JAMES CLAYTON LECTURE

The Aviation Engine

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This lecture is an attempt to give a cross-section of aviation engine development and progress during the last two decades.

Such a lecture is now opportune, since great progress has been made during the last ten years—mainly due to the 1939–45 war and the advent of the aviation gas turbine. The rapid development of this prime-mover has started a new era in aviation and the well-tried piston engine is already being relegated to second place and even eliminated in the more important aircraft applications.

The lecture is divided into two parts. The first deals with the piston engine and the second with the gas turbine. In the first part, the development of the piston engine is discussed; with emphasis upon those factors, such as supercharging and improved fuel, which have particularly contributed to its present high performance.

Cooling developments of both the air- and the liquid-cooled engine are discussed, and also the influence of such detail refinements as the sodium-cooled valve and sparking plugs having sintered aluminium oxide insulators.

The second part of the lecture describes the present forms of aviation gas turbine, with some details of its specific performance and the author's views concerning the future development and application of this new engine.

At the request of the Council, and where relevant, the author has shown how the development of the aviation engine has influenced that of the automobile engine and he has also given—in Appendix I—a summary of his views on the application of the gas turbine to road-vehicle propulsion.

This lecture would not be complete without reference to the great part played, in the 1939–45 war, by the aviation engine and automobile industries of Britain, in combining to produce large numbers of aviation engines for the Royal Air Force. In this particular connexion, the author is of the opinion that the experience and the production methods of the automobile industry, as a whole, contributed largely to the uniform excellence and economical production of the aviation engine, and fully vindicated the faith of those who originated the "Shadow" scheme in the rearmament period before the war.

PART I—THE AVIATION PISTON ENGINE

BRISTOL ENGINES

(*a*) *Mercury and Pegasus.* The first engines to be put into large-scale production by the Bristol Aeroplane Company and the "Shadow" group of factories, during the 1939–45 war, were the "Mercury" and "Pegasus". These were both 9-cylinder, air-cooled, radial, four-valve engines. The Mercury had a capacity of 1,525 cu. in. or 25 litres, the Series XII version being rated at 890 b.h.p. at 2,750 r.p.m., at an altitude of 6,000 feet. The (dry) weight of this engine was 995 lb. The war-time Mercury had a higher rating than its racing predecessor, which was originally built in 1927 as a Schneider Trophy engine and then gave 800 b.h.p.

The Pegasus was an engine of 1,752 cu. in. or 28·79-litres capacity, and the Series XXVII had a rating of 925 b.h.p. at 2,600 r.p.m. at an altitude of 10,000 feet, for a (dry) weight of 1,035 lb.

These two engines gave excellent service in the war but, eventually, the demands for still greater power rendered them obsolete and their place was taken by the "Hercules", which was the first sleeve-valve engine to go into large-scale production.

The Bristol Aeroplane Company first became interested in the sleeve-valve engine in the middle 'twenties and, in 1926, had already started the development of a single-cylinder "Burt" type of sleeve-valve unit.

(*b*) *Perseus.* Single-cylinder development continued until 1932 when, in the middle of that year, Bristol built the first multi-cylinder, sleeve-valve engine, the "Perseus", which had the same bore and stroke and cubic capacity as the Mercury.

The Perseus VIII, which was type tested in 1936, was the first sleeve-valve engine to go into service and, at the beginning of the 1939–45 war, the fully supercharged Perseus X gave a power of 880 b.h.p. at 2,750 r.p.m., at 15,500 feet.

(*c*) *Hercules.* In 1935, with the Perseus in the final development stage, the Bristol Aeroplane Company were already engaged upon the design and development of a 14-cylinder, two-row, sleeve-valve radial engine, the "Hercules". This had the same cylinder size as the Perseus, but the total displacement was 2,360 cu. in. or 38·7 litres.

The Hercules passed its first military type test early in 1937 and, during the war, was developed to give 1,600–1,700 b.h.p. for a total (dry) weight of about 2,000 lb.

The power of the post-war Hercules has since been increased to more than 2,000 b.h.p.

The Perseus engine, which had also been developed for civil use, was dropped in the early stages of the war because the Mercury and Pegasus had already been elaborately tooled-up for large-scale production and the Bristol Aeroplane Company were then pressing along with the larger and more powerful Hercules.

There was also at that time a 14-cylinder, two-row, sleeve-valve engine of 1,000–1,100 b.h.p., the "Taurus", which had smaller cylinders than the other engines, with a 5-inch bore and 5·625-inch stroke. But the production of this engine was discontinued, since it had limited application.

(*d*) *Aquila.* Yet another Bristol sleeve-valve engine was the small 9-cylinder "Aquila", which had the same bore as the Taurus but a slightly shorter stroke—5·375 inches. The civil version of this engine, the Aquila IV, had a take-off power of

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about 600 b.h.p. but, owing to the war, it did not go into production.

(*e*) *Centaurus*. During the war, the Bristol Aeroplane Company produced another and somewhat larger engine than the Hercules. This was the 18-cylinder "Centaurus", which had the same cylinder bore as the Hercules (5·75 inches) but a longer stroke—7 inches instead of 6·5 inches. This gave a total cylinder capacity of 3,270 cu. in. or 53·6 litres. The take-off power of the present Centaurus is about 2,450 b.h.p. at 2,700 r.p.m., and 2,700 b.h.p. at 2,900 r.p.m.

The power ratings of the Hercules and Centaurus engines are given on 100–130 grade fuel.

ROLLS-ROYCE ENGINES

(*a*) *Merlin*. The principal engine built by Rolls-Royce during the 1939–45 war was the "Merlin".

There was the "Kestrel", which had been very successful in the biplane fighters such as the Hawker "Fury", but this engine was obsolescent at the beginning of the war and only saw very brief service in a twin-engined fighter, the Westland "Whirlwind". The Kestrel was a 12-cylinder, 60 deg. "V" type of liquid-cooled engine with a displacement of 1,295 cu. in. or 21·24 litres. In its final development, it gave a take-off power of 765 b.h.p. with 6 lb. per sq. in. boost, and had a maximum power rating of 745 b.h.p. with 3·25 lb. per sq. in. boost at 14,500 feet.

The first experimental model of the Merlin was completed in the latter part of 1933 and, in nine months after the first run on the test bed, the engine had passed a type test at a rating of 790 b.h.p. at 2,500 r.p.m. with 2 lb. per sq. in. gauge boost, at 12,000 feet equivalent altitude. The take-off power was 625 b.h.p. and the (dry) weight was 1,177 lb.

The Merlin was a 12-cylinder, 60 deg. "V" type of liquid-cooled engine having a cylinder bore of 5·4 inches and a stroke of 6 inches, giving a total displacement of 1,649 cu. in. or 27 litres.

In 1935 a later model, the Merlin B, had been developed, to give 950 b.h.p. at the equivalent altitude of 11,000 feet. These first engines originally had integral cylinder-block and crankcase construction and a combustion chamber of the ramp or turbulent type. Eventually, Rolls-Royce discarded the integral construction in favour of a separate cylinder block and crankcase, with the cylinder block and head in one piece. The cylinder head design was also changed from the ramp to the Kestrel type, with a flat, four-valve, combustion chamber.

Odd-shaped combustion chambers, designed to promote turbulence and improve combustion, are undesirable for the engine cylinder of the aeroplane, as distinct from that of the automobile. In a cylinder having such relatively high heat flow, a combustion chamber of non-uniform section will probably produce results the reverse of those desired, owing to the difficulty of obtaining uniform and adequate cooling—leading to hot spots and distortion, and causing valve trouble. Certainly the combustion chamber must be compact and, after providing for adequate valving and two sparking plugs, there is little scope left for creating a combustion chamber of fancy shape. In no circumstances should the piston be other than uniform in shape and section.

In the Merlin one-piece cylinder head and block the cylinder liners acted as compression struts, the whole assembly being held in position and pre-loaded against explosion loads by through-bolts. This proved to be a light and satisfactory construction in the early Kestrel engines, but the intensive development of supercharging by Rolls-Royce gave rise to trouble in the later Kestrel engines and in the Merlin, owing to gas leakage occurring between the top joint of the cylinder liner and the coolant space. Early in 1938 a new type block of two-piece construction was put in hand, in which the top flange of the cylinder liner was nipped or held between the separate cylinder head and the upper surface of the cylinder block. Owing, however, to the heavy commitments of the re-armament programme, followed by the war, the two-piece block was not put into production until 1942.

In 1936 the Merlin Mark I and II engines were developing a take-off power of 880 b.h.p. and had an altitude rating of 1,030 b.h.p. at 16,250 feet, on 87 octane fuel. Their net (dry) weight was 1,335 lb. The modern Merlin engine, on 100–130 grade fuel, gives 1,760 b.h.p. for a (dry) weight of about 1,740 lb., with an altitude rating of about 1,150 b.h.p. at 23,500 feet. Special military types of Merlin were developed during the war having an altitude rating of 1,500 b.h.p. at 25,000 feet.

Some interesting tests were done by Rolls-Royce during the war period, to explore the possibilities of the Merlin in regard to absolute maximum sea-level power.

Using an engine with a two-stage supercharger in high gear, on the test bed, a maximum corrected power of 2,620 b.h.p. was obtained at 3,150 r.p.m. with a boost pressure of 101·46 inches of mercury (35·73 lb. per sq. in. gauge) on a fuel of 100–150 grade, plus water injection. A ten-hour flight approval test was made with a similar engine in a "Mustang" and, at a limiting boost of 91 inches of mercury (30·5 lb. per sq. in. gauge) on 100–150 grade fuel only, the maximum power developed was 2,129 b.h.p. at 3,150 r.p.m. The former power is equivalent to a b.m.e.p. of 400 lb. per sq. in. and the latter to one of 326 lb. per sq. in., calculated on net power delivered at the propeller shaft.

The Merlin proved to be the most versatile of the war engines, and was used with equal success in both fighters and bombers. It was also built in large numbers by the Packard Motor Company of America.

The development of the Merlin engine is described by Lovesey (1946)*.

(*b*) *Vulture*. In 1939 Rolls-Royce had built another engine, the "Vulture". This was a 24-cylinder, liquid-cooled, 90 deg. "X" type of engine, on a single crankshaft. It had the equivalent displacement of two Kestrels—2,590 cu. in. or 42·5 litres. The Vulture Mark II, which was fitted with a 2-speed supercharger, had a combat rating in low blower gear of 1,845 b.h.p. at 3,000 r.p.m. with 6 lb. per sq. in. boost, at 5,000 feet altitude. In high blower gear, 1,710 b.h.p. was developed at the same boost at 15,000 feet. The take-off power of the Vulture was 1,800 b.h.p. for a (dry) weight of 2,450 lb. The engine was at that time type tested on 87 octane fuel but, in 1941, it was run experimentally on 100 octane fuel and gave a take-off rating of 2,010 b.h.p. at 3,000 r.p.m. with 9 lb. per sq. in. boost.

The Vulture engine was the power unit of the Avro "Manchester" bomber and it was also installed experimentally in the Hawker "Tornado" fighter, but its production ceased when the "Manchester" was replaced by the "Lancaster" with four Merlins. This enabled Rolls-Royce to concentrate their efforts upon the Merlin and the "Griffon".

(*c*) *Griffon*. The "Griffon," a larger engine than the Merlin, was of the same size as the earlier "H" or "Buzzard" engine, that had been so successfully developed as the "R" or racing engine for the seaplanes which won the Schneider Trophy contests of 1929 and 1931. It had a bore of 6 inches and a stroke of 6·6 inches, giving a displacement of 2,239 cu. in. or 36·7 litres —about 36 per cent greater capacity than the Merlin. In 1939, when the design of the Griffon was first considered, it was originally intended to be a simple scale-up of the Merlin. But, since this would have meant a somewhat longer engine and because it was intended that the Griffon should be capable of replacing the Merlin in the same aircraft, an entirely new design was made, incorporating the lessons learned from the Kestrel and the Merlin, with particular attention paid to ease of production.

Griffon production began early in 1942, and it replaced the Merlin in some of the earlier "Spitfire" models; but it was not until the middle of 1944 that the Spitfire XIV, fitted with the Griffon 61, first showed real superiority over the Spitfire-Merlin combination which had been so successful up to that time.

The post-war Griffon 64 produces a power of 2,375 b.h.p. for a net (dry) weight of 2,090 lb. The Griffon 57 develops 2,450 b.h.p. on 100–130 grade fuel, plus water-methanol injection, which permits a boost of 25 lb. per sq. in. The specific (dry) weight of the Griffon at this rating is 0·825 lb. per b.h.p.

* An alphabetical list of references is given in Appendix II.

(*d*) *Eagle*. Another engine designed and built by Rolls-Royce during the war, although it did not see war service, was the 24-cylinder, double-crankshaft, sleeve-valve "Eagle". This engine, which revived a famous name, was larger but similar in general arrangement to the Napier "Sabre". The Eagle had the same bore as the Merlin but a shorter stroke (5·125 inches) giving a displacement of 2,808 cu. in. or 46 litres. The maximum power rating of the latest Eagle Series 22 is 3,500 b.h.p. at 3,500 r.p.m. and 3,250 feet altitude. The net (dry) weight is 3,900 lb. The Series 22 model is fitted with a contra-rotating propeller reduction gear.

All Rolls-Royce engines have single-point fuel injection at the supercharger eye.

THE NAPIER "SABRE" ENGINE

In 1936, encouraged by the Air Ministry, Halford designed for Napier an advanced, liquid-cooled, 24-cylinder, sleeve-valve, double-crankshaft engine of flat "H" configuration, the "Sabre".

This engine first ran in 1937. It had a bore of 5 inches and a stroke of 4·75 inches, giving a total displacement of 2,239 cu. in. or 36·7 litres, and was designed for a power of 2,000 b.h.p. During the war, in successive stages of development, the take-off power was raised to 2,600 b.h.p. An interesting feature of the Sabre was that it fired as a 12-cylinder engine, that is, two cylinders firing at the same time.

In 1946 the Sabre 7 was type-tested at a take-off power of 3,000 b.h.p. at 3,850 r.p.m. with a maximum boost of 17·25 lb. per sq. in. and a compression ratio of 7/1. The fuel used was 100–130 grade, plus water-methanol injection for the take-off power. A power of 1,975 b.h.p. was developed at 3,700 r.p.m. with 10·5 lb. per sq. in. boost, at 18,250 feet. The (dry) weight of the engine was 2,540 lb.

The Sabre was used in the Hawker "Typhoon" and "Tempest" fighters, but its production was discontinued after the war. Since it was a specialized engine primarily intended for fighter or pursuit type of aircraft, it had to give way to the jet propulsion gas turbine in this particular sphere.

GERMAN DEVELOPMENTS

When Hitler came to power there was a rapid revival and intense development of the high-duty aviation engine, all work on which had ceased after the 1914–18 war owing to the restrictions imposed by the Treaty of Versailles. With the Nazis in power, firms such as Daimler-Benz, Junkers, B.M.W. and Bramo-Fafnir were encouraged to design and develop new engines of higher power. Other firms such as Argus and Hirth were already active in building small air-cooled engines of relatively low power.

Daimler-Benz and Junkers both concentrated upon the liquid-cooled engine, while B.M.W. and Bramo built the air-cooled radial type.

To some extent, the Germans had the advantage in starting their development of the high-duty engine after a good deal of experience had already been accumulated in Britain and the United States—although, naturally, the Germans were devoid of the necessary techniques. But the experience of other countries, whose activities in the field of aviation engine development had continued after the 1914–18 war, gave the Germans a fairly good idea of the shape and size of engine upon which to start.

Both Daimler-Benz and Junkers concentrated upon the inverted type of liquid-cooled gasolene engine, with the latter also developing a two-stroke Diesel or compression-ignition aviation engine of the characteristic twin-crankshaft opposed-piston type.

A licence for the latter engine had been granted to Napier before the 1939–45 war. But the development of this compression ignition engine was eventually discontinued, although a number were produced for the JU.86 German military transport at the beginning of the war.

B.M.W. and Bramo-Fafnir air-cooled radial engines generally followed United States practice, having 2-valve cylinders.

In common with all high-duty German engines, the B.M.W. and Bramo-Fafnir made an early change from the float type of carburetter to direct fuel-injection to the individual cylinders.

The Daimler-Benz "D.B." and Junkers "Jumo" series of engines were all of the supercharged, 12-cylinder, liquid-cooled, inverted 60 deg. "V" type, with direct fuel injection.

These German liquid-cooled engines were of relatively low weight, the specific weights being 1·06 lb. per h.p. and 0·91 lb. per h.p. for the Jumo and 1·15 lb. per h.p. and 1·09 lb. per h.p. for the Daimler-Benz models.

If the Germans had properly developed their larger-capacity, liquid-cooled engines, they would have had the advantage of the Merlin fairly early in the war and forced the British to intensify the development of the Griffon. But their lack of technique prevented them from exploiting this potential advantage, and the Merlin kept ahead.

DEVELOPMENT HIGH-SPOTS

The development of the aviation engine was very much accelerated during and because of the two world wars. Many millions of pounds have been expended upon its improvement, and the modern examples represent the highest form of specialized engineering technique.

The piston type of aviation engine has advanced in performance from 25–30 b.h.p., in 1909, to 3,500–4,000 b.h.p. at the present time—more than a hundred-fold power increase in just over forty years. Equally important is the fact that the sea-level power can now be reproduced at an altitude of more than 30,000 feet.

The power developed per cubic inch of piston displacement has increased from

about, 0·12 h.p. in 1909
to 0·3 „ „ 1919
to 0·4 „ „ 1929*
to 0·59 „ „ 1939
and 1·34 „ „ 1949–50†

In the same period, the specific (dry) weight on the sea-level or take-off basis has been reduced from 3–4 lb. per b.h.p. to 1 lb. per b.h.p. and less. The specific weight of the Griffon 57 is 0·825 lb. per b.h.p., and that of the Sabre 7 is 0·85 lb. per b.h.p.

Fuels. The improved engine performance in the last twenty years has been mainly due to supercharging and the use of better fuels. Comparing the performance in 1930 on 73–77 octane fuel —specified by the Air Ministry and consisting of a mixture of approximately 80–20 straight-run aviation gasolene and benzol—with that on present-day 115–145 grade fuel, the increase in power has been between 300 and 400 per cent, with about 20 per cent improvement in fuel consumption.

Leaded fuel, 87 octane, was first introduced into Britain early in the 'thirties and was used for all British military engine development from 1932, onwards. Later came the 100 octane fuel specification, which entirely replaced the 87 octane material in the 1939–45 war. Full credit for both these fuels goes to the United States.

Superchargers. The mechanically driven supercharger, a British development, was first used by Armstrong Siddeley on their "Jaguar" engine and was brought later to a high state of development by the Bristol Aeroplane Company and Rolls-Royce. It has been built in a number of forms and variations such as single-speed single-stage, two-speed single-stage, two-speed two-stage and, even, three-speed two-stage. The first two-speed single-stage supercharger was built by Armstrong Siddeley.

The adiabatic efficiency of a modern two-stage blower is between 73 and 74 per cent at a pressure ratio of 4/1, dropping to 71 per cent at 6/1 ratio. Higher figures than these have been obtained more recently on experimental blowers.

Variable-pitch Propeller. The full benefit of both supercharging and the use of the propeller reduction gear could not,

* The value for the Rolls-Royce type of Schneider engine in 1931 was more than 1 h.p. per cu. in.

† The 1949–50 value is that of the Napier Sabre, obtained on 100–130 grade fuel plus water-methanol injection.

however, be realized until the advent of the variable-pitch propeller. The first practical example of this was the American Hamilton-Standard, two-position, controllable-pitch propeller, which permitted great improvement in take-off and, also, in performance at altitude. Later, Hamilton-Standard developed the controllable-pitch, constant-speed propeller, which allowed the propeller to work at a chosen speed almost independent of engine power and altitude.

Although a variable-pitch propeller had already been built in this country—the Hele-Shaw Beacham—its development was not encouraged, because the use of a reduction gear and low wing-loadings already permitted good take-off performance. This appeared to blind those concerned to the great possibilities of the propeller to give improved aircraft performance at higher wing-loadings. Most of the United States engines of that time had direct propeller drives and, when the prototype of the classic Douglas civil aircraft was built, the "D.C.1.", its considerably higher wing-loading demanded the controllable-pitch propeller for successful operation.

To Rolls-Royce can be credited the development of the (liquid-cooled) after-cooler, which was the natural result of their work on supercharging, when highly supercharged engines demanded charge cooling to avoid fuel depreciation and to keep clear of detonation and pre-ignition. The after-cooler reduces the charge temperature by about 100 deg. C., but this depends upon the operating conditions, and is mainly a compromise between desired manifold temperature and the "drag" of the after-cooler radiator.

The Exhaust Turbo-blower. The exhaust turbo-blower was originally developed by the Royal Aircraft Factory (now the R.A.E.), at Farnborough. The Factory became interested in this type of blower during the 1914–18 war, and built one on the lines originated by Rateau in France. Their effort was, in fact, the first practical application of Rateau's ideas.

The turbo-blower was tried experimentally on the test-bed and in flight but, because of the formidable temperature and material problems, it was eventually dropped.

However, the United States Army Air Corps (now the U.S.A.F.) became very interested in the exhaust turbo-blower; and Wright Field, in collaboration with the General Electric Company of America, started to investigate its practical possibilities during the 'twenties.

After many years of painstaking development they succeeded in producing a satisfactory turbo-blower, although it had a somewhat short operating life. But this was enormously improved during the 1939–45 war; and its maintenance is now regarded as reasonable, when compared with the normal operating period of the engine.

The apparent lack of British interest in the turbo-blower, apart from questions of temperature and metallurgy, was probably because the propeller reduction-gear already existed in Britain, whereas the United States engines then had direct propeller drive. This, and the mechanically driven supercharger, offered considerable improvement in aircraft performance without resort to the turbo-blower.

The United States application of the turbo-blower, which was in the first place a military one, made use of the exhaust gases by collecting them in a common manifold and discharging them through a turbine to increase operational altitude and improve the performance of the aircraft at altitude. A waste-gate was provided in the exhaust system to cut out the turbo-blower at sea-level or take-off and discharge the exhaust direct to the atmosphere. Those engines fitted with the turbo-blower only then became naturally aspirated for take-off; but, generally, there was a single-speed, single-stage, mechanically driven blower for take-off, which acted as a second compressor stage when the turbo-blower came into operation.

Since the 1939–45 war, the United States have applied the turbo-blower in other ways—fully to compound the piston engine. Instead of using only a proportion of the exhaust energy, as previously, they now employ the exhaust turbine (without blower) to absorb the bulk of the exhaust gas energy, and return this to the engine by connecting the turbine mechanically —through double-reduction gearing—to the engine crankshaft.

For example, the 18-cylinder Wright "Cyclone" 3,350 cu. in. or 55-litre engine, has three blow-down or impulse-exhaust turbines, arranged radially at the rear of the engine, each turbine being fed by a group of six cylinders. The turbines feed back power to the engine crankshaft through couplings and gearing, each turbine having its own cooling arrangement. This engine produces a maximum of 3,250 b.h.p., and is reported to have a cruising fuel consumption of less than 0·4 lb. per b.h.p. at a power of approximately 1,400 b.h.p., available at 20,000 feet. Blow-down or impulse turbines are employed to avoid high back-pressure.

In this particular application there is a two-speed, two-stage, mechanically driven blower, which is of sufficient capacity to provide all the supercharging required.

Another compounding scheme is under development by Pratt and Whitney. In this, the 28-cylinder engine of 4,360 cu. in. or 71·5 litres is fitted with full capacity exhaust turbo-blowers, and the mechanically driven supercharger has been completely discarded. This allows an increase in available shaft power of the amount previously absorbed by the mechanically driven supercharger, the full capacity exhaust turbo-blowers providing the degree of supercharging required. This engine's cruising power, at about 40,000 feet, is reported to be a very high proportion of its take-off power. Since it is a military development actual figures are not yet available, but it is said to give considerably more than 4,000 b.h.p. for take-off. It is not known whether there are engine cooling difficulties at altitude for such high power output.

IMPROVEMENTS IN DETAIL

In regard to materials of engine construction. The forged light-alloy piston, which is now universally used, first came into use in the early 'twenties, when Messrs. Peter Hooker Ltd. developed the method of forging "Y" alloy and first used it in the Monosoupape "Gnome" engine, which was built under licence by them. The forged piston was later fitted in all Jaguar engines and then in the racing Napier "Lion" which won the 1927 Schneider Trophy contest. Shortly afterwards, this piston became established for all aviation engines.

Copper-lead and silver bearings, with a flash coating of lead—both American developments—are in general use today.

In the poppet-valve engine, Heron's sodium-cooled valve has contributed largely to engine reliability at the powers permitted by supercharging and better fuel. Valve facing materials such as "Stellite", "Brightray" and "Eatonite" are a valuable contribution by the material specialists to improve exhaust-valve condition and life. Stellite No. 6, which is the grade generally used for facing exhaust valves and the insert seat rings in the cylinder head, is an alloy of approximately 65 per cent cobalt, 30 per cent chromium and 5 per cent tungsten. Brightray is an alloy of approximately 80 per cent nickel and 20 per cent chromium.

Stellite was first used to face exhaust valves but, later, it was found that Brightray offered superior resistance to hot corrosion attack by the combustion products of leaded fuel, and it replaced Stellite for this purpose in British engines. It is usual to coat the whole valve head and the face or seat with Brightray. Stellite, because of its hardness, is used mainly to face the steel insert ring in the cylinder head, and it is particularly resistant to corrosion at the relatively low temperature of the ring.

The single sleeve (Burt) type of valve, pioneered by Ricardo and so successfully developed by Bristol for the aviation engine, is another and very satisfactory method of ensuring freedom from trouble at the combustion end of the engine.

THE CARBURETTER VERSUS THE INJECTOR

The first single-point injector or injection carburetter was originally conceived by the Royal Aircraft Establishment at Farnborough, and was further developed during the 1939–45 war. Bendix of the United States had also developed an injection carburetter, which was very largely used on United States engines and also on some British engines during the war.

The reason why British engines did not use direct injection—to the individual cylinders—which was general practice for German war engines, is explained by the reason why the

Germans discarded the carburetter in favour of direct injection. When Daimler-Benz originally designed their engine it incorporated the French idea of a "*moteur-canon*", with a 20 mm. gun in the "V" of the cylinder blocks, firing through the hollow propeller shaft. As a result, the supercharger had to be mounted on one side of the engine, at 90 deg. to the crankshaft.

It was not then possible to fit the normally aspirated (suction) carburetter to the supercharger without putting a bulge in the cowling line of the aircraft. Therefore, a pressure carburetter was used, on the discharge side of the blower, fitted between the cylinder blocks. But, since there were no well-developed carburetters available and because a pressure carburetter tended to make things more difficult, considerable carburation trouble was experienced. So much so that the Germans naturally turned to the direct-injection system, because of their considerable knowledge of injection technique on the "Junkers" compression-ignition aviation engine.

On the other hand, the British had brought the normal float type of carburetter to a high state of development, and it gave little trouble in its normal functioning. But the limitations of this carburetter soon became apparent during the war, and particularly in the case of air fighting, where the pilot wished to attack an enemy flying below him. When the nose of the machine was put down quickly the resultant gravity effect caused continued fuel flow and the engine suffered what was known as a "rich mixture cut". After various expedients had been tried to overcome this trouble, it was decided to use a single-point injector, injecting into the eye of the blower.

Rolls-Royce had, in fact, already considered single-point injection, quite apart from the carburetter limitations mentioned, since the carburetter Venturi offered considerable restriction to the air flow in the highly supercharged engine and, therefore, limited the performance. As a result, Rolls-Royce developed a single-point injector operating on the "speed-density" principle —based on the fact that the mass of air taken into the engine cylinder is proportional to the difference between the pressures in the inlet and exhaust manifolds and inversely proportional to the absolute temperature in the inlet manifold, the total mass flow being determined by these factors and the engine speed (r.p.m.). The speed-density fuel-metering system directly governs the fuel flow by mechanical means sensitive to the difference between the two pressures, the temperature and engine speed.

SPARKING PLUG DEVELOPMENT

A complete paper could be written on the subject of the sparking plug and its development during the last two decades, but it is sufficient to say that the progress made in the last ten years or so is such that the sparking plug no longer presents a real problem in the case of the high-duty engine running on leaded fuel.

In the early 'thirties, with the advent of supercharging and 87 octane leaded fuel, the limitations of mica as an insulating material soon became apparent, necessitating improvement in the operating condition of the sparking plug, by better cooling of the central electrode and that of the complete plug in the cylinder head of the engine.

The troubles experienced with mica were due to the ever-increasing engine duty and consequent heat increase, which tended to dehydrate the mica and render it more susceptible to attack by the combustion products of leaded fuel.

During this period there was developed by Siemens, in Germany, a material known as *Sinterkorund*, basically, sintered aluminium oxide, which proved highly resistant to attack by the combustion products of leaded fuel.

Eventually both the Siemens and Bosch Companies were experimenting with plugs having insulators of this material; and, in the years immediately prior to the 1939–45 war, they had produced suitable aviation sparking plugs. The Bosch material was a variant of that used by Siemens, principally, it is understood, to avoid certain basic patents of Siemens.

The author was responsible for introducing some of the original samples of these plugs to Britain and, later, with the encouragement of the Air Ministry and the engine manufacturers, three British plug manufacturers, A.C., K.L.G., and Lodge, started to develop the aluminium oxide insulator for aviation plugs. The A.C. Company made use of the original Siemens' material, while K.L.G. took a licence for the Bosch insulator. Lodge developed a very successful insulator of the same basic material along their own lines.

At first there was considerable alchemy involved in the development of a suitable aluminium oxide insulator. Many mixtures were tried, which looked good at first but gave considerable trouble when the finished plug was tried in the engine. However, with the pressure of war, a very reliable insulator was eventually evolved, and British plugs undoubtedly led the world at that time.

Each plug company has its own manufacturing technique in regard to the preparation and firing of the aluminium oxide and its binder. The latter is generally some form of clay and, therefore, contains silica. Great care must be taken to keep the silica content as low as possible, since silica can combine with lead to form a glass or glaze and produce a conductive surface which will cause electrical breakdown under running conditions, even although at room temperature the plug may spark quite satisfactorily. Most manufacturers of aviation plugs try to keep the silica content down to four or five per cent.

This type of insulator is fired in a gas kiln at a temperature somewhere between 1,500 and 1,700 deg. C.

Another important British development was the fine-wire electrode of platinum alloy, originally introduced by K.L.G. This type of electrode has been a feature of high-duty British sparking plugs for more than ten years—the principal aim was to reduce corrosion and erosion and consequent gap increase. The platinum-alloy wire electrode together with a carbon resistor of about 1,000 ohms—also used to reduce gapping, and developed by K.L.G. in conjunction with Napier—brought about a great improvement in sparking plug efficiency and life.

When the United States came into the 1939–45 war their mica type of plug gave considerable trouble, which had a serious effect upon the early operations of the Eighth Air Force. This resulted in a demand for British plugs and these, practically, saved the day.

The sintered aluminium oxide insulator is now universally used for aviation plugs, and some very good examples are also being made in the United States.

Peculiar as it may seem, the German plugs developed during the 1939–45 war were not outstanding, and would have given a poor account of themselves in high-duty British engines—principally due to poor design and manufacture.

MERITS OF AIR- OR LIQUID-COOLING

There were many arguments between the wars on the relative merits of air- and liquid-cooling.

The air-cooled engine was largely favoured in the nineteen-twenties when the plumbing troubles of the liquid-cooled type seriously affected reliability of engines. Since that time the air-cooled engine has predominated in world civil aviation. In the military field, immediately after the 1914–18 war, the liquid-cooled engine retained its place in such machines as the two-seater fighter and reconnaissance aircraft and, also, in bombers. But, later in the 'twenties, it was being replaced by the air-cooled engine, even before the Napier "Lion", the successful liquid-cooled engine of that period, was rendered obsolete.

In fact, the cooling-system troubles of the liquid-cooled engine were so formidable that, at one time, there appeared to be some risk of this type of engine becoming obsolete.

That this did not happen was due, primarily, to two factors. First, the introduction by Fairey of a very "clean" biplane 2-seater fighter and reconnaissance machine (the "Fox"), built around the 400 b.h.p. Curtiss "D.12" engine. Fairey had visited the United States and had seen the first of the really "clean" biplanes built by Curtiss; and, upon his return to England, had designed and produced the Fox, fitted with the (imported) D.12 engine. This stimulated other British aircraft firms and, also, engine builders, encouraged by the Air Ministry, with the result that the Hawker "Fury" and "Hart" machines of the early 'thirties were built, fitted with the Rolls-Royce Kestrel engine.

The second factor was the influence upon military thinking of the Schneider Trophy contests of 1929 and 1931. These contests demonstrated the potentialities of supercharging and high specific power output, in conjunction with low-drag cooling, to improve fighter performance; although the particular wing surface cooling used on the racing Supermarine seaplanes was scarcely a practical proposition for more normal aircraft.

But even with the advent of the Kestrel engine, which proved very successful, there was no rush to the liquid-cooled engine. In fact, in 1935, British aircraft manufacturers were still complaining that they were paying too much in drag and weight for liquid cooling, although the fixed radiator, with its shutter control, had been replaced by the retractable type.

The air-cooled engine had already derived benefit from an important development in the United States in 1927 devised by the National Advisory Committee for Aeronautics of America, in conjunction with the United Aircraft Corporation: the N.A.C.A. cowl. This was a complete engine cowling which was designed, in the first place, to reduce drag and was later developed, with the cylinder baffle system, to improve the cooling of air-cooled radial engines. The N.A.C.A. cowl eventually became the accepted and basic type of cowling for the air-cooled radial engine. In the latest examples of radial engine installation little of the engine can be seen, the air entering and leaving through relatively small slots in the leading and trailing edges of the cowling.

In parallel with, but quite separate from, this development was the Townend ring, devised by Townend of the National Physical Laboratory. This was a simple ring of aerofoil section fitted around the radial engine, to reduce its drag. The Townend ring was used for some time in Great Britain and in the United States, on military aircraft, in preference to the N.A.C.A. cowl, owing to an early prejudice against the latter by United States military pilots who, it is alleged, complained that it interfered with their forward view. Eventually, however, the Townend ring went out of use in favour of the N.A.C.A. cowl.

As a result of the complaints against the liquid-cooled engine or, rather, against its installational and cooling problems, Rolls-Royce, the only active manufacturer of this type of engine in the 'thirties, intensified their efforts to improve the reliability and efficiency of the system and reduce installed drag. They tried evaporative cooling but found it unsatisfactory, particularly for fighter type of aircraft which were subjected to considerable manoeuvres, which often put the steam where the water should have been and *vice versa*.

In 1935, a most important discovery was made by Meredith at the Royal Aircraft Establishment. He found that the energy which was lost in forcing air through the radiator could be compensated for, to some extent, by putting the radiator in a properly designed duct where the heat could then be converted into useful thrust. This discovery, leading to the use of the ducted radiator, together with later improvements in radiator design and construction, finally helped to bring the liquid-cooled engine back into full favour.

In 1936, Rolls-Royce acquired for their flight test department a Heinkel "70" machine. This was the cleanest low-wing monoplane in existence at that time and it was used by Rolls-Royce as a flying test-bed for their engine installation experiments, many tests being made with different radiators in various positions.

To avoid an increase in radiator size when engine power was increased, and for R.A.F. machines operating in tropical climates, Rolls-Royce tried ethylene glycol as a coolant. It was thought that, in replacing water by a material of higher boiling point, it would be possible to run at relatively high engine-jacket temperature and use a smaller radiator. The United States had also tried glycol, but they had discarded it because of the difficulties of maintaining a leak-proof cooling system, and because the higher coolant temperature—about 20 deg. C. higher than water—caused excessive cylinder temperatures; these, in turn, gave rise to lubrication troubles and sticking piston rings, and also depreciated the anti-knock value of the fuel.

A mixture of 97 per cent glycol and 3 per cent of water was used, which had a boiling point of 163 deg. C. Glycol has very searching qualities and it was exceedingly difficult to maintain leak-proof joints in the engine and in the cooling system as a whole. As a result, Rolls-Royce reverted to water or, rather, to an anti-freeze mixture of 70 per cent water and 30 per cent glycol.

Another trouble of liquid cooling was the loss of coolant by boiling and evaporation, with serious risk of lowering the coolant level in the engine jacket and causing overheating and engine failure. To avoid this, Rolls-Royce developed the integral vapour separator and header tank which, in its final form, was fitted at the front of the engine and above the jacket level. They next developed a pressure cooling system, which not only ensured against loss of coolant, due to a sealed circuit, but also permitted the engine to operate satisfactorily over a wide range of conditions, without alteration of the radiator size—since the coolant temperature could then be raised without risk of boiling or loss.

The pressure system came into general use for Rolls-Royce engines immediately prior to the 1939–45 war. The maximum pressure at present permitted in the system is 40 lb. per sq. in., and the working pressure, at altitudes between 10,000 and 20,000 feet, at a coolant temperature of about 105 deg. C., is from 10 to 20 lb. per sq. in.

Probably the most important change, which came as a result of cooling research by Rolls-Royce, was the substitution for the old honeycomb type of radiator of the secondary-surface radiator, in 1940. The automobile industry had discarded the honeycomb radiator many years before, in favour of the secondary-surface type. The result of this change was a reduction in frontal area and in the weight of coolant carried, by about 10 per cent and 40 per cent, respectively.

With the honeycomb radiator there is about an equal area of metal exposed both to air and water and, since heat is carried much more rapidly through a film of water than through a film of air, an unnecessary amount of water was being carried in the cooling system. In conjunction with Morris Radiators, a radiator was developed having a much larger surface exposed to air than to water—between five and ten to one. This was very similar to the ordinary automobile radiator, except that the radiator tubes of the aviation type were of greater chord.

The advantages of the secondary-surface radiator, apart from its improved efficiency, are that it is lighter and much cheaper to produce than the honeycomb type. Also it is more easy to inspect and repair.

Since then, as the result of further development by Rolls-Royce and Marston-Excelsior, a light-alloy radiator has been evolved which is now in production for all civil types of Merlin engine. The aluminium radiator is, naturally, much lighter than the copper tube type, the difference being about 110 lb. The actual weights are 71·5 lb. for the former and 181·5 lb. for the latter.

This development followed that of the light-alloy radiator used for the supercharger after-cooler of the Merlin, which originally weighed 78·5 lb., when constructed of copper and brass, and now weighs 37·5 lb. in light alloy—a saving of 41 lb.!

The honeycomb type of oil cooler, having axial or parallel flow and with circular instead of hexagonal tubes, is still used but has been much improved in recent years. One of the principal factors contributing to oil-cooler reliability is the cyclic pressure test—substituting for the steady pressure test one employing fluctuating pressures—to meet the high oil-pressures which may build up in the cooler, particularly under cold weather starting conditions.

Another Rolls-Royce development, to offset drag and further improve performance, was the rearward-facing ejector type of exhaust system, which made use of the exhaust gas energy by suitable proportioning of the exhaust pipe stacks, to give useful thrust. Some tests on a "Spitfire", fitted with a two-stage Merlin 61 type of engine, showed that the ejector system was responsible for an increase of 35 m.p.h. at 26,000 feet. The actual speeds were 375 m.p.h. with the normal exhaust system and 410 m.p.h. with ejectors, or equivalent to an increase of 310 b.h.p. at that altitude and speed. This particular application of the ejector principle is, however, too noisy for civil aircraft.

The author has attempted here to show how the manufacturers of the air- and the liquid-cooled engines have competitively improved the efficiency and reliability of their respective cooling systems and engine installations. The result has been to narrow the argument of their respective merits down to a consideration of aircraft requirements, that is, whether air or liquid cooling best fits the operating needs of a particular aircraft.

Although there appear to be no strictly comparable data available for air-cooled and liquid-cooled engines in the same aircraft, war experience has shown that the liquid-cooled engine can be installed and cooled for somewhat less bulk and for a lower cooling and form drag than an air-cooled engine of similar power. The weight of the two installations is now very similar but slightly favours the liquid-cooled engine for altitudes above 25,000 feet, where the loss in cooling efficiency is, generally, less for this type. Its specific fuel consumption at high cruising power and high altitude will also be superior to that of the air-cooled engine.

The liquid-cooled engine appears to have the greater margin of safety for emergency running at maximum power and overload conditions. But, from the military point of view, it is more vulnerable to battle-damage than the air-cooled engine. There were instances in the 1939–45 war when air-cooled engines had been repeatedly hit by bullets but the pilots were able to fly back to base; whereas any damage to the liquid-cooled engine, which resulted in loss of coolant, caused complete engine failure in a minute or so. Therefore, the air-cooled engine was favoured for ground support and low-flying ground attack and the liquid-cooled engine preferred for air-to-air combat where maximum performance was necessary.

With the piston engine the author would favour the air-cooled engine for civil type of aircraft, where the unit power required does not exceed 3,000 b.h.p. and the operating altitude is below 25,000 feet. For large civil aircraft, flying at very high altitude and speed and requiring a power of more than 3,000 b.h.p. per engine, the liquid-cooled engine is recommended. This would also apply to the long-range heavy bomber. But it is now most unlikely that any new high-duty piston engines will be developed and built; and the present tendency is to leave the future to the gas turbine and concentrate upon the improvement in detail of existing piston engines, to enhance their reliability and ensure a more or less healthy obsolescence during the life of the aircraft in which they are fitted.

THE POPPET VALVE VERSUS THE SLEEVE VALVE

As with air-cooling and liquid-cooling, the relative merits of the poppet valve and sleeve valve have been argued for some years. But only since the 1939–45 war has it been possible to give a more considered opinion on these valves, because no truly comparable engines existed before this time.

Although the Bristol poppet-valve Mercury and Perseus sleeve-valve engines might at first appear comparable, because of their equal cylinder size and total displacement, the author does not consider the Mercury representative of best poppet-valve practice. Therefore, it was not until the Rolls-Royce Eagle was built, during the war, that two comparable engines existed.

The Eagle had the same bore as the Merlin (5·4 inches) and a shorter stroke (5·125 inches), but it ran at about the same piston speed as the latter.

The single-cylinder tests of the Eagle confirmed that, although it could operate at half a ratio of compression higher than the Merlin, on the same fuel, its b.m.e.p. was no greater; that is, the boost response curve of the Eagle cylinder—b.m.e.p. plotted against specific fuel consumption with increasing boost at constant knock—was practically the same as that of the Merlin.

Rolls-Royce consider this as due, possibly, to the higher mechanical losses of the sleeve valve or its lower volumetric efficiency, or a combination of both. The author considers that the higher mechanical or frictional losses are the probable cause.

Apparently, no air consumption tests were done at the time and, therefore, it is difficult to judge whether or not the volumetric efficiency of the Eagle cylinder was lower than that of the Merlin. But the orifice coefficient of the sleeve valve is generally considered inferior to that of the poppet type, although some compensation is possible by providing greater inlet port area in the sleeve. This, however, will apply more to the radial engine with separate cylinders than to one having monobloc cylinder construction. Even with the former there is a limit to the amount of porting which can be cut in the cylinder, on account of its strength to withstand high explosion loads. Further, the sleeve valve is more limited than the poppet valve in regard to the degree of overlap which can be given for optimum operation over a reasonably wide range of engine conditions.

The piston temperature in a sleeve-valve engine is somewhat higher than that of its poppet valve counterpart and, while this higher temperature might not be of real importance in an automobile engine, it has considerable significance in a high-duty aviation engine. If any new sleeve-valve engines were to be produced, the author would recommend oil cooling of the pistons by static oil pipes, attached to the crankcase structure immediate to each cylinder, which directed a solid jet of oil on to the underside of the piston crown. For piston sizes of between five and six inches, the rate of oil flow required would be somewhere between thirty and fifty gallons per hour at maximum power and speed.

The advantage of the sleeve valve is, of course, its immunity from trouble due to the combustion products of leaded fuel.

The development of the sleeve valve has been long and arduous, and it took the Bristol Aeroplane Company ten or twelve years of hard work before a satisfactory sleeve-valve cylinder could be produced and a complete engine built. The principal problem of this type of valve has been its manufacture, and Bristol's success is the result of concentrated effort to evolve a satisfactory manufacturing technique. There was, also, considerable trouble due to sticking of the junk head rings, but the difficulties have now been largely overcome by the introduction of the taper- or wedge-section ring and the copper cooled junk head, with its steel ring carrier.

The taper- or wedge-section piston ring has been one of the most important detail developments in the last fifteen years, and was first used by Napier in their "Rapier" and "Dagger" engines. This ring, which usually has an included angle of taper between ten and fifteen degrees, is very satisfactory in dealing with piston-ring sticking in high-duty engines.

A disadvantage of the sleeve valve, which mainly concerns the in-line engine, is that it makes for a somewhat heavier and longer engine, compared with the poppet-valve arrangement. It was estimated, from a design study by Rolls-Royce during the war, that a poppet-valve version of the Eagle would have saved about 200 lb. in weight and quite a few inches in length. But the geometric layout of the sleeve-valve engine is good as regards installation and accessibility, of sparking plugs, etc.

A further disadvantage of the sleeve valve but, at the same time, one which has not apparently retarded its development, is the higher sparking voltage at the plug. This higher voltage requirement is due to two reasons. The first is the higher compression ratio of the sleeve-valve cylinder, compared with that of its poppet-valve counterpart; and the second, the relatively high degree of directional turbulence, caused by the charge entering the cylinder tangentially, which also helps to reduce the degree of spark advance.

It has recently been demonstrated that the sleeve-valve engine—the Hercules—can equal the best poppet-valve example, in operating time between maintenance periods—this particular engine being now cleared for 1,000 hours' operation in transport aircraft. In fact, there has recently been a resurgence of valve trouble with some of the larger poppet-valve radials, due to the higher cruising powers and leaner mixtures required for the economical operation of certain modern civil aircraft.

For those specifically interested in the automobile engine, the sleeve valve does not appear to offer any advantages for this type. One big disadvantage, which would be almost impossible to overcome, is that of a smoky exhaust at idling and low loads. While this feature is of little importance in the aviation engine, a large number of sleeve-valve-engined automobiles operating in towns would almost constitute a public nuisance!

The author considers that, while the piston engine exists, the sleeve valve is the most suitable for operation in civil aircraft, and where long running periods are required before maintenance becomes necessary. This also applies to long-range bombers of, say, the "B.36" type. The poppet-valve engine would, in these circumstances, have best application in the high-performance fighter and for small transport aircraft where, in the latter case, first cost is of importance. The sleeve valve is more costly than the poppet valve for small production quantities.

These remarks are somewhat academic since the gas turbine will take the place of the piston engine in many civil and military aircraft types and has, in fact, already done so for the fighter.

DEFINITION OF SPECIFIC POWER

The question has arisen of whether the specific power output of an engine should be quoted in terms of b.h.p. per cubic inch of piston displacement or per square inch of piston area. The former value has been generally accepted for many years but, since the late 'thirties, the latter term has crept into use.

The author contends that b.h.p. per cubic inch is the proper measure of the capabilities and state of development of a particular cylinder design.

While it is appreciated that *total* piston area counts in the full-scale engine and, all other things being equal, the greater the area the more power the engine will produce, the power per square inch of piston area appears to mean little as a definition of specific engine performance or as a measure of its state of development.

Two examples which prove the fallacy of quoting specific performance in terms of power per square inch of piston area, are the 12-cylinder Griffon and the 24-cylinder Sabre.

Both these engines have the same displacement—36·7 litres or 2,239 cu. in. The Griffon has a total piston area of 339·3 sq. in. and produces 2,450 b.h.p., whereas the piston area of the Sabre is 471 sq. in. and it develops 3,000 b.h.p. This represents a specific power of 1·09 b.h.p. per cu. in. for the Griffon and 1·34 b.h.p. per cu. in. for the Sabre. But the Griffon produces 7·23 b.h.p. and the Sabre 6·37 b.h.p. per sq. in. of piston area.

The maximum boost of the Griffon for this power is 25 lb. per sq. in., compared with 17·25 lb. per sq. in. for the Sabre; so that, on the face of it, the Sabre pistons would appear to have the easier time, thermally, even when developing 550 b.h.p. more than the Griffon.

It seems from these examples, that it is still necessary to know the b.h.p. per cu. in. in order properly to compare engines and to get the measure of the efficiency of a cylinder in making the best use of a given amount of air.

For a new engine, the designer will generally first consider the cylinder size and then the arrangement of cylinders—radial, in-line, with multi-bank and "V" or "H" variations—to give the maximum practicable piston area for the power required. In a well-established firm, the choice of cylinder size is usually a relatively simple if somewhat conservative procedure; because it is unlikely that the designer will depart from an existing cylinder size, which has probably had many thousands, or even millions, of engine-hours in actual service, unless the total power required makes it impracticable to use the cylinder. Any such departure from a known design, particularly to a cylinder of larger dimensions, would first have to be very seriously considered, since an entirely new cylinder would involve considerably more initial development to ensure piston reliability, etc., at the increased heat flow.

Thus it would appear that a large number of cylinders is advantageous but, as with all things in engineering, there is a compromise and a practical limit to the number of cylinders which can be arranged around a crankshaft, on the grounds of mechanical efficiency, complexity and producibility.

The comparison of the Griffon and Sabre engines is also interesting in showing the different approach by two firms to achieve the same result, that is, high performance with compactness. And, since the criterion by which every aviation engine is judged is its fitness for a particular purpose, it cannot be said that either firm were wrong, in view of the success attending each engine in war service.

It will be appreciated, however, that the Griffon was the natural and logical development of an engine type which had already proved very successful, whereas Napier had not built a successful line of high-duty engines since the Lion and, in consequence, they started with a "clean sheet of paper". The double-crankshaft arrangement did, however, follow as the result of experience with the Dagger, which was a 24-cylinder, air-cooled, poppet-valve, twin-crankshaft engine of 1,000 b.h.p., designed by Halford for Napier.

The ingenuity displayed in the design of the Sabre, to make a compact engine, naturally had to be paid for by some undesirable features. For instance, the cooling jackets were not extended to the equivalent full travel of the piston and, therefore, the pistons and sleeves depended to a considerable degree upon oil cooling. Consequently, the heat rejection to the oil was relatively high and the rate of oil circulation was about 2,400 gal. per hour, or more than twice that of an engine of similar size. Owing to the cramped nature of the engine, with the crankcase "full of machinery", there was little clearance between the moving parts themselves or between them and the main structure. This involved a relatively large power loss due to "oil interference".

The oil and fuel consumptions were both high. With the latter, this was partly due to the somewhat lower mechanical efficiency of the sleeve valve, relative to the poppet valve, and partly to the twin-crankshaft system, where more pistons and bearings gave higher total frictional losses—all of which infers increased installation weight for a given aircraft operating range, and higher cooling drag. The Sabre was, also, a somewhat heavier engine than the Griffon.

Another important consideration, particularly where war production is concerned, is that the manufacturing time for the double-crankshaft sleeve-valve engine is more than twice that for a single-crankshaft poppet-valve engine of similar displacement.

But, for maximum powers of 3,000 b.h.p. and over, the 24-cylinder double-crankshaft engine would undoubtedly have been one of the accepted piston engine types today, had not the gas turbine intruded.

FUEL

Influence on Performance. Fuel, with supercharging, has been mainly responsible for the great improvement in engine performance in the last two decades. An excellent technical review of aviation fuel development, to which those directly interested are referred is given by Schlaifer and Heron (1950).

In the 1914–18 war, when anti-knock value was not fully appreciated or understood, gasolene (petrol) was distilled direct (straight-run) from the crude oil, there being no such thing as synthesis at that time. Its octane number, as now determined, varied considerably, according to the source of the crude oil, but was in the region of 58.

As an illustration of the influence of fuel upon engine performance Heron cites the "Liberty" engine of the 1914–18 war, which developed 400 b.h.p. on about 58 octane fuel, and the Merlin of the 1939–45 war that produced more than 2,000 b.h.p. on a fuel of 150 grade or performance number. Both engines had the same displacement, of 1,649 cu. in., or 27 litres. He estimates that a little less than half this increase can be credited to fuel development.

Aviation fuels are now classified, according to their anti-knock value, in terms of grade numbers. If a single number is given this refers to the weak mixture rating only, and generally applies to 87 octane fuel, where the rich mixture value is not given. All other fuels of 100 octane and over are graded in terms of their weak and rich mixture ratings, that is, W-R, for instance, 100–130 grade.

A convenient and arbitrary formula has been evolved in the United States to convert octane number to performance number and *vice versa* for fuels below and above 100 octane—100 ON being equal to 100 PN:

$$\text{ON} = \frac{128}{1} - \frac{2{,}800}{\text{PN}} \text{ and } \text{PN} = \frac{2{,}800}{128 - \text{ON}}$$

The performance number scale has been based on the average performance of a variety of fuels in a number of full-scale engines and 3C supercharged knock-test engines, related to the performance of clear iso-octane (PN 100). It will be appreciated that the rich mixture rating of a fuel is an approximate indication of the increase in power permitted by richening the mixture at, say, the take-off condition, thus 100–130 grade would indicate an increase of about 30 per cent in engine m.e.p. at rich mixture.

Further examples given by Heron show what these ratings mean in permissible improvement in engine power. He considers four fuels, two of 72 and 100 ON and two consisting of iso-octane plus 1 cu. cm. and plus 6 cu. cm. of lead per (U.S.) gallon. The performance numbers of these four fuels are 50, 100, 125, and 161, respectively. In a supercharged engine, increasing the PN from 50 to 100 permits 100 per cent increase in power, whereas there is only a 30 per cent increase from 125 to 161 PN.

Heron has stated that, in doubling the PN from 50 to 100, the compression ratio can be approximately doubled, at constant manifold pressure. This will increase the power by about 25 per cent and reduce the specific fuel consumption by some 20 per cent.

Fuel of 100–130 grade is in general use today for high-duty piston engines in both military and civil aircraft. But, for those engines requiring to operate at yet leaner cruising mixtures and greater take-off power, 115–145 grade is now available. Fuel of still higher rating can be made, but it is doubtful if present engine requirements will warrant any large-scale production, on account of its cost.

Water-methanol Injection. The mixture generally used is in the proportion of 50:50, which is injected in the blower eye to allow more boost to be given for increasing the maximum or take-off power. The use of water-methanol injection is not, of course, new and its effect has been known for many years—certainly in the form of water injection.

The Americans favoured its use more than the British, during the war, mainly due to the beneficial effect it had in improving the take-off power of their air-cooled radials, and because of the near-critical condition of such large cylinders at maximum boost. British engines, particularly the liquid-cooled Merlin, appeared to be milder than United States engines in respect of their fuel requirements; and it was their relative mildness that allowed them to make full use of temperature-sensitive fuels of good rich-mixture characteristics.

Water-methanol injection permits considerable power increase, and an air-cooled engine giving 2,400 b.h.p. on 100–130 grade fuel will give, with the addition of water-methanol, about, 2,800 b.h.p., with open exhaust, and 2,700 b.h.p. when fitted with an exhaust manifold or as installed in the aircraft.

Distribution. Considerable trouble was experienced during the latter part of the 1939–45 war due to unequal fuel distribution in the engine, when operating at very low manifold temperatures. This occurred when the war was being taken into Germany, and fighters and other aircraft with relatively limited fuel capacity, such as the "Mosquito", had to cruise at very lean mixtures and low engine power in order to "stretch" their range.

The result of this low power operation with leaded fuel in the in-line type of engine, such as the Merlin and the American Allison, was to upset the distribution and cause the heavier and less volatile parts of the fuel to pile up in some of the cylinders. Since the vapour pressure of tetraethyl lead was similar to that of these heavier fuel ends, the lead accompanied them to the same cylinders. There was, therefore, an excess of lead present without a compensating amount of halide or scavenger, ethylene dibromide, which, because of its higher volatility, was fairly evenly distributed throughout the engine cylinders. In consequence, severe plug fouling occurred.

Various expedients to overcome this trouble were tried and, finally, a new lead scavenger was evolved by the Thornton Laboratory of Shell, working in collaboration with the Ministry of Aircraft Production: acetylene tetrabromide. This material, which had a vapour pressure close to that of the lead and lower than that of ethylene dibromide, remained with the lead when operating under these engine conditions, thus more effectively disposing of the combustion products. Most unfortunately, however, it was found that acetylene tetrabromide was unstable in storage and, therefore, it could not be generally used. The work on scavengers is proceeding but it is a lengthy testing process and as yet there are no results to report.

To improve distribution at these engine conditions, Rolls-Royce developed an automatic control which, virtually, converts the after-cooler into an after-heater, when cruising at very low power, to permit a minimum manifold temperature of 40 deg. C. This has given considerable improvement and has largely eliminated sparking-plug fouling.

PART II—THE AVIATION GAS TURBINE

The new era of aviation may be said to have started with the first flight of the "Gloster" E28/39 aircraft fitted with a Whittle jet-propulsion gas turbine, on 15th May 1941.

Actually, the first flight ever made with a jet-propulsion gas-turbine engine was that of the German Heinkel "178" machine, in August 1939; but this had not, apparently, the early impact upon the German aviation industry as had the British effort on the industry in Britain—although it was due largely to the Ministry of Aircraft Production that the pace was forced, since the engine firms were already loaded almost to the limit of their development capacity, to meet war needs.

The aviation gas turbine, because of its characteristics and peculiar limitations, has altered the parameters of aviation; and some years must elapse before a completely clear view is possible in the new and advanced field of aerodynamics and engineering opened up by this prime mover.

It should not be forgotten that the success of the aviation gas turbine, and the swiftness of this engine's practical development and application in the world of flight has been due largely to the specialized engineering technique built up by those who have devoted the best part of their lives to the advancement of the piston type of aviation engine.

There are three distinct types of aviation gas turbine. The first is the jet-propulsion unit or turbo-jet, in which all the air for combustion and propulsion passes through the engine. The second is known as the ducted-fan engine, which is a turbo-jet incorporating a multi-bladed turbine type of fan or propeller enclosed in a duct. This fan can be located in front and ahead of the main engine compressor, driven by gearing, so that it takes a proportion of the air and passes it through a separate duct surrounding the engine proper, the rest going through the engine in the normal manner. Conversely, it can be an integral part of a free turbine located behind the main turbine which drives the compressor. But this latter arrangement makes the engine awkward to install and, unlike the former type, it does not provide the engine compressor with any additional "ram".

The third type is the propeller turbine, in which the ratio of turbine to compressor size is such that sufficient power is available to drive a propeller in addition to that absorbed by the compressor.

In all three types the propulsive efficiency is highest when there is least difference in speed between their respective "discharges" and the speed of the aircraft in which they are fitted. This assumes that efficiency is assessed on a basis of change of kinetic energy, where input per pound of air flow is equal to the jet or discharge kinetic energy less the kinetic energy of forward speed:

$$\left[\frac{V_J^2}{2g}-\frac{V^2}{2g}\right] \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (1)$$

where V_J is jet velocity and V is aircraft velocity, both expressed in feet per second.

The ducted-fan engine falls between the turbo-jet and the propeller turbine, in that it offers improved propulsive efficiency over the turbo-jet at take-off and climb and at all speeds below about 650 m.p.h.

It is impracticable, within the scope of this lecture, to discuss all the possible applications of these three engine types and how one may overlap the other in meeting particular aircraft requirements.

The turbo-jet has already taken the place of the piston engine in fighters and also in light bomber aircraft such as the English Electric "Canberra". It also has application in the passenger transport field, the De Havilland "Comet" being the first practical example.

The propeller turbine has a wide field of application; but its potentialities do not yet appear to be fully appreciated, probably because the turbo-jet has caught the imagination owing to its apparent simplicity of installation and lack of mechanical complexity.

Civil aircraft of medium size, between 50,000 and 60,000 lb. all-up weight, having four propeller turbines totalling 4,800 or 6,000 e.h.p. (equivalent horse-power, that is, shaft power plus jet thrust), can give economical operation in the range of 400 to 600 miles; whereas the operating costs of the equivalent aircraft with turbo-jets would be some 50 per cent higher according to a private communication by Mr. Edwards of Vickers-Armstrongs. For instance, the former would have a fuel-consumption during stand-off at 5,000 feet, before landing, of about 250 gal. per hr., which figure would be increased to 600 gal. per hr. for the turbo-jet installation. At 20,000 feet, the specific mileage of the propeller turbine machine is about 1·2 nautical m.p.g. and that for the turbo-jet 0·4 nautical m.p.g.

The economics of the turbo-jet aircraft become feasible for a machine of not less than 100,000 lb. all-up weight and a stage length (range) of over 1,250 miles, since the reserve fuel carried for stand-off purposes is then a more reasonable proportion of the total fuel capacity.

The second application of the propeller turbine concerns large high-speed civil or bomber type of aircraft, having an operating range of between 3,000 and 5,000 miles in still air, where engines of large unit power—about 10,000 e.h.p. each—would be required.

It was first thought that propeller efficiency would be relatively low at high aircraft speeds, but more recent information suggests that a propeller could now be built having a propulsive efficiency better than 80 per cent at 0·9 Mach number. In fact, it has been suggested that high propulsive efficiencies are possible with propellers designed for aircraft speeds up to Mach No. 1·5.

A propeller to absorb the power of a large gas turbine of, say, 10,000 e.h.p. would be of the multi-blade contra-rotating type, very restricted in diameter, and running at high tip speed Such a propeller would have blades of very thin section.

It is often erroneously supposed that the propeller turbine is the intermediate step in power plants, between the piston engine and the turbo-jet. In fact, it has a place of its own and, if the propeller efficiencies just quoted can be realized in practice, the propeller turbine is seen in quite a different light. An aircraft fitted with propeller turbines should give better all-round flexibility of operation than the turbo-jet machine, with greater latitude or choice of operating height.

There is already, in the United States, a very promising propeller turbine of high power and relatively low fuel-consumption. This is the "Allison" T.40, which produces more than 2 h.p. per lb. of engine weight—an impressive power/weight ratio. The T.40 consists of two separate (T.38) units, each of 2,750 s.h.p. (shaft horse-power, neglecting jet thrust), geared together and driving a contra-rotating propeller through a double-reduction gear ratio of 15·7/1. Incorporated in the gear system is an automatic device to de-clutch an engine in case of its failure, and so allow the other engine to continue driving the contra-rotating propeller, relieved from the "drag" of the defective unit.

The total weight of the T.40 twin unit is reported to be only 2,618 lb., which includes the gearbox and the short extension shafts between the power units and the box.

The single power unit of this engine, the T.38, has a 17-stage-axial flow compressor of steel construction, driven by a four-stage turbine, having a pressure ratio in excess of 6/1. There are eight combustion chambers.

The twin T.40 engine is now installed in the "Consolidated" XP5Y-1 flying boat for the United States Navy, and it is also fitted in prototype Douglas and North American attack bombers. In the "Consolidated" boat, a new type of Aero-products six-bladed contra-rotating propeller is used. An experimental installation, with two (single) T.38 units, has also been made in a "Convair" airliner.

The specific fuel consumption of the engine has not yet been disclosed, but is stated to be equivalent to that of a piston engine of comparable power. On this basis, and because the gas turbine cruises at a high proportion of its maximum speed and power, compared with the piston engine which cruises at some 50 per cent of its maximum, the specific fuel consumption should be less than 0·6 lb. per h.p. per hr. !

Details of another American propeller turbine have just been released. This engine, the Pratt and Whitney T-34, produces about 5,700 h.p. in a single unit for a (dry) weight of about 2,550 lb., representing a specific weight of less than 0·45 lb. per h.p. It is understood that approximately 90 per cent of the power goes to the propeller and the remainder to jet thrust.

The specific fuel consumption is quoted as 0·62 lb. per h.p.-hr.; which is presumably at or near maximum power, static sea-level conditions.

The respective fuel consumptions of both the Allison and Pratt and Whitney engines are remarkable, particularly when compared with that of a piston engine at equivalent full power output. Since the mixture strength of the latter has to be increased at take-off, a specific fuel consumption of 0·85 lb. per b.h.p.-hr. and greater is quite usual.

The Pratt and Whitney T-34 engine is also interesting because it is mainly of steel construction, and this does not appear to have imposed any weight penalty.

The T-34 has a thirteen-stage, axial-flow compressor and a three-stage turbine. Apparently, an annular-type combustion chamber is used. The engine is 155 inches in length and has a basic diameter of 30 inches.

AXIAL OR CENTRIFUGAL COMPRESSOR

Future development of the turbo-jet and the large propeller turbine will favour the axial compressor, because the never-ceasing efforts further to improve aircraft performance, particularly in the military field, demand still greater reduction in the bulk of all aircraft components; and the axial compressor affords the best opportunity for obtaining high power from minimum dimensions, since it can handle a much larger mass air flow for a given diameter than can the centrifugal compressor.

It was first thought that the axial type of turbo-jet engine would always be the heavier, but this has not proved so in practice, and there are already axial engines in existence of lower weight than centrifugal engines of similar thrust.

The axial compressor is considerably more expensive to manufacture than is the centrifugal but, to view this in the correct perspective, the production cost of each should be specifically related to engine thrust. The cost of the axial, which is due to its many blades, will not remain at the present high differential, particularly when more is known of its aerodynamics, and the designer has all the information necessary to design a compressor which will readily meet requirements without excessive "trial and error" testing.

The difficulty, so far, has been to design an efficient axial compressor that will give, say, 6/1 pressure ratio in eleven or twelve stages, with flexibility of operation and without surging. It is relatively easy to design and build a compressor which will meet performance estimates, if the number of stages is unlimited and if no restrictions are imposed on length or weight. But, since the whole secret of building a successful aviation engine lies in the ability to reduce bulk and weight, the designer will naturally attempt to get the desired performance and maximum work from the minimum number of stages.

Therefore, until further aerodynamic knowledge is available, "trial and error" tactics must prevail and these naturally take time. To reduce the time factor, which is principally due to the calculations involved in designing new blade forms and manufacturing fresh blades, it is necessary to have in the design office a number of high-level aerodynamicists with good mathematical training and also, possibly, an electronic calculating machine, plus a very efficient pre-production department in the works.

COMPRESSOR EFFICIENCY

At present, the adiabatic efficiency of the axial is between 83 and 85 per cent, and it should be possible to obtain 87 per cent efficiency in the course of a few years. The efficiency of a well-designed centrifugal compressor is about 77 or 78 per cent, which is also capable of further improvement by two or three per cent. But the high pressure-ratios possible with the axial compressor are very important in relation to overall operating economy.

It is true that the axial compressor is more sensitive and susceptible to dirt and to damage by ice, etc., than the centrifugal, but this will not influence its general acceptance where high economy and performance are concerned.

The centrifugal compressor turbine will, however, have a further obsolescence of some years in military aircraft of fairly high subsonic performance and will, probably, continue to be used in those machines designed for short-range low-level operations such as ground attack. It will also see reasonable length of service in civil aircraft of the "Comet" type, and in the smaller-propeller turbines such as the Rolls-Royce "Dart".

FUTURE DEVELOPMENTS

The specific thrust of centrifugal turbo-jet engines now produced is between 315 and 460 lb. per sq. ft. of frontal area, and that of an axial engine is about 700 lb. per sq. ft. When a satisfactory annular type of combustion chamber has been developed, it should be possible to build an axial engine of 10,000 lb. thrust for an overall diameter of about 40 inches, or a thrust of 1,150 lb. per sq. ft.

The largest centrifugal type of turbo-jet so far built is the Rolls-Royce "Tay", of static thrust 6,250 lb. and diameter about 50 inches.

To boost the power of turbo-jet engines, it is now usual to employ after-burning or reheat. This comprises a set of fuel burners placed in the jet pipe, some distance aft of the turbine. The burners are controlled by the pilot, to give additional power for take-off or for combat. After-burning, which can be likened to an assisted ram jet, makes use of the excess or unburnt air that has passed through the engine, to increase the total thrust. Naturally, burning fuel in this way is very expensive and the fuel consumption per pound of additional thrust rises considerably, which limits the period of after-burning to only a minute or two.

The increase in thrust by after-burning is about 25–30 per cent at sea-level static conditions, which increase is nearly doubled at high altitude and speed.

Fuel Economy. At the present time (1950), the specific fuel consumption of turbo-jet engines in service is in excess of 1 lb. per lb. thrust per hr. at static sea-level conditions. This has, until recently, applied to both the centrifugal and axial types of engine, but there are axial engines now running which give a specific consumption below 0·9 lb. per lb. thrust per hr. and it should be possible to reduce this to 0·85 lb. within the next five years. The latter figure can be obtained with a compressor adiabatic efficiency of 85 per cent, a pressure ratio of 6·7/1, and a turbine efficiency of not less than 87 per cent.

Whereas the fuel consumption figures quoted are on the basis of full static thrust at sea level, at normal temperature, it is also necessary for the aircraft designer to know the engine performance at the operating altitude of the aircraft.

To achieve a specific fuel consumption of 0·8 lb. per lb. thrust per hr., and less, will probably necessitate a compounded engine, that is, one having two compressors in series on separate (co-axial) shafts, driven by separate turbines, to obtain the desired pressure ratio which, in this case, will be between 8/1 and 10/1. The alternative is, of course, the ducted-fan engine but, whereas this type will give high thrust and improved propulsive efficiency at take-off and climb conditions, its best performance is limited to aircraft speeds of about 650 m.p.h.

Relighting at Altitude. One of the difficulties experienced with current turbo-jet engines is the relighting of a stopped engine at altitude. At present, the effective relighting altitude varies from about 10,000 to 20,000 feet, according to the particular engine. However, experimental relighting has now been achieved at an altitude in excess of 40,000 feet, and, therefore, it is not considered that the present difficulties are fundamental.

Another drawback, after getting a relight at altitude, is that of accelerating the engine to full speed from its starting speed, which is usually less than 2,000 r.p.m. This is due to the difficulty in getting the flame to propagate from one combustion chamber to another.

The annular combustion chamber should reduce or eliminate the difficulties of altitude relighting and also ensure good flame propagation. The high pressure-ratio compressor will also help in altitude restarting.

Engine Weight. The question of engine weight is always important, and the specific weight of the turbo-jet should not exceed 0·35 lb. per lb. thrust. This might be reduced to 0·3 lb. per lb. thrust in about five years.

The specific weight is usually based on the "dry" weight of the engine and includes the accessories necessary for its functioning, but not those driven by the engine for the aircraft services.

An aviation engine must always be designed for low weight in the first place, and careful checks made during the design stages to ensure that the prototype engine will not exceed its designed weight. It is rarely possible appreciably to reduce the weight of a heavy engine after it has been built; but a light engine can be strengthened considerably without too much penalty in added weight. The practical way of dealing with weight in a heavy engine is to increase its power and, therefore, improve the power/weight ratio. This is not, however, always desirable in the gas turbine, since the engine will have been designed in the first place to fit a particular aircraft and the greater power may be embarrassing to fuel consumption and range, particularly if there are also Mach number limitations. Any extra engine power can then only be made available to improve take-off and climb, with the risk that the engine has to run at below its economical speed in level flight.

The Propeller Turbine. The propeller turbine can be designed to give any desired proportion of propeller thrust relative to jet thrust, and the ratio between the two will vary according to the operating requirements of the aircraft. For instance, in the Allison T.40 unit, the combined output is, apparently, divided to give 5,100 s.h.p. at the propeller, with 1,600 lb. of thrust at the engine exhaust or jet pipe.

For powers of 6,000 e.h.p. (s.h.p. plus jet thrust) and higher, the duplex engine arrangement is preferable, consisting of two medium size engines coupled through gearing to the same propeller, since the scantlings and stresses in the engine components can be kept to a reasonable figure. A single engine of higher unit power would probably have a high specific weight, due to the disproportionate increase in scantling size. The specific weight of a 6,000 e.h.p. propeller turbine should not exceed 0·50 lb. per e.h.p.

It is possible that within seven years, a propeller turbine of this power may be built, to give a specific fuel consumption of about 0·55 lb. per e.h.p. per hr. Such an engine would probably be of the compounded type, with two compressors in series on separate (co-axial) shafts, having an overall pressure ratio of 12/1.

Gas Turbine Fuel. The supply of aviation fuel, both in regard to quality and quantity, is largely controlled by the strategic military position.

At present, British and United States military aircraft use kerosene; but all gas turbine engines intended for military use are also tested on aviation gasolene—usually 100–130 grade leaded fuel—in case they have to use it in an emergency and because there are still many piston-engined aircraft in the Services of both countries. The United States Navy fly their turbo-jet machines on aviation gasolene, because of the fuel problem in aircraft carriers and the difficulty of dual-fuel storage.

Since gasolene represents the major requirement from the barrel of crude oil, the supply of any other refined product of the crude oil will be controlled in a direct relation to the world demand for gasolene.

In the catastrophic event of another world war within, say, the next five years, the aviation gas turbine would be committed to a fuel having a predominance of hydrocarbons in the gasolene boiling range, to ensure adequate supplies. This would result in a loss of operating range of anything up to ten per cent, for a given aircraft tankage, since a gasolene type of fuel would reduce the total volumetric heat value. It might also cause operating difficulty and further loss due to its relatively high vapour pressure. It should, however, be quite possible, by a change in refinery processing, to produce a fuel of 1–2 lb. Reid vapour pressure, suitable for military use and available in quantity within five years.

In regard to turbine fuel for civil aviation purposes, the gasolene type of fuel of high vapour pressure should be avoided at all costs. If, however, it is possible to produce a military fuel of not more than 2 lb. Reid vapour pressure, this should also be acceptable for civil needs.

For the first time in the history of aviation, an engine has been produced—the gas turbine—which does not need a fuel of high vapour pressure, and in the interests of safety in passenger carrying aircraft a fuel of low volatility is most desirable.

For the next five years, there should be sufficient kerosene to meet the relatively small demands of turbine-engined civil aircraft; but one of the principal difficulties of the kerosene type of fuel is to ensure a sufficiently low freezing point, which must not be higher than −40 deg. C. This is particularly important when it is realized that most civil aircraft fitted with gas turbines will fly at considerable altitude, between 35,000 and 40,000 feet. Military aircraft will be required to operate at still greater altitudes, and will require fuel having a freezing point of 20–30 deg. lower than −40 deg. C.

Engine Maintenance. Even during its relatively short existence the gas turbine has shown that it requires less day-to-day maintenance than the piston engine in normal aircraft operation, although it has not yet achieved the same total operating time between overhauls as the latter. But there is no fundamental reason why the gas turbine cannot eventually achieve the same running time as a well-developed piston engine which, for the latter type in airline service, is now 1,000 hours.

Present turbo-jets in military use, according to their state of development, are giving between 100 and 350 hours of service before removal for maintenance. Combustion chamber inspection is still necessary at intervals of 50 or 100 hours, but further experience with this type of engine should render such intermediate inspections unnecessary.

Since performance is the criterion of the military engine it is unnecessary for this type to go beyond 500 hours' operation between overhauls, but it should be completely reliable during this period. In peace-time, military aircraft are slow to accumulate flying hours and, for one reason or another, the airframe will eventually require maintenance before this time. Therefore, there is no particular advantage in having a very long-lived engine, since it will probably be removed in any case when the aircraft is withdrawn for maintenance. It is then more easy to put in a new or serviced engine, so that engine and airframe maintenance do not get out of step. In war-time an engine is unlikely to see out its full running time, and will probably only average 100 hours' flying before it is removed on account of battle damage.

INFLUENCE OF AVIATION ENGINES ON AUTOMOBILE ENGINES

Since aviation and automobile engines operate under different conditions, and have diverged considerably with time and experience, the contribution of the aviation engine to the improvement of the automobile engine has been mainly a matter of detail—but important detail. These contributions include improved coolant flow in the cylinder head, pressure cooling, better bearing technique with new bearing materials, the development of heat and corrosion-resistant valve steels and valve-facing materials, sodium-filled valves and greatly improved sparking plugs.

The principal difference between the aviation engine and that of the automobile is that the former operates at constant speed and load, whereas the latter works at widely varying power and speed and, generally, at relatively low power.

The specific power output (b.h.p. per cu. in.) of the aviation engine, at a cruising power equivalent to 50 per cent of the maximum or take-off power, is about the same as that of the average automobile engine at full power. Under these conditions, the aviation engine in airline service is now expected to operate for 800 or 1,000 hours before it is removed for overhaul. Assuming a block-to-block speed of 200 m.p.h., this represents a total mileage between overhaul periods of between 160,000 and 200,000 miles.

The automobile industry gave the liquid-cooled aviation engine the secondary-surface radiator in place of the obsolete honeycomb type.

Design and Development. The development of a successful aviation engine is a long and expensive undertaking, and occupies between four and five years from the first line on the drawing board before an entirely new engine is ready for production. The gas turbine demands more time spent in calculation, before and during the actual design stages, than is necessary for the piston engine, but there is generally less time on the test-bed. Therefore, the total time to make the gas turbine ready for production will be very similar to that needed for the piston engine, with a possible reduction, later, as the result of more experience. For the simple Whittle type of turbo-jet, with a centrifugal compressor, the development period, of about four and a half years, can be nearly halved.

The gas turbine has brought a change in the design and engineering departments. In addition to the staff needed for proper mechanical design and testing, there must now be included some high-level mathematicians and aerodynamicists.

While such lengthy and expensive development is not required for a new automobile or vehicle engine, many manufacturers could with benefit enlarge their engineering departments—some of which exist only in name, and consist of a small design office of limited capacity, largely subordinated to the whims of management and sales.

An engineering department will not only include facilities for engine study but also those for investigating chassis problems, etc. This is a particularly important consideration, since the life or obsolescence of the average automobile is between ten and fifteen years, and it is most necessary to ensure that every advantage is taken of any technical advance to improve the type in detail and, also, to ensure that the design office has all the information possible to enable the designers to make the best of a new design.

For instance, an engine designed today should have inherent in it the possibility of raising the compression ratio, without a major change or distortion of the combustion chamber shape, when better fuel becomes available. Also included in the basic design will be increased scantlings of the main cylinder block and crankcase structure, to accommodate a stiffer crankshaft and ensure a smooth engine, free from vibration when running with high compression ratio.

CONCLUSION

The first half-century of aviation is now practically completed and, as the gas turbine becomes established, the second half of aviation's century may bring big changes in the world by the increasing rate of air transport, further to reduce space in terms of time.

The piston engine, which has served aviation well, should continue to do so for about another ten years, but will by that time have assumed second place. Its future will then be controlled by the obsolescence of the aircraft in which it is fitted and, also, by the gas turbine's suitability or otherwise for such machines as feeder line and executive aircraft, where first cost and operating flexibility are of chief importance.

In Appendix I are some comments on the application of the gas turbine to road vehicles.

It has not been possible to give a full discussion of aviation engine development in all its aspects, and those wishing for more detail are referred to an excellent treatise on the subject by Schlaifer and Heron (1950).

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APPENDIX I

THE GAS TURBINE FOR ROAD VEHICLES

The gas turbine is not an impossibility for the automobile; experimental engines are being designed and some are already under test, but the turbine does not appear to have advantages compared with the normal piston engine, for this purpose. It may well have useful application in certain large commercial and military vehicle types and, perhaps, for the long-distance passenger coach.

In the first place, the most suitable size of gas turbine to give reasonable efficiency is better fitted to the needs of the large vehicle rather than the automobile—since it is more easy to build an efficient gas turbine of 250 s.h.p. than one of 50 or 100 s.h.p. Scaling the engine down to these comparatively low powers demands lengthy and expensive development, to obtain the required efficiency of components such as the compressor, the combustion chamber, and the turbine.

Smoothness of operation and lack of vibration, inherent in the gas turbine, are now so good in the piston engine that a change to the former on these grounds alone could hardly be justified.

The idea that the gas turbine will use "any old fuel" may not be confirmed in practice, in the relatively low-power-vehicle engine. While anti-knock or detonation value is unimportant, the fuel must be fluid at all temperatures likely to be encountered in cold climates; and it must also give clean combustion, so that frequent cleaning or decarbonizing of the combustion chamber is not required. This would seem to rule out the so-called boiler or heavy fuels, because of their relatively high freezing point and their ash and asphaltic contents, etc. Light Diesel or gas oil, or kerosene, are, therefore, likely to be used, but these may be in short supply if any large number of turbine-engined vehicles emerge on the highways.

To compete with the piston type of petrol engine, the thermal efficiency of the gas turbine cannot be much less than 25 per cent. Although the efficiency of the former is relatively low at part-throttle and with the constant accelerations and decelerations of the automobile in ordinary usage, that of the gas turbine is likely to be considerably less. The gas turbine is a constant-speed or "full-throttle" engine and, therefore, its efficiency falls off seriously at engine speeds much below the maximum. The provision of a heat exchanger may, however, help in this respect.

The gas turbine will also have to meet the competition of the very high-compression piston engine, the development of which has now been brought to a practical stage (General Motors Research of America (Kettering 1947; Campbell, Caris, and Withrow 1948)).

An advantage of the gas turbine for automobile propulsion is that it should be possible to dispense with the multi-speed gear-box; since the usual arrangement, of a free turbine separated mechanically from the turbine-compressor system of the engine, will provide its own very satisfactory torque converter. A low-speed gear, for very steep gradients, and a reverse gear, will probably be the only mechanical speed changes required. But a multi-speed box may still be necessary for the heavy road vehicle.

The vehicle gas turbine will have a centrifugal compressor, since the axial type is somewhat impracticable and expensive to produce. Therefore, the pressure ratio of a small two-stage centrifugal compressor is hardly likely to be much more than 5/1 and it will do well to give 74 per cent adiabatic efficiency. The matter of turbine efficiency is the most important for fuel economy, since 1 per cent improvement in turbine efficiency will be equivalent to about 3 per cent improvement in fuel consumption. Since the efficiency of the aviation turbine is about 87 per cent, a similar efficiency will be difficult to achieve with such a comparatively small turbine. In view of the low pressure-ratio of the compressor and the probable limitations in component efficiencies, a heat exchanger will be required.

Since the gas turbine will pump anything between six and ten times more air than the piston engine, for equal power output, the problems of silencing the air intake and exhaust disposal are not inconsiderable although, perhaps, not insuperable.

If the gas turbine eventually challenges the piston engine in the automobile, as distinct from its use in the heavy vehicle, this may come about purely as the result of an urge to be different, and not because it shows any better economy than the piston engine. Peculiar as it may seem, engineers often follow fashion for fashion's sake; but it is to be hoped that any such change to the turbine will be made only on the grounds of sheer merit, before discarding the many decades of experience of the piston engine.

Finally, if the gas turbine were to be used in the automobile, it would have to be dealt with as a "packaged power unit", because the average garage or motor engineering concern would be unable, for some time, to handle any of the major or even minor maintenance problems of this type of engine. This would involve a considerable extension of the supplier's manufacturing and servicing facilities, to ensure a cushion of power units to meet such contingencies.

APPENDIX II

REFERENCES

CAMPBELL, J. M., CARIS, D. F., and WITHROW, L. L. 1948 Society of Automotive Engineers Meeting, San Francisco.

LOVESEY, A. C. 1946 *Aircraft Engineering*, vol. 18, p. 218, "Rolls-Royce Merlin Development".

KETTERING, C. F. 1947 Trans. Soc. Auto. Eng. America, vol. 55, p. 669.

SCHLAIFER, R., and HERON, S. D. 1950 Harvard Business School, "The Development of Aircraft Engines and Aviation Fuels".