Part 3: Gas Turbine Engines

Learning Outcomes

After studying Part 3 of the module you will be able to:

- Calculate changes in thermodynamic properties through gas turbine components, and the associated work, heat and fuel flows, explaining your assumptions.
- Explain how high-bypass ratio and high turbine entry temperature technologies contribute to increasing overall efficiency.
- Identify gas turbine architectures that address competing design requirements, including noise, emissions, weight, fuel consumption and cost.
- Optimise the specification of a turbofan engine on the basis of the fan pressure ratio.
- Explain the ways in which fan pressure ratio affects the effective net thrust of a turbofan.

Introduction

In Parts 1 and 2 of the module you have covered all of the fundamental thermodynamics and fluid dynamics needed for the rest of this module: from now on the focus is on applying that understanding to a practical system – the gas turbine engine.

In a gas turbine, gas flows through a series of components, and you already know how to analyse each of them.

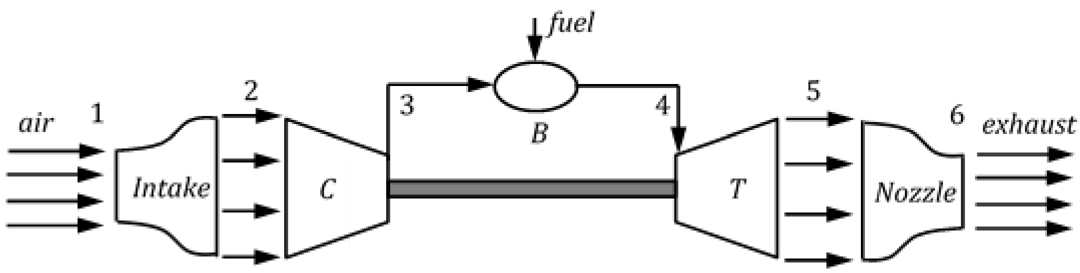

Figure 1. Schematic diagram of the components making up a turbojet engine.

The intake and propulsion nozzle can be analysed using your new-found understanding of compressible flows. You have recently learned how to analyse combustion processes in greater depth. And you know how to account for irreversibilities in compressors and turbines using the isentropic efficiency.

The goal now is to put these components together and to analyse the behaviour of the overall system. In Part 3 you will develop analysis of turbojet and turbofan engines. This will allow you to understand important design choices such as the bypass or fan pressure ratio of a turbofan engine. In part 4 you will look in more detail at the operation of turbomachinery (compressors, turbines and propellers), so that you can start to understand how choices in the design of these components affect their efficiency and the overall system performance.

Perf

- Thrust
- operating
  - speed
  - altitude
- fuel efficiency
- weight

Econ

- fuel cost
- maintenance
- development
- manufacturing
  - material
  - methods

Env

- efficiency
- exhaust + types
- sound

NOx
- air quality pollutant
- greenhouse gases

Safety

- reliability
- safety factors
- procedure
- bird strike/debrie resilience
- redundancy
- altitude relight
  - restarting engines mid flight.

7 ANALYSIS OF JET ENGINES (WEEK 7)

This Section looks at the operation of simple gas turbines and outlines the method of calculating the thrust and efficiency. We introduce the analysis of an ideal jet engine, but go on to account for thermodynamic losses, for more realistic thermodynamic properties, and technologies such as blade cooling and reheat that are sometimes used in real engines.

7.1 LEARNING OUTCOMES

After completing this section of the course you will be able to:

- Explain how technical, societal and economic factors influence design of aeronautical gas turbines.
- Calculate changes in thermodynamic properties through gas turbine components, and the associated work, heat and fuel flows, explaining your assumptions.
- Assess how alternative design choices contribute to overall efficiency.

7.2 DESIGN REQUIREMENTS

A propulsion system must meet the essential technical requirements of providing sufficient thrust and sufficient efficiency for the intended aircraft. But, beyond the essential technical requirements, the design of propulsion system must satisfy a range of additional requirements, including ones relating to safety, reliability, life-cycle cost and operability. Engine design is therefore informed and constrained by the thermodynamic analysis developed in this section, but also involves a trade-off between a much wider set of considerations, some of which are suggested in the table below.

Table 1

Technical performance	**Safety and reliability**	**Competitive life-cycle cost**	**Operational requirements**
Sufficient thrust for aircraft operations	Complete take-off after 1 engine failure	Development costs	Runway length and airport infrastructure
Sufficient efficiency to achieve specified range	Extended operation over sea	Materials and manufacturing costs	Maintenance intervals
	Altitude relight	Fuel costs	Noise
	Containment of mechanical failures	Maintenance and operating costs	Emissions

7.3 GAS TURBINE PRINCIPLES

The essential components of gas turbines are a compressor, a combustor (or another device to increase the temperature of the flow), and a turbine which powers the compressor. These components make up the *core* of the engine. The exit flow from the core engine may be accelerated through a nozzle, resulting in a pure turbojet, or passed through an additional low-pressure turbine in order to generate additional shaft power for a fan, resulting in a turbofan engine. The extra power from the low-pressure turbine may alternatively be used to power other devices, such as propellers on a ship or turbofan engine, pumps, or generators in a power station. A range of possibilities is illustrated in Figure 2.

when working with aircraft turbines W_{out} is a sad metric, since useful work is done on the air not a shaft

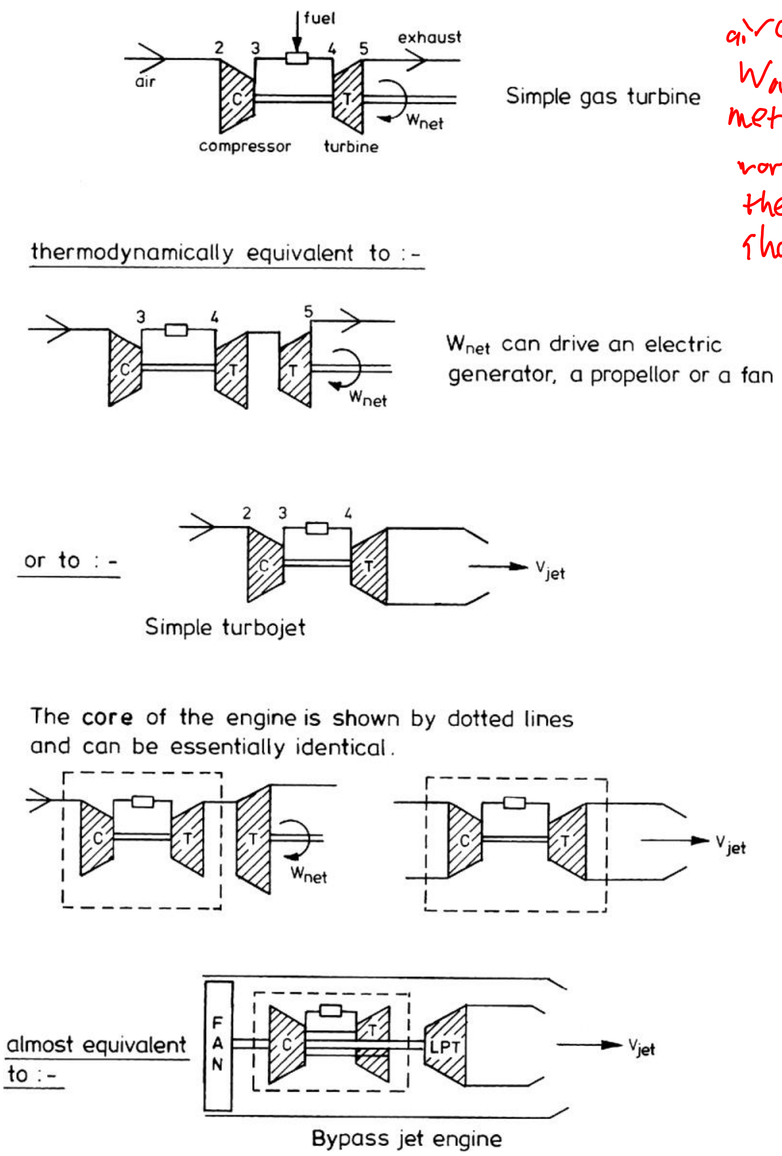

Figure 2. Gas turbines -- variations on a core theme [Image from Jet Propulsion, N. Cumpsty]

7.4 THE IDEAL BRAYTON CYCLE VERSUS REALITY

In the Part 1 Thermofluids module you analysed an ideal cold-air-standard cyclic heat engine following the Brayton cycle[1]:

- isentropic compression,
- constant pressure heating,
- isentropic expansion,
- constant pressure cooling,
- the working fluid is a perfect gas
- the perfect gas has properties of cold air ($c_p = 1.005\ kJ.kg^{-1}.K^{-1},\ \gamma = 1.40$).

We can take the ideal Brayton cycle as a simple model for a gas turbine engine. Assuming that the working fluid is a perfect gas with ratio of specific heats γ, the thermal efficiency of the ideal Brayton cycle is:

$$\eta_{th} = \frac{w_{x,net}}{q_{in}} = 1 - \frac{1}{r_p^{\frac{\gamma-1}{\gamma}}}.$$

$w_{x,net}$ is the net specific shaft work output from the engine and q_{in} is the specific heat input. This shows that the thermal efficiency depends only on the properties of the working fluid through the ratio of specific heats, γ, and the stagnation pressure ratio of the engine, r_p.

An aeronautical engine produces thrust rather than shaft work output and, strictly, combustion engines consume the chemical energy of the fuel, rather than a heat input, so the concept of *thermal* efficiency is not directly applicable to an aircraft engines. The ideal Brayton cycle also is not an accurate model for a real gas turbine engine. Nonetheless, the simple Brayton cycle analysis tells us that increasing pressure ratio increases gas turbine efficiency, and that is also for real gas turbine design in many cases.

Real gas turbines differ from the ideal Brayton cycle in the following respects:

- The turbo-machinery is not isentropic.
- There is a pressure loss in the combustor.
- c_p and γ vary with temperature.
- c_p for combustion products is higher than c_p for air.
- Combustion occurs internally instead of heat being supplied externally.
- The addition of fuel in the combustor increases the mass flow rate in the turbine.
- The turbine exhaust is released to the atmosphere (i.e. it is not a cycle).

In addition, in gas turbines used for aircraft propulsion:

- the engine inlet conditions depend on altitude and flight speed (Mach number).
- there is no net work output from each gas turbine spool[2] (unless there is a power take-off).

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[1] The cycle is also sometimes referred to as the Joule cycle.
[2] A spool is the collective name for a compressor and turbine, and their connecting shaft. Modern engines often have two or three separate spools.

- The back-pressure downstream of the turbine may be different to atmospheric pressure due to the pressure difference across the propulsion nozzle.

Using elementary thermodynamics and compressible flow theory it is possible to derive useful models[3] for gas turbine systems with a wide range of architectures (e.g. turbojet, turbofans, etc.). Thermodynamic analysis tells us about changes in properties between the inlet and outlet of each component, and this allows us to understand and optimise the overall engine design. Thermodynamic analysis (on its own) does not tell us what happens within each component – that is the domain of fluid mechanics, heat transfer and mechanics. Design of gas turbine components is a highly-developed multi-disciplinary enterprise. This module is restricted mostly to system-level thermodynamic analysis, however we will start to look at some of the fluid-mechanic aspects of compressors and turbines in Part 4 of the module.

7.5 Worked Example: Analysis of a turbojet in flight

An aeronautical gas turbine can be analysed by considering each component in sequence starting from a point in the engine where you know the thermodynamic conditions (often that is conditions in the atmosphere upstream of the engine).

Here we analyse a turbojet engine in flight at Mach 2.0 and 31,000 ft altitude, in order to find the thrust, specific fuel consumption and efficiencies. This worked example is long but is broken into analysis for each sub-component.

The turbojet engine is shown schematically in Figure 1[4]. It is conventional to label the entry to the fan (or compressor) as station number 2, the entrance to the combustor as station 3, and entrance to the turbine as station 4. Additional numbers are introduced when considering more complex engine architectures.

In the following analysis we will make the following assumptions:

- Flow through the intake is isentropic and adiabatic[5].
- The compressor has a stagnation pressure ratio of 30, and isentropic efficiency of 90 %,
- The fuel has calorific value of 43 MJ/kg at 298 K. The fuel enters the burner at 298 K and has negligible heat of vaporisation. The burner is adiabatic and combustion is complete.
- Assume a 4 % stagnation pressure loss through the combustor.
- The turbine entry temperature is 1500 K and the turbine isentropic efficiency is 90 %.
- Flow through the propulsion nozzle is isentropic, adiabatic and fully expanded to the ambient pressure.
- Air behaves as a perfect gas with heat capacity c_p=1005 J.kg^{-1}.K^{-1} and γ=1.40.
- Combustion products behave as a perfect gas with heat capacity c_p=1100 J.kg^{-1}.K^{-1} and γ=1.33.

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[3] Note that for a model to be 'useful' it does not necessarily have to be quantitatively accurate. The simple ideal Brayton cycle provides important and valid insight, even though, in absolute terms, the efficiency values it predicts are not representative of real gas turbine engines.

[4] Figure 1 shows a convergent nozzle however the present worked example has a converging-diverging nozzle.

[5] The fact a flow is isentropic does not imply it is adiabatic.

Tip: it will help you, and those reviewing your work, if you draw a schematic diagram for the engine you are analysing, and follow the standard numbering convention. Then apply the steady-flow energy equation and relevant thermodynamic state relationships to each component in sequence.

7.5.1.1 Inlet conditions and flight velocity

The ambient conditions at 31,000 ft can be modelled by reference to the International Standard Atmosphere. From the Thermofluids Data Book[6] Table 24, the *static* temperature and pressure in the at 31,000 ft are:

$$T_1 = T_{atm} = 226.73\ K, \qquad p_1 = p_{atm} = 28.7\ kPa$$

We conduct the engine analysis in a frame of reference moving with the engine. Relative to the engine, the atmosphere is moving at Mach 2.0. The stagnation properties at inlet can be calculated from the perfect gas relationships for compressible flow (page 13 of the Thermofluids Data Book) or using the gas flow tables for air (Table 19 of the Thermofluids Data Book):

$$\frac{T}{T_0} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-1}; \ \frac{p}{p_0} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-\frac{\gamma}{\gamma-1}}$$

giving

$$T_{01} = 226.73\left(1 + \frac{1.40 - 1}{2} 2.0^2\right) = 408.1\ K$$

$$p_{01} = 28.7\left(1 + \frac{1.40 - 1}{2} 2.0^2\right)^{\frac{\gamma}{\gamma-1}} = 224.6\ kPa.$$

The flight velocity V also follows from the Mach number and the speed of sound in the atmosphere, $a = \sqrt{\gamma R T_{atm}}$ (note that speed of sound is a function of *static* temperature).

$$V = M.a = 2.0 \times \sqrt{1.40 \times 287 \times 226.73} = 603.7\ m.s^{-1}.$$

7.5.1.2 Intake 1 → 2

The intake is assumed to be adiabatic and isentropic. Applying the steady flow energy equation:

$$q_{12} - w_{x,12} = h_{02} - h_{01}$$

Since the flow is adiabatic, $q_{12} = 0$. Since there is no shaft work device[7] in the flow, $w_{x,12} = 0$, from which the preceding equation gives $h_{02} = h_{01}$.

Since the flow is assumed to be a perfect gas, the heat capacity is a constant and we can relate changes in enthalpy to changes in temperature:

$$h_{02} - h_{01} = c_p(T_{02} - T_{01})$$

Since the stagnation enthalpy does not change, the stagnation temperature is also constant:

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[6] This can be accessed at Course Content / Reference Material / Data Book

[7] A shaft work device is a machine driven by a rotating shaft, such as a compressor, or a machine like a turbine that turns a shaft.

$$T_{02} = T_{01} = 408.1\ K$$

If the flow is isentropic and adiabatic, the stagnation pressure is constant,

$$p_{02} = p_{01} = 224.6\ kPa.$$

7.5.1.3 Compressor 2 → 3

The compressor has pressure ratio $p_{03}/p_{02} = 30$, giving $p_{03} = 6736.9$ kPa.

The isentropic and the actual compression processes are illustrated on the following temperature-entropy plot:

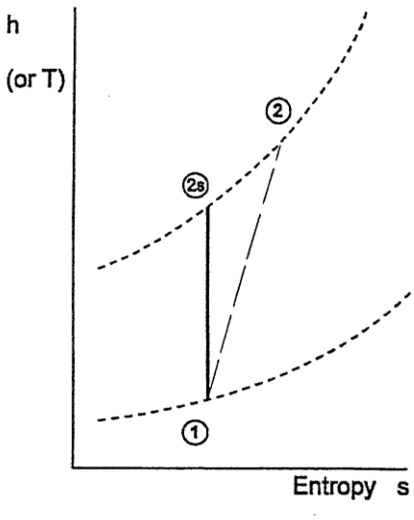

Figure 3. Temperature-entropy plot for isentropic and non-isentropic compression.

First we analyse the compression process as if it were isentropic, using the isentropic relation[8]:

$$T_{03s} = T_{02}\left(\frac{p_{03}}{p_{02}}\right)^{\frac{\gamma-1}{\gamma}} = 1078.5\ K$$

The actual temperature is then found using the definition of isentropic efficiency[9]:

$$\eta_c = \frac{h_{03s} - h_{02}}{h_{03} - h_{02}}$$

For a perfect gas, heat capacity is constant, giving

$$\eta_c = \frac{c_p}{c_p} \cdot \frac{T_{03s} - T_{02}}{T_{03} - T_{02}}$$

$$T_{03} = T_{02} + \frac{T_{03s} - T_{02}}{\eta_c} = 1153.0\ K.$$

The work input to the compressor is obtained from the steady flow energy equation written for a perfect gas:

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[8] The isentropic relations for perfect gas are provided in the Thermofluids Data Book on page 9.
[9] The isentropic efficiencies are defined in the Thermofluids Data Book on page 16.

$$\dot{Q}_{23} - \dot{W}_{x,23} = \dot{m}_{air} c_p (T_{03} - T_{02})$$

Treating the compressor as adiabatic ($\dot{Q}_{23} = 0$), the compressor specific work input is given by

$$w_{x,c} = -\frac{\dot{W}_{x,23}}{\dot{m}_{air}} = c_p (T_{03} - T_{02}) = 748.6\ kJ.kg^{-1}$$

Tip: When using the isentropic efficiency to analyse a compressor or turbine it is a good idea to sketch the compression or expansion process on a T-s diagram, as it helps sense-check the result.

7.5.1.4 Burner 3 → 4

The stagnation pressure drops by 4 % in the burner:

$$p_{04} = 0.96 p_{03} = 6467.4\ kPa$$

The fuel flow rate can be determined by considering the balance of energy entering and leaving the combustor (the steady flow energy equation):

$$\dot{Q} + \dot{W}_x = 0 = \sum\nolimits_{reactants} \dot{m}_i \left(h_{i,0} - h_{i,in}\right) + \dot{m}_{fuel}(-CV) + \sum\nolimits_{products} \dot{m}_j \left(h_{j,out} - h_{j,0}\right).$$

For a perfect gas the enthalpy changes can be related to temperature changes. Since the fuel enters at the reference temperature T_{ref} and since we do not need to consider any heat of vaporisation, the equation becomes:

$$\dot{Q}_{34} + \dot{W}_{x,34} = 0 = \dot{m}_{air} c_{p,air} \left(T_{ref} - T_{03}\right) - \dot{m}_{fuel}(LCV) + \dot{m}_{prod} c_{p,prod} \left(T_{04} - T_{ref}\right)$$

Defining the fuel-air ratio, $f = \dot{m}_{fuel}/\dot{m}_{air}$, and noting that the mass flow rate of products $\dot{m}_{prod} = \dot{m}_{air} + \dot{m}_{fuel}$, gives

$$f = \frac{c_{p,air}\left(T_{ref} - T_{03}\right) + c_{p,prod}\left(T_{04} - T_{ref}\right)}{LCV - c_{p,prod}\left(T_{04} - T_{ref}\right)} = 0.01111.$$

Note that there is no heat addition to the combustion chamber – we assumed it is adiabatic. If anything, there is likely to be heat loss since the combustion chamber surfaces get very hot.

7.5.1.5 Turbine 4 → 5

We do not yet know the pressure ratio for the turbine. What we do know is that the turbine pressure drop must be sufficient to produce enough power to drive the compressor:

$$\dot{m}_{air} c_{p,air} (T_{03} - T_{02}) = \dot{m}_{prod} c_{p,prod} (T_{04} - T_{05}).$$

Rearranging and recalling that $\dot{m}_{prod} = (1 + f)\dot{m}_{air}$ gives

$$T_{05} = T_{04} - \frac{c_{p,air}(T_{03} - T_{02})}{(1+f)c_{p,prod}} = 826.9\ K.$$

We can use the isentropic relation to find pressure changes only for isentropic processes. To proceed we need to know what the turbine outlet temperature would have been if the turbine had been isentropic. Using the definition of isentropic efficiency for a turbine, applied to a perfect gas gives:

$$\eta_t = \frac{T_{04} - T_{05}}{T_{04} - T_{05s}},$$

and

$$T_{05s} = T_{04} - \frac{T_{04} - T_{05}}{\eta_t} = 752.2\ K.$$

The isentropic relation then gives

$$p_{05} = p_{04}\left(\frac{T_{05s}}{T_{04}}\right)^{\frac{\gamma_{prod}}{\gamma_{prod}-1}} = 400.4\ kPa.$$

7.5.1.6 Nozzle 5 → 6

The flow through the nozzle is assumed to be isentropic and perfect gas, so $T_{06} = T_{05}$ and $p_{06} = p_{05}$, and fully-expanded, so that $p_6 = p_{atm}$. By definition, the process between static and stagnation conditions is isentropic, so temperature T_6 can be determined from the isentropic relation:

$$T_6 = T_{06}\left(\frac{p_6}{p_{06}}\right)^{\frac{\gamma_{prod}-1}{\gamma_{prod}}} = 430.0\ K.$$

From the definition of stagnation enthalpy and the perfect gas assumption it follows that,

$$T_{06} = T_6 + \frac{V_j^2}{2c_p} = T_{05} = 826.9\ K,$$

from which the jet velocity is given by

$$V_j = \sqrt{2c_p(T_{06} - T_6)} = 934.5\ m.s^{-1}.$$

7.5.1.7 Performance

Assuming that the flow from the nozzle is fully expanded (so that there is no pressure thrust), the specific net thrust, per kg of air entering the engine is given by

$$X = \frac{F_N}{\dot{m}_{air}} = (1+f)V_j - V = 341.2\ m.s^{-1}$$

The thrust specific fuel consumption is given by

$$sfc = \frac{\dot{m}_{fuel}}{F_N} = \frac{\dot{m}_{fuel}}{\dot{m}_{air}}\frac{\dot{m}_{air}}{F_N} = \frac{f}{F_N/\dot{m}_{air}} = 32.55\ g.s^{-1}.kN^{-1}$$

The overall efficiency is given by

$$\eta_{oa} = \frac{\text{Power to aircraft}}{\text{Thermal power}} = \frac{VF_N}{\dot{m}_{fuel}LCV} = 43.12\ \%$$

The thermal efficiency is given by

$$\eta_{th} = \frac{\text{Power to jet}}{\text{Thermal power}} = \frac{\frac{1}{2}\left((1+f)V_j^2 - V^2\right)}{fLCV} = 54.29\ \%$$

The propulsive efficiency is given by

$$\eta_p = \frac{\text{Power to aircraft}}{\text{Power to jet}} = \frac{\eta_{oa}}{\eta_{th}} = \frac{VF_N}{\frac{1}{2}\left((1+f)V_j^2 - V^2\right)} = 79.44\ \%$$

7.6 Effects of modelling assumptions

Compared to the ideal cold-air standard Brayton cycle, the more realistic assumptions used in Section 7.5 tend to reduce the fuel efficiency of the engine. The effects that each assumption has on efficiency is illustrated in Table 2 and Figure 4 by switching the assumptions one at a time. The analysis starts with an ideal cold-air-standard Brayton cycle with pressure ratio of 30 and turbine entry temperature of 1500 K. Later these predictions are compared to those for a turbojet engine in flight at Mach 0.8 at 31,000 ft.

- Replacing isentropic turbomachinery with 90% efficient turbomachinery reduced the efficiency by 14.3 percentage points[10]: the irreversibility reduces how much power is recovered in the turbine and increases the power required by the compressor.
- Switching from the irreversible Brayton cycle to a turbojet at Mach 0.8 increased the efficiency by 13.1 percentage points for two reasons. First, part of the expansion in the turbojet is through the nozzle, which we have assumed to be isentropic, whereas all the expansion in the closed cycle is through an irreversible turbine. Second, the forward motion of the turbojet adds ram compression[11] resulting in a greater overall pressure ratio and higher efficiency.
- If the working fluid is treated as the same perfect gas throughout the cycle, the only difference between combustion and external heat addition is due to the additional mass flow through the turbine caused by fuel addition, which is relatively small. In reality combustion significantly changes the heat capacity of the working fluid.
- In reality the heat capacity of air increases substantially over the range of temperatures in a gas turbine, and there is a further increase when air reacts to produce CO_2 and H_2O. The cold-air perfect gas assumption has a large impact on predictions. Efficiency reduced by 2.7 percentage points when using more representative heat capacities for combustion products.

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[10] A percentage point is an absolute change in percentage value. For example, a change from 10 % to 20 % is a change of 10 percentage points in absolute terms, but a change of 100 % in relative terms.
[11] At Mach 0.8 and Mach 2.0, the ram compression increases the stagnation pressure at entry to the engine by a factor of 1.5 and 7.8 respectively.

- The commercial GasTurb13 software provides an alternative model for the combustion process and for the mixture thermodynamics, taking effects of incomplete combustion and the temperature-dependence of the heat capacity into account. Generally it is a good approximation to treat gas turbine combustion as complete, but it is important to account for the composition and temperature-dependence of heat capacities (this can be done by using a semi-perfect gas model, or by judicious choice of constant values for the heat capacity of air and combustion products).
- A 4% stagnation pressure drop in the combustor resulted in a 0.8 percentage point reduction in efficiency, and diverting 10% of the compressor air for cooling the turbine rotor reduced efficiency by a further

For simplicity in this module we often (but not always) use the cold air-standard assumption (c_p=1.005 kJ.kg^{-1}.K^{-1}; γ=1.40) and treat combustion as an external heat addition related to the fuel flow rate by

$$\dot{Q} = \dot{m}_{fuel}.LCV.$$

where LCV is the lower calorific value of the fuel (typically 43 MJ/kg for kerosene). It is emphasised that these approximations are not quantitatively accurate, however they still give correct trends and the simplicity helps explanation and understanding of the reasons for those trends. The simplicity also makes hand calculations feasible. But, if you go on to design gas turbines yourself, you will want to use more accurate thermodynamic modelling and that will require you to use a computer program, such as the GasTurb software used in your turbojet laboratory activity.

Table 2. Effects of modelling assumptions on efficiency and specific work predictions

Configuration	Process assumptions	Working fluid assumptions	η_{fuel}	$w_{x,net}$ [12] (kJ.kg^{-1})
Brayton cycle	ideal[13]	perfect gas: (cold air)[14]	62.2 %	515.
Brayton cycle	irreversible[13]	perfect gas: (cold air)[14]	47.9 %	468.
Turbojet	irreversible[13] & external heating	perfect gas: (cold air)[14]	61.0 %	477.
Turbojet	irreversible[13] & complete combustion[15]	perfect gas: (cold air)[14]	61.1 %	496.
Turbojet	irreversible[13] & complete combustion[15]	perfect gas: air/products[16]	58.4 %	533.
Turbojet	irreversible[13] & equilibrium combustion[17]	semi-perfect gas[18]	58.6 %	584.

[12] The effective net work output of the turbojet engines has been calculated as 0.5($m_{\text{out}}V_j^2$-$m_{\text{in}}V^2$).
[13] Ideal: η_c=η_t=100 %; irreversible: η_c=η_t=90 %
[14] Properties of cold air: c_p=1.005 kJ.kg^{-1}.K^{-1}; γ=1.40.
[15] Complete combustion of kerosene/air with 43 MJ/kg calorific value.
[16] Assumes air has properties of cold air and that combustion products have $c_{p,ex} = 1{,}100\ kJ.kg^{-1}.K^{-1}, \gamma_{ex} = 1.35$.
[17] Combustion model accounting for incomplete combustion/chemical equilibrium as modelled by GasTurb13.
[18] Semi-perfect (i.e. temperature- and composition-dependent) heat capacities as modelled by GasTurb13.

Turbojet	irreversible[13] & equilibrium combust.[17] & 4 % pressure drop[19]	semi-perfect gas[18]	57.8 %	576.
Turbojet	irreversible[13] & equilibrium combust.[17] & 4 % pressure drop[19] & blade cooling[20]	semi-perfect gas[18]	55.7 %	501.

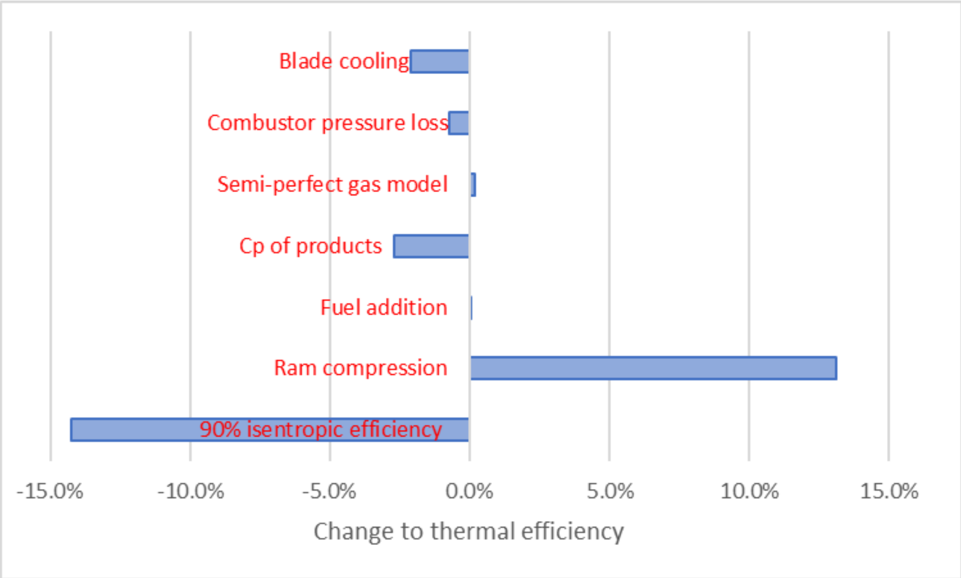


Figure 4. Effects of engine configuration and modelling assumptions on thermal efficiency. Data as per Table 2.

7.7 Turbine Entry Temperature

7.7.1 Effect on efficiency and power

The efficiency of the ideal Brayton cycle depends on pressure ratio, but is independent of the turbine entry temperature. However irreversibility changes that. Figure 5 shows how the cycle efficiency and non-dimensional work output depend on cycle temperature ratio (T_{04}/T_{02}) and on the isentropic efficiencies η_c and η_t. The optimum pressure ratio for maximum net power (or thrust) is achieved at a much lower pressure ratio than for optimum efficiency. For this reason, engines for fighter aircraft (which prioritise thrust) typically use a compressor pressure ratio around 15, compared to pressure ratios closer to 45 in modern civil turbofans (which prioritise range and fuel efficiency).

[19] 5% drop of absolute total pressure in combustion chamber.

[20] Assumes 10 % of compressor air is used for blade cooling, modelled by re-introducing the cooling flow after the high pressure turbine.

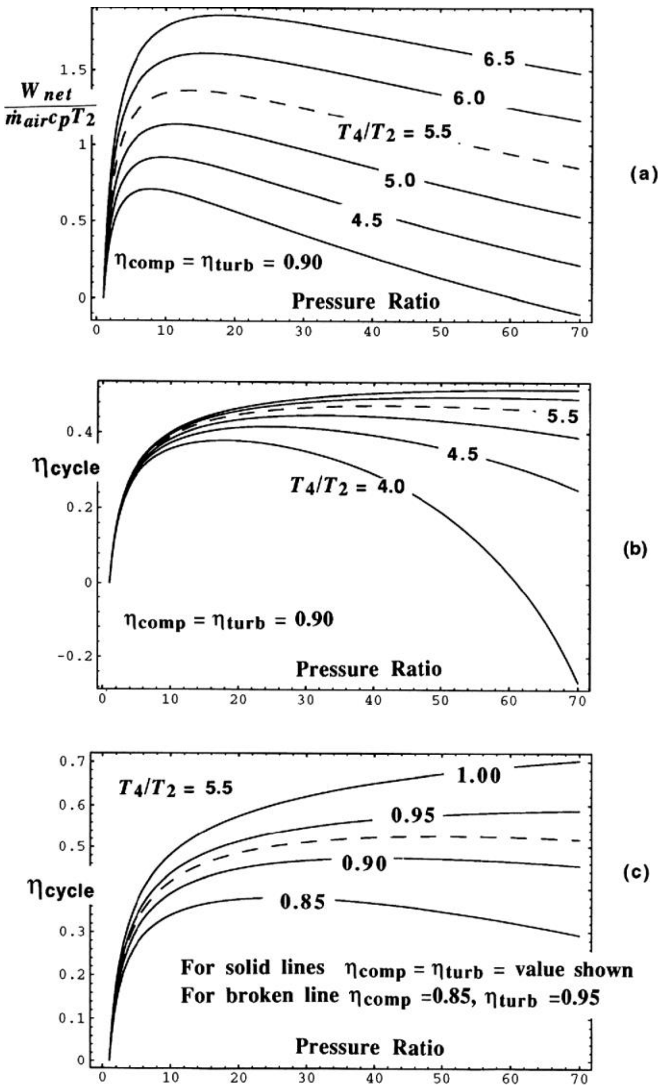


Figure 6. Non-dimensional power and cycle efficiency for an idealised gas turbine.

7.7.2 Blade cooling

Increasing T_{04}/T_{02} is generally beneficial for the efficiency and power output, however the turbine entry temperature $T_{04}$is limited by the need to keep the turbine at a safe operating temperature, and values of $T_{04}/T_{02} \approx 6$ are typical for cruise conditions.

Typical values for turbine entry temperature (TET) in a high-bypass ratio turbofan engine are presented in Table 3 at take-off, top-of-climb and at cruise. The engine rotational speed (which depends primarily on (T_{04}/T_{02})) is greatest at top-of-climb. However the TET is hottest at take-off due to the higher atmospheric temperature at sea level. The TET at cruise is critical for engine life since a civil aircraft engine spends most of its operational live at the cruise condition.

Table 3. Engine inlet and turbine entry temperatures for a high-bypass ratio turbofan engine at take-off, top of climb and start of cruise.

	Engine inlet	**Turbine entry**	
	T_{02}	T_{04}	T_{04}/T_{02}

Take-off (standard day, sea level)	288.15 K	1750 K	6.07
Top-of-climb (35,000 ft, *M*=0.78)	245.4 K	1600 K	6.52
Start of cruise (35,000 ft, *M*=0.78)	245.4 K	1500 K	6.11

Turbine entry temperatures of 1500 K are now usual for cruise, and 1750 K may be sustained for the limited period of take-off. The melting temperature of current single-crystal nickel super-alloy turbine materials is around 1500 K, however considerations of oxidation, thermal fatigue and creep limit the safe-operating temperature of the material to lower temperatures, requiring that the turbine must be cooled. For aircraft applications, compressed air from before the combustor which, at a mere 800-900 K, is cool in comparison to the combustor outlet temperature, is bled off from the compressor and passed through the turbine structures. The cooing air may be passed along cooling channels within the turbine structure (so-called *convective* cooling), and bled out of small holes and slots on the turbine surface to provide a protective layer of cooler air (so-called *film* cooling), as indicated in Figure 7. The development of cooling-air technology has enabled an increase in the turbine entry temperature substantially faster than the improvement in the capabilities of the materials available, as shown in Figure 8.

Adoption of ceramic turbine structures remains some way off due to the unacceptably low toughness that has been achieved by ceramic components to date, although the use of ceramic thermal barrier coatings which have low thermal conductivity has contributed around 100 K of the improvement in turbine entry temperature.

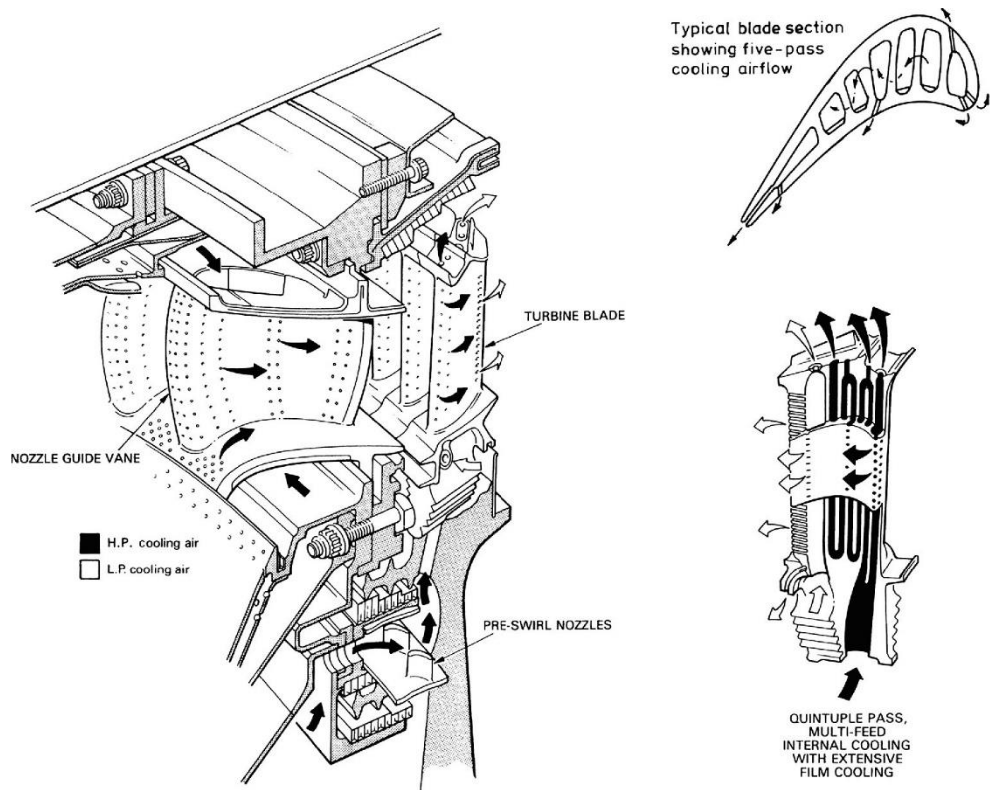


Figure 7. Arrangements for cooling an HP turbine and rotor blade. It should be noted that the rotor blade configuration is unique to Rolls-Royce, with a shroud on the tip. (From the *Jet Engine*, 1986)

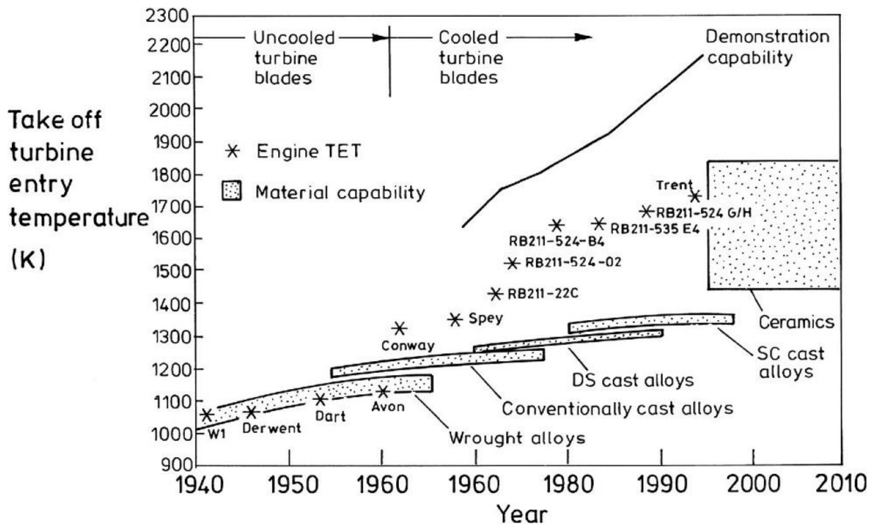
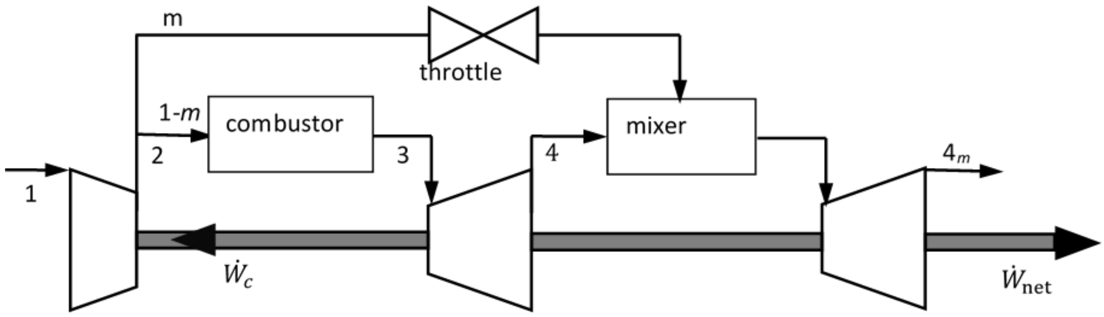


Figure 8. Turbine entry temperature for Rolls-Royce engines since 1940: a figure drawn in about 1993. (For cooled turbines this figure shows the mixed-out temperature at exit from the first turbine stator row.)

While high TET benefits efficiency, there is an efficiency penalty associated with blade cooling. The cooling air is compressed to a high pressure and then looses pressure as it passes through the narrow cooling channels without contributing to the engine power output. The impact on efficiency can be analysed in a highly simplified way by constructing thermodynamic models of the kind illustrated in Figure 9, and this modelling approach is employed in the GasTurb software. Work is lost because flow through ha throttle and also the mixing of hot and cold gases are both irreversible processes.

There is a trade-off between the additional efficiency that comes from increasing TET and the loss of efficiency due to provision of cooling air. Typically 15-25 % of the compressor air is used for cooling the combustor and high-pressure turbine in civil turbofan engines.



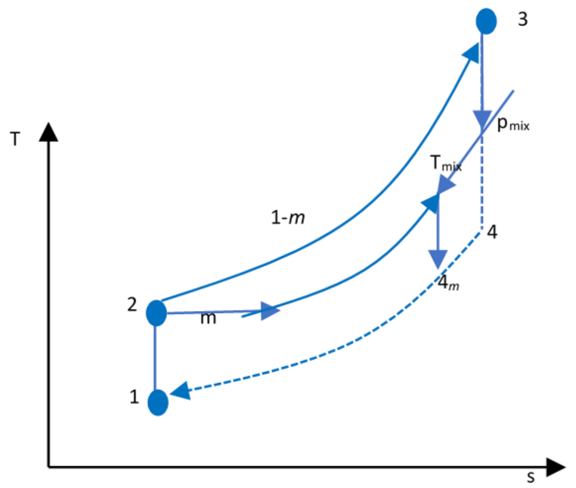


Figure 9. Schematic diagram (top) and corresponding *T-s* diagram (bottom) for a gas turbine with blade cooling modelled by a fraction *m* of compressor air bypassing the combustor, and achieving cooling by mixing-in this cooler gas part-way through the turbine.

7.8 Reheat (Afterburning)

In propulsion systems, reheat is a method for augmenting the thrust of a jet engine by burning additional fuel downstream of the turbine. Raising the temperature increases the speed of sound and the flow velocity through the choked propulsion nozzle. Reheat systems are often combined with variable-area propulsion nozzles that can be adjusted during flight to allow for the increased volume flow rate when the reheat burner is ignited.

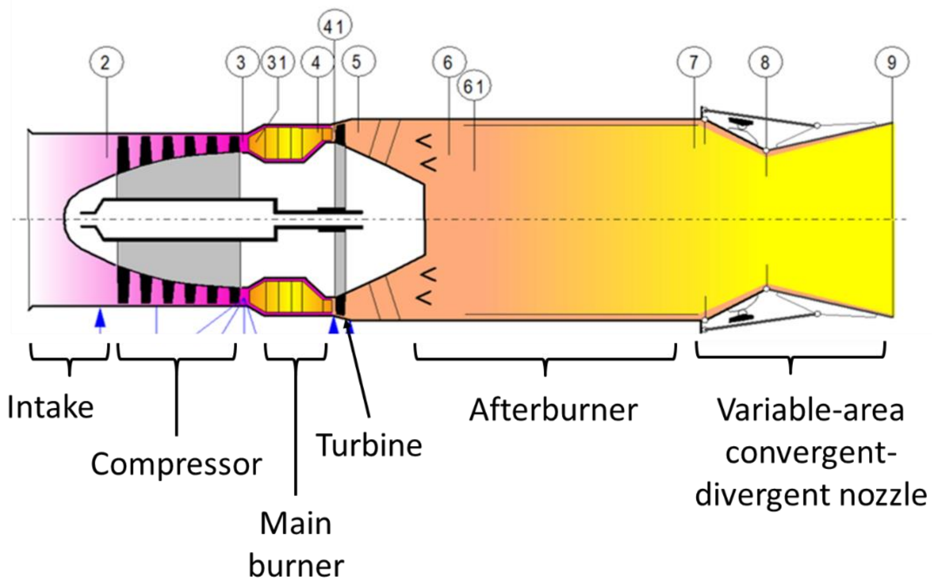


Figure 10. Schematic of a turbojet with reheat.

Reheat is inherently less efficient than adding heat in the primary combustor, so typically it is only used in a short portion of a mission when maximum thrust is required. If increased thrust is required throughout a mission, it would be more efficient to install a more powerful gas turbine without reheat.

7.9 Summary

The thermodynamic analysis of the gas turbine engine shows that the fuel efficiency can be increased primarily by:

- increasing the compressor pressure ratio,
- increasing the temperature ratio T_{04}/T_{02},
- increasing the isentropic efficiencies of the compressor and turbine,
- reducing other losses, such as pressure drops in the combustor, or due to cooling flows.

A number of simplifying assumptions have been considered in the various analyses in this section:

- omission of cooling flows (air taken from the compressor to cool or shield the turbine blades);
- treating the working fluid as perfect gas with properties of ambient air (y = 1.40) in some or all of the engine;
- neglecting mass of fuel in calculating power from the turbine;
- omitting pressure drop in the combustor, which may be 4 % of maximum pressure;
- simple assumptions have been made for compressor and turbine efficiency;
- neglecting electrical power off-takes and air for cabin pressurisation.

The suitability of these simplifications depends on the purpose of your analysis, and you may be expected to use different sets of assumptions for different problems. Despite these shortcomings the trends predicted by the simpler assumptions are correct and the magnitudes for efficiency and available power are plausible.

Note that a number of the equations used in this chapter are given in the databook, including the steady flow energy equation, isentropic relations for perfect gases, and definitions of isentropic efficiencies.