Preparing for the Future: Reducing Gas Turbine Environmental Impact—IGTI Scholar Lecture

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In the long term, the price of fuel will rise and it is now urgent to reduce carbon dioxide emissions to avoid catastrophic climate change. This lecture looks at power plant for electricity generation and aircraft propulsion, considering likely limits and possibilities for improvement. There are lessons from land-based gas turbines, which can be applied to aircraft, notably the small increases in efficiency from further increase in pressure ratio and turbine inlet temperature. Land-based gas turbines also point to the benefit of combining the properties of water with those of air to raise efficiency. Whereas the incentive to raise efficiency and reduce CO_2 will force an increase in complexity of land-based power plant, the opportunities for this with aircraft are more limited. One of the opportunities with aircraft propulsion is to consider the whole aircraft operation and specification. Currently the specifications for new aircraft of take-off and climb thrust are not fully consistent with designing the engine for minimum fuel consumption and this will be addressed in some depth in the lecture. Preparing for the future entails alerting engineers to important possibilities and limitations associated with gas turbines which will mitigate climate change due to carbon dioxide emissions. [DOI: 10.1115/1.4001221]

1 Introduction

"Prediction is very difficult, especially about the future," wrote the physicist Niels Bohr. Nevertheless some predictions seem pretty secure. One is that the future price of fuel will rise for a number of reasons. Another, in a way more compelling, is the need to reduce carbon dioxide, CO_2, emissions to mitigate the change in climate. In fact, the pressure to reduce CO_2 emission may be felt through taxes and charges as an effective increase in the cost of fuel burned. Some of the engineering responses to this are the subject of this lecture.

The first part of the lecture will be devoted to land-based power generation and the second to aviation. The issues are very different. Land-based power generation can adapt by a number of routes, with increased complexity and cost of plant. The treatment of land-based engines will be used to bring out some general points, particularly about thermodynamics, with a view to opening up possibilities for gas turbines. In addition, some conclusions can be reached which are applicable to aircraft engines. Aircraft propulsion can be expected to remain with the gas turbine and to experience comparatively little increase in complexity. This resistance to complexity in the aircraft engine reflects the need to keep the power plant weight low and hold down development cost. In addition, greater complexity is normally associated with a reduction in reliability, which is not tolerable in a commercial aircraft application. For aircraft, it is also necessary to maintain a sleek overall shape to minimize aerodynamic drag and this limits the scope for adding certain types of device, which would be possible on land.

For aircraft it seems improbable that hydrocarbon fuel will be replaced and it is inconceivable to capture the CO_2, so the contribution to climate-change mitigation has to be the increase in efficiency of the engine, aircraft, and aircraft operation. Fortunately this also leads to a minimum fuel cost. One of the complications is that the lowest fuel cost does not in general correspond to the lowest airline operation cost, since they can raise their revenue by increasing utilization, for example, by flying at higher speed. For the aircraft engine there are conflicting requirements. One intense and obvious conflict is between lower fuel consumption and engine weight. In terms of engine cycle choice there is a conflict between requirements to minimize fuel burn during cruise (when the air is cold) and the ability to take-off without exceeding allowable metal temperatures when the incoming air is warm. To understand the selection of an aircraft engine cycle, one therefore needs to understand the operation away from the design point and to do this some background for the aircraft at different operating conditions is needed. Because these off-design topics are not a part of the general education and background, some coverage of basic ideas is given here. As may be inferred from this, the character of lecture relating to aircraft engines will be different from that for land-based engines.

2 Power Generation: Land-Based Engines

2.1 Background. Another quotation, attributed to Louis Pasteur, is "Fortune favours the prepared mind." This fits the gas turbine industry rather well. For a long time heavy-frame gas turbines were available and their manufacturers did much to support the International Gas Turbine Institute and its annual conference in the 1970s and 1980s. There was relatively little uptake of their products, for gas turbine efficiency was not particularly high and they required high quality fuel. When, however, natural gas became abundant and regulations in many countries were altered to allow it to be burned for electricity generation, a wonderful business opportunity arose for those manufacturers of large gas turbines who were prepared. Because the industry had already developed large gas turbines, it was ready and able to take advantage.

For over a century, the principal method of generating electricity has consisted of burning coal or oil to generate steam and expanding the steam through a series of turbines before admitting

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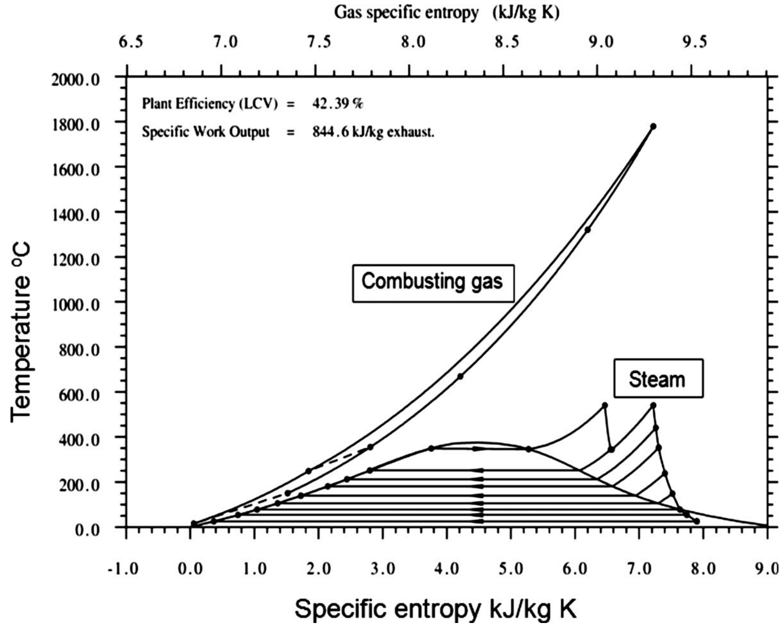

Fig. 1 Temperature-entropy chart for steam power plant [1]

it to a condenser. Steam power plant had reached a considerable degree of sophistication by 1950, with several feed heaters, reheaters, and an economizer. With few exceptions, the steam conditions leaving the superheater changed little from about 1950 to 2000: about 165 bars and 565°C, the latter fixed by the upper level of affordable steel for the steam tubes. The plant efficiency was about 40% based on lower calorific value. Recently the steam is being raised at supercritical conditions, around 300 bars and 600°C, giving a plant efficiency around 43%, but efficiencies close to 46% are possible if cold sea water can be used to cool the condenser. Steam has certainly not been diminished in importance by the gas turbine and it is relevant to ask what are the particular features of steam plant and gas turbines, which are advantageous and which are a hindrance.

The steam power plant has two principal advantages and one principal drawback. The advantages are the ability to burn any fuel and the fact the compression work in the feed pump is typically two orders of magnitude less than the power output from the turbines. The disadvantage is a thermodynamic one: although the flame temperature in the furnace may be over 2000 K, the temperature of the steam is only approaching 900 K. This is illustrated in Fig. 1 in which the combustion products are superimposed on the steam cycle, with the entropy (abscissa) of the combustion products expanded to match that of the steam. Thinking in terms of Carnot efficiency, by dropping the temperature between combustion products and steam the potential for high efficiency has been thrown away.

3 Simple-Cycle Gas Turbine

The gas turbine can be said to have two principal disadvantages and one principal advantage. The first disadvantage is the requirement for high quality fuel to avoid corroding the hot components or clogging the turbine with slag. (This remains a serious inhibition for the use of gas turbines on merchant ships, which typically burn very low-cost bunker oil unsuitable for gas turbines.) The other fundamental disadvantage is that the work of compression is a substantial fraction of the work from the turbine: the net power being the difference between two large quantities is sensitive to the efficiency of both turbine and compressor.

The principal advantage of the gas turbine is a thermodynamic one: the gas is used to generate work at a temperature relatively close to the flame temperature. A modern large (say, 250 MW) gas turbine on its own might have an efficiency between 35% and 40%, while an aeroderivative, operating at higher overall pressure ratio, can be about 42%. Figure 2 shows the thermal efficiency for different simple-cycle gas turbines as a function of turbine entry temperature, derived from Wilcock et al. [2] for combustion with natural gas. The results are presented for three overall pressure ratios, 30, 45, and 60 (all high for heavy-frame land-based machines) and for three cases:

(a) compressor and turbine efficiencies of unity (η_s=100%), an idealization that helps put an upper bound on what is achievable, and no turbine cooling (shown black in Fig. 2);
(b) compressor and turbine efficiencies of 0.9 (η_s=90%), which is not far from what might be achieved in practice, and no turbine cooling (shown blue in Fig. 2);
(c) compressor and turbine efficiencies of 0.9 (η_s=90%) and cooling (shown red in Fig. 2); the cooling was referred to

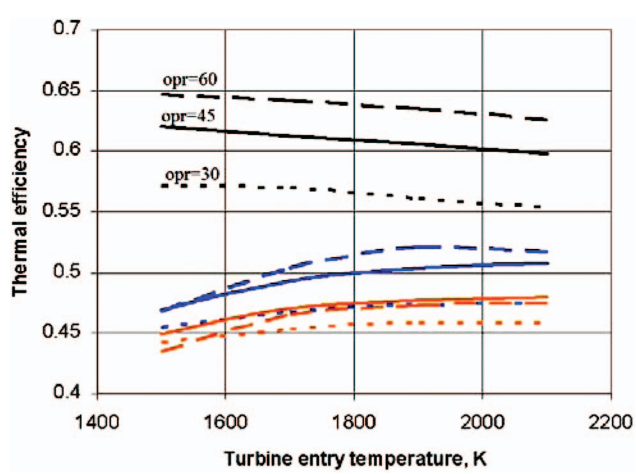

Fig. 2 Thermal efficiencies of simple gas turbines from Wilcock et al. [2]. Overall pressure ratios of 60 for broken line, 45 for solid line, and 30 for dotted line. (*a*) black is η_s=100%, no cooling; (*b*) blue η_s=90%, no cooling; (*c*) red η_s=90%, "advanced" cooling.

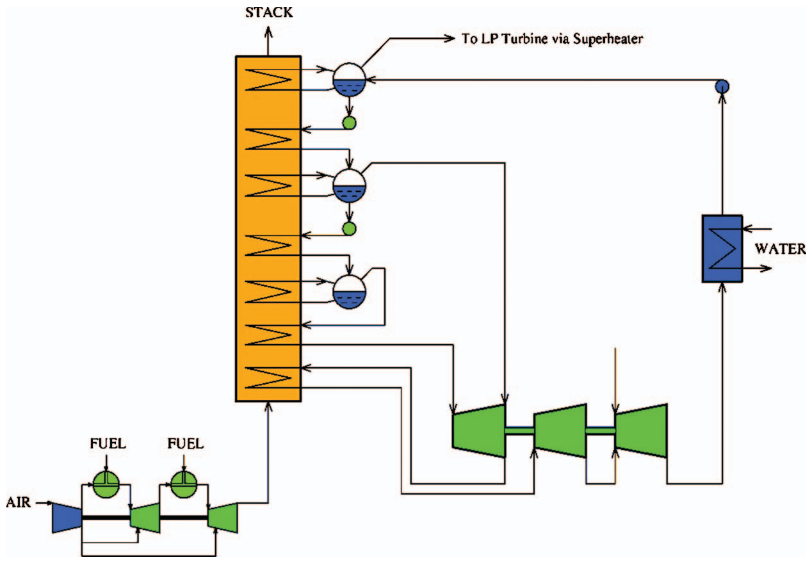

Fig. 3 Alstom GT24/26 combined-cycle gas turbine, with reheat of gas in a second combustor after HP turbine [1]

as "advanced cooling" by Wilcock et al. [2] with the amount of cooling air increased as the gas became hotter so as to maintain a constant turbine metal temperature.

It is observed that for the most idealized cycle, case (a) in Fig. 2 (compressor and turbines isentropic, no cooling), the thermal efficiency falls as temperature rises. The fall in efficiency with increase in temperature is a result of the exhaust gas properties changing as fuel fraction rises. This fall is a surprise to many people because for a cycle with an ideal compressor and turbine, using ideal gases with constant specific heat, the thermal efficiency is independent of turbine entry temperature. Furthermore, for more realistic cycles the efficiency is commonly expected to increase with turbine entry temperature, which as shown in Fig. 2, is not always realistic.

When the compressor and turbine isentropic efficiencies are reduced to 0.9, Fig. 2 case (b), the thermal efficiency only falls with temperature for the highest pressure ratio and then only above about 1900 K. Overall the effect of nonisentropic compression and expansion is to reduce the efficiency substantially and to reduce the benefits of increased pressure ratio on efficiency.

Cooling the turbine, Fig. 2 case (c), continues the downward trend on thermal efficiency and the benefit for efficiency of increased pressure ratio and higher temperatures is small. There is virtually no benefit in terms of efficiency beyond a pressure ratio of about 45 and for combustor exit temperature beyond about 1800 K (3700°F). The temperature referred to here is that at entry to the high pressure (HP) turbine nozzles, before the cooling air is mixed in. For aircraft engines it is normal to refer to the temperature downstream of the nozzles, assuming full mixing of the cooling air; this temperatuer is typically about 100 K lower than at nozzle entry.

The absolute levels of efficiency in Fig. 2 are somewhat higher than observed in practice because a number of small effects and leakages are neglected. The simple gas turbine cycle, even in its modern advanced form, does not offer high efficiency and a figure of around 40%–42% is broadly appropriate when the pressure ratio is allowed to rise to the optimum, around 40.

In discussing plant the sole criterion discussed here is efficiency. For a real machine power output per unit mass flow through the engine is important, for this fixes the size that strongly affects the cost. The economic optimum will therefore not be at the highest thermal efficiency.

4 The Combined-Cycle Gas Turbine

It is the combined cycle, where the exhaust from the gas turbine is used to generate steam, which makes the gas turbine so attractive. A combined-cycle efficiency near 60% is possible for the most advanced plant, known as H-class. Combined cycles are considered in some depth by Horlock [3].

Figure 3 shows a schematic of one arrangement for combined-cycle plant, the Alstom GT24/26. This is unusual in having two combustors in the gas turbine, for reasons discussed below. The corresponding temperature-entropy diagram of the Alstom GT24/26 is shown in Fig. 4, where the entropy scale of the gas is stretched to match that of the steam. (Dots on the lines in Fig. 4 correspond to corner points in gas and steam.)

The crucial step is the matching of the useful properties of gas and water. The relatively low efficiency of the gas turbine on its own results in a high exhaust temperature. It is therefore beneficial to take the highest possible work from the steam cycle, so in a combined cycle the gas turbine is designed so that the exhaust gas is at about 600°C and therefore able to generate steam at the highest practical temperature. For most manufacturers of heavy-frame gas turbines for combined-cycle application, this means keeping the overall pressure ratio of the gas turbine down near to 20, so that the expansion in the turbine does not drop the exhaust gas temperature below the required level. Alstom use a higher pressure ratio but have a second combustor to raise the gas temperature after the HP turbine and again arrive at an exhaust temperature high enough to make best use of a steam cycle.

Once having decided to have the complication of a steam cycle, there are other effective ways to use the water, and General Electric, for example, uses steam to cool the turbine blades of their H-class engine.

5 Heat Exchangers

There is another way to improve the performance of the simple gas turbine (i.e., without the steam cycle), which is to use heat exchangers. The outline of such a scheme is shown in Fig. 5 for the WR21 engine developed by Rolls-Royce for the U.S. Navy.

TheWR21 engine is based on a three-spool aircraft engine, the RB211, but the low pressure (LP) turbine, which would drive the fan, instead drives the ship propulsion through a gearbox. The layout of the three-spool engine made it relatively easy to split the compression process so that the air is cooled in a heat exchanger

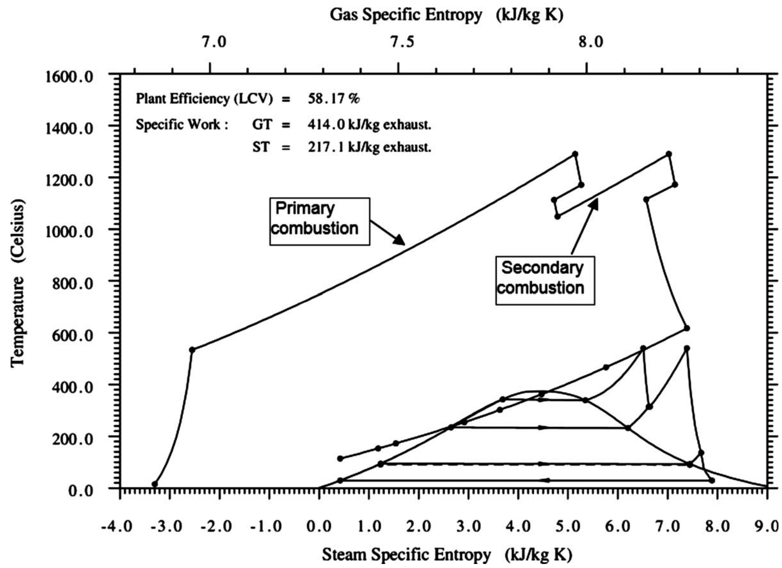

Fig. 4 Temperature-entropy diagram of Alstom GT24/26 with reheat of gas after HP turbine [1]

using sea water before entering the HP compressor—this intercooler primarily reduces the compression work in the HP compressor and increases net power. The gas leaving the turbine enters the recuperator a heat exchanger which uses the exhaust gas out of the turbine to raise the temperature of the air leaving the compressor before it enters the combustor, reducing the amount of fuel required to reach a given turbine inlet temperature. This raises the efficiency somewhat at full power but, more significantly for this marine application, prevents the efficiency falling very substantially at much reduced power.

Figure 5 is so simple, and heat exchangers are so conceptually straightforward, that one might wonder why all gas turbines do not incorporate recuperators. As Fig. 6 shows, the recuperator with its pipework dominates the picture and is bigger than the gas turbine itself, yet space is very restricted on ships so every effort has been made to minimize the size of this component.

Moreover, given the conceptual simplicity of the heat exchanger, it may be surprising to some that the recuperator is the component that caused most development difficulty for the WR21. In particular, transient thermal stress is a challenge for all recuperators. This goes some of the way to explain why heat exchangers are not much used in large gas turbines and why, in particular, they do not seem an attractive idea for aircraft engines. The recuperator offers an advantage when the gas turbine is asked to operate at lower pressure ratios and temperature ratios than those for maximum power, as in a marine propulsion application, because the efficiency does not fall as sharply as it does for a simple cycle. (As will be shown below, although the thrust for an aircraft engine at take-off is much higher than at cruise, the pressure and temperature ratios are similar at both conditions. Therefore this off-design benefit by intercooling and recuperating would be much smaller for aircraft engines.)

Simple cycle calculations show that there is a pressure ratio at which the recuperator no longer offers an advantage for efficiency; this pressure ratio is increased when an intercooler is used. The General Electric LMS100 is based on the intercooling between the LP compressor and the HP compressor, the latter an aeroderivative. With this the overall pressure ratio can be raised to 42 and the overall efficiency to about 46%. The schematic drawings offered by the company again show that the heat exchanger is large in relation to the turbomachinery parts of the plant.

6 Water Addition to Gas Turbines

The use of water as an addition to the gas stream in a gas turbine has been advocated quite frequently and in many different ways. It is possible, for example, to generate steam using the hot exhaust gas from the turbine and inject this steam where the pres-

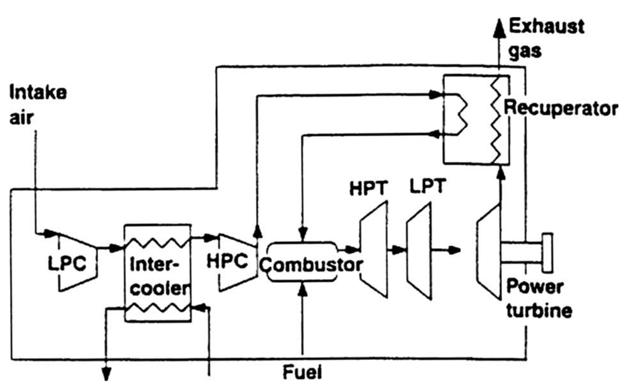

Fig. 5 Rolls-Royce WR21 engine for marine propulsion. VAN refers to variable area nozzle, a means to vary power output and speed [4]

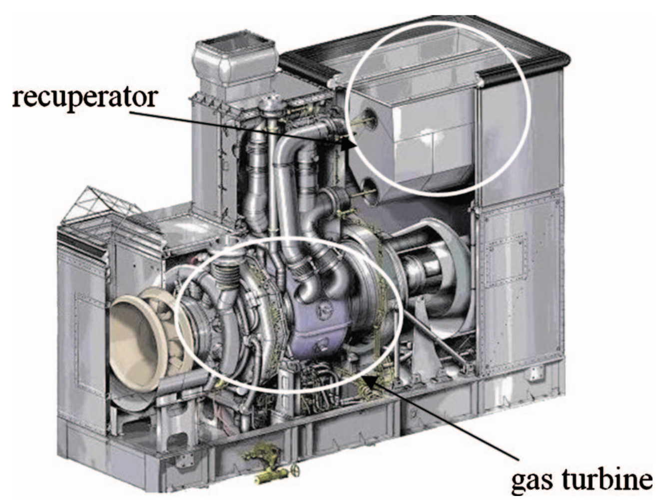

Fig. 6 The Rolls-Royce WR21 marine gas turbine [4]

sure is high somewhere upstream of the turbine or in the turbine. This is referred to as the stream injected gas turbine (STIG) and is offered by GE with the LMS100, described above. Steam injection increases the power output of about 15% and raises efficiency to nearly 50%. STIG provides a way of extracting extra work from the hot gas leaving the gas turbine without having the cost of a separate steam cycle.

An alternative scheme is to evaporate water in the warm compressed air, referred to as the humid air turbine (HAT). This has been the subject of considerable work in Lund University, where they operate a small gas turbine in this mode [5], but is now of interest across the world [6,7].

The benefits of adding water or steam to the gas stream appear substantial, while risk and uncertainty are comparatively low. It does appear to be a promising way of adapting the gas turbine for land-based use, particularly where the scale is not large enough to justify the costs associated with the separate steam cycle of the combined-cycle plant. Despite the long standing theoretical and academic interests in gas turbines wherein water or steam is added to the gas stream, they have not been widely adopted. Part of the problem is that although in a schematic diagram it seems that the steam or water can be added to an existing machine (since the pressure ratio normally stays the same), in fact, if the power increase is large, the flow capacity of the turbine is radically altered and a wholly new machine is required.

Despite the small take up for adding water inside engine, water is widely used as a mist (inlet fog) at gas turbine inlet to lower the inlet temperature by evaporation, thereby restoring or raising the power output. This is particularly attractive when the engine is operated in hot, dry regions. One of the biggest uses of water injection into gas turbines is not to raise efficiency but to reduce NO_x from combustion as an alternative to "dry" premixed lean combustion. Aeroderivative land-based gas turbines with premixed low-emission combustors were subject to development problems, but these difficulties were avoided by running conventional combustors "wet," which involves injecting liquid water with the fuel into the combustor. Not only is the combustion free from damaging oscillations and the NO_x level low, but there is a worthwhile increase in power and reduced installed cost. This approach is now widespread. Evidently, when the incentive is large enough, the complications and cost associated with providing demineralized water and injecting it into an engine can be accommodated.

7 The Future for Power Generation

Standing back to consider developments in power generation up to the present day, the innovation represented by the combined-cycle gas turbine is quite modest. Even raising plant efficiency from under 40% to around 60% has not gone far to solve the problem that too much CO_2 is getting into the atmosphere. This pessimistic view is particularly true since the combined cycle is dependent on burning highly refined fuel, usually natural gas. Coal is far more abundant than natural gas or oil and sober assessments conclude that coal is going to remain the primary fuel for electricity generation for many years to come.[1] Currently an 80% reduction in greenhouses gas emissions by 2050 is viewed by many in western Europe and the United States as essential to avoid the risk of catastrophic climate change.

7.1 Carbon Dioxide Separation and Sequestration. An essential technology to reduce the emission of CO_2 will be by capturing it and sequestering it underground, so-called carbon capture and sequestration (CCS). The storage is likely to be either in depleted oil and gas wells or in saline aquifers; less satisfactorily it might be placed on the sea bed. The pressure required for the CO_2 prior to injection is around 150 bars.

There are numerous schemes for using fossil fuel, often coal, and separating the CO_2 before storing it. In precombustion capture, the CO_2 can be removed from the fuel by chemical processing before it enters the combustor. Alternatively, using postcombustion capture, the CO_2 can be removed by collecting it from the flue gas. Because with postcombustion processing most of the flue gas is nitrogen, the partial pressure of CO_2 in the exhaust is low, typically no more than about 0.15 bar. The pressure of the CO_2 must be raised from this value to the high value for underground injection. The separation of the CO_2 is equivalent to raising its partial pressure to ambient and, because this is done chemically, the energy cost is several times the thermodynamic minimum. On the other hand, the technology for postcombustion processing is well established and it can, in principal, be retrofitted to coal burning steam power plant which were built without this facility always assuming that the infrastructure of pipes and storage wells is available. The International Energy Agency [8] anticipated in their 2006 report that the overall plant efficiency burning coal will be decreased by about 9 percentage points with postcombustion capture, with an increase in capital cost and consequent increase in electricity cost of about 39%. (In all the comparisons for power plant, the data from which the changes in efficiency and cost are derived are modern supercritical steam plant burning pulverized coals.)

An important variation of postcombustion capture is to replace air by oxygen in the combustor, so-called oxy-fuel combustion, so the exhaust gases are not dominated by nitrogen. One solution is shown schematically in Fig. 7, using a gas turbine for the combustion to create a combined (i.e., gas and steam) cycle. In all cases of oxy-fuel combustion it is necessary to dilute the combusting gases to prevent excessively high temperatures in the absence of nitrogen as a dilutant. In Fig. 7, this is done by recycling some of the CO_2, which is compressed and mixed with the oxygen, but in other schemes the oxygen is diluted with steam. Steam can be removed easily from the exhaust gas by condensation to leave CO_2. In all oxy-fuel plant there is a requirement for an air separation unit to produce the oxygen, with consequent energy cost. Currently the energy cost of producing pure oxygen is quoted by Praxair to be around six times the theoretical minimum, pointing up the need for research and development in many underlying technologies.

Other oxy-fuel schemes have combustion in a furnace to raise steam, like a current steam power plant, but with oxygen diluted with CO_2 in place of air. For a supercritical steam power plant with oxy-fuel combustion, the International Energy Agency [8] estimate about a 9 percentage points loss in overall plant efficiency with an increase in electricity cost of about 44%.

For the precombustion route, the solution is inherently more radical and an example is shown in a simplified form in Fig. 7, where the coal is turned into gas and then by shift conversion into hydrogen and CO_2. The CO_2 is then removed to be stored and the hydrogen is burned in a gas turbine, which forms part of a combined-cycle plant. The International Energy Agency [8] estimates that the thermal efficiency of a scheme such as that in Fig. 8 would be reduced relative to that for a modern coal fired plant by about 12 percentage points, with the cost of electricity raised by about 28%.

Both Figs. 7 and 8 show highly simplified diagrams of the plant, in which the gas turbine is an essential part. If either of these two approaches is adopted as a primary route to the low-carbon use of coal, there is going to be an enormous increase in the application of gas turbines to the base-load generation of electricity. There will be differences from current gas turbines. In the precombustion route to CCS, the gas turbine would be compressing air and burning hydrogen as the fuel. In the oxy-fuel gas turbine, the compressor could be working with CO_2 and the com-

[1]The U.S. Government Energy Information Administration released data in 2008. Assuming 2006 consumption, it can be calculated that accessible coal can meet consumption for about 200 years, while natural gas can meet consumption for about 60 years. If, however, natural gas replaced coal the proven reserves of natural gas would be exhausted in about 30 years.

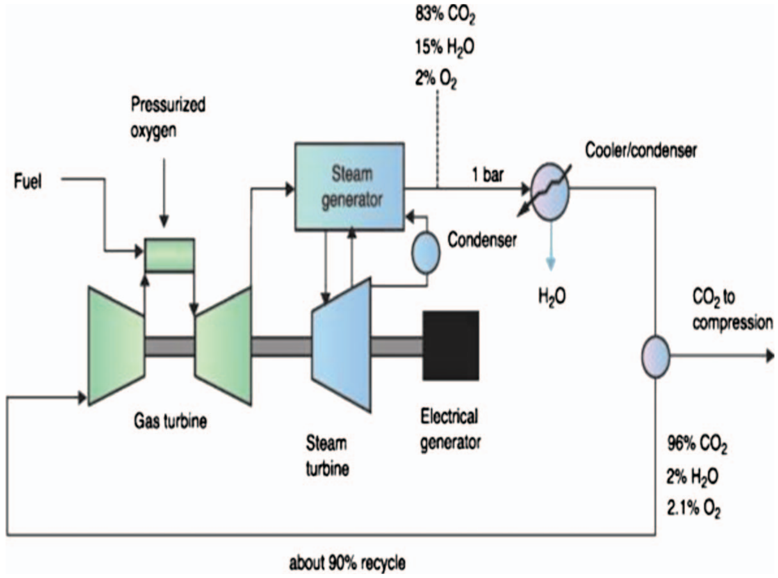

Fig. 7 A simplified scheme using a gas turbine for power generation with CO_2 separation with oxy-fuel combustion [9]

bustor would have pure oxygen injected to burn gaseous fuel in the O_2/CO_2 atmosphere. Either way, the gas turbine is now only a small part of the overall plant, which has become more like a chemical engineering site than a conventional current power plant. The pressures and temperatures of the gas streams need to be altered around the plant so there will be numerous heat exchangers and compressors.

The loss in plant overall efficiency and increase in the cost of electricity estimated by the International Energy Agency [8] have already been referred to. It is to be emphasized that these are merely estimates and, because the schemes are so capital intensive, are susceptible to errors from many sources. As a result, one cannot know at this stage which CCS approach will be the best and there is a natural hesitancy to commit to one scheme. The grounds for uncertainty are not of the scientific kind (it is known that the processes required can be carried out) but rather how will they scale up to work with the hundred or thousand MW plants whih need them.

One set of figures on cost does seem to be relatively sound: when coal is the fuel the cost of CO_2 release avoided is estimated by the International Energy Agency [8] to be around \$30/ton of CO_2, whichever method of separation is used. For natural gas, however, the cost per ton of CO_2 release avoided is two or three times higher. This is because natural gas burning plants produce less CO_2 per unit of electrical energy produced. This reinforces what can be concluded merely by considering the amount of coal available compared with the amount of natural gas: the principal target for CO_2 separation and storage should be plants burning coal to produce electricity.

8 Key Observations Related to Land-Based Power

The simple gas turbine is not very efficient and the combined cycle with a steam cycle transforms the gas turbine into the most efficient large plant available. In general, the combination of the gas turbine with steam or water offers successful power plant with opportunities for further improvement.

The difficulties and cost of heat exchangers should not be underestimated: They tend to be bulky and heavy, to introduce significant pressure losses (not least in the pipe work to and from them) and to be subject to cracking. In dirty environments heat

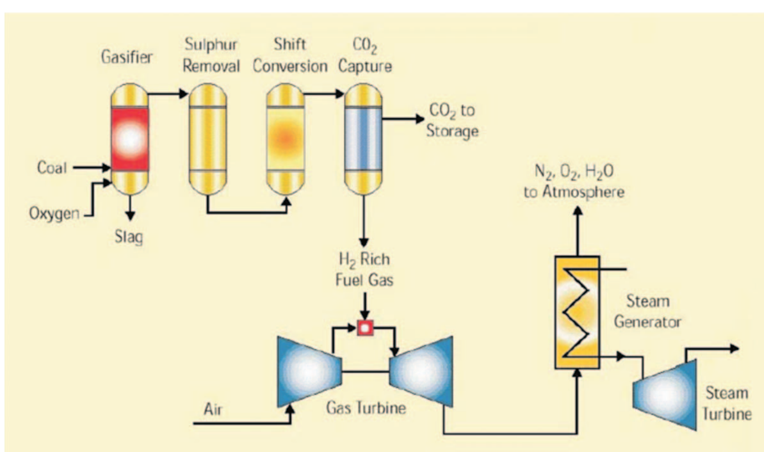

Fig. 8 A simplified scheme for power generation based on gasification of coal and separation of CO_2 from the fuel [8]

exchangers are liable to foul. If the incentive is sufficient, for example, in the boilers to produce steam, these issues and costs are accepted.

The future of large-scale electrical power generation must lie with plants where the CO_2 is collected and then stored, CCS. It is overwhelmingly preferable to direct efforts for CCS towards coal burning plant. The most appropriate scheme is not yet identified, but in some, such as precombustion gasification of coal or some types of oxy-fuel, there would be an expanded role for gas turbines.

A most striking thing about most schemes for power generation in which CO_2 is separated and stored is that the key discipline has shifted from current conventional power engineering to become chemical engineering. There is nothing wrong with this, but the significance should not be overlooked: power engineers will need a broader background in thermodynamics and chemistry to understand the significance of the designs and to be able to contribute usefully.

Several lessons can be taken to the aircraft engine. With current levels of component efficiency and cooling technology there is little gain in efficiency for simple gas turbine cycles with an increase in overall pressure ratio beyond about 40. Also efficiency becomes a weak function of turbine entry temperatures, after mixing of the nozzle cooling air, above about 1600 K if the requirement for increased turbine cooling is included. Higher pressure ratios and turbine entry temperatures than these will raise the power output per unit mass flow through the engine, which does have a real benefit for the aircraft application by reducing weight, but the direct benefit in efficiency is small. Heat exchangers are typically bulky and heavy and are prone to in-service problems. The use of water can markedly improve the operation of the gas turbine.

9 Aircraft Engines

9.1 Background. Numerically, the jet engine is by far the largest application of the gas turbine. The engines vary from small power producing engines in light helicopters to large jet engines propelling commercial airliners. It is the engine for the large airliner, which will be discussed here.

Currently global carbon dioxide emissions from the aviation sector are only around 2% of the total from all sectors. Air transport has, however, been growing rapidly, at about 5% per annum. Aviation is also a conspicuous use of fuel, which attracts attention. Furthermore, many governments are committed to, or are considering the need to commit to, steep reductions in greenhouse gas emissions from all sectors of their economies. As a result the emissions from aviation are likely in future to contribute several times more than 2% of the total CO_2.

Compared with most land-based power generation, the jet engine is simple in layout. On the other hand, the range of operating conditions, which the engine has to accommodate is large–consider the range from take-off in a hot desert, perhaps on occasion up to 50°C, to cruise, which may go down to as low as −70°C.

There is a particular issue with the design of the engines. For medium- or long-range commercial aircraft, the principal requirement is for low fuel consumption at cruise, where well over 90% of the fuel is burned. Logic would put the design point at cruise but the constraints on the engine, in the form of temperature at compressor delivery and turbine entry, occur principally at take-off. With few exceptions, academic treatments of aircraft engine design do not cover the way in which the take-off affects what can be selected for design at cruise and this requires calculations other than cruise.

Results of calculations will be used to show how engines designed for cruise in steady, level flight operate away from design at take-off and during climb. The calculations need to make some assumptions about component efficiencies, cooling, and temperature capability; these assumptions will seek to make the discussion relevant but not to reveal any company level of technology. Although the calculations to be described have been carried out for a three-spool unmixed engine, nothing in the results would be altered were similar calculations to be carried out for a two-spool engine. New commercial engines now invariably have a high bypass ratio, defined as the mass flow through the bypass duct divided by the mass flow of air through the core of the engine. Most of the thrust is produced by the bypass stream.

The different operating conditions (restricted in the present lecture to cruise, take-off, and climb) put different constraints on the design. It is convenient and logical to take the cruise condition as the reference. It will be shown that the ratio of thrust for take-off to thrust for cruise gets bigger as the aircraft aerodynamic design for cruise improves; that is, the aircraft has lower drag at cruise. Likewise, for a given rate of aircraft climb, the increment in thrust required from the engine relative to the thrust at cruise becomes larger as the aerodynamic design of the aircraft gets better. Furthermore, engines designed to give low fuel consumption at cruise have a low jet velocity (giving high bypass ratios) and for these the take-off condition becomes more difficult for the fan. Together these mean that for newer aircraft-engine combinations, the take-off length should be increased and the climb rate at altitude should be reduced relative to the older versions they replace, whereas the actual trend is in the opposite direction. These are addressed below in more detail and form an important part of this discussion.

The pressure ratio of the fan in the bypass stream (fpr) has particular importance for the behavior of the engine and determines the way it can be designed. The obvious effect of lowering fpr is to reduce the jet velocity, increase the bypass ratio, and lower the fuel consumption. There is an indirect effect of lowering fpr, which is to alter the off-design behavior, particularly at take-off. As will be shown, by reducing fpr it is possible to design the core for higher overall pressure ratio (opr) and higher turbine entry temperature (TET). The fan will therefore be treated here in some detail.

It is, of course, a requirement that an aircraft engine produces acceptable levels of noise and emissions. Fortunately, the steps to reduce fuel consumption by lowering jet velocity also reduce noise, or have done until the open-rotor engine is considered, and noise will not be considered further here. Raising the overall pressure ratio tends to increase NO_x because of the dependence on compressor delivery temperature. Combustor design is not going to be discussed here, but it should be noted that the regulations normally ensure that for each new aircraft engine some technology advance is required. For land-based gas turbines the levels of emissions permitted are much lower and premixed lean-burn combustion has been adopted, but the requirement for exceptional combustion stability in aircraft, and for relight at high altitude, has so far kept the design of the aircraft combustor relatively conservative. Above all aircraft engines must be reliable: suffice it to say that the current level of reliability is such that a typical commercial pilot is likely to go through his whole career without a single compulsory in-flight engine shutdown.

10 Efficiency and Specific Fuel Consumption

The specific fuel consumption (sfc), sometimes referred to as thrust specific fuel consumption, can be written as

$$sfc = \mathrm{const}/(\eta_{\mathrm{th}}\eta_{\mathrm{tr}}\eta_p)$$

The thermal efficiency, η_{th}, like that discussed in conjunction with the land-based gas turbines, depends on overall pressure ratio, temperature ratio, cooling technology, and the component efficiencies. The most convenient temperature ratio is the ratio of turbine entry stagnation temperature TET (out of the HP nozzle) to the stagnation temperature into the fan T_{02}. The thrust of the present paper is to find how the performance depends on the overall specification, not on component efficiencies, which are held constant.

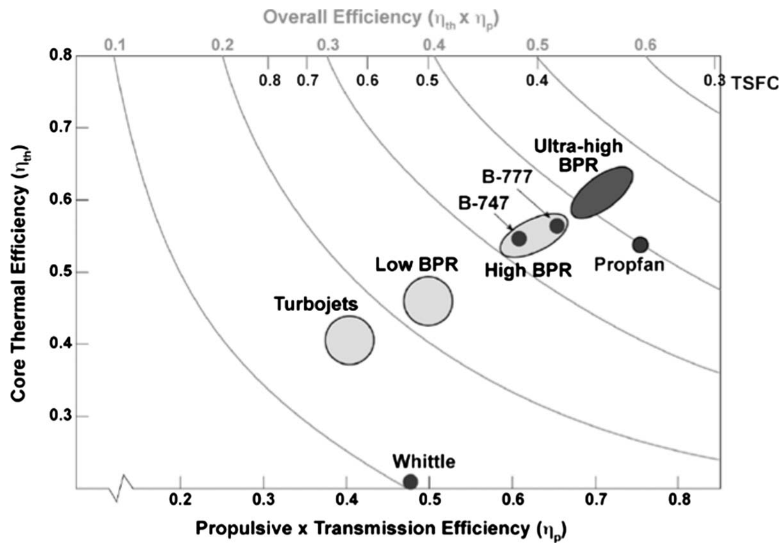

Fig. 9 The link of *sfc* and overall efficiency with thermal efficiency and propulsive efficiency [10]

The transfer efficiency, η_{tr}, being the product of LP turbine efficiency and fan efficiency, is taken for the present purpose as fixed. The efficiency, which lends itself to improvement, is the propulsive or Froude efficiency, η_p. Neglecting the mass fraction of fuel, which is only about 0.15% of the mass flow through a high bypass ratio engine, the propulsive efficiency is given by

$$\eta_p = \frac{2V}{V + V_j}$$

where V is the flight velocity and V_j is the average jet velocity. The only way to increase propulsive efficiency at a given flight speed is to decrease jet velocity, recognizing that the mass flow must then be increased to maintain the same net thrust. Specifying jet velocity is equivalent to specifying the fpr, more precisely the pressure ratio of the bypass section of the fan. The interaction of the various efficiencies and the thrust specific fuel consumption, with the experience of a range of different engines, is shown in Fig. 9 from Ref. [10]. Although prepared by Pratt&Whitney, this reflects general experience and use.

11 Aircraft and Engine Operating Points

To carry out any assessment of engines, it is necessary to relate conditions to the different conditions of flight. For this lecture, it will be assumed that the engine is designed for cruise; that is, the cruise condition sizes the engine. This is the logical approach but, as will be seen, some adaptation may be required for achieving climb thrust as the aircraft approaches cruising altitude, referred to as top of climb (TOC). This will often require some compromise at cruise conditions and may actually fix the size of the engine. Long ago, when turbojets were used for civil aircraft, take-off was the hard condition to achieve and engines were sized for take-off and throttled back at cruise. Now it is more relevant to think of the take-off condition merely setting limits on the overall pressure ratio and turbine entry temperature at cruise.

11.1 Cruise. To understand the design and performance of the engine, it is necessary to know something of the aircraft. In steady level flight, the lift of the wings will be equal to the weight of the aircraft and the drag of the aircraft will be equal to the net thrust of the engines. The lift-drag ratio of the aircraft, L/D, is a function of the flight Mach number M and the lift coefficient. To minimize drag for a given weight at the desired cruise Mach number, that is to maintain maximum L/D, it is necessary to adjust the lift coefficient. This is done by altering the cruise altitude to vary ambient air density. For a long range aircraft, the weight of fuel at take-off can be as much as 45% of the maximum take-off weight, comparable to the empty weight of the aircraft. As the fuel is burned during a long flight, the change in weight is large and to maintain the optimum lift coefficient, which gives the highest L/D, the aircraft must climb to lower density air. Nowadays air traffic control normally sets steps of 2000 ft when flying in one direction so the climb is not continuous but incremental.As a good approximation the ambient pressure in the standard atmosphere falls by 10% for each 2000 ft increase in altitude.

For a given Mach number and lift coefficient, the lift (equal to weight) is proportional to the ambient static pressure. Likewise the net thrust (equal to aircraft drag) from the engine at constant Mach number is also proportional to ambient static pressure if the engine is maintained at the same operating condition (meaning the same pressure *ratios* and temperature *ratios*). Therefore, if the engines and the wings are properly matched, each will be at their most efficient when the altitude is changed during cruise. For a new commercial airliner, it is reasonable to take an optimum lift/drag ratio of 21 at a cruise Mach number of 0.85. This, of course, requires that the thrust from the engines is equal to 1/21 times the weight. (For the previous generation, aircraft were nearer $L/D \approx 18$.) Cruise is assumed here to start at 35,000 ft (a typical value in design studies). The weight at start of cruise will be only 1% or 2% less than the take-off weight, and this change in weight is below the accuracy of the treatment presented here. As an approximation, therefore, the thrust at start of cruise will be taken here to be 1/21 times maximum take-off weight, that is,

$$Fn_{cr} = MTOW/21$$

11.2 Take-Off. In operation the thrust used for take-off is selected depending on the actual weight (how much fuel and payload is being carried), but the engine must be capable of giving the thrust specified for the maximum take-off weight ($MTOW$). For a range of recent[2] aircraft, the take-off thrust Fn_{TO} is equal to about 0.275$MTOW$ with a standard deviation of 0.02. More recently maximum take-off thrust of 0.3$MTOW$ has been demanded for some new aircraft, and this more onerous value will be adopted here. Together the cruise and take-off stipulations imply for the net thrust,

$$Fn_{TO} = 0.3MTOW \quad \text{and} \quad Fn_{cr} = MTOW/21$$

so that

[2]A380-800, A340-500, A340-600, B777-200ER, B787-base, B787-stretch, B777–300, and B747-400.

$$Fn_{TO} = 0.3 \times 21 \times Fn_{cr} = 6.3Fn_{cr}$$

This ratio of 6.3 between initial take-off thrust and cruise thrust will be used throughout this paper. Some calculated results will also be shown for the lower level of take-off thrust, $Fn_{TO}=0.275MTOW$, giving $Fn_{TO}=5.77Fn_{cr}$.

It is immediately apparent that, as the aircraft lift-drag ratio improves, the ratio take-off thrust to cruise thrust increases. A previous generation of aircraft would have had L/D closer to 18, so as a result of improvement in aircraft aerodynamic performance there has been about a 16% increase in the ratio Fn_{TO}/Fn_{cr} in addition to the 9% increase going from $Fn_{TO}=0.275MTOW$ to $Fn_{TO}=0.3MTOW$.

It is common now to offer engines with take-off thrust rated to higher temperatures than the International Standard Atmosphere (ISA) at sea level, so in line with this take-off thrust is specified for this lecture to be at ISA+15°, which is 303 K or 86°F.

11.3 Top of Climb. An additional requirement on thrust is at top of climb, just as the aircraft approaches the cruising altitude at a given flight speed. The angle of climb θ equal to the velocity of climb divided by the forward speed of the aircraft is less than about 0.6 deg near cruising altitude, so the top-of-climb thrust Fn_{TO}/Fn_{cr} is well approximated by

$$Fn_{TOC} = Fn_{cr} + W\theta$$

where W is the weight. Climb thrust can be rewritten as

$$Fn_{TOC} = Fn_{cr}(1 + \theta L/D)$$

As for take-off, the ratio of climb thrust to take-off thrust Fn_{TO}/Fn_{cr} increases as the lift-drag ratio increases. In other words, as the quality of the aircraft aerodynamics gets better, the *range* of thrust the engine has to produce increases.

As the aircraft approaches its cruising altitude, the climb rate is required to be at least 300 ft/min. In fact, some new aircraft specify 500 ft/min and require that this be produced at cruising altitude. For 2000 ft steps, this means that ideally at start of cruise the aircraft would be heavy by an amount appropriate to an altitude difference of 1000 ft, an amount equivalent to about 5%. For $L/D=21$, the ratio of climb-to-cruise thrust is

$$\text{for 300 ft/min,} \quad Fn_{TOC} = 1.17Fn_{cr}$$

$$\text{for 500 ft/min,} \quad Fn_{TOC} = 1.26Fn_{cr}$$

Some of the latest aircraft, in addition, require this top-of-climb thrust when the temperature is higher than that for the standard atmosphere, at ISA+10 K, and this will be used here.

12 The Datum Engine

Little can be learned about engines by discussing generalities: one needs to look at concrete examples. A datum engine is considered here with representative values specified, which are intended to be reasonably compatible with new large engines currently offered for service.

The datum engine is specified for cruise at 35,000 ft and $M=0.85$. The *opr* is selected to be 40 and the *TET* is taken to be 1475 K. In line with aircraft-engine practice, this is the temperature downstream of the HP nozzles where the cooling air is assumed to have fully mixed out. For the sake of simplicity, round-number efficiencies have been assumed:

core air compression polytropic efficiency	=0.9
core turbines isentropic efficiency	=0.88
fan outer stream polytropic efficiency	=0.94
LP turbine isentropic efficiency	=0.925

Suitable losses, cooling, and bleed air have been used. The trends and arguments of the paper, and the conclusions drawn, are not sensitive to these values.

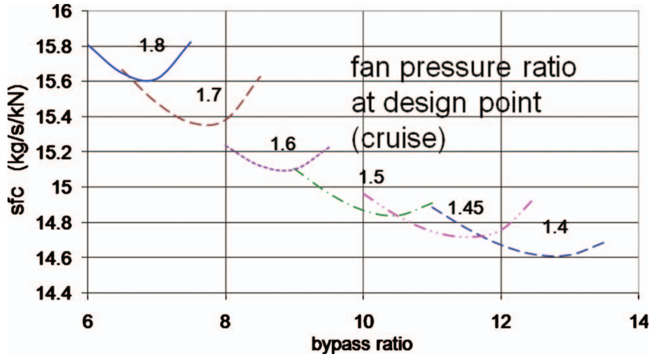

Fig. 10 Specific fuel consumption versus bypass ratio at cruise for different fan pressure ratios. Core conditions held constant, *opr*=40, *TET*=1475 K.

A decision was made to adopt for this study values which are representative of the present rather than aspirations or expectations of the future. The same is true of the range of fan pressure ratios to be adopted. The rationale for this is that the optimization of an engine, and of an aircraft-engine combination, is complex and it is not clear how the various parameters will change.

13 Calculations

The calculations have been carried out with the commercial software package GASTURB. This has been found to give results compatible with those used in large companies and some description is given by Kurzke [11,12]. For off-design calculations, the program needs to have values of efficiency when operating away from the design point and for the present purpose, the performance maps in the program suite were used to calculate off-design behavior.

13.1 Cruise Design Point: Choice of Fan Pressure and Bypass Ratios. Figure 10 shows for the datum engine core the specific fuel comsumption, *sfc*, at cruise versus bypass ratio, *bpr*, for a range of fan pressure ratio, *fpr*, from 1.4 up to 1.8. For each value of *fpr*, there is a *bpr* that gives the lowest *sfc*. The minimum *sfc* falls as *fpr* is reduced, which is because the propulsive efficiency is increasing: η_p increases because jet velocity, V_j, is reduced and bypass stream jet velocity depends only on *fpr*. (For a high bypass engine, the thrust is overwhelmingly produced by the bypass stream.) With a constant datum core and a given *fpr*, varying *bpr* is equivalent to varying the velocity from of the core jet. The value of *bpr* which gives the lowest *sfc* depends on the overall pressure ratio of the engine and the turbine entry temperature as well as *fpr*; raising *opr* or *TET* would raise optimum *bpr* for the same *fpr*. Because η_p is a function of *fpr* and flight speed, it is more appropriate to treat *fpr* as the independent variable and *bpr* as the dependent variable, the opposite of most academic treatments of the turbofan engine. The benefit of using *fpr* as the independent variable is even more apparent for off-design consideration.

For a modern engine, V_j is chosen to be as low as possible to give high propulsive efficiency η_p and this requires the mass flow to be large to give the required thrust; in other words, the fan diameter must increase as V_j is reduced. The principal constraints on reducing fan pressure ratio are the fan diameter (will it fit between the wing and the ground?), the weight of the fan and its containment, the higher drag associated with a larger nacelle, and the losses associated with the greater mass flow through the bypass duct. Since the diameter is constrained by such things as ground clearance or weight, optimization for the aircraft, including weight and nacelle drag, so will not be attempted here.

From Fig. 10, for the datum engine core and the aircraft flying at M=0.85, the optimum combination of fan pressure ratio and bypass ratio for an engine typical of the latest designs is

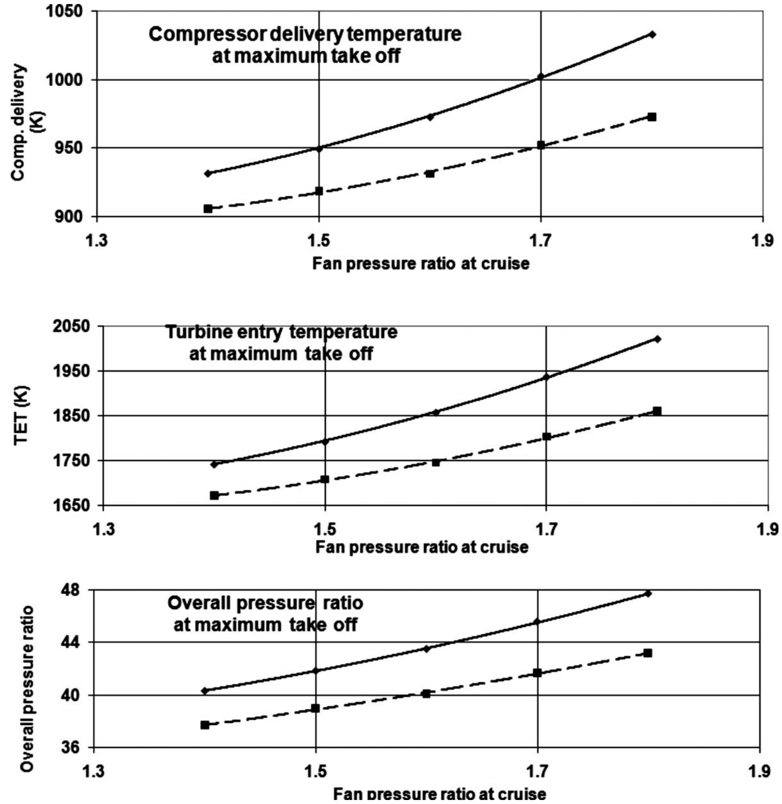

Fig. 11 Core conditions for maximum take-off at ISA+15 K for data core with different fan pressure ratios at cruise. The solid line is take-off thrust of 0.3*MTOW*, and the broken line is 0.275*MTOW*.

$$fpr = 1.5 \quad \text{and} \quad bpr = 10.4$$

while for an engine typical of the previous generation,

$$fpr = 1.8 \quad \text{and} \quad bpr = 6.8.$$

These bypass ratios are rather higher than would be selected for a real engine design, probably because the pressure loss in the bypass duct has not been included.

With the core, fan pressure ratio and bypass ratio specified at the cruise condition, the net thrust for a given mass flow through the engine (i.e., specific thrust) is fixed. The required cruise thrust then determines the mass flow, so the diameter of the engine fan can be specified using the mass flow per unit area capability known inside the company designing the engine.

13.2 Maximum Take-Off (MTO) Condition: Off-Design Calculations. From the discussion in Sec. 11.2, it will be recognized that with the specification $Fn_{\mathrm{TO}}=0.3MTOW$ and $L/D=21$, the maximum take-off thrust at sea-level static conditions should be 6.3 times the cruise were thrust at 35,000 ft and M=0.85. The on-design calculations at cruise used to find the *bpr* to give lowest *sfc* shown in each *fpr* from 1.4 to 1.8, as shown in Fig. 10. The off-design conditions were calculated for sea-level static conditions; in these the turbine inlet temperature was increased until the thrust was equal to 6.3 times the corresponding cruise thrust (corresponding to $Fn_{\mathrm{TO}}=0.3MTOW$). The calculations were repeated for the $Fn_{\mathrm{TO}}=5.77Fn_{\mathrm{cr}}$ (corresponding to the lower take-off thrust, $Fn_{\mathrm{TO}}=0.275MTOW$).

The overall pressure ratio, turbine entry temperature, and compressor delivery temperature are shown for maximum take-off in Fig. 11 as functions of design fan pressure ratio at cruise: solid lines are for $Fn_{\mathrm{TO}}=0.3MTOW$ and broken lines are for $Fn_{\mathrm{TO}}=0.275MTOW$.

Whereas all the hypothetical engines had the same core conditions at cruise ($opr=40$, $TET=1475$ K) at the off-design condition for take-off, the core conditions are radically altered with the alteration depending on the fan pressure ratio. At the higher take-off thrust, the core pressure ratio is almost equal to the cruise value for $fpr=1.4$ and significantly below the cruise value for the lower take-off thrust. In fact, for the lower take-off thrust, the *opr* does not exceed that for cruise until design *fpr* exceeds about 1.6; in other words, at take-off the core is operating at a nondimensional loading below design up until $fpr=1.6$. The compressor delivery temperature and the temperature into the turbine, *TET*, at take-off are higher than the cruise values for all take-off conditions because the inlet air is so much warmer.

For a design fan pressure ratio of 1.8 for both levels of thrust at take-off, the overall pressure ratio is higher than design, indicating that the engine core is operating at a higher operating point. For the higher take-off thrust, the compressor delivery temperature 1033 K is well above allowable metallurgical limits, as is the turbine entry, $TET=2021$ K. For the lower take-off thrust, *TET* is about acceptable nowadays, but compressor delivery temperature is high. Expressed another way, the data core would not be acceptable with $fpr=1.8$.

For design $fpr=1.5$, the core parameters at the higher level of take-off thrust are at about the current limits. Thus at this fan pressure ratio, the data engine core specifications ($opr=40$, $TET=1475$ K) would be acceptable even at the higher level of take-off thrust. Put another way, the core is too small for take-off when the cruise $fpr=1.8$ but the core is large enough when $fpr=1.5$. It is important to appreciate that designing the core for cruise with a pressure ratio and turbine inlet temperature as high as $opr=40$ and $TET=1475$ K is only possible when the fan pressure ratio is held down to a low value, such as 1.5.

This may be exploited in a different way. Were the take-off thrust to be restricted to $Fn_{\mathrm{TO}}=0.275MTOW$, it would be possible to increase the *opr* and *TET* at cruise further with $fpr=1.5$. For the reasons given in the discussion of land-based engines, this would give little direct benefit in terms of thermal efficiency, but because the power output per unit mass flow rate through the core would be increased, it would allow the core to be smaller and therefore lighter.

Reducing fan pressure ratio at cruise, therefore, does more than lower the jet velocity and raise propulsive efficiency. The lower cruise *fpr* makes possible higher overall pressure ratio and turbine inlet temperature at cruise, thereby increasing thermal efficiency and making the core relatively smaller and lighter. The explanation lies with the so-called lapse rate. For a low *fpr*, the jet velocity is low ($Vj \approx 357$ m/s at cruise for $fpr=1.5$) compared with flight speed $V=252$ m/s and the specific net thrust, $(Vj-V) \approx 105$ m/s, is small. If the fan pressure ratio were held constant, the specific thrust with the engine stationary at sea level would be about 270 m/s giving a ratio of specific thrust of about 2.6. For $fpr=1.8$, however, the cruise specific thrust is $(Vj-V) \approx 146$ m/s, while sea-level static specific thrust at this pressure ratio is about 330 m/s and the specific thrust ratio is only 2.25. The higher thrust ratio for the lower cruise fan pressure ratio means that the engine with the same cruise thrust gives proportionately more thrust at take-off.

The decision to use ISA+15 K (i.e., 303 K, 86°F) to specify take-off conditions has a large effect on temperatures in the engine though pressure ratios are essentially unchanged. Take as an illustration the data engine core with design $fpr=1.5$ for the higher take-off thrust, $Fn_{\mathrm{TO}}=0.3MTOW$. If ambient temperature were restricted to the ISA value (288 K), the compressor delivery temperature is reduced by 41 K and *TET* is reduced by 80 K, a bit over five times the rise in ambient inlet temperature.

13.3 TOC Condition: Off-Design Calculations. Top of climb is taken here as synonymous with maximum climb (MCL). As explained in Sec. 11.2, the climb rate at the top of climb is a choice and the present rate is higher than that historically regarded as necessary. At 500 ft/min maximum climb thrust $Fn_{\mathrm{TOC}} = 1.264Fn_{\mathrm{cr}}$. Some results for the datum core at the TOC conditions are shown below for both 500 ft/min and 300 ft/min climb. The column shown as *mr* is the ratio of corrected mass flow into the fan at maximum climb condition to corrected mass flow at cruise $mr=(m_{\mathrm{cor}})_{\mathrm{TOC}}/(m_{\mathrm{cor}})_{\mathrm{cr}}$. Corrected mass flow is defined by

$$m_{\mathrm{cor}} = \dot{m}\sqrt{\theta}/\delta$$

where θ is the ratio of stagnation temperature entering to the stagnation temperature at a reference condition and δ is the ratio of stagnation pressure entering to stagnation pressure at a reference condition. The reference condition is usually ISA at sea-level static conditions.

(It is worth noting that the values of turbine entry temperature are very sensitive to component efficiencies used so for these off-design calculations, variations in temperature of less than, say, 20 K are not meaningful here.)

Design	TOC, 500 ft/min ISA+10			
fpr	*fpr*	*opr*	*TET*	*mr*
1.5	1.59	47.2	1660	1.057
1.8	1.93	47.5	1680	1.065

Design	TOC, 300 ft/min ISA			
fpr	*fpr*	*opr*	*TET*	*mr*
1.5	1.545	43.5	1540	1.028
1.8	1.873	43.7	1510	1.039

Whereas at take-off the *opr* and *TET* varied markedly with fan pressure ratios, for top of climb the conditions are remarkably similar for $fpr=1.5$ and 1.8. At this top of climb, the limit is essentially on producing enough power from the core and, apart from the difference in propulsive efficiency, the fan pressure ratio does not affect this.

Although the fan occupies most discussion below, it is worth mentioning that achieving top-of-climb thrust requires the core compressor to swallow about 110% of core mass flow at design (cruise). Achieving this large increment will compromise efficiency at cruise to some extent. The turbine entry temperature for the high rate of climb is also an issue. Although *TET* is about 100 K below the take-off value, take-off lasts 1 or 2 min whereas maximum climb condition can be maintained for more like 20 min. As a result, a significant part of the life of HP turbine blades could be used up in the climb.

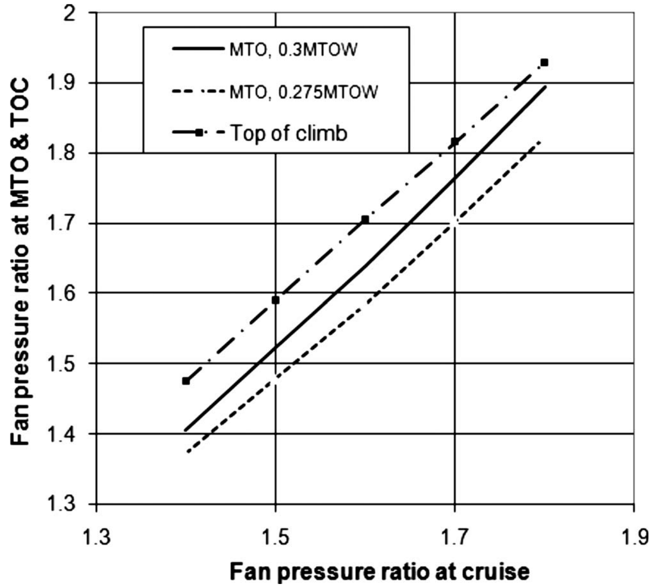

Fig. 12 Fan pressure ratio at MTO and TOC (data core)

13.4 Fan Operating Point at Take-Off and Top of Climb. As noted above, reducing the fan pressure ratio for cruise not only improves the propulsive efficiency and directly reduces the *sfc*, but it also makes it possible to operate at higher *opr* and *TET* at cruise and raise thermal efficiency. This is because these values do not increase so much for take-off with a low fan pressure ratio. There are, however, special problems that arise for the fan as the pressure ratio is reduced and these are addressed in this section.

Figure 12 shows for a range of fan pressure ratios at cruise the variation in fan pressure ratio for maximum take-off, when the take-off thrusts are $0.3MTOW$ and $0.275MTOW$. Also shown are the fan pressure ratios for top of climb at a rate of 500 ft/min. All the core conditions at cruise are the same. For all design values of *fpr*, the fan pressure ratio during the maximum climb is significantly higher than design. The take-off value of *fpr* moves to be closer to the design value as the fan pressure ratio for cruise is reduced and by $fpr=1.4$ the pressure ratio is almost exactly equal to the design value at the higher take-off thrust and below the cruise value for the lower take-off thrust. As will be shown below, the lower fan pressure ratio does not make the design of the fan for low design pressure ratios easy. To see why there are difficulties for the fan when design *fpr* is low, one needs to consider the corrected mass flow into the fan for both maximum take-off and top of climb. In Fig. 13, the ordinate is the ratio of corrected mass flow to the design value at cruise *mr*.

For top of climb, the increase in corrected mass flow relative to design point at cruise is similar at all design fan pressure ratios. For take-off, however, the differences in *mr* depends strongly on fan pressure ratio. For design *fpr* above about 1.6, the corrected mass flow *increases* for take-off, whereas below about $fpr=1.6$,

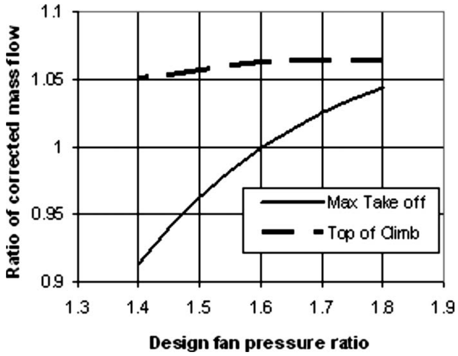

Fig. 13 Corrected mass flow for max. take-off and top of climb divided by corrected mass flow at cruise (data core), Fn_{TO} =0.3*MTOW*

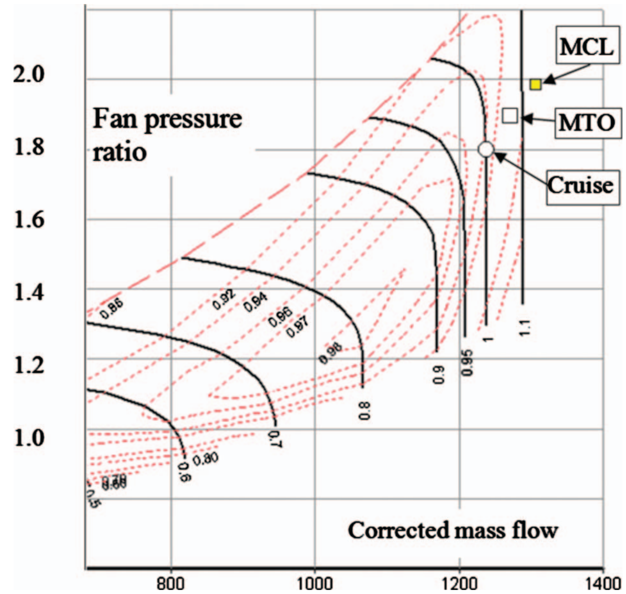

Fig. 14 Fan operating characteristic with design *fpr*=1.8 at cruise, data core. MTO is for Fn_{TO}=0.3*MTOW*, and MCL is TOC at 500 ft/min.

the corrected mass flow falls. To understand the fall in the corrected mass flow *into* the fan for take-off when *fpr* is below about 1.6, one needs to consider the corrected mass flow *out* of the fan.

Although the corrected mass depends on temperature as well as pressure, as a good approximation the behavior can be understood in terms of just pressure. The bypass nozzle expands the air from the stagnation pressure in the bypass duct to the static pressure in the surrounding atmosphere. For cruise at M=0.85, the ratio of stagnation pressure into the fan to the static pressure in the atmosphere is 1.60. The combined pressure ratios across the nozzle for the design cruise condition are therefore 2.4 for fpr=1.5 and 2.88 for fpr=1.8. In both cases, the nozzle is choked and, since the corrected mass flow is fixed by the nozzle area, the mass flow is proportional to the stagnation pressure. For the engine stationary at sea level, the pressure ratios across the nozzle at maximum take-off (Fn_{TO}=0.3*MTOW*) are 1.524 when the design is for fpr=1.5, see Fig. 12, and 1.895 when the design is for fpr=1.8. In the case of the higher *fpr*, the nozzle will be choked at take-off (since $fpr>1.892$, the value of pressure ratio at which a convergent nozzle chokes) and the corrected mass flow leaving the fan will be unaltered relative to cruise. For the fpr=1.5 fan the nozzle is unchoked and the corrected mass flow at nozzle exit will be reduced relative to the design point at cruise. The lower the design fan pressure ratio the more the corrected mass flow is reduced at take-off.

The significance of the change in corrected mass flow at take-off can be seen from the operating maps for the fan. Figure 14 shows an operating map produced by GASTURB for fpr=1.8 and Fig. 15 shows the corresponding map for a fan designed for fpr =1.5. These maps, created by GASTURB from its stored data, are similar to those that would be measured by testing modern fans designed for these pressure ratios. In these figures, the abscissa is the corrected mass flow into the fan, while the ordinate is the *fpr*. The black lines represent lines of constant nondimensional speed. The black lines end at the broken red line, which indicates where the fan would stall or surge. Shown on the plots are the operating points for cruise (design), maximum take-off and top of climb (denoted here as MCL). The contours of efficiency, shown in red in Figs. 14 and 15 , indicate the type of variation expected, but the absolute values are not meaningful.

For fpr=1.8 fan, the operating points for take-off and climb are both to the right of the design point. That is, they have higher pressure ratio, higher corrected mass flow, and the fan would operate at higher nondimensional rotational speed. For fpr=1.5, however, the climb condition is to the right of design, whereas the maximum take-off condition is to the left. The movement of the operating point to the left, where the pressure rise is relatively high for the mass flow and nondimensional rotational speed, puts it near to the surge line, where the fan is also more prone to aero-elastic instability. Adapting the fan to cope with this proximity to the surge line makes it harder to position the fan near its peak efficiency at cruise, which is exacerbated by the need to allow the fan to work satisfactorily at the higher speed case for maximum climb. The off-design complications for the fpr=1.5 fan are therefore more acute than those for the earlier fpr=1.8 fans.

Although the operating point of the fan for a lower take-off thrust, Fn_{TO}=0.275*MTOW*, is not shown, this reduction does not greatly ease the operating condition of the fan designed for lower pressure ratio. The fan would operate at lower corrected mass flow and lower fan pressure ratio, as may be found from Figs. 12 and 13 , but the proximity to the surge line is hardly altered.

Reducing the maximum climb rate would ease conditions for the fan, since swallowing capacity at the increased rotor speed at

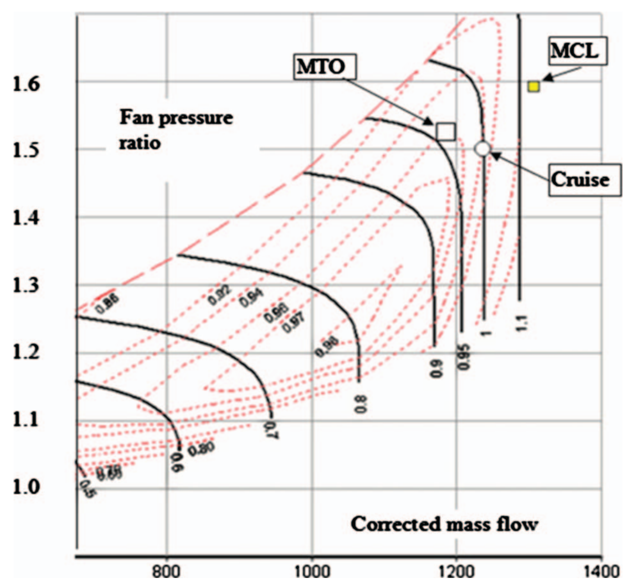

Fig. 15 Fan operating characteristic with *fpr*=1.5 at cruise, data core. MTO is for Fn_{TO}=0.3*MTOW*, and MCL is TOC at 500 ft/min.

top of climb, denoted MCL here, is the dominant issue. The extent to which efficiency of the fan is compromised at cruise by top-of-climb thrust requirements depends on the technology of the company designing the engine and cannot be quantified here.

What can be said is that a successful design must allow the required mass flow for top of climb to pass through the fan and that this, rather than cruise optimization, often comes to determine the choices.

As Fig. 13 shows, the excursion in corrected mass flow for the fan from take-off to maximum climb gets worse as the design fan pressure ratio decreases. Fan pressure ratios at cruise significantly lower than, say, 1.45 will probably have to wait until variable nozzles for the bypass stream become available or acceptable, a point that has been recognized in the industry for some time. In other words, the variable area nozzle is an enabling technology for lower fan pressure ratio.

What does not appear to have been recognized is that the requirements for the fan at take-off and climb have recently been specified to be more onerous (higher take-off thrust as a ratio of maximum take-off weight and higher rates of climb), while the higher lift-drag ratio of the aircraft have made the proportional increases in thrust relative to cruise larger for both take-off and climb.

14 Routes for Improvement

14.1 Within the Current Layout and Style. The first observation relates back to Sec. 11. To improve the engine, one needs to improve the complete system, which includes the whole aircraft and the air traffic management. Maximum take-off thrust and maximum climb rates should be chosen so that they do not unduly compromise the engine for its cruise condition. The high take-off and climb thrusts are giving the airlines, or some airlines, greater operational flexibility: maximum take-off weight on hot days, operation on short runways, and the ability to climb to the cruising altitude and take the slots allocated by air traffic control in some parts of the world. Although these may on some occasions offer an advantage, of necessity they compromise fuel burn in the majority of operations. It is possible that many customers do not need these facilities and are unaware of the cost in terms of cruise fuel consumption. It is probable too that some of the extra engine thrust for take-off and climb specified for all new aircraft is an insurance against future growth in aircraft weight.

Nevertheless the trend is in exactly the wrong direction: the higher cruise lift/drag ratio of the aircraft makes the increment in thrust for take-off and climb greater. At the same time low design fan pressure ratio moves the fan operating point for take-off near to the surge line, to the opposite side of design point from that for climb. It is natural to expect with a new product that the performance will exceed the older one in all respects. Certainly it is expected that a new aircraft will be more economic, quieter, and with lower levels of emissions. But would airlines be prepared to accept some reduction in take-off thrust to weight (at least until some forward speed had been built up) or a reduction in rate of climb? The wish for high rate of climb is related to air traffic management, but it must surely be better to adapt that rather than compromise the design of the aircraft and engine.

In general, the management of air traffic also offers a means of reducing fuel burn. The European Advisory Council for Aeronautical Research in Europe (ACARE) [13] has goals which call for a 50% reduction in fuel burn per passenger mile by 2020 relative to 2000. It is intended that 20–25% should come from the airframe, 15–20% from the engine *sfc*, and 5–10% from better operations and air traffic management Generally the impact of air traffic management is more significant for shorter flights but allowing aircraft on long flights to approximate to their optimum altitude is important and the reduction in the climb steps from 4000 ft to 2000 ft is a welcome sign of this happening. (In practice, it is still not uncommon for aircraft to cruise for long distances at altitudes below their optimum for reasons associated with air traffic control.)

A good example of the scope for improvement brought about by redefining operations was given quite recently by Green [14]. A result in this, which surprised many, was that for long journeys savings in fuel burned per passenger mile on the order of 20% may be achieved by designing aircraft for "short" range. Short range here means distances up to about 4000 nautical miles. Longer journeys would be achieved in a number of stages. The explanation, of course, is that much of the aircraft weight at the start of a long flight is fuel and therefore much of the drag is produced by the fuel. To get the full benefit of this, one needs to design the aircraft from the start with this range in mind. Simple as the logic is, the trend is in the opposite direction, with new large aircraft generally being designed for much longer range[3] than 4000 nm.

How fast do we need to travel? Few would notice a reduction in flight Mach number by, say, 10%, since for most journeys the overall journey time is much longer than that spent in the air. But such a reduction in Mach number could allow increased wing L/D and lower wing sweep with thicker wings could lead to reduced structural weight and further savings in fuel burn.

14.2 Open Rotor. Within the confines of the engine, what can be done to reduce fuel burn? As the fan pressure ratio is reduced the drag of the nacelle becomes an ever bigger proportion of the generated thrust, while the weight of fan rotor and containment system increases. One solution, of course, is to take the bypass engine to its limit and have an open rotor. To be most efficient, it should be a contrarotating rotor to remove the kinetic energy associated with the swirling flow from one rotor.

There are two technical issues and one nontechnical issue holding the open-rotor concept back. Of the technical, the most pressing is the need to have a different configuration of aircraft, so the gamble on the future of the open rotor needs to start with the laying down of a wholly new aircraft capable of carrying the large diameter rotor blades, which will be required. For example, a twin-engine aircraft of the size of a Boeing 737 or Airbus A320 with open-rotor engines would require rotors 4.3 m (14 ft) in diameter. The second technical issue is the noise and at the moment there does not seem to be a clear understanding of how low the noise might be made with development. Nevertheless the present feeling is that is essential is that the aircraft meet the ICAO Annex 16 Chapter 4 levels, whereas new large aircraft with ducted fans are cumulatively 15–20 dB below this. The nontechnical issue is whether the public will accept aircraft, which could be described as propeller aircrafts and seen as a retrograde step, although studies carried out in 2007/2008 show that the traveling public are very positive about the open rotor's low level of CO_2 production.

14.3 Engine Core Improvements. What can be done to improve the core of the engine? As already noted, the scope for improving the thermal efficiency by raising overall pressure ratio or turbine entry temperature is small; the values of these parameters are at or close to the level at which no gain in efficiency is produced. Higher *TET* does produce more power per unit mass flow through the core, making the core smaller and lighter (thereby increasing the bypass ratio) but it does little for efficiency, especially if cooling has to be increased to cope with the increased temperature. The efficiency of the fan, compressors, and turbine will increase, but after decades of work on these, the additional scope is probably small. Likewise the scope for better cooling or better materials is, after many years work, modest.

[3]From the Boeing website, it may be seen that the B787-8, B787-9, and B787-3 are designed for ranges of 7500–8200 nm, 8000–8500 nm, and 2500–3050 nm, respectively. From the Airbus website, the ranges for the A350-800, A350-900, and A350-1000 are 8300 nm, 8100 nm, and 8000 nm, respectively

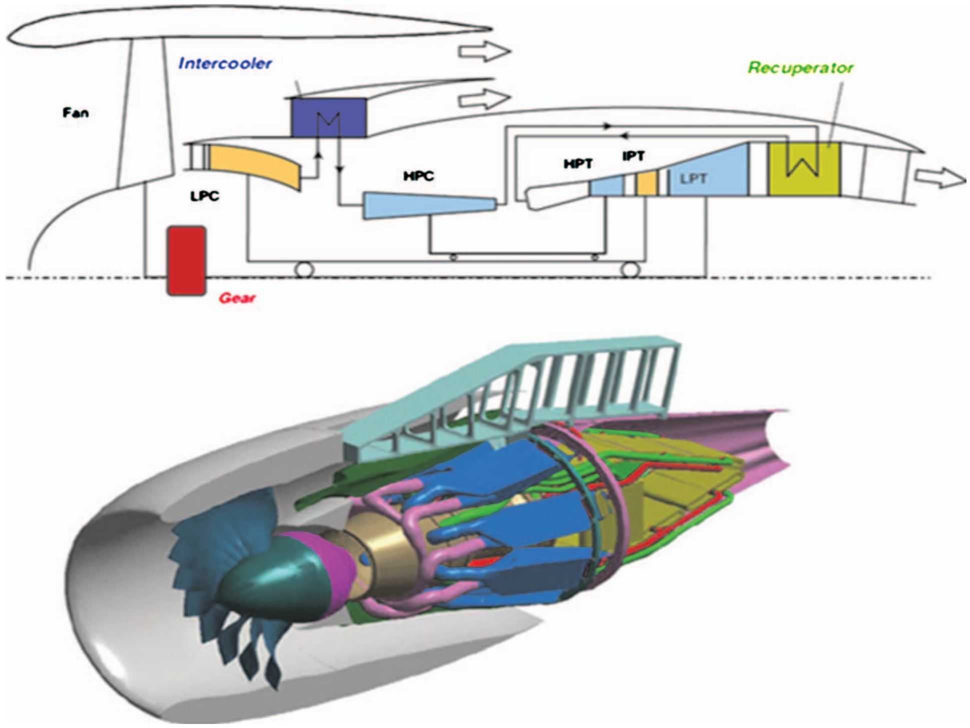

Fig. 16 The proposed CLEAN engine, part of EU EEFAE program [15]

Heat exchangers. An idea that has floated around for many years is to incorporate a recuperator, taking heat from the exhaust and giving it to the air entering the combustor and an intercooler, reducing the temperature into the HP compressor. A scheme for this has been advanced recently in the "European Efficient Eco-friendly Aircraft Engine" program under the name component validation for ecofriendly aero-engine (CLEAN). The idea is like the marine engine in Figs. 5 and 6, but enclosed in a flight nacelle. A schematic and a perspective drawing are shown in Fig. 16.

Much work has gone into the design of the heat exchangers and there is reason to believe that they would provide the necessary heat exchanges if the flow is sufficiently uniform. Fitting heat exchangers into the sleek outlines of a nacelle requires, as Fig. 16 shows, a network of pipes and manifolds. Achieving flow uniformity without excessive pressure loss, given the pipe work, can be envisaged to be difficult and tests have already shown this to be a problem. Achieving low pressure loss would drive the layout toward a large, bulky, and heavy engine.

Cooling air and water injection. One of the things taken for granted so far in this paper is the cooling of the turbine. The cooling requirements are predominantly set by the take-off condition, and somewhat by the requirements for climb, but at cruise the HP turbine needs significantly less cooling and the later turbine rows probably none at all.

There is scope for modulating the cooling air. To quantify the potential benefits, one can carry out GASTURB calculations on the data engine. For the data engine at the design cruise condition ($opr=40$, $TET=1475$, and $fpr=1.5$), halving the cooling air to the HP turbine, here a single stage, leads to a 2% reduction in specific fuel consumption at cruise. Modulation has major implications for the architecture of the engine. Furthermore, reducing the pressure of cooling air entering the blades, in order to reduce cooling flow, can lead to hot gas entering the blade in regions near stagnation points on the blades.

If the temperature of the cooling air could be reduced, the cooling system could be designed to have smaller mass flow. Complications of architecture (how to get the cooling air out of the core and then back in) and issues of heat exchanger reliability have made it unattractive to cool the cooling air, with few exceptions to date. In fact, the cooling of the cooling air is only needed for take-off and occasionally, perhaps, during climb. An elegant solution would be to cool the cooling air by evaporating water in it during the take-off phase. As a guide, 1% by mass of water evaporated reduces the temperature of the air by about 30 K. This has been used with the datum engine described earlier in which the metal temperature for the HP turbine during take-off was held equal with and without the added water. It has been assumed that the cooling efficiency for the blades is unaltered. With the assumptions adopted for cooling, 7% by weight of water to the cooling air (210 K temperature drop) leads to a halving of the cooling air required to the HP turbine, with a consequent reduction of sfc at cruise of about 2%.

Water injection was used a long time ago at take-off to increase the thrust for some commercial engines and it is regularly used to increase thrust from the Pegasus lift engine by injecting water ahead of the combustor. Water injection in the cooling air was proposed as a way of obtaining contingency power (i.e., above normal maximum power) from helicopter engines by Biesiadny and Klann [16]. More recently water injection into the main flow at combustor entry has been proposed as a way to reduce NO_x emissions at take-off [17–19]. As noted earlier, water injection is widely used in some land-based engines, mainly for the reduction of NO_x but also to increase power. In all proposals and applications except that of Biesiadny, the water is injected in the main gas stream, but the proposal here is the injection into only the cooling air, which is a small fraction of the core air stream, so the quantities of water involved are much less. Airlines will not welcome the extra complexity for operations, but complexity may be part of the price for lower fuel burn. Daggett et al. [19] showed that for an all-new engine and airframe combination, it was financially plausible to have water injection, but not as a retrofit for an existing aircraft. In Ref. [19], the financial benefit came from lower NO_x and lower turbine deterioration (because the gas was cooler) and not from reduced fuel burn.

For a modern engine with a take-off thrust of about 70,000 lb, the core mass flow of air is about 100 kg/s during take-off. For 2 minute of water cooling at the highest rate, each engine would require some 34 kg of water. In a 6 hour flight, each engine would burn around 16×10^3 kg of fuel, so the 2% reduction is worth about 300 kg fuel saved, representing an excellent rate of return. Cooling to other blade rows could likewise be reduced with consequent reductions in *sfc* if water were injected during take-off.

14.4 More Radical Ideas. Two more radical ideas will be discussed briefly. One is the idea of fans or open rotors driven by a remote core, i.e., the core is no longer concentric with the fan(s). The other is the pressure-gain combustion, which encompasses such ideas as pulse-detonation combustion and wave rotors.

It is well known that if an aircraft could be propelled by engines able to ingest the wake off the wings and fuselage, the propulsive efficiency could be raised to a high level, in some cases above unity. To get high thermal efficiency, the core needs to be relatively large (to enable the pressure ratio to be high without having excessively small blades) and this in a conventional configuration is inconsistent with ingesting the wake and boundary layer. One proposal for achieving this is to have the core separate from the fans and to drive the fans via shafts or electric connection, see Ref. [20]. From a thermal or aerodynamic point of view, this is fine but from a practical point of view it sounds implausible. With shafts to move high levels of power around the aircraft weight could increase rapidly. Electrical coupling would be attractive if copper and iron were not so heavy: superconductivity may be possible, but is many years away. Weight increases could rapidly remove any advantage in propulsive efficiency It should never be forgotten that safety is paramount and it would be hard with shafts taking large amounts of power around the aircraft, or electrical systems, to match the current self-contained engine systems for reliability. Though schemes along these lines are talked about, they are a long way from being considered seriously.

Pressure-gain combustion continues to attract funding and produce many papers. It is well known that combustion that takes place in detonations gives rise to high levels of pressure, whereas combustion of the type in current engines, deflagration, gives rise to a drop in pressure. Funding appears to remain favorable for research in this area, with the gas turbine being one of the major applications. There are a number of difficulties, but one which is usually given cursory treatment is how the pressure pulse from the detonation wave can be converted to a quasisteady flow through the turbine.

A paper on detonation combustion by Zel'dovich [21] published in Russian in 1940 was republished in English in 2006. This shows, with elegance and simplicity, that the benefits from detonation combustion are small. Although one does obtain very high pressures in the waves, these require, of necessity, low pressure waves and the averaging out leads to only a small net gain. Zel'dovich's calculations were carried out assuming perfect gases, but recent calculations by Shepherd and Wintenberger [22] modeling real gas effects have confirmed the findings. In other words, the benefits from unsteady combustion are small, while the complications required to install the necessary device would be considerable, not to say heavy. It seems likely that the current system of combustion, with a small net loss in pressure and large loss in availability (i.e., a large rise in entropy), will continue to be the necessary approach.

Though it is customary to think of pressure rise combustion as a new or promising idea, it is salutary to quote from the paper written by Zel'dovich [21] in the USSR in 1940. " It is noteworthy that over the last 20 years, engineers have primarily studied gas turbines with constant-pressure combustion even thought their efficiency is lower than that of the explosion turbine (given the same initial pressure). The decisive factors were the simplicity of implementation and reduction of losses in a continuously operating machine."

15 Concluding Remarks

15.1 Land-Based Power Generation. While the true rate of change for land-based power generation in the past 50 or 60 years has been quite slow, there are reasons to believe that change now will be rapid and radical. In terms of saving the planet from devastating climate change, the significance of developments in land-based power dwarfs developments for aviation.

Almost all schemes for advanced thermal power plants require heat exchangers. These are deceptively simple to draw, but a frequent cause of grief. The pitfalls should always be born in mind and if a way exists to avoid them it should be explored.

The successful use of the combined cycle (gas and steam) has created highly efficient power plant. There are some general lessons here about combining air and water for other applications, and one is proposed for the aircraft engine. Combined-cycle plant burning natural gas or other high quality fuel cannot form a large part of the strategy for reducing CO_2 emissions, which will predominantly come from burning coal.[4]

It is probable that thermal power engineering will be much closer to chemical engineering than it has been in the past. It is to be expected that power plants will get much more complicated, partly to squeeze the last bit of efficiency, but also because of the need to capture and store the CO_2 resulting from the burning of coal.

CCS seems an essential route to utilizing coal while minimizing damage to the climate change impact. With CCS, there could be wide application for gas turbines in the new plant. With oxy-fuel the compressor could be handling CO_2 and the combustor using a CO_2 and O_2 mixture, whereas with precombustion separation the gas turbine could be conventional except the combustor would be burning hydrogen.

No one can know for sure what will represent the best route to power generation and CO_2 capture, but the essential learning process is to build prototypes to find out, specifically to find what are the issues when carbon capture is attempted for plant with electrical output in the range from hundreds to thousands of MW. There will certainly be new and interesting technical innovations.

The pressing requirement at the present time is *not* for more research; we know what needs to be done. The most urgent requirement is to build prototypes to find out the problems in practice. This is expensive. From the experience of the prototypes, the needs for research will become clearer.

Those contemplating a career directed to power plants would be well advised to become familiar with advanced thermodynamics, and chemical thermodynamics, in particular. Issues related to chemistry (for example, what are the unavoidable losses with different chemical processes and routes) will play a much bigger part in determining design choices.

15.2 Aircraft Engines. Improved engine performance is only one of the steps to reduce fuel burn per passenger mile and there needs to be improvements in the complete system, which includes the whole aircraft and the air traffic management. Maximum take-off thrust and maximum climb rates should be chosen so that they do not unduly compromise the engine for its cruise condition. Take-off and maximum climb thrusts should be just enough in relation to maximum take-off weight, anything more than this implies that the cruise efficiency will be compromised relative to the optimum.

Aircraft engines are a special problem because of all the conflicting requirements. Weight and safety probably require configurations not very different from the present ones; it is unlikely, for

[4]Note added to Journal paper in proof. Bohr's observation, quoted at the opening of the paper, is born out here. It now looks as if unconventional sources of natural gas (shale gas, tight gas, coal-bed methane) are abundant and more accessible than had been widely realized. Reports suggest that these unconventional sources of natural gas may be more abundant than the conventional sources used hitherto, and as a result the use of coal for electricity generation may be much reduced.

example, that the core will not be concentric with the fan. Aircraft engines for commercial aircraft are also special because the lowest *sfc* is required for cruise, but the limit on temperature (at compressor delivery and turbine inlet) occurs during take-off. Rational design for cruise therefore cannot be carried out without considering the off-design (take-off) condition.

The complex interrelation of components in the design specification of the aircraft engine is easier to understand if the fan pressure ratio is taken as the key independent variable. For a given fan pressure ratio, the bypass ratio is then adjusted so that the core jet velocity gives the lowest *sfc*. The benefit of taking fan pressure ratio as the independent variable is more obvious for off-design consideration, particulary at take-off.

For highest propulsive efficiency and lowest *sfc* the fan pressure ratio should be as low as possible. It is the impact of fan diameter, including the effect on weight and drag, which currently limits how low *fpr* can go. Recent engines have already taken advantage of the benefit from low *fpr*, which allows higher overall pressure ratio and turbine entry temperature for cruise because for take-off (off design) *opr* and *TET* increase less than for earlier engines with higher *fpr*.

If *fpr* is reduced below about 1.45, it is likely that a variable bypass nozzle will be needed to prevent the take-off working point getting too close to the fan surge line. Put another way, a reliable and light variable area nozzle, which has aerodynamic performance as good as the current fixed nozzles, is an essential enabling technology for much lower *fpr*.

For an improved new aircraft with a higher lift-drag ratio at cruise, there is an increase in both the ratio of thrust for take-off relative to cruise and thrust for climb relative to cruise. The logical design decision would be to reduce the ratio of take-off thrust to maximum take-off weight and to accept lower climb rate with new aircraft the very opposite of what is actually happening.

As fan pressure ratio is reduced, the impact of meeting the take-off and top-of-climb ratings increases, probably with a consequent drop in fan efficiency at cruise. The logical design choice for an aircraft with low *fpr* engines intended to minimize fuel burn at cruise would be to reduce take-off thrust and climb rate, again the opposite of what is happening. How much the *fpr* at cruise can be reduced depends to some extent on the specified take-off and climb thrust because these determine the excursion in operating point for the fan.

The scope for improvement in core thermal efficiency from further increase in overall pressure ratio or turbine entry temperature relative to the values currently in large engines is small.

The component efficiencies play a large role in determining the fuel consumption of aircraft engines. The greatest influence is in the efficiency of the fan and the LP turbine, both of which give nearly 1% *sfc* reduction for 1% efficiency improvement. However, the scope for raising component efficiency, after many years of active work, is limited. Likewise, although reducing the cooling air to the turbine would reduce *sfc* (halving the cooling flow to the HP turbine during cruise reduces *sfc* by about 2%), there is not much scope by conventional methods for this after many years work.

Any scheme for aircraft engines, which relies on heat exchangers for a large part of the flow (like all the core flow), seems problematic. The pressure losses in the heat exchanger are likely to be significant and problems of flow nonuniformity in any compact configuration will compromise heat exchanger performance. Moreover the increased weight of the complex pipe work and plenum ducts may undo much of the potential benefit. Lastly, heat exchangers can be a serious cause of unreliability. In summary, heat exchangers are difficult to accommodate in the envelope of an engine with weight, flow nonuniformity, and reliability all major challenges.

Pressure-gain combustion is a topic full of interesting avenues for research, but practical application seems remote now as it did in the past.

The appropriate use of water in aircraft engines should be seriously explored as a way to reduce fuel burn. Quite small quantities of water evaporated in the HP turbine cooling air would reduce the amount of cooling air required as a proportion of the total flow through the core. Adding water at a rate of 7% of cooling air mass flow rate would reduce cooling air temperature by about 210 K and this would by approximately halve the required cooling air to the HP turbine, therefore reducing *sfc* at cruise by about 2%. For a possible long-range flight, this might entail using 34 kg of water during take-off to save about 300 kg of fuel burned during cruise.

15.3 And Finally. The title of this lecture, *Preparing for the Future*, may have led some to expect a prescription for what to do. This is not realistic: What seems possible to me is to prepare by creating understanding of key issues and drivers, and that is what has been attempted here in this lecture. To return to Louis Pasteur, the industry will be in a more favorable position to respond if the appropriate preparation of the thinking processes, to create the understanding and appreciation, has taken place. Many people are now specialists and too few are solid generalists. Surprisingly few people have a broad view of the issues and possibilities for energy, power, and power plant. Few seem properly aware of the challenges and opportunities which will arise as climate change is addressed. This lecture has attempted to address some of these issues.

So let me conclude with two statements of belief. First, we are entering a period of great uncertainty, when the consequences of climate change are being appreciated and action is being planned to tackle this. Plans on their own are not enough and our political leaders are going to need sound advice. The involvement of engineers in the political and legislative processes for dealing effectively with climate-change mitigation is essential. I strongly urge engineers to be involved because decisions should not be left to economists and bankers.

My final point: it is my belief that young engineers associated with thermal power are in for a very exciting future. In some sense, the destiny of the human species, or the way of life we have come to enjoy, depends on what they can accomplish. In particular, those entering the power industry who have the techniques and skill to "keep the lights on" while not destroying the planet may be perceived as heroes to an extent that engineers have not enjoyed for a long time. The opportunities that will open up for interesting and rewarding work are enormous: I believe it will be a glorious age to be in the field.

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Nomenclature

D = drag
Fn = net thrust
L = lift
M = flight Mach number

$\dot{m}$ = mass flow rate
m_{cor} = corrected mass flow
mr = ratio of corrected mass flow
V = flight speed
V_j = jet velocity
δ = pressure/standard pressure
η = efficiency
θ = temperature/standard temperature
θ = angle of aircraft climb

Subscripts

cr = cruise
p = propulsive, polytropic
s = isentropic
TO = take-off
TOC = top of climb
th = thermal
tr = transfer
02 = engine entry condition

References

[1] Young, J. B., 2008, *New Directions in Power Engineering*, Cambridge University, Cambridge, London.
[2] Wilcock, R. C., Young, J. B., and Horlock J. H., 2005, "The Effect of Turbine Blade Cooling on the Cycle Efficiency of Gas Turbine Power Cycles," ASME J. Eng. Gas Turbines Power, **127**, pp. 109–120.
[3] Horlock, J. H., 2002, *Combined Power Plant*, Krieger, Malabar, FL.
[4] Cool, T., 2002, "Using Genetic Algorithm to Optimise Heat Exchangers," http://www.msm.cam.ac.uk/phase-trans/2002/Harrypresarial/
[5] Lindquist, T., Thern, M., and Torisson, T., 2002, "Experimental and Theoretical Results of a Humidification Tower in an Evaporative Gas Turbine Cycle Power Plant," ASME Paper No. GT2002-30127.
[6] Macri, F., 2002, "Next Generation Gas Turbine System Study," Phas1 Final Report No. DE00806992.
[7] Araki, H., Higuchi, S., Marushima, S., and Hatayami, S., 2006, "Design Study of a Humidification Tower for the Advanced Humid Air Turbine System," ASME J. Eng. Gas Turbines Power, **128**(3), pp. 543–550.
[8] 2006, International Energy Agency (IEA) Capture as a Factor in Investment Decisions, Greenhouse Gas Programme, Report No. 2006/8.
[9] Intergovernmental Panel on Climate Change, 2005, Special Report of Working Group III, Carbon Dioxide Capture and Separation.
[10] Epstein, A. A., 2008, "Primer: Aircraft Emission and Environmental Impact," Air Transport Association Meeting: Aviation and the Environment, Washington, DC.
[11] Kurzke, J., 2005, "How to Create a Performance Model of a Gas Turbine from a Limited Amount of Information," ASME Paper No. GT2005-69537.
[12] Kurzke, J., 2008, "The Importance of Component Maps for Gas Turbine Performance Simulations," presented at the 12th International Symposium on Transport Phenomena and Dynamics of Rotating Machinery, Paper No. ISROMAC12-2008-20186.
[13] Advisory Council for Aeronautics Research in Europe, 2008, Strategic Research Agenda, Addendum, http://www.acare4europe.com/
[14] Green, J. E., 2002, "Greener by Design—The Technology Challenge," Aeronaut. J., **106**, pp. 57–113.
[15] Wilfert, G., Kriegl, B., Wald, L., and Johanssen, O., 2005, "CLEAN Validation of a GTF High Speed Turbine and Integration of Heat Exchanger Technology in an Environmental Friendly Engine Concept," International Symposium on Air Breathing Engines, Paper No. ISAB-2005-1156.
[16] Biesiadny, T., and Klann, G., 1992, "Contingency Power for a Small Turboshaft Engine by Using Water Injection into Turbine Cooling Air," NASA Tech Memo 105680/AVSCOM Technical Report No. 92-C-019.
[17] Daggett, D. L., Fucke, L., Hendricks, R. C., and Eames, D. J. H., 2004, "Water Injection on Commercial Aircraft to Reduce Airport NO_x," 40th AIAA.ASME-.SAE.ASEE Joint Propulsion Conference and Exhibit, Fort Lauderdale, FL, AIAA Paper No. 2004-4198.
[18] Becker, A., 2006, "Engine Company Evaluation of Feasibility of Aircraft Retrofit Water-Injected Turbomachines," NASA Report No. CR-2006–213871.
[19] Daggett, D. L., Hendricks, R. C., Mahashabde, A., and Waitz, I. A., 2005, "Water Injection–Could it Reduce Airplane Maintenance Costs and Airport Emissions," 17th International Symposium on Airbreathing Engines, Munich, Germany, Paper ISABE-2005–1249.
[20] de la Rosa Blanco, E., Hall, C. A., and Crichton, D., 2007, "Challenges in the Silent Aircraft Engine Design," AIAA Aerospace Sciences Meeting, Reno, NV, Jan., AIAA Paper No. 2007-454.
[21] Zel'dovich, Ya. B., 2006, "To the Question of Energy Use of Detonation Combustion," J. Propul. Power, **22**(3), pp. 588–592.
[22] Shepherd, J. E., and Wintenberger, E., 2006, "Introduction to "To the Question of Energy Use of Detonation Combustion" by Ya. B. Zel'dovich," J. Propul. Power, **22**(3), pp. 586–587.