fuel is pumped through a heat exchanger in the air-intake, simultaneously heating the liquid hydrogen and cooling the incoming air. This cooling of the incoming air is critical in achieving a reasonable efficiency. The hydrogen then continues through a second heat exchanger positioned after the combustion section, where the hot exhaust is used to further heat the hydrogen, turning it in a very high pressure gas. This gas is then passed through the tips of the fan providing driving power to the fan at subsonic speeds. After mixing with the air, the hydrogen is burned in the combustion chamber.

Bypass ratio

In aeronautical engineering, and jet engine design in particular, **bypass ratio** is a common measurement that compares the amount of air deliberately "blown past" the engine to that moving through the core. For instance, an engine that blows two kilograms of air around the engine for every kilogram that passes through it is said to have a bypass ratio of 2 (or 2 to 1). Higher bypass ratios generally infer better specific fuel consumption as an increasing amount of thrust is being generated without burning more fuel.

Jet engines are generally able to create considerably more energy than they can use in moving air through the engine core. This is because the limiting factor is the temperature at the turbine face, and that is a function of the total amount of fuel burned. Increasing airflow, and thus thrust, would imply burning more fuel and generating higher temperatures. It is possible to increase the airflow by burning "too much" fuel or adding water in front of the turbine to cool it, but both methods lead to incomplete combustion and very poor fuel efficiency. This was nevertheless common for some time to produce added thrust on takeoff, which is why the exhaust plumes of older aircraft appear so smoky in films.

Rolls-Royce came up with a better use of the extra energy in their Conway turbofan engine, developed in the early 1950s. In the Conway an otherwise "normal" axial-flow turbojet was equipped with an oversized first compressor stage (the one closest to the front of the engine) and centered inside a tubular nacelle. While the inner portions of the compressor worked "as normal" and provided air into the core of the engine, the outer portion blew air around the engine to provide extra thrust. In effect the Conway was a much larger engine which only burned some of the air flowing through it, but did so in an efficient way. The Conway had a very small bypass ratio of only 0.3, but the fuel economy was nevertheless improved enough that it, and its derivatives like the Spey became some of the most popular jet engines in the world.

Through the 1960s the bypass ratios grew, making jetliners competitive in fuel terms with piston-powered planes for the first time. Most of the very-large engines in this class were pioneered in the United States by Pratt & Whitney and General Electric, which for the first time was outcompeting England in engine design. Rolls-Royce also started the development of the "high-bypass turbofan", and although it caused considerable trouble at the time, the RB.211 would go on to become one of their most successful products.

Turbofans are typically broken into one of two categories--low-bypass ratio and high-bypass ratio. In a low-bypass turbofan, only a small amount of air passes through the fan ducts and the fan is of very small diameter. The fan in a high-bypass turbofan is much larger to force a large volume of air through the ducts. The low-bypass turbofan is more compact, but the high-bypass turbofan can produce much greater thrust, is more fuel efficient, and is much quieter.

Today almost all jet engines include some amount of bypass. For "low speed" operations like airliners modern engines use bypass ratios up to 17, while for "high speed" operations like fighter aircraft the ratios are much lower, around 1.5.

Bleed air

Bleed air in gas turbine engines is compressed air taken from within the engine, after the compressor stage(s) and before the fuel is injected in the burners. This compressed air can be used in many different ways, from de-icing to pressurising the cabin to pneumatic actuators. However, bleed air is quite hot and if being used in the cabin or other low temperature areas it must be cooled, even refrigerated. Bleed air is valuable in an aircraft for two properties: its high temperature and its high pressure.

Since most gas turbine engines use multiple compressor stages, some newer engines for new aircraft designs have the bleed air inlet between compressor stages to reduce the temperature and reduction in the need for compressed air in more electric aircraft.

Merits of bleed air

In civil aircraft, its primary use is to provide pressure for the aircraft cabin by supplying air to the Environmental Control System. Additionally, bleed air is used to keep critical parts of the aircraft (such as the wing leading edges) ice-free.

When used for cabin pressurization, the air from the engine must first be cooled (as it exits the compressor stage at temperatures as high as 300 °C) by passing the bleed air through an air-to-air heat exchanger cooled by cold outside air. It is then fed to an air conditioning unit which regulates the temperature and flow of air into the cabin, keeping the environment comfortable.

A similar system is used for wing de-icing. In icing conditions, water droplets condensing on a wing's leading edge can freeze at the ambient temperatures experienced during flight. This build-up of ice adds weight and changes the shape of the wing, causing a degradation in performance, and possibly a fatal loss of lift. To prevent this, warm bleed air is pumped through the inside of the wing's leading edge. This heats up the metal, preventing the formation of ice. Alternatively, the bleed air may be used to inflate a rubber boot glued to the leading edge, breaking the ice loose.

Recent developments in civil aircraft

Bleed air systems have been in use for several decades in passenger jets. Recently, Boeing announced that its new airplane, the 787 would operate without use of bleed air (and the two engines proposed for the airplane, the General Electric GEnx and the Rolls-Royce Trent 1000, are designed with this in mind). This represents a departure from traditional airplane design, and proponents state that eliminating bleed air improves engine efficiency, as there is no loss of mass airflow and therefore energy from the engine, leading to lower fuel consumption. Additionally, eliminating bleed air may reduce the aircraft's mass by removing a whole series of pumps, heat exchangers and other heavy equipment. Lastly, advocates of the design say it improves safety as heated air is confined to the engine core, as opposed to being pumped through pipes and heat exchangers in the wing and near the cabin, where a leak could damage surrounding systems.

Skeptics point out that by eliminating bleed air there is then a requirement to find alternative methods of providing cabin heating, de-icing and other functions previously covered by bleed air, which require additional systems which take up space, weight, and electrical energy. Therefore this approach is less efficient from an overall point of view (taking the entire airplane into consideration and not just the engines), as it involves drawing in very cold ambient air and heating it, a very energy-intensive process.

Airbus does not currently (as of November 2004) have any plans to eliminate bleed air from its 787 competitor, the A350, while Boeing is actively pursuing this technology, touting it as one of the main advantages of its design.

Compressor stall

A **compressor stall** is a situation of abnormal airflow through the compressor stage of a jet engine, causing a stall of the vanes of the compressor rotor.

All compressor stalls result in a loss of engine power. This power failure may only be momentary (occurring so quickly it is barely registered on engine instruments), or may shut the engine down completely (that is, causing a flameout). When a compressor stall affects the airflow through the entire engine it is also known as a **compressor surge**.

Types

There are two general types of compressor stall.

The first, less severe type of stall, the "axis-symmetric stall", is a straightforward expulsion of air out the intake due to the compressor's inability to maintain pressure on the combustion chamber.

In the second, more-severe "rotational stall", the air flow disruption of the stall causes standing pockets of air to rotate within the compressor without moving along the axis. Without fresh air from the intake passing over the stalled compressor vanes they overheat, causing accelerated engine wear and possible damage.

Causes

The most likely cause of a compressor stall is a sudden change in the pressure differential between the intake and combustion chamber. Jet aircraft pilots must take this into account when dropping airspeed or increasing throttle.

The following factors can induce compressor stall:

- Engine over-speed
- Engine operation outside specified engineering parameters
- Turbulent or disrupted airflow to the engine intake
- Contaminated or damaged engine components

One of the most common causes of 1st stage compressor stalls in commercial aviation aircraft is a bird strike. On take off, while manuvering on the ground or while on approach to landing; these planes operate in close proximity to birds. It is not uncommon for birds to be sucked into the intake of the engine and often times can cause a 1st stage compressor stall. Because birds are combustible material, a fire or flames described as "shooting" out of the engine are common reports during this type of compressor stall.

Effects

Compressor stalls can result in one or more extremely loud bangs emanating from the engine as the combustion process "backfires". This may be accompanied by an increased exhaust gas temperature, and yawing of the aircraft in the direction of the affected engine.

The effects of a stall can vary. A minor stall may create an alarming noise but have little other effect. On the other hand, a violent compressor surge might completely destroy the engine and set it on fire.

The appropriate response to compressor stalls varies according the engine type and situation — but usually consists of immediately and steadily decreasing thrust on the affected engine.

As a bit of trivia, the crews of the Lockheed (now Lockheed Martin) SR-71 Blackbird were witness to arguably some of the most spectacular compressor stalls ever. These usually happened at high Mach numbers, and were normally the result of shockwave detachment on the conic diffusers on the intakes. The resulting loss of thrust on the affected engine (and its accompanying yaw moment), combined to the long fuselage of the airplane should have made for an interesting ride for the crew, specially taking into account that they had a rather short time to correct the problem before the mission or the airframe were compromised.

De Laval nozzle

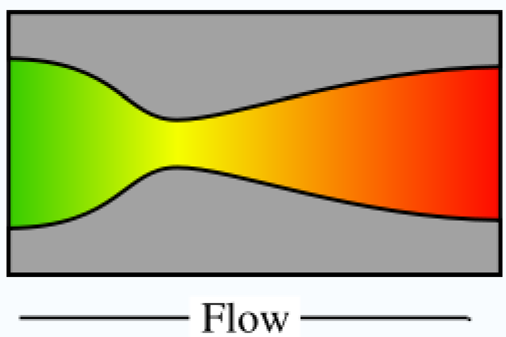


Diagram of a de Laval nozzle, showing approximate flow velocity increasing from green to red in the direction of flow

A **de Laval nozzle** (or **convergent-divergent nozzle**, **CD nozzle** or **con-di nozzle**) is a tube that is pinched in the middle, making an hourglass-shape. It is used as a means of accelerating the flow of a gas passing through it. It is widely used in some types of steam turbine and is an essential part of the modern rocket engine and supersonic jet engines.

The nozzle was developed by Swedish inventor Gustaf de Laval in the 19th century. Its operation relies on the different properties of gases flowing at subsonic and supersonic speeds. The speed of a subsonic flow of gas will increase if the pipe carrying it narrows because the mass flow rate is constant (grams or pounds per second). The gas flow through a de Laval nozzle is isentropic (gas entropy is nearly constant) and adiabatic (heat loss or gain is nearly zero). At subsonic flow the gas is compressible; sound, a small pressure wave, will propagate through it. Near the nozzle "throat", where the cross sectional area is a minimum, the gas velocity locally becomes transonic (Mach number = 1.0), a condition called choked flow. As the nozzle cross sectional area increases the gas continues to expand and the gas flow increases to supersonic velocities where a sound wave will not propagate backwards through the gas as viewed in the rest frame of the nozzle (Mach number > 1.0).

A de Laval nozzle using hot air at a pressure of 1,000 psi (6.9 MPa or 68 atm), temperature of 1470 K, would have a pressure of 540 psi (3.7 MPa or 37 atm), temperature of 1269 K at the throat, and 15 psi (0.1 MPa or 1 atm), temperature of 502 K at the nozzle exit. The expansion ratio, nozzle cross sectional area at exit divided by area at throat, would be 6.8. The specific impulse would be 151 s (1480 N·s/kg).

This principle was used in a rocket engine by Robert Goddard. Walter Thiel's implementation of it made the V2 rocket possible.

Conditions for operation

A de Laval nozzle will only choke at the throat if the mass flow through the nozzle is sufficient, otherwise no supersonic flow is achieved.

In addition, the pressure of the gas at the exit of the expansion portion of the exhaust of a nozzle must not be too low. Because pressure cannot travel upstream through the supersonic flow, the exit pressure can be significantly below ambient pressure it exhausts into, but if it is too far below ambient, then the flow will cease to be supersonic, or the flow will separate within the expansion portion of the nozzle, forming an unstable jet that may 'flop' around within the nozzle, possibly damaging it.

In practice ambient pressure must be no higher than roughly 2.7 times the pressure in the supersonic gas for supersonic flow to leave the nozzle.

Analysis of gas flow in de Laval nozzles

The analysis of gas flow through de Laval nozzles involves a number of concepts and assumptions:

- For simplicity, the combustion gas is assumed to be an ideal gas.
- The gas flow is isentropic (i.e., at constant entropy), frictionless, and adiabatic (i.e., there is little or no heat gained or lost)
- The gas flow is constant (i.e., steady) during the period of the propellent burn.
- The gas flow is along a straight line from gas inlet to exhaust gas exit (i.e., along the nozzle's axis of symmetry)
- The gas flow behavior is compressible since the flow is at very high velocities.

Exhaust gas velocity

As the combustion gas enters a nozzle, it is traveling at subsonic velocities. As the throat contracts down the gas is forced to accelerate until at the nozzle throat, where the cross-sectional area is the smallest, the linear velocity becomes sonic. From the throat the cross-sectional area then increases, the gas expands and the linear velocity becomes progressively more supersonic.

The linear velocity of the exiting exhaust gases can be calculated using the following equation:[1] [2] [3]

$$V_e = \sqrt{\frac{T\,R}{M} \cdot \frac{2\,k}{k-1} \cdot \left[1-(P_e/P)^{(k-1)/k}\right]}$$

where:

$\boldsymbol{V_e}$ = Exhaust velocity at nozzle exit, m/s

$\boldsymbol{T}$ = absolute temperature of inlet gas, K

$\boldsymbol{R}$ = Universal gas law constant = 8314.5 J/(kmol·K)

$\boldsymbol{M}$ = the gas molecular mass, kg/kmol (also known as the molecular weight)

$\boldsymbol{k}$ = c_p/c_v = isentropic expansion factor

$\boldsymbol{c_p}$ = specific heat of the gas at constant pressure

$\boldsymbol{c_v}$ = specific heat of the gas at constant volume

$\boldsymbol{P_e}$ = absolute pressure of exhaust gas at nozzle exit, Pa

$\boldsymbol{P}$ = absolute pressure of inlet gas, Pa

Some typical values of the exhaust gas velocity V_e for rocket engines burning various propellants are:

- 1.7 to 2.9 km/s (3800 to 6500 mi/h) for liquid monopropellants
- 2.9 to 4.5 km/s (6500 to 10100 mi/h) for liquid bipropellants
- 2.1 to 3.2 km/s (4700 to 7200 mi/h) for solid propellants

As a note of interest, V_e is sometimes referred to as the *ideal exhaust gas velocity* because it based on the assumption that the exhaust gas behaves as an ideal gas.

As an example calculation using the above equation, assume that the propellant combustion gases are: at an absolute pressure entering the nozzle of $\boldsymbol{P}$ = 7.0 MPa and exit the rocket exhaust at an absolute pressure of $\boldsymbol{P_e}$ = 0.1 MPa; at an absolute temperature of $\boldsymbol{T}$ = 3500 K; with an isentropic expansion factor of $\boldsymbol{k}$ = 1.22 and a molar mass of $\boldsymbol{M}$ = 22 kg/kmol. Using those values in the above equation yields an exhaust velocity $\boldsymbol{V_e}$ = 3181 m/s or 3.18 km/s which is consistent with above typical values.

The technical literature can be very confusing because many authors fail to explain whether they are using the universal gas law constant R which applies to any ideal gas or whether they are using the gas law constant R_s which only applies to a specific individual gas. The relationship between the two constants is $R_s = R/M$.