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Gas Turbine Technology Evolution: A Designer's Perspective

Bernard L. Koff
TurboVision, Inc., Palm Beach Gardens, Florida 33418

Many of the design and manufacturing process innovations responsible for the remarkable evolution of the gas-turbine engine, which has been credited with making the world much smaller, as well as with being a great benefit to humankind are presented. Many of the technologies presented show, from a design engineer's perspective, the levels of insight and foresight put forth by government agencies, the engineering community, and their leadership. The information presented offers a detailed understanding of technical issues that are not well known even within the industry. Universities can use this paper to bring the art of design into the classroom for applications of thermodynamics, materials, and structural design.

Introduction

DURING the past 60 years the aircraft gas turbine has evolved into the world's most complex product which has made an astoundingly positive impact on mankind. Jet powered aircraft have provided the United States with unprecedented air power supremacy for defense and global reach to help promote worldwide peace and aid. Large turbofan powered transport and commercial aircraft have spanned the Globe, making the world much smaller while clean burning gas turbines are used worldwide for power generation. Lessons learned and design innovations developed for gas turbines have also been transitioned to rocket engines including the oxygen and hydrogen pumps for the space shuttle main engines. This presentation highlights key technologies created and developed by engineers and which have been responsible for the extraordinary evolution of state-of-art advances in gas turbine propulsion.

In the Beginning

The United States got a late start in the development of the gas-turbine engine because responsible leaders did not believe that the gas generator cycle consisting of a compressor, combustor, and turbine was practical. Their reasoning was that after the power was extracted from the turbine to drive the compressor, there would not be enough residual energy in the exhaust gas for useful work. This reasoning was partially dispelled in the United States after the first flight of the Gloster aircraft in 1941 powered by the Whittle jet engine. Whittle was a visionary genius with outstanding design engineering skills and remarkable determination. His 1930 engine patent (Fig. 1) shows a compressor with two axial stages, followed by a centrifugal stage, an axial cannular combustor with fuel nozzles, and a two stage axial turbine.[1]

It would be an understatement to say that Whittle had great difficulty in getting support to pursue his revolutionary invention. However, after persisting with great courage and personal sacrifice, he was able to test the kerosene-fueled Whittle unit (W.U.) successfully on 12 April 1937, the world's first jet engine. Whittle selected kerosene, also used today as jet fuel, as the fuel of choice and conducted extensive combustion testing to develop a fuel system that could be controlled for his experimental engine.

Four years later, in 1941, General Hap Arnold was briefed on the progress of the Whittle engine and the Gloster E.28/39 aircraft shortly before the first flight in 15 May 1941, and work began to obtain production rights for the United States. Later that year, the W.1.X engine and drawings for an improved W.2.B engine were delivered to the General Electric Company in Lynn, Massachusetts.[2] The improved W2 engine model (Fig. 2) incorporated a double-sided centrifugal compressor, an axial reversed-flow cannular combustor, and a single-stage axial turbine. It produced a thrust of 1560 lb. The exhaust gas energy or specific power reached approximately 50 hp for each pound per second of airflow.

The General Electric Company (GE) was selected because they had already designed and manufactured turbosuperchargers for reciprocating engines. Frank Whittle was also sent to the United States in 1942 to help the GE engineers to design, improve, and build his latest W.2 jet engine. This effort resulted in the production of the GE I-A, the first U.S. jet engine that was an improved version of the Whittle W.2.B. Two I-A engines (Fig. 3) powered the Bell XP-59A aircraft to the United State's first jet flight in late 1942.[3]

Unknown to the Allies, Hans von Ohain, a brilliant engineering student, with continuing support from Ernst Heinkel the aircraft manufacturer, had completed testing of the hydrogen-fueled He S 1 turbojet engine in March 1937. Hans explained that he used

Bernard L. Koff is a pioneer whose leadership in the gas-turbine industry produced a host of innovative breakthroughs in design and development. With General Electric Company (GE) and Pratt & Whitney (P&W) from which he retired as Executive Vice President of Engineering and Technology, his contributions impacted the design and development of over one-half of all jet engines flying. His patents and highly regarded technical papers cover the entire spectrum of jet engine design and manufacturing technology. The score of honors and awards he has received are among the highest that his industry can bestow and include the ASME/AIAA/SAE Daniel Guggenheim Medal, Air Force Association Theodore von Kármán Award, AIAA Reed Aeronautics Award, AIAA Air Breathing Propulsion Award, AIAA Engineer of the Year, AIAA and SAE Littlewood Lecture Award, ASME Tom Sawyer Award, SAE Franklin Kolk Award, the GE Perry Egbert Award, and the P&W George Mead Medal. He is also Fellow and Honorary Member of the ASME, Fellow of both the AIAA and SAE, and member of the National Academy of Engineering. Mr. Koff graduated from Clarkson University with a B.S. in mechanical engineering (1951) and earned an M.S. in the same field from New York University (1958). Clarkson University granted him an Honorary Doctor of Science Degree, having also recognized him with the Golden Knight Award as a Most Distinguished Alumnus. E-mail: benlkoff@aol.com.

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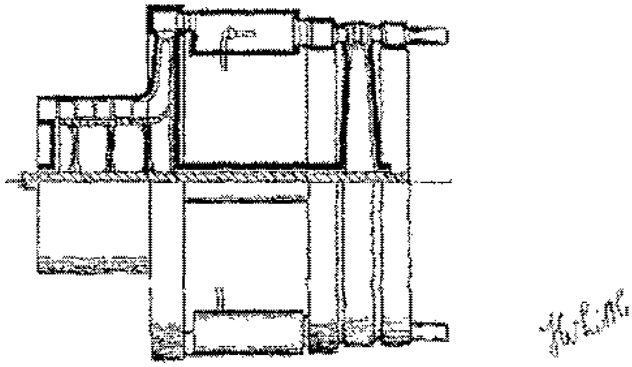

Fig. 1 Whittle Gas-Turbine Drawing, British Patent 347206, courtesy of Rolls Royce plc.

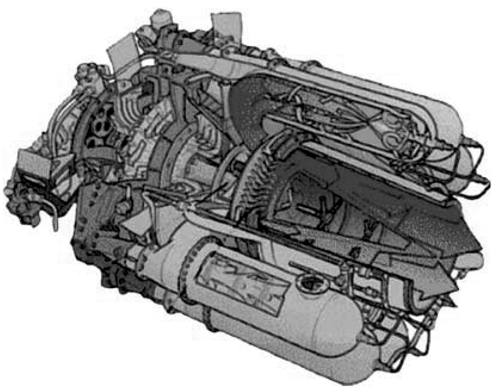

Fig. 2 Whittle W.2 Engine.[1]

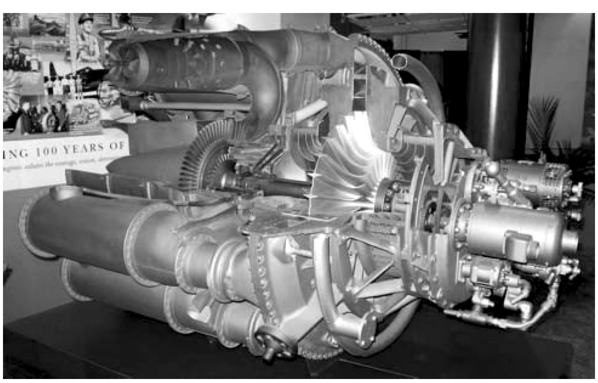

Fig. 3 3 GE/ Whittle I-A Turbojet, first U.S. engine, courtesy of GE.

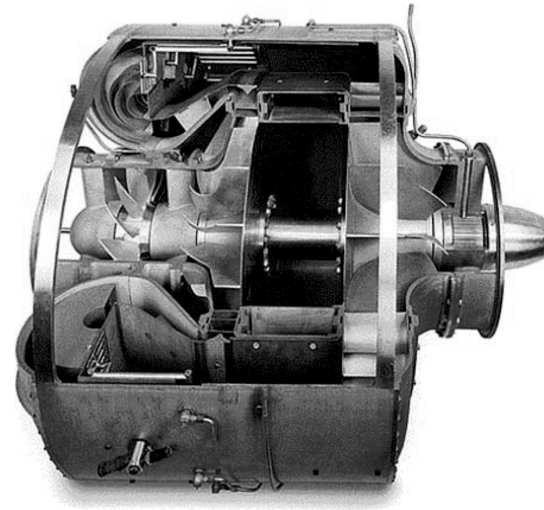

Fig. 4 Hans von Ohain's jet engine, He S 3B (Ref. 2).

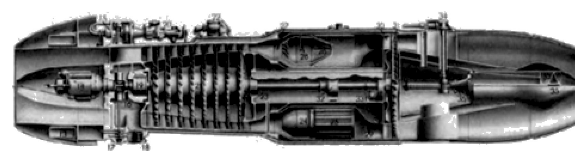

Fig. 5 Junkers Jumo 004 Turbojet.[3]

Fig. 6 Speakers von Ohain, Koff, and Whittle, 50th anniversary of first jet-engine flight (Dayton Engineer's Club 1989).

hydrogen, which he considered impractical for flight, only as an interim step to start and operate the engine because Heinkel was very impatient, and a petroleum-fueled combustion system had yet to be developed. The He S 3B petroleum-fueled jet engine was tested successfully in March 1938, shortly after the test of Whittle's engine. In von Ohain's own words, "Heinkel was crazy for speed and gave us everything." This focused effort resulted in the historic first jet-powered flight in 27 August 1939, on the eve of World War II and almost two years ahead of the British. The von Ohain engine (Fig. 4) had an axial flow inducer ahead of the centrifugal impeller stage, a reverse-flow annular combustor, and a radial inflow turbine. The exhaust gas energy was approximately 50 hp for each pound per second of inlet airflow, similar to Whittle's engine.[4]

During World War II, the Junkers Jumo engine powering the famous Messerschmitt Me 262 jet aircraft was developed in a competition with Heinkel. The Jumo engines (Fig. 5) for this aircraft were mounted in nacelles, rather than internal to the aircraft fuselage, by the use of an axial flow compressor, axial flow turbine, and a straight through-flow cannular combustor to reduce frontal area and increase performance. This early configuration became a forerunner of how future jet engines would be configured relative to overall design arrangement.[5]

The inventor–designers Whittle and von Ohain talked about past achievements at the symposium commemorating von Ohain's 50th anniversary of the first jet flight (Fig. 6). They suggested having this author included who spoke on the future of jet propulsion.

Advancing the Technology

Materials

The identification of key technologies responsible for the evolution of the gas-turbine over the past 50 years appropriately begins with the achievements of the materials and manufacturing process engineers. The chronological progress (Fig. 7) of turbine airfoil material capability over the past 50 years shows an improvement exceeding 500°F (468°C). This achievement was the result of many innovative scientific breakthroughs in materials research, processing, and manufacturing technology.

The early jet engines were severely limited by hot section materials, which motivated the research and invention of improved alloys. The concept of the development of superalloys was derived in the 1940's, but the air melting process produced low ductility with the

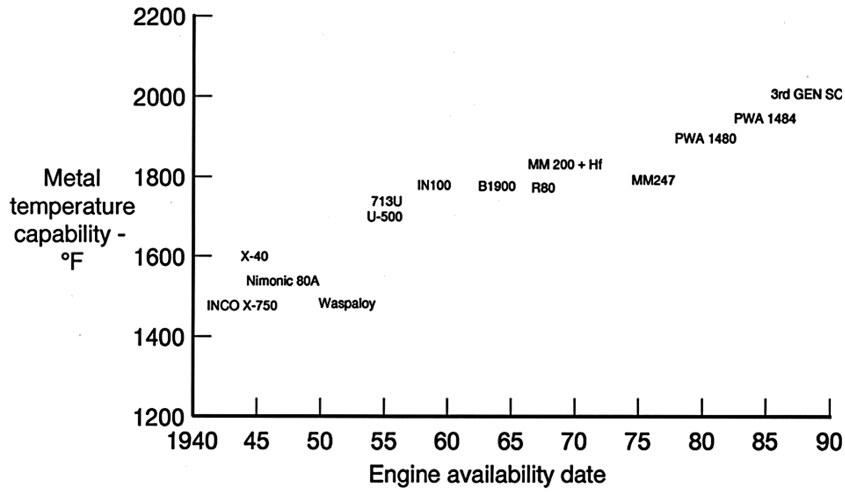

Fig. 7 Turbine airfoil materials progress [°C = (°F–32)/1.8].

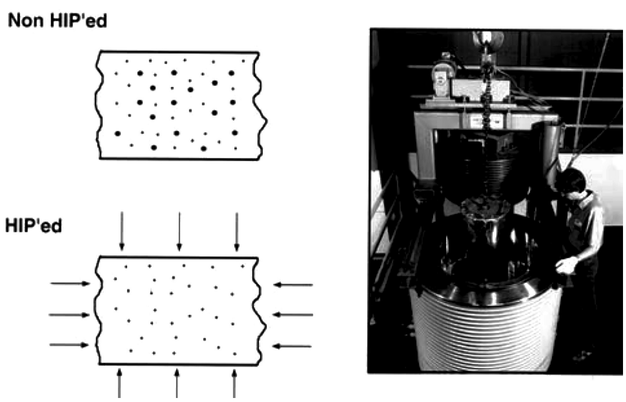

Fig. 8 HIP process.

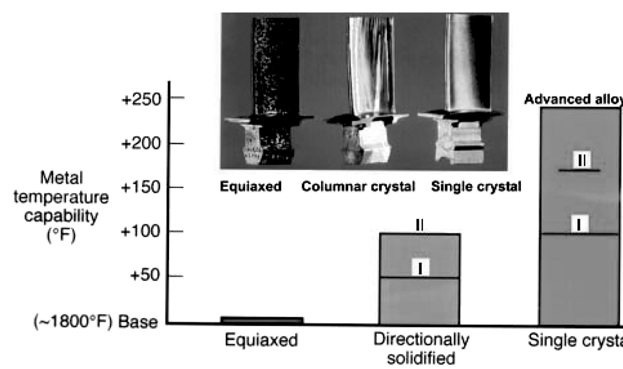

Fig. 9 Turbine airfoil material evolution [°C = (°F–32)/1.8].

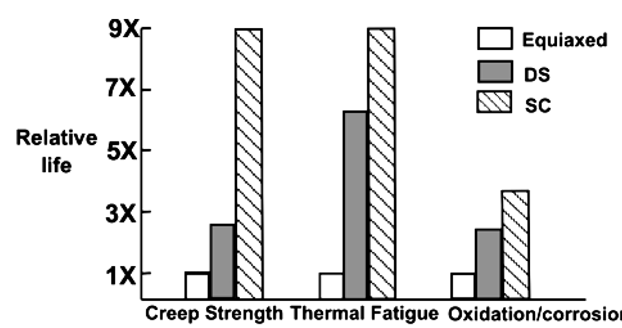

Fig. 10 Improved life using DS and SC turbine superalloys.

addition of the key strengthening elements such as aluminum and titanium. The breakthrough came in 1953 when vacuum induction melting was developed, a process heralded as having made the jet engine what is it today. This innovative process boosted alloy capability by 200°F (93.3°C) for turbine airfoils in the 1955 time period. The vacuum arc remelting process followed in 1958 to produce large forgings for disks. These innovative manufacturing processes launched the development of today's generation for disks, shafts, bolts, and structures.

An interesting observation can be made that if an alloy can be used to produce a bolt requiring an upset forged head for strength, it can usually be used for most other engine components such as compressor airfoils, disks, casings, and frames.

The combustor hot streak gas entering the turbine made it necessary to air-cool the stationary vanes on the early engines, and X-40 castings were a standard for some 40 years. Because the rotor blades pass through the hot streaks, they experience a lower average temperature. Also, because the rotor blades are moving, they are subjected to a lower relative temperature. This allowed the use of forged alloy blades with both higher fatigue strength and mechanical properties for more than 25 years. The introduction of air-cooled blades required complex internal passages more readily provided by lower-strength castings but with higher temperature capability. Dampers under the platforms of the cast blades also became standard features to suppress vibratory response. Hot isostatic pressing (HIP) (Fig. 8) was introduced in the early 1970s by a creative materials processing engineer to reduce porosity and increase both ductility and fatigue strength for castings. The parts are placed in an autoclave and subjected to heat and pressure, which effectively removes the internal porosity by compression. The significant reduction in material flaws reduces internal stress concentrations, resulting in improved ductility. This process is responsible for reducing the failure rate of cast parts subjected to high local stresses, while remaining cost effective.

The introduction of directional solidification (DS) and single-crystal (SC) superalloys (Fig. 9) produced a breakthrough providing a 200°F (93.3°C) increase in metal temperature capability over conventional multigrain equiaxed cast materials.

Equiaxed castings have many grain boundaries surrounding the crystals of the superalloy, forming failure initiation points in fatigue, creep, and oxidation. The DS castings arrange the crystals in the form of radial stalks, eliminating the weaker grain boundaries in the tensile direction, providing improved resistance to creep, thermal fatigue, and oxidation/hot corrosion. The SC casting process goes one step further by completely eliminating all weaker grain boundaries and providing further improvements in resistance to creep, fatigue, and oxidation (Fig. 10). This highly innovative process, originally invented by a visionary materials research engineer whose ideas were not initially accepted, has made it possible to cast a complete turbine airfoil, dovetail, and platform in an SC superalloy.

An additional benefit of the DS and SC alloys is that they can be tailored in the casting process to exhibit a lower Young's modulus

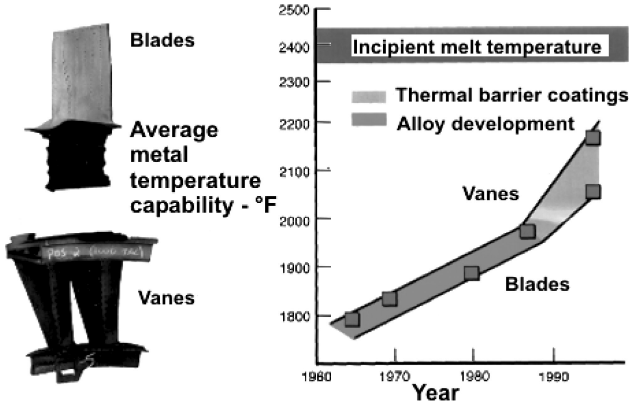

Fig. 11 Thermal barrier coatings [°C = (°F-32)/1.8].

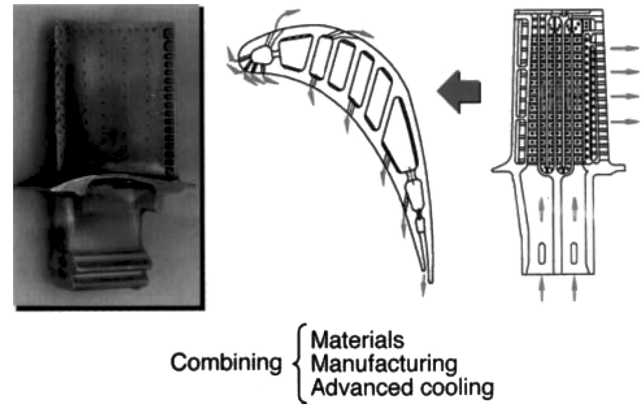

Fig. 12 F100-PW-220 SC turbine blade with film and convection serpentine cooling (courtesy of Pratt & Whitney).

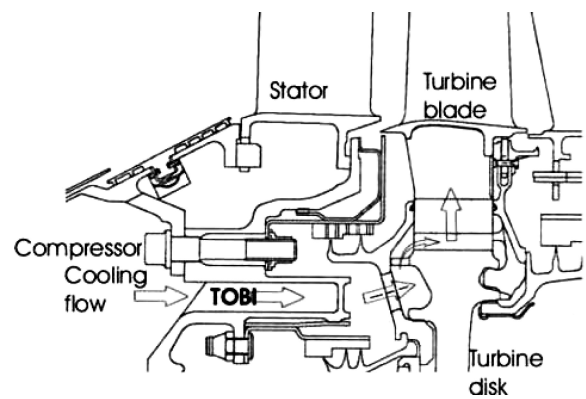

Fig. 13 F100-PW-229 turbine cooling the blade cooling air (courtesy of P&W).

directionally, resulting in a lower stress for the same strain range. This feature allows the designer to pass the lowest temperature cooling air up through the internal blade passage at the hot airfoil leading edge. Higher airfoil cooling efficiency can then be achieved without encountering thermal fatigue cracks experienced in equiaxed cast blades.

With the realization that nickel-based superalloys encounter incipient melting at 2400°F (1316°C), work began to develop thermal barrier coatings for hot section airfoils to prevent oxidation. While superalloy development proceeded, aluminide coatings were first applied in the mid-1970s to meet the demand for increased hot section life. Ceramic thermal barrier coatings (Fig. 11) were applied in the mid-1980s after reaching within 400°F (204°C) of incipient melting (shown in red) with the best DS and SC alloys.

Thermal barrier coatings are also applied to the blade outer air seals, which are subjected to higher combustor temperatures than the moving blades, which are not subjected to the hot streak stagnation temperatures.

Turbine Airfoil Cooling

Today, the jet engine turbine blade is the world's most sophisticated heat exchanger. Until the mid-1960s, there were three basic schools of thought for the design of the first-stage high-pressure turbine blade: uncooled, convection cooled, and film cooled. Many advocated that putting cooling holes in the highly stressed turbine blades would lead to failures. Others demonstrated in the late 1950s that convection cooled blades using radial holes drilled into the core of the airfoil with a shaped tube electrolytic machining (STEM) process would not compromise fatigue strength. Material removal via the STEM process did not leave a brittle recast layer subject to cracking.

The film-only group maintained that a film of cool air should be used as a barrier between the hot gas and metal. Turbine blades were designed and manufactured in the late 1950s using a forged radial strut and dovetail with a brazed on porous sheath forming the airfoil. The concept of the porous sheath airfoil was to discharge air on the airfoil surface to achieve transpiration cooling while protecting the load carrying strut. This concept was not successful because of backflow when the hot gas flowed in and mixed with the airfoil internal cooling air.

It is interesting that for many years, both cast and fabricated turbine stator vanes successfully used film holes on the airfoil leading edge to cool the combustion hot streaks. Why the film-cooled vane leading-edge technology was not adapted to blades has been a contentious issue. A likely reason for not considering leading-edge film holes to cool the rotating blades was the fear of encountering high cycle fatigue failures.

A breakthrough was made in the early 1970s when the U.S. Air Force funded industry to develop a turbine blade using both convection and film cooling. The engineers worked closely with casting suppliers to produce a one-piece casting of a convection/film cooled blade, as well as methods for producing both round and shaped film cooling holes.

Finally, in the early 1980s, the design, materials, and processing came together to produce a one-piece convection/film cooled blade that used SC materials (Fig. 12).

Internal cored passages form the compartments where compressor cooling air flows in a five-pass serpentine and a single-passage flowing air to a cavity adjacent to the leading edge. The internal passages have cast-in trip strips to promote turbulent flow and increase the convection heat transfer coefficient. Film cooling is provided by strategically discharging air on the airfoil concave pressure surface. The film cooling holes at the leading edge are densely spaced to provide both convection and film cooling where the hot gas heat transfer rate is highest. The suction (convex) side of the airfoil has shaped cooling holes to help keep the film attached to the surface. All suction surface cooling holes are also upstream of the airfoil passage throat to minimize mixing losses. The trailing edge has pin-fin pressure side discharge cooling to minimize thickness and reduce the wake loss. A milestone was achieved in the early 1980s when the blade shown successfully passed an accelerated 4000 Tactical Air Command (TAC) cyclic endurance test involving rapid hot starts and throttle retards.

In the 1960s, the tangential onboard injector (TOBI) concept (Fig. 13) was invented by an engineer at Pratt & Whitney (P&W) to lower the temperature of the compressor discharge cooling air before it entered the first-stage turbine blade. The concept developed into an annulus with turning vanes to accelerate the airflow from axial to tangential rotor speed, decreasing the temperature while minimizing the pressure loss entering the rotor. The pressure drop across the combustor and first stage turbine vane accommodates the TOBI pressure drop and also allows the discharge pressure to be set higher than the turbine flowpath to avoid backflow into the airfoil.

The TOBI decreases the temperature of the compressor cooling air by 125°F (52°C) for the two-stage turbine shown. The TOBI also decreases the turbine pump work required in getting the air up to rotor speed before entering the blade. Engines with cooled turbine blades and interstage vane cavities have used the TOBI concept for the past 25 years to reduce the temperature of the cooling air, increase efficiency, and improve durability.

The forward and aft turbine blade retainers shown at the rim also seal the cooling air and are attached without bolts through the disk rim slots. These boltless blade retainers increase life by reducing rim stress concentrations. They were invented by an innovative GE design draftsman in the late 1970s and are in wide use today.

The turbine vanes must accommodate combustor hot streaks, depending on the pattern factor, which can be 400°F (204°C) higher than the average gas temperature. Because the vanes are stationary, they are subjected to the total gas stagnation temperature. The turbine vanes (Fig. 14) used in the 4000 TAC cycle accelerated endurance test have extensive internal convection and external film cooling to meet the higher gas temperatures. These SC film/convection cooled vanes set a milestone in durability for high-temperature gas-turbine engines and are used worldwide.

Typically, the stationary turbine vanes for aircraft engines require approximately 10% of the inlet compressor flow for cooling with combustor discharge temperatures in the range of 2800–3200°F (1538–1760°C). At the average relative gas temperature, the rotating turbine blades typically use 4% of the compressor flow for cooling. Although the turbine blades operate at lower gas temperatures than the vanes, the metal temperatures must be reduced to account for centrifugal and vibratory stresses.

A chronological evolution of higher turbine rotor inlet temperature (RIT) capability as a function of the cooling effectiveness using SC materials is shown in Fig. 15.

The cooling effectiveness is represented by the ratio of the airfoil heat load to cooling flow, a measure of how well the airfoil is cooled between the hot gas and cooling air temperatures. The RIT base for the solid uncooled blade is 1800°F (982°C), representing the mid-1950s technology. Note that without cooling, there is only a 50°F (10°C) improvement in RIT in going from the first to the latest generation of SC material. Progressing to convection cooled blades with a cooling effectiveness of 0.4 allows a 400°F (204°C) increase in RIT. The payoff for increased material temperature capability is amplified as the cooling effectiveness level increases. The SC film/convection cooled blade family with an effectiveness of 0.6+ has an RIT capability to 3000°F (1649°C). This operating temperature is 1200°F (649°C) above the solid uncooled blades and 600°F (1316°C) above the 2400°F (1316°C) incipient melting temperature of nickel-based superalloys.

This spectacular progress resulted from a dedicated team effort that combined SC material manufacturing processes, innovative casting suppliers, creative designers, and government support.

Compressor Design and Engine Configuration

The compressor has often been referred to as the heart of the engine. Air must be pumped to discharge pressure at high efficiency without encountering failures or stall instability induced by inlet distortion, Reynolds number effects, engine transients, acceleration back pressure, and control tolerances. The normal compressor operating line and stall line (a compression limit) is shown as a function of pressure ratio and airflow in Fig. 16.

Lessons learned have repeatedly demonstrated that it is essential for the engine compressor to accommodate the factors shown in the stability audit that have the potential for causing a flow breakdown.

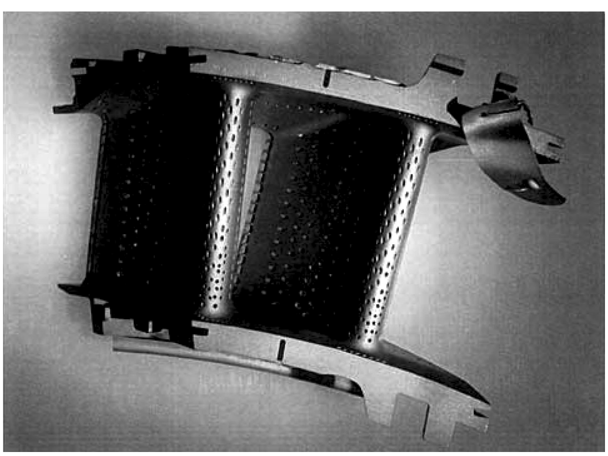

Fig. 14 F100-PW-220 SC turbine vanes using film and convection cooling (courtesy of P&W).

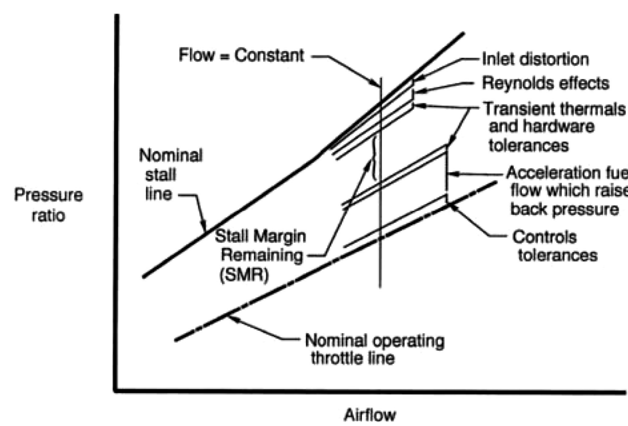

Fig. 16 Compression system stability audit.

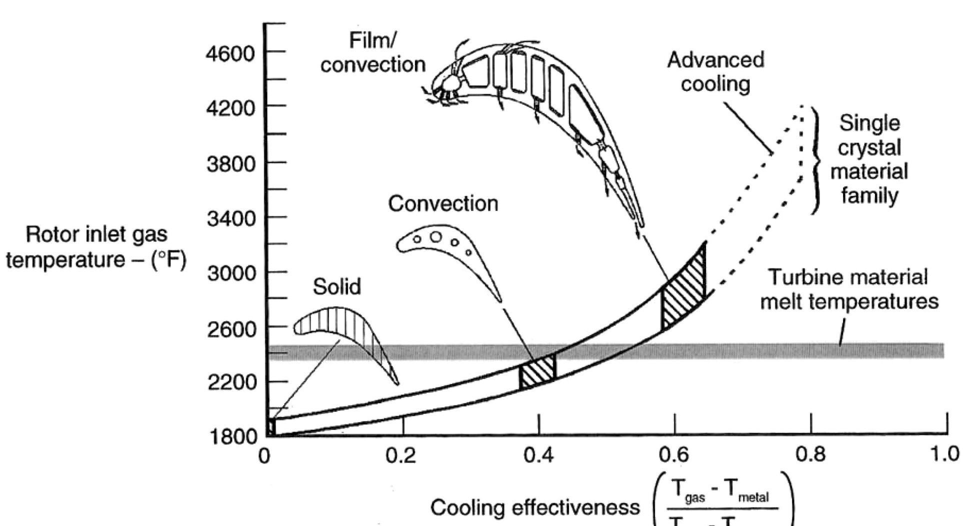

Fig. 15 Turbine blade cooling technology: inlet gas temperature vs effectiveness [°C = (°F-32)/1.8].

Inlet distortion caused by flow separation and Reynolds effects due to the lower air density at altitude decreases the stall line. Control tolerances and acceleration fuel flow that increases combustion backpressure both raise the operating line. Transient thermals, deterioration, and hardware tolerances drop the stall line and raise the operating line. The key is to have enough compressor stall margin remaining for safe engine operation. Higher rotor tip speeds, low aspect ratio airfoils, and axial inlet velocity to wheel speed ratios (C_x/U) in the range of 0.4–0.5 have substantially raised stall margin.

Tip treatment such as grooved slots in the casing shroud have also been used to improve stall margin. When the grooves are circumferential, a 3–5% increase in stall margin has been demonstrated under distorted inlet flow without efficiency loss. When the grooves are axial and also skewed in the circumferential direction, stall margin is further increased, but with an accompanying loss in efficiency. In 1987, a P&W engineer developed an highly innovative casing treatment concept that increased fan stall margin by a staggering 20% under distorted inlet flow and with little loss in efficiency. This advanced casing treatment is being used successfully in aircraft applications with high inlet flow distortion.

The early engines were all single-rotor turbojets with fixed geometry compressors. In the late 1940s, GE developed the J47 turbojet (Fig. 17) with a pressure ratio of 5 in 12 stages driven by a single-stage turbine. A Curvic coupling compressor rotor was used to reduce engine vibration resulting from shifting parts experienced on the earlier J35. The turbine rotor disk had a Timkin 1625 alloy rim tungsten inert gas (TIG) welded to an AMS 4340 hub to provide higher strength at the flowpath. Heating of the aluminum inlet guide vane struts with compressor bleed was later developed and added to prevent ice buildup. In 1948, a J47 turbojet powered the F-86A fighter to a new world's speed record of 671 mph.

By the early 1950s, two schools of thought had developed. In a major step forward, with the goal to leapfrog the industry, P&W successfully developed the J57 (Fig. 18), an axial flow dual spool turbojet with a nine-stage low-pressure (LP) compressor, and a seven-stage high-pressure (HP) compressor driven by a single-stage HP and two-stage LP turbine. Thrust was 10,500 lb (4764 kg), with an initial pressure ratio of 11 and a thrust to weight of 2.7.

In 1953, a North American YF-100A powered by a J57 became the first fighter aircraft to reach supersonic speed in level flight.

Meanwhile, GE developed the single-rotor J79 turbojet (Fig. 19), with a variable-geometry 17-stage compressor at a pressure ratio

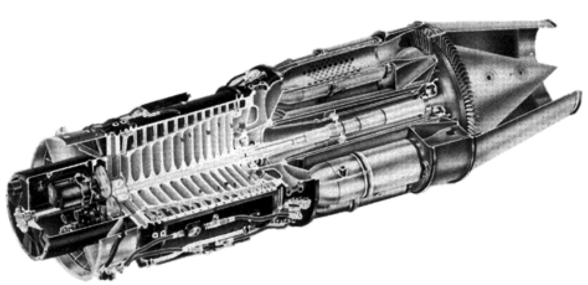

Fig. 17 GE J47 single-rotor turbojet engine (courtesy of GE).

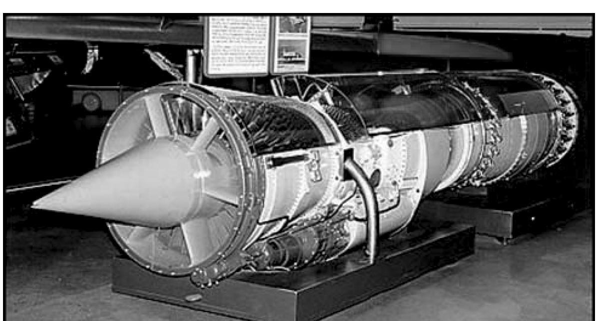

Fig. 18 P&W J57 dual spool turbojet.[4]

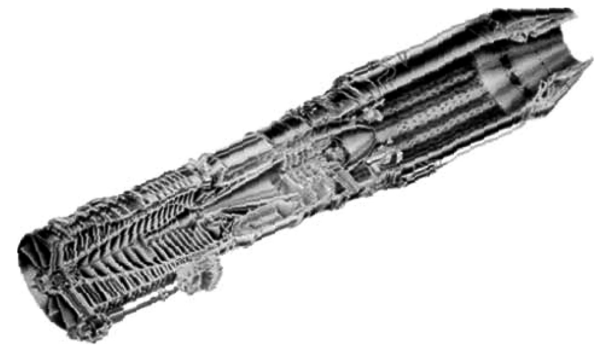

Fig. 19 GE J79 turbojet with afterburner.[5]

of 13.3 and driven by a three-stage turbine.[5] The first-stage turbine nozzle diaphragm was air cooled, and the three rotor stages used forged uncooled blades. The compressor rotor incorporated smooth spool rim drive spacers between the disks attached with dowel bolted flanges for excellent alignment and balance retention. This was the first production application for dowel bolted flanges that used precision machined bolts assembled into disks that had accurate jig bored holes at the rim. This design was developed by an engineer after the initial compressor rotors with rabbet or piloted flanges encountered severe vibration problems resulting from shifting parts. The dowel bolted rotor could resist radial shifting between the spacers and disks, whereas the rabbeted rotors offered resistance to radial shifting only in one direction.

Behind the first two shrouded stators, the vanes were cantilever supported to match with the smooth spool rotor. Generous radial clearances between the vane tips and rotor spool were required to avoid heavy rubs that could melt holes in the spacers.

A hydromechanical control scheduled the variable inlet guide vanes and front six stators to match front and rear stages during acceleration and engine operational transients.

The thin, light conical shaft connecting the compressor and turbine rotors used an innovative flow turning process invented by a materials processing engineer.

P&W argued that having two compressors on separate shafts with bleed for matching stages would operate closer to their optimum corrected speeds, producing higher pressure ratios and operational flexibility. GE insisted that the J79 single rotor turbojet with variable stators was less complex, with fewer parts, and lower manufacturing cost. For a considerable time, both companies were polarized in their views. Eventually and with compelling reasons, engine configurations using dual spool, bleed matching variable stators and cast air-cooled blades were adopted by both companies.

For subsonic flight, the exhaust velocity of the turbojet engine is significantly higher than the aircraft flight speed. The wasted jet energy reduced propulsion efficiency and increased fuel consumption:

$$\text{Propulsive efficiency} = \frac{2}{1 + \text{jet velocity}/\text{aircraft velocity}}$$

Rolls-Royce (RR) is credited with developing the concept of first using fan stages to bypass air around the core engine to reduce the exhaust jet velocity for improved subsonic performance. The initial reaction of P&W and GE was negative and seemingly confirmed when the Rolls-Royce Conway turbofan engine did not outperform the turbojet. It has been speculated that the fan had to have low efficiency to produce such a result.

In an effort to enter the commercial aircraft engine market, GE added an aft fan module to the J79 turbojet, and this became the first U.S. turbofan engine. The GE aft fan rotor consisted of a turbine blade supporting a tip-mounted fan blade separated by a transition platform and seals. This configuration change was straightforward and converted the J79 turbojet to a fan engine by adding a separate module at the engine exhaust. However, the temperature gradient across the transition shroud between the tip of the turbine blade and root of the fan blade was 400°F (204°C), requiring considerable development to eliminate low cycle fatigue in the shroud junction.

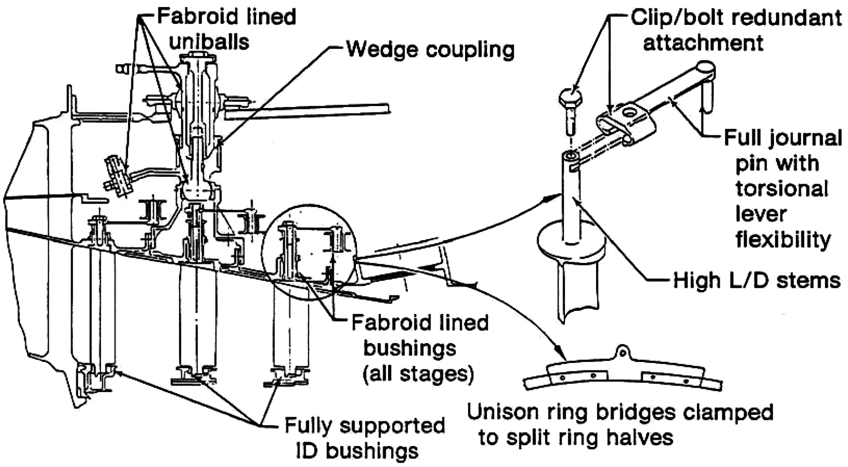

Fig. 20 F100-PW-220 core compressor variable vane actuation system.[6]

P&W countered the GE aft fan with the successful JT3D/TF33 front fan engine and succeeded in capturing the market on the B52, B707, DC-8, and many other aircraft. It soon became evident that P&W's front fan configuration was superior because it supercharged the core compressor and also produced lower nacelle drag.

The mechanical design of the multistage axial compressors proved to be a formidable problem for the engineers from the beginning for the following reasons:

1) The airfoils have relatively thin edges for performance and are subject to damage.

2) High aspect ratio airfoils have been easier to design for high efficiency, but have lower stall margin and durability.

3) First-stage blades can encounter flutter vibration at low corrected speed (high Mach) caused by high angle of attack.

4) Blade dovetails must be stronger than the airfoils, and disk dovetails even stronger.

5) Rotor disk and shaft assemblies must not change balance after assembly.

6) Close operating clearances are required for high efficiency and stall margin.

7) The variable vanes must track accurately to prevent performance loss, stall, and blade fatigue failures.[6]

It took many years to develop a lightweight, accurate and durable compressor variable vane actuation system, in combination with the engine control, to match airflow and physical rotor speed as a function of corrected speed and inlet temperature (Fig. 20).

The early F100-PW-100/200 first line fighter engines experienced a number of durability problems, including the variable vane actuation system, which had wear and hysteresis problems in service. The engine was upgraded to the -220 model, and the vane actuation system was part of the redesign. Increased length/diameter ratio of the vane stems spread the load to reduce the forces in the couple resisting the vane bending moment. The combination of lower stem forces with Fabroid-lined vane bushings and uniballs resulted in a considerable increase in durability and life. Uniballs in the vane levers were replaced by a journal pin to eliminate wear in the connection to the half rings driven by the actuator. Longer vane levers were used to provide the torsional flexibility between the rigid vane stem connection and the journal pin mounted in the half ring. Although this design requires additional actuation force to overcome the torsional spring resistance of the lever arms, it is considered a good tradeoff for light aerodynamically loaded stages due to the improved durability and life. However, with higher aerodynamic stage loading, higher vane lever stiffness is required to absorb the higher forces. The attachment of the vane arm to the stem is redundant, in that the vane arm stays attached in the event the clamping bolt is broken or missing. This avoids having a disconnected vane rotate off schedule. An off-schedule variable vane usually results in a downstream blade airfoil failure from a one-per-revolution excitation.

Supersonic aircraft engines began encountering first-stage blade flutter (self-excited vibration) at higher flight Mach numbers. Airfoil torsional flutter can occur when the compressor operates at low corrected speed due to the high ram inlet temperature. In the late 1950s, the GE YJ93 turbojet engine encountered first-stage blade failures at Mach 2.2 and used loose fitting pins from blade to blade to damp the flutter vibration. The engineers realized that another solution was needed for the preliminary flight rating test (PFRT). Increased chord to add beam stiffness was rejected because of the excessive weight increase. The break came when a GE design engineer saw a scrapped P&W JT3D fan blade with a midspan shroud at a vendor site. J93 blades were then designed and manufactured by the use of the P&W midspan shroud idea to pass PFRT. The JT3D midspan shroud blade design concept is still used worldwide for moderate aspect ratio fan and first-stage compressor blades (Fig. 21).

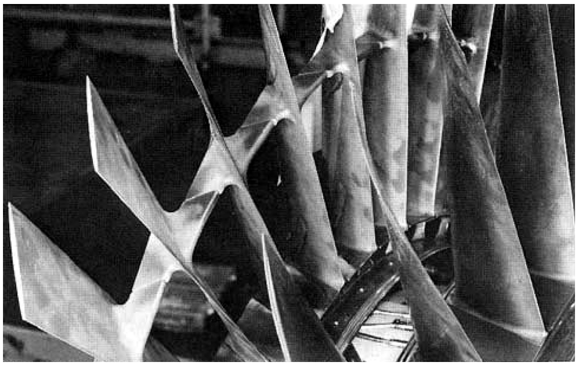

Fig. 21 GE YJ93 first-stage compressor blade with midspan shroud (1960).

The shroud is formed by "angel wing" extensions integrally forged with the blade that butt together midway in the flowpath, providing stiffness against high incidence angle flow and flutter. The contact areas are coated with tungsten carbide to resist fretting and wear. The original midspan shrouds had flat-plate cross sections. A streamlined airfoil cross section for the midspan shroud was later adopted to reduce shroud drag and efficiency loss. When the first J93 shrouded compressor blades were tested at Mach 3 inlet conditions, the hydromechanical control allowed the inlet guide vanes to drift 12 deg off schedule, causing high incidence flow. The shrouded blades were instrumented with strain gauges, and vibration levels reached 80% of the stress limits without damage to the blades. This was a very convincing durability demonstration for the part span shrouded blades.

Newer turbofan engines incorporate low aspect ratio and low radius ratio fan blades without shrouds to improve efficiency, stall margin, and the ratio of flow per unit annulus area. The largest of the commercial turbofan engines use both hollow diffusion bonded titanium and composite blades to reduce weight with the low aspect ratio airfoils.

Over the years, aircraft engine rotor configuration varied with company experience and included the following: 1) Curvic coupling teeth machined into integral disk spacers with tie bolts to clamp the assembly; 2) Spacers bolted to disks using rabbets (to pilot) or close-fitting precision dowel bolts for axial clamping and radial positioning; 3) forged titanium drum with internal rings to carry the blades and bolted to disks on the forward and aft flanges for radial rigidity; 4) TIG, plasma arc, and electron beam welding to attach disk spacers; and 5) inertia welding to attach disk and spacers (developed at GE in 1968).

In the early 1960s a GE engineer invented a forged titanium drum rotor using internal rings to support the blades. Because the rotor was a smooth spool design like the J79 and YJ93 compressors, the blades used circumferential dovetails, so that assembly could be made after the rotor drum was machined. This novel concept was used successfully in the GE TF39, CF6-6, CF6-50, and CF6-80 engines, including derivatives for power generation and marine applications (Fig. 22).

The 16-stage TF39 compressor was a half scale aerodynamic copy of the 11-stage YJ93 with five additional stages to bring the pressure ratio up to 16.8, a milestone for aero engines in the mid 1960s. During this period, development efforts began to increase the aerodynamic loading of compression systems to reduce the number of stages and axial length. Shortly after, design and manufacturing engineers also began working on developing a welded rotor construction.

Over a 15-year period, the superiority of inertia welding for compressor and turbine rotors was established and implemented, reducing structural complexity and the number of parts. A cross section comparison of the 1955 GE J79 and the 1970 GE F101 compressor rotors (Fig. 23) shows how the design engineers capitalized on being able to weld both titanium and IN718 alloys.[7]

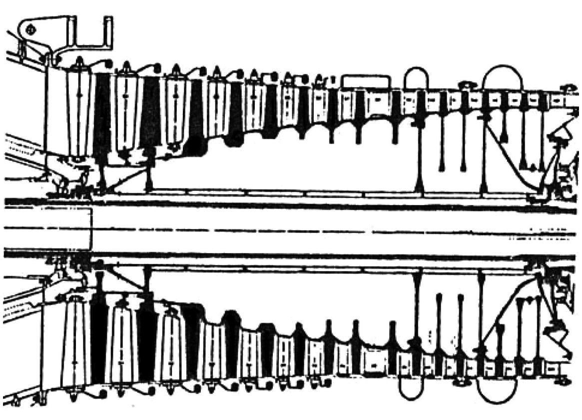

Fig. 22 GE TF39 ring drum compressor rotor design using, titanium and IN718 alloys.

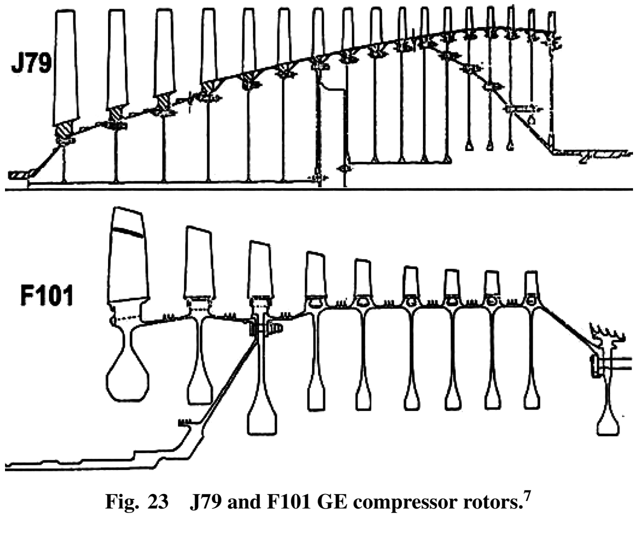

Fig. 23 J79 and F101 GE compressor rotors.[7]

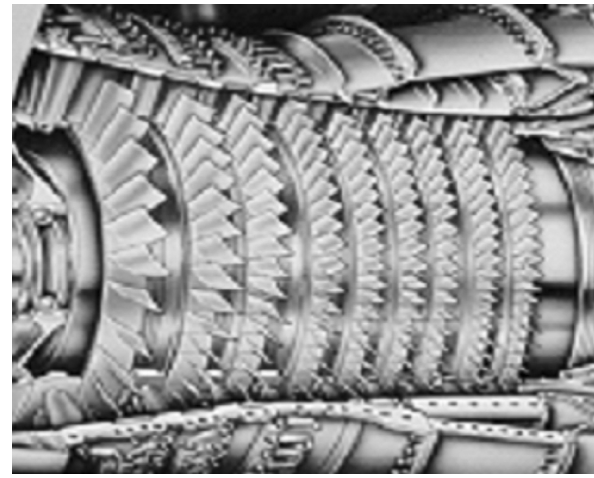

Fig. 24 GE F101 inertia welded compressor rotor in titanium and Inconel 718 superalloy.

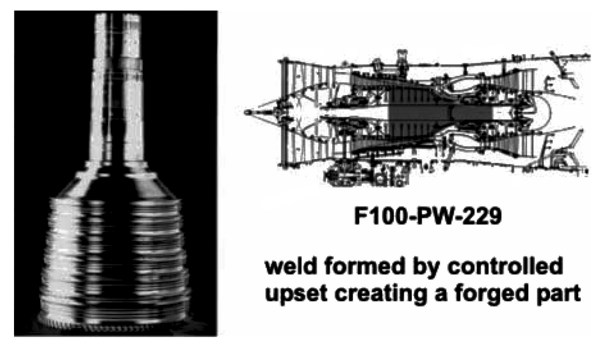

Fig. 25 F100-229 inertia welded compressor rotor spool and shaft in Inconel 718.

The 17-stage J79 compressor rotor has 39 structural parts and 22 flange joints bolted together at the disk rims on the forward 11 stages, with dual bolted flanges and redundant load paths on the rear stages. The blades have axial dovetails, and aft of stage four, are retained by the smooth spool spacers forming the flowpath. Blade removal and replacement requires a complete disassembly of the rotor.

The inertia welded 12-stage F101 (and the CFM 56) compressor rotors have only five structural parts and two bolted flange joints. The first three stages have axial dovetails, whereas stages 4–12 have circumferential dovetails, and all of the blades can be removed without disassembly of the rotor spool. The nine-stage F101 compressor has a pressure ratio of 12, which is only 10% less than the J79, with almost twice the number of stages.

Today, inertia welded rotors are being used by many major engine manufacturers (Figs. 24 and 25) to provide maximum beam rigidity in bending and shear for balance retention while ensuring defect-free welds and low maintenance.

Inertia welding is a forging process that can eliminate defects and achieve strength and ductility within the weld joint that is higher than the parent metal. Energy is stored in a flywheel where a rotor part is mounted, rotated, and then moved into contact with a stationary part. Forging of the rotor takes place as the flywheel energy is dissipated. When the parameters are set properly, there is no melting and resolidification to produce defects such as voids and microcracks.

Metal upset or weld flash on both spacer surfaces should be machined and shot peened for surface enhancement. Except where the titanium stages are attached to the higher-temperature nickel alloys, welded rotor spools eliminate bolt holes and stress concentrations in the disk web and rim.

Aircraft core engine compressors (without low spool supercharging) have increased in pressure ratio from 2 to 20, while efficiencies increased from 78 to 90%. Typical fighter engine compressors (Fig. 26) have pressure ratios of 8, whereas some large commercial engines, such as the GE90, have compressor pressure ratios of over 20.

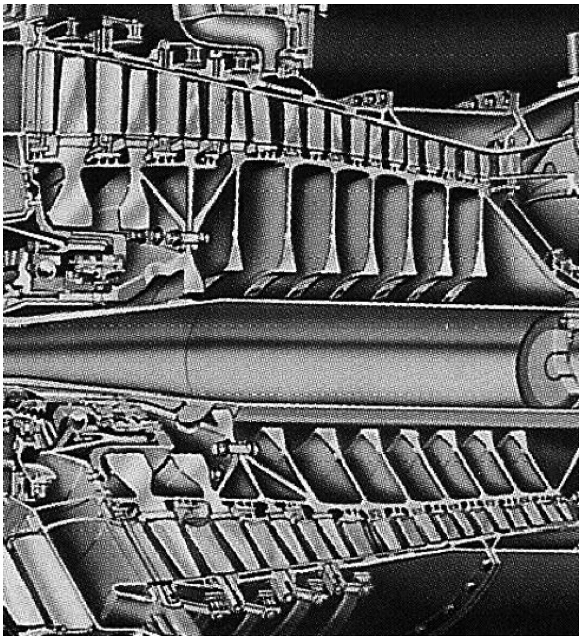

Fig. 26 F100-PW-229 core engine compressor inertia welded rotor in titanium and IN718.

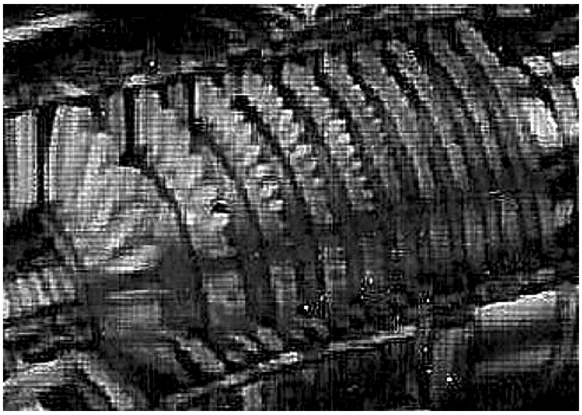

Fig. 27 GE90 core engine compressor (courtesy of GE).

The PW F00-229 core compressor rotor is inertia welded with only one bolted joint between the forward two-stage titanium spool and a seven-stage IN718 spool. The variable inlet guide vanes are followed by three variable vane stages. The internal rotor drum is vented and cooled by third-stage air to improve the thermal match with the outer casing for improved clearance control. This concept, first implemented at GE in the mid 1960s, is now used worldwide to reduce turbomachinery radial clearances for improved performance and durability.

The GE90 commercial core engine compressor has the world's highest pressure ratio at more than 2.5 times the military engine for the same number of stages. This is a tribute to the engineers who worked for many years to increase the average stage pressure rise without incurring a serious efficiency penalty (Fig. 27).

As pressure ratios increased, radial clearances for blades and vanes became increasingly important. Because metal to metal rubs can induce failure, both abradable and abrasive coatings were developed and put into service during the past 40 years. The abradable coatings were rough, causing a performance loss, and also spalled, leaving craters in the blade tip shrouds that reduced stall margin. The abrasive coatings loaded up with metal debris during a rub causing a metal to metal rub with local overheating.

In the mid-1980s, cubic boron nitride (CBN) blade tip coatings for compressors and turbines were developed to prevent blade tip wear during light rubbing. The application of CBN to the blade tips

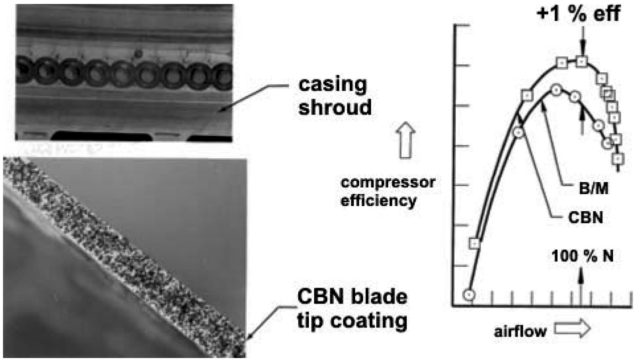

Fig. 28 Compressor blade tip coating CBN grits.

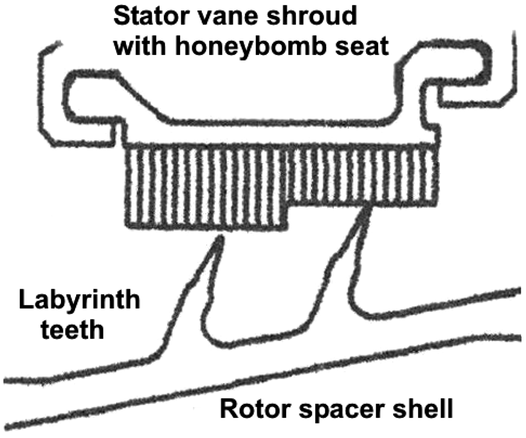

Fig. 29 Honeycomb labyrinth seal.

increases initial cost, but allows radial clearances to be reduced for increased efficiency without encountering rub damage (Fig. 28).

The CBN grits improved the compressor efficiency by 1% over the standard bill of materials design by allowing the blades to rub into the stator shroud in the range of 10–15 mils. Without CBN coatings, the radial clearance at assembly would have to be increased to insure a safe margin against rubs. Several turbomachinery components use CBN airfoil tip coatings in production on high-pressure turbines for increased efficiency.

In the early engines, it was common for interstage knife edge or labyrinth seals between the rotor and stator vanes or casing structure to run opposite a solid metal seat. It took considerable experience to set a radial gap that would avoid a rub that could melt metal and also minimize leakage. To avoid excessive heating during a rub, many labyrinth seal teeth were machined with a thin ribbon of material at the tip. After numerous failures, a GE engineer invented a honeycomb seal seat in the 1950s that could be rubbed without causing distress to the labyrinth teeth (Fig. 29). It took time for the engineers to gain confidence in setting the labyrinth teeth to rub, but this configuration was universally adopted by all engine manufacturers.

In the 1960s, the engineers also began to coat the rotating labyrinth teeth with Al_2O_3 in critical locations to prevent excessive wear during initial break-in and extreme transient operation. Coated teeth have become standard design practice to improve durability and reduce maintenance cost.

The J79, J93, TF39, and CF6 family of GE engine compressors used the smooth spool configuration that eliminated interstage seals by having the vane tips run close to the rotor shell. As late as the early 1970s, a majority of engineers and managers believed that a smooth spool rotor design gave superior performance over one with interstage shrouds, even with honeycomb labyrinth seals. However, in a back-to-back test on the GE F404 engine compressor, it was established that shrouded stator vanes and rub in labyrinth seals resulted in higher performance. As the aerodynamic stage loading increased, it also became evident that shrouded stator vanes were more durable than cantilever vanes and provided higher resistance to vibratory modes.

Efforts to improve compressor performance and operability highlighted the need to improve control of radial clearances. Unequal thermal heating and cooling of turbomachinery rotors result in increased radial buildup clearances to avoid excessive rubbing during transients. Compressor casings that are scrubbed by the flowpath air reach a stabilized temperature in a relatively short time. Since the rotor disks are shielded from the flowpath, thermal stabilization takes much longer than the casing. This difference in thermal response increases radial clearances and reduces stall margin, limiting the acceleration rate during a throttle advance from idle to full speed. A large difference in thermal response of the casing and rotor causes the opposite effect during a throttle retard from full speed. In that case, the casing cools faster than the rotor, causing a reduction in radial clearances. Compressor designers have struggled for many years to achieve the optimum compromise that avoids stalls during an acceleration and rubs on a decel.

Compressor thermal excursions have been reduced by the use of a combination of circulating cooler air within the rotor cavities and slowing the stator thermal response by the use of shielding and lower expansion alloys (Fig. 30).

In the case of the turbine, an active clearance control concept (Fig. 31) was developed by P&W engineers using compressor cooling air on internal parts adjacent to the flowpath and cooler modulated fan air on the turbine outer casing.

The introduction of computational fluid dynamics (CFD) (Fig. 32) allowed the aerodynamicists to model the complex flowfield in three dimensions.

CFD-three-dimensional modeling for steady and unsteady flow provides a more complete physical understanding of the airfoil flowfield from root to tip and has replaced many empirical correlations with physics. The CFD modeling has also been very useful in the understanding and solution of complex structural dynamics problems involving turbulent and separated flow.

The average compressor discharge temperature T_3 has been limited approximately to 1200°F (649°C) for the past 40 years, limiting the maximum pressure ratio and flight speed (Fig. 33). At 1200°F (649°C), nickel superalloys used in the high-speed rotating disks can encounter creep and rupture. At sea-level takeoff, Mach 0.25 (aircraft rotation), pressure ratios of 40–42 reach this T_3 limit. At Mach 3 flight speed in the altitude range between 36,000 and 69,000 ft, the air is already compressed to 632°F (333°C) at the compressor inlet. This high inlet temperature limits engine pressure ratios to a range of 3.5–3.8 without exceeding the peak profile temperature limit at the compressor discharge. For the future, higher-temperature superalloys are critically needed to extend pressure ratio, flight speed, and performance of airbreathing turbine engines. The compressor discharge temperature limit is the only barrier that limits airbreathing turbojet engines from operating significantly above Mach 3. Airbreathing ramjet engines can operate at very high Mach because they are not limited by the turbomachinery components. However, because the ramjet cycle depends on high speed for the ram compression process, these systems cannot take off without turbojet or rocket assist up to speeds of Mach 2.

Commercial aviation has spanned the globe using evolutionary advances in jet propulsion. The increase of compressor pressure ratios has been a key in the achievement of higher engine cycle efficiency for longer-range aircraft. Pressure ratios for subsonic aircraft applications have increased by a factor of 20 during the past 50 years (Fig. 34). The more recent engines have reached pressure ratios of 40 at sea level and are already pushing the compressor discharge temperature limit with current nickel superalloys.

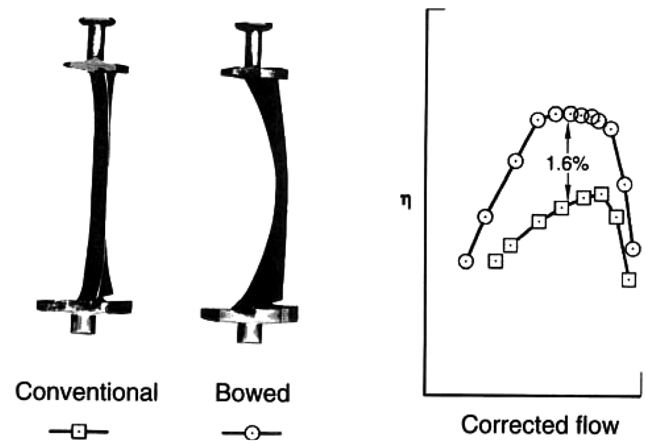

Fig. 32 Bowed compressor stators provide higher efficiency and stall margin.

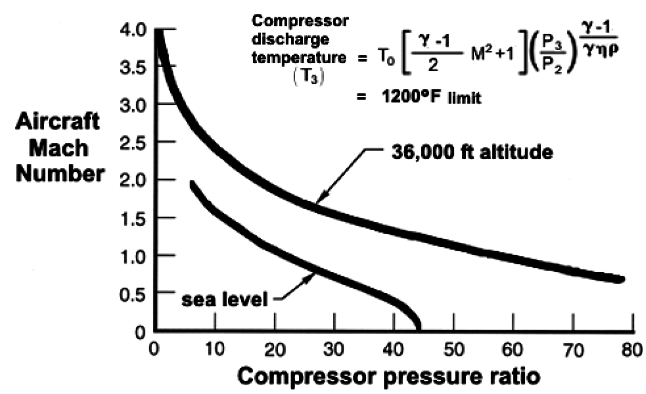

Fig. 33 Compressor exit temperature limits pressure ratio and flight Mach number.

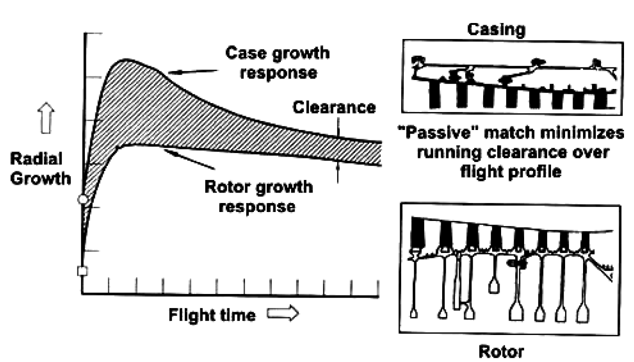

Fig. 30 Matching compressor rotor and stator radial clearances to reduce leakage.

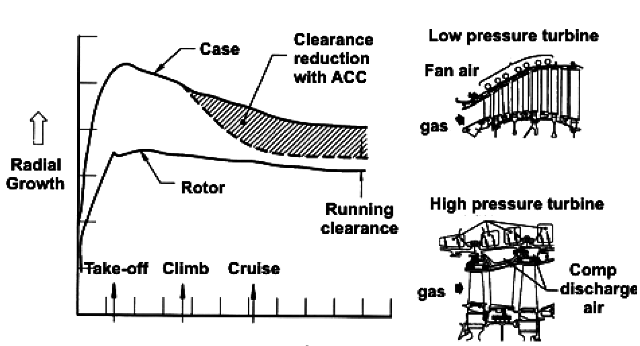

Fig. 31 Active clearance control to improve thermal match and reduce leakage.

Combustion Design

In the late 1950s, efforts began to eliminate visible smoke particulates, and, by 1970, work was in progress to reduce unburned hydrocarbons (HC) and carbon monoxide (CO). Combustion efficiency at high power has always been near 100%, but during the next 10 years, it was improved dramatically at or near idle power. Additional progress was made in the 1980s in understanding and developing technologies to reduce oxides of nitrogen (NO_x) emissions. With gas-turbine combustion efficiencies above 99%, little opportunity existed for further reductions in HC and CO. Smoke was reduced below the threshold of visibility with lean combustion, while still providing adequate windmilling air start. The focus in the 1990's was on NO_x emissions due to its contribution to ground level ozone and smog, acid rain, and atmospheric ozone depletion (Fig. 35). As we progress further into the 21st century, reducing NO_x emissions remains a primary challenge for gas-turbine applications including aircraft, marine, and power generation.

Reduced emissions combustors have also been significantly reduced in axial length, fuel delivery nozzles have been improved to eliminate fuel coking, and overall durability has been increased by orders of magnitude (Fig. 36).

The axial length of the early engine combustors was increased to provide adequate mixing of the fuel and air to avoid hot streaks into the turbine. However, improved fuel/air distribution nozzle systems allowed shorter combustors to reduce engine length and rotor bearing span, along with a reduction in liner cooling air while still improving the combustion pattern factor.

The durability of the combustion liners enclosing the hot gases became an issue with increased engine life requirements. For many years, combustion liners were manufactured using spot and seam welded overlapping sheet metal louvers with relatively short life (Fig. 37).

The machined ring liners provided a 10-times life improvement over the sheet metal liners at the same combustion temperatures. In the early 1970s, the U.S. Air Force Materials Laboratory funded a new concept that provided removable liner panels shielding the outer liner shell from the hot gases. This concept was developed and resulted in a 10-times life improvement over machined ring configurations in use and a 100-times life improvement over the earlier sheet metal liners. Improved machined ring liners have also been developed by the use of film/convection cooling.

High-Mach Engines

In the mid-1950s there was considerable U.S. Air Force and U.S. Navy interest in high-Mach flight as the next frontier for reconnaissance, interceptor, and bomber aircraft. GE was funded to develop the YJ93 engine (Fig. 38) for the XB-70 Mach 3 supersonic bomber. Two prototype aircraft were built and flight tested before the program was canceled. The original compressor had 12 high aspect ratio stages with insufficient stall margin and durability. Stalls would occur often during testing, causing blades and vanes to "clash and clang" as a result of insufficient flexural rigidity. This experience promoted the use of lower aspect ratio airfoils for both stall margin and durability.

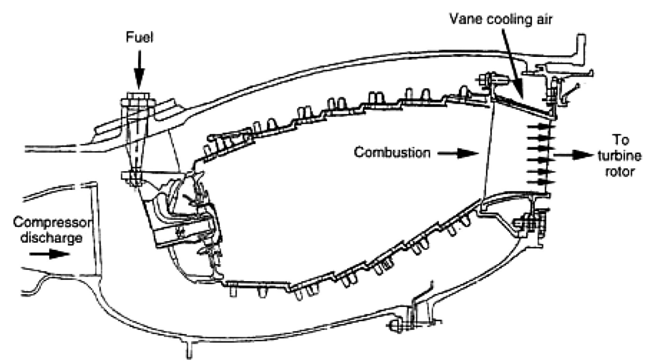

Fig. 36 Combustor durability improvements.

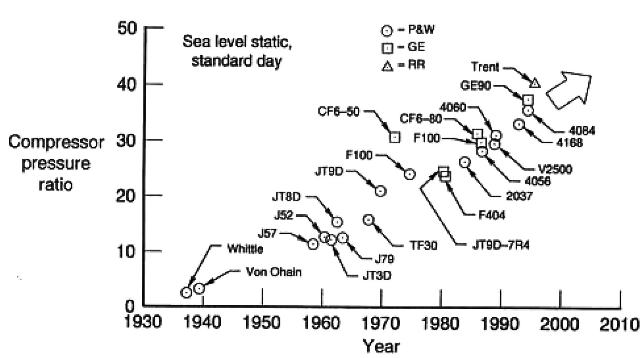

Fig. 34 Compressor pressure ratio growth.

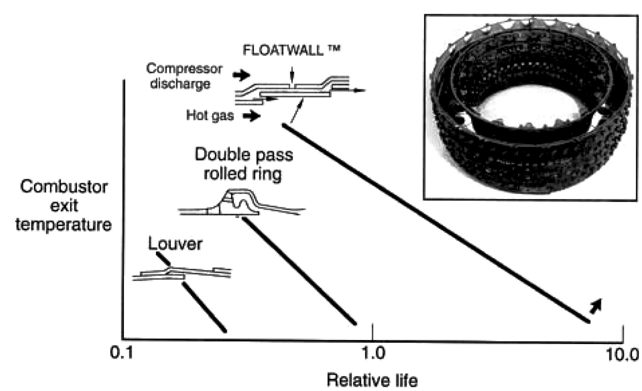

Fig. 37 Increasing combustor liner durability.

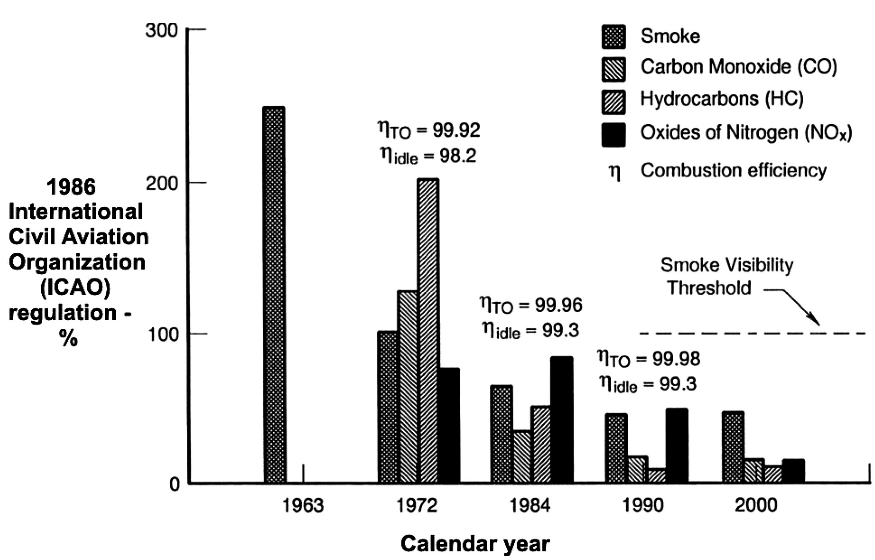

Fig. 35 Emissions reduction progress.

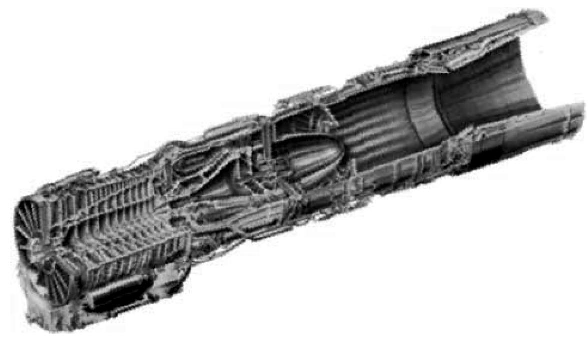

Fig. 38 GE J93 Mach 3 turbojet engine (courtesy of GE).

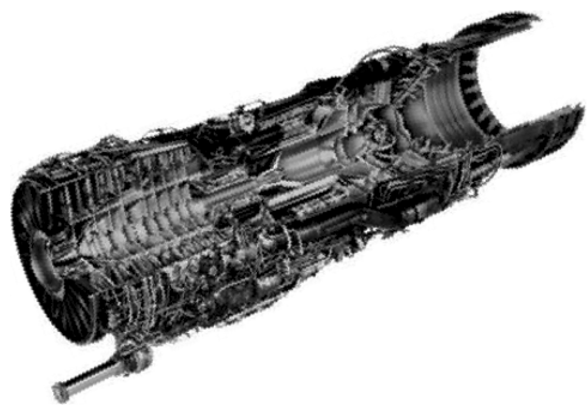

Fig. 39 PW J58 Mach 3-plus turbojet engine shown with primary nozzle (courtesy of P&W).

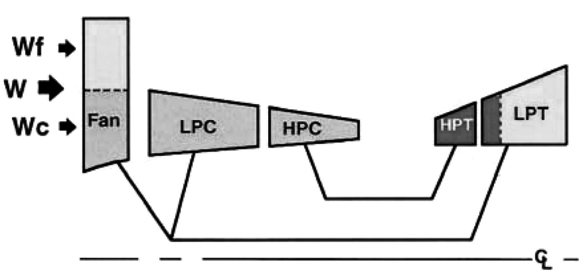

Fig. 40 Core engine with dual spools and fan.

A PFRT redesign was initiated that used 11 "long chord" stages in titanium and A286 at a pressure ratio of nine in the same axial length. The J93 engine had variable inlet guide vanes (IGVs), three front and five rear cantilevered variable stators, an annular combustor, convection-cooled turbine blades operating at 2000°F RIT (1093°C), and a large converging/diverging exhaust nozzle. The rear variable stators were opened above Mach 2 to decrease overall pressure ratio and increase airflow at the higher Mach numbers. The last stage of variable stators could be closed in the event of an engine failure at high flight speed to prevent the rotor from windmilling at high rpm due to the pressure drop from inlet to exhaust. Extensive development and lessons learned on the J93 resulted in the production of turbomachinery technology that was transitioned to all future GE engines.

In the same time period, P&W was also funded to develop high-Mach engine technology. This effort led to the development of the J58 Mach 3-plus turbojet (Fig. 39) for the twin-engine Lockheed SR-71 high-altitude reconnaissance aircraft. The J58 engine used variable IGVs, a nine-stage compressor, a midstage bleed bypass, a cannular combustor, the first P&W convection-cooled turbine rotor at 1780°F (971°C) RIT, and an afterburner. Compressor bypass bleed doors after stage four opened at Mach 2.5, decreasing the compressor pressure ratio and discharge temperature while increasing airflow. This produced a similar result as the J93 with rear variable stators, introducing a variable pressure ratio cycle. The exhaust nozzle consisted of a P&W primary section and a Lockheed blow-in door ejector section for the secondary, which was integrated into the aircraft structure. A new generation of advanced superalloys and processing was also developed to meet the extended high-temperature operation. The J58 was a resounding success, and was the only operational Mach 3-plus engine with a service life for over 30 years.

Core Engine Evolution

The introduction of either dual spool turbojets or turbofans redefines the traditional core engine used to produce propulsive energy (Fig. 40).

The core engine now includes the high-pressure compressor (HPC), low-pressure compressor (LPC) and fan hub with the drive high-pressure turbine (HPT) and low-pressure turbine (LPT). The fan bypass W_f using LPT-drive turbine energy provides a lower exhaust velocity for improved performance.

Gas turbine evolution is best represented by a comparison of the power of the core engine with the turbine rotor inlet temperature (Fig. 41). This comparison is also a guide to future development opportunities for an increase in core engine performance and has become a benchmark for all gas-turbine engines. The extended core

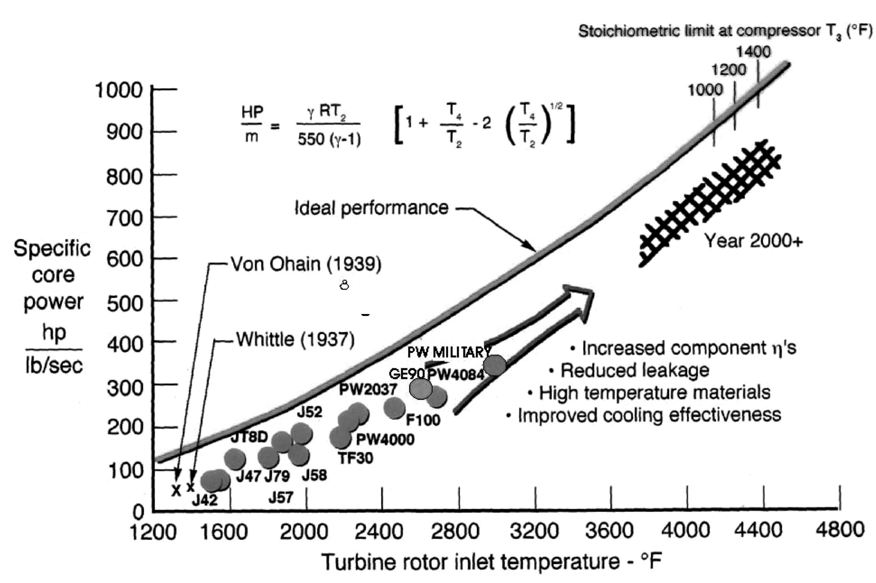

Fig. 41 Core engine performance evolution [°C = (°F-32)/1.8].

engine power is normalized by division by the mass flow to eliminate the size effect representing the specific core engine power. By 1990, the core engine power increased steadily with turbine temperature by more than five times over the early engines. The ideal Brayton cycle performance (100% efficiency and no cooling air) is represented by the formula showing that specific power is a function only of turbine rotor temperature.

The engines shown with losses and cooling air produce about 30–35% less power than the ideal. It takes higher-efficiency, higher-temperature materials and improved cooling effectiveness to move closer to the ideal performance. This roadmap for an increase of core power was first introduced in 1983 to counter the idea that gas turbines were a sunset industry and that little additional technology funding by the government or industry was required. It took a young engineer to develop the theoretical formula showing that specific power is a function only of the turbine rotor inlet temperature T_4. This chart was easily understood and convinced the Congressional staffers and members of Congress to fund the Department of Defense to improve gas-turbine performance. This initiative launched the highly successful Integrated High Performance Turbine Engine Technology (IHPTET) program responsible for many of the technology advances in use today.

The ideal specific power increases with turbine temperature until it reaches the fuel stoichiometric temperature. It is not well known that the stoichiometric temperature of hydrocarbon fuel is a function of the compressor discharge temperature T_3 (Fig. 42). With the average compressor discharge temperature T_3 limited to 1200°F (649°C) because of nickel superalloy limitations, the stoichiometric temperature is 4300°F (2371°C) and remains a barrier for gas turbine engines.

Control Systems

The control system is the brain of the engine, translating the throttle signal request into single or multiple specific commands to the various engine components.

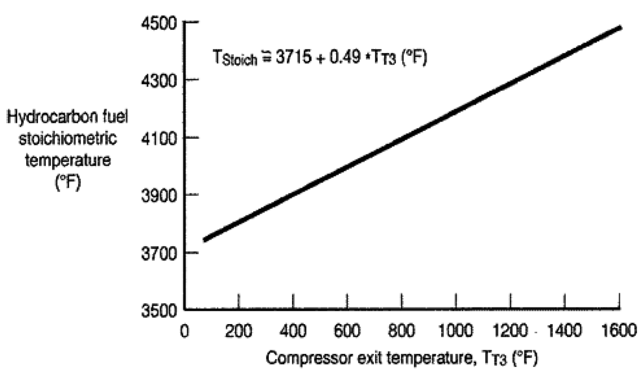

Fig. 42 Fuel stoichiometric temperature vs T_3 [°C = (°F–32)/1.8].

Hydromechanical analog control systems use cams, springs, levers, and valves to deliver the correct fuel flow and position the engine's variable-geometry devices for a given combination of throttle setting, engine speed, and ambient conditions.

During the past 40 years, control systems have seen a remarkable evolution from hydromechanical analog to multifunction full authority digital electronic controls (Fig. 43).

In the 1960s, the hydromechanical control served as an analog computer with as many as seven functions involving combustor and afterburner fuel flow, inlet guide vane, stator vane position, bleed/cooling valve settings, exhaust nozzle position, and rotor speed feedback. By the 1970s, the complexity of these controls made them both difficult and costly to maintain. The development of a reliable digital integrated circuit at P&W allowed engineers to design electronic trimmers for the B727 aircraft to reduce hydromechanical complexity. In the mid-1980s, a full authority, single-channel digital engine electronic control (DEEC) with partial redundancy, was developed by the use of a small hydromechanical control as a backup for "get home" capability. The DEEC could self trim the engine to maintain thrust level in flight and also made it possible to develop vectoring nozzles for fighter aircraft. The mid 1980's F15 demonstrator aircraft (Fig. 44) with pitch vectoring nozzles and electronic controls led a new generation of engines providing greater fighter aircraft maneuverability and capability.

In the 1990s, the P&W designers developed a full authority digital electronic control (FADEC) with sufficient redundancy and reliability to eliminate the hydromechanical backup unit. FADEC controls now have the ability to detect engine failures and both isolate and accommodate them for safety while providing "get home" capability and quick maintenance.

New Generation of Fighter Engines

A new requirement for the design engineers was to explore means of reducing the radar signature of new fighter aircraft for

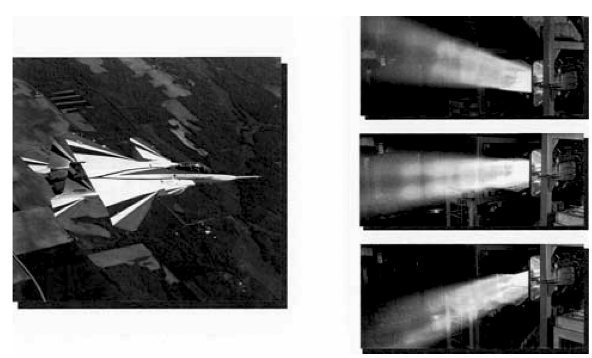

Fig. 44 NASA F15 flight test with electronic controls and pitch vectoring nozzles.

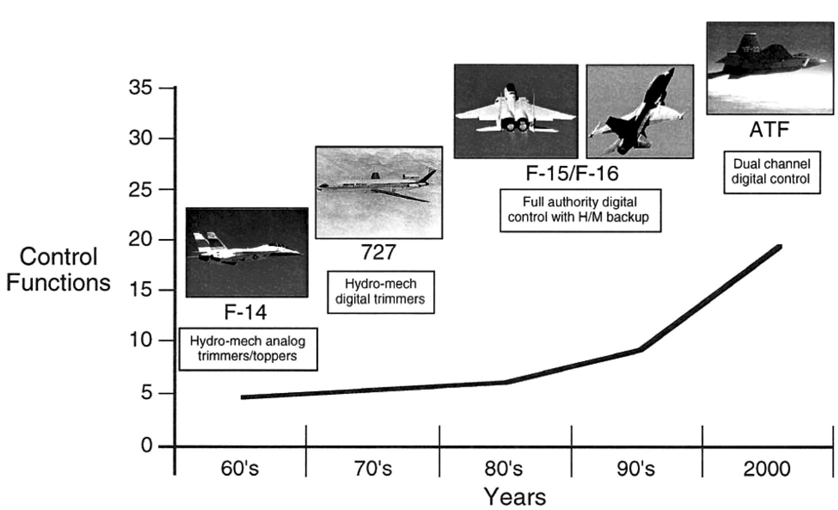

Fig. 43 Engine control system evolution.

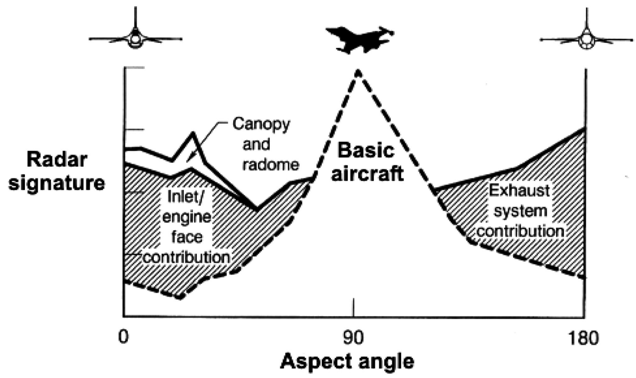

Fig. 45 Engine and aircraft radar signature characteristics.

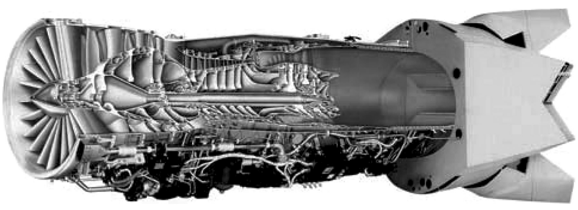

Fig. 46 F119-PW-100 fighter engine with dual FADEC for the F-22 aircraft (courtesy of P&W).

Fig. 47 PW F100 pitch/yaw nozzle demo.

Fig. 48 NASA flight test of pitch/yaw nozzles in F-15 aircraft with P&W F100 engines.

survivability and stealth (Fig. 45). Capital investment was provided to build radar facilities to study configurations that would meet requirements for the engine/airframe interface. Technologies developed were coordinated closely with the airframe manufacturers and transitioned into all new systems.

The flight-test pitch vectoring nozzle (Fig. 44) is a two-dimensional configuration requested by Lockheed and Northrop for stealth applications to reduce radar signature and is based on a prototype nozzle tested in the early 1980's. The square vectoring nozzle cross section was favored for integration with the aft section of the aircraft, although the engineers were somewhat reluctant to design a lightweight noncircular pressure vessel with moveable surfaces. It took great ingenuity and persistence to design and manufacture the structure for this pitch vectoring nozzle that could withstand afterburner exhaust temperatures and provide high performance.

The PW F119 engine for the F22 stealth fighter incorporates all key technologies developed over the past 50 years (Fig. 46) including dual channel FADEC controls and pitch vectoring nozzles.

The dual FADEC control units are mounted aft on both undersides of the lower fan duct. All accessories, including pumps, valves, oil tank, and gearbox are mounted on the underside with the ability for quick maintenance with hand tools. The low-radius ratio fan blades are hollow diffusion bonded titanium and linear friction welded to the disk, making a one-piece rotor stage. The compressor blades are machined integral with the disks to eliminate dovetail attachments, reducing leakage and weight. The rotor spools are counter-rotating, with the high-pressure rotor piggy back mounted on roller bearings inside a flange disk on the low-pressure shaft. Placement of the high-speed rotor roller bearing inside a race mounted on the low-speed shaft reduces the normal radial clearance caused by unequal centrifugal growth and provides a tight fit to maximize concentricity. The piggy back bearing mounting arrangement eliminates a hot strut frame aft of the HPT and results in having three frames instead of four to support the four bearing sumps. After it passes through the Mach 1 drag rise, the high turbine inlet temperature allows the F-22 fighter to shut down the afterburner in the supersonic cruise mode for increased fuel efficiency and range. The vectoring nozzle increases maneuverability and operational flexibility at both military (no afterburner) and maximum power (with afterburner) throttle positions.

Meanwhile, development of a vectoring nozzle was initiated with a conventional circular cross section incorporating both pitch and yaw. NASA and airframe company studies showed that providing both pitch and yaw would result in exceptional aircraft maneuverability. The nozzle design engineers were determined and developed a lightweight, low-cost pitch/yaw nozzle with a relatively small amount of resources (Fig. 47), and NASA flight tested it on an F-15 aircraft (Fig. 48).

After the success of the pitch/yaw vectoring nozzle, the design engineers proceeded to develop stealth features, so that this nozzle would be ready when needed.

During the late 1980s, studies were initiated by the U.S. Marine Corps to provide the next generation of short takeoff and vertical landing (STOVL) aircraft to replace the AV-8B Harrier powered by the Rolls-Royce Pegasus turbofan. Both GE and P&W developed advanced engines with higher temperature and higher core power for the F-22 competition in 1991. Initially, the engine studies were based on the direct lift system used in the Pegasus where fan bypass and engine exhaust was diverted downward to generate the required lift during STOVL operation. However, after the initial preliminary design studies, the U.S. Marine Corps requested a significant increase in the thrust margin over the AV-8B during the lift and hover modes, which was very difficult to achieve with the direct lift concept. Lockheed/Martin engineers conceived the idea of adding a shaft-driven fan in the forward section of the fuselage to provide higher airflow in the STOVL mode. This fan moved a larger volume of air at lower velocity, increasing propulsive efficiency and lift margin. The Joint Strike Fighter (JSF) program was formed, and a derivative of the F119 engine, the F135, was developed and selected for the propulsion system. Changes included increasing the high-pressure turbine temperature and adding a low-pressure turbine stage to provide the added power for the lift fan (Fig. 49).

Transport Engines

The DeHavilland Ghost 50 turbojet, the world's first commercial aircraft engine (Fig. 50) was introduced into service in 1952 to power the DeHavilland Comet (D.H.106), the world's first commercial airliner. Although the Comet aircraft was short lived due to a low cycle fatigue failure of the fuselage, it introduced the jet age into commercial aviation. It was five years later, in late 1957, before the Boeing B-707 powered by four PW JT3 turbojets entered

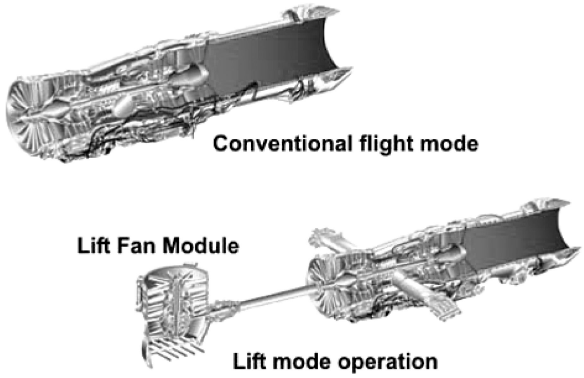

Fig. 49 F135-PW-600 STOVL propulsion system for the F-35 JSF (courtesy of P&W).

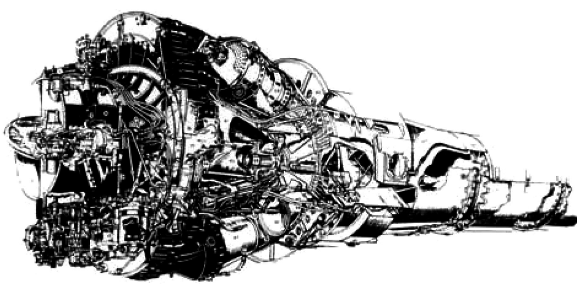

Fig. 50 DeHavilland Ghost 50 turbojet: the world's first commercial engine.

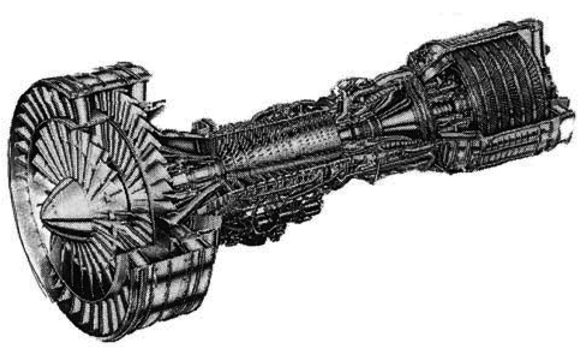

Fig. 51 GE TF39 turbofan for the C5-A (courtesy of GE).

commercial service and launched jet passenger service in the United States.

The Ghost 50 was a 2.3 thrust-to-weight single spool turbojet with a centrifugal compressor, 10 axial flow combustor cans, and an axial flow turbine. At standard sea-level conditions, the engine developed 5000 lb of thrust with an airflow of 87 lb/s, a pressure ratio of 4.87, and a turbine rotor inlet temperature of 1472°F (800°C). The technology was based on a smaller military engine known as the Goblin, which was patterned after the Whittle engine.

In the early 1960s, a U.S. Air Force initiative to define the next generation of transport engines was taken seriously by GE management, and studies began. Technical requirements such as the fan bypass ratio were discussed with the Director of Turbine Engine Division in the Aero Propulsion Laboratory. The U.S. Air Force pushed for the next engine to have a bypass ratio of 10, whereas GE wanted a bypass ratio of 6. A compromise was made, and bypass ratio of eight was agreed on. A half size demonstrator engine was built and tested in late 1964, and GE won the contract for the full size TF39 engine in late 1965. The TF39 launched a new generation of high bypass engines and is still in service today in the Lockheed C5-A transport. The CF6 engine family with bypass ratios in the five range is based on the TF39 core turbomachinery (Fig. 51).

The large stage and a half fan was a unique split inlet flow configuration and the only one ever to be used for a high bypass engine. The design was very complex, requiring shrouded inlet guide vanes over the one-half stage rotor and a platform splitter built into the large fan blade to isolate the bypass flow. It was conceived for the one-half scale demonstrator to overcome the top management edict that no GE engine would use a front fan like P&W. Such "not invented here" rules for the engineers at both P&W and GE were not uncommon in the 1960s. Both stages supercharged the core engine for an overall pressure ratio of 26, setting a milestone for the lowest demonstrated fuel consumption. Turbofans with bypass ratios of eight were not built for another 25 years because both higher pressure ratios and turbine temperatures were required to offset weight and installation losses. Engine bypass ratios of six were introduced in the early 1970s using the P&W configuration of a single-stage front fan without IGVs and with booster stages on the fan rotor to supercharge the core compressor.

The GE F100 and PW F100 engines were offered for the F-15 fighter during an 18-month competition in the late 1960s. P&W won by displaying superior experience and understanding in transient flow compatibility between the aircraft inlet and engine. GE's engine also failed a fan blade in a critical test at the Arnold Engineering Development Center. The GE engineers increased the compressor pressure ratio from 8 to 12 by adding a stage, increased the bypass ratio, and were awarded a contract in 1969 to power the North American B-1 bomber. This engine core used all of the technology that GE aggressively accumulated during the 1960s and became the GE F101 (Fig. 52).

The nine-stage, more highly loaded compressor was driven by a single-stage high-temperature turbine with an unheard of tip speed of 1800 ft/s. The entire core rotor was straddle mounted on two bearings, which was a radical departure from all of the other GE engines that used three bearing systems. Until 1968, it had been a company rule that each compressor and turbine be supported on its own set of bearings. To convince the group vice-president, the compressor, short combustor, and turbine was scaled to the 17-stage J79 compressor size to show that all of the turbomachinery fit within the J79 compressor bearing span.

A very creative and persistent preliminary design manager conceived the idea of using the F101 core engine to create a 10-ton commercial engine to compete with the popular PW JT8D. After being rejected by the GE chairman for funding, he proceeded to find a foreign partner for the low spool portion of the turbofan engine. After approval to partner from the U.S. Department of State, this engine became the GE/SNECMA CFM56 and went on to be the most successful engine program in commercial engine history.

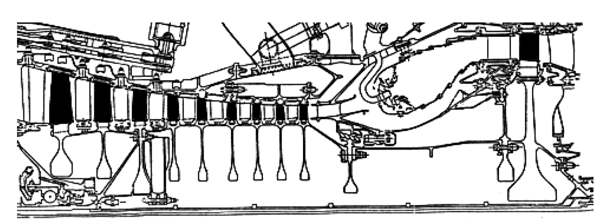

Fig. 52 F101/CFM56 common core engine: a new concept in turbomachinery (circa 1970) (courtesy of GE).

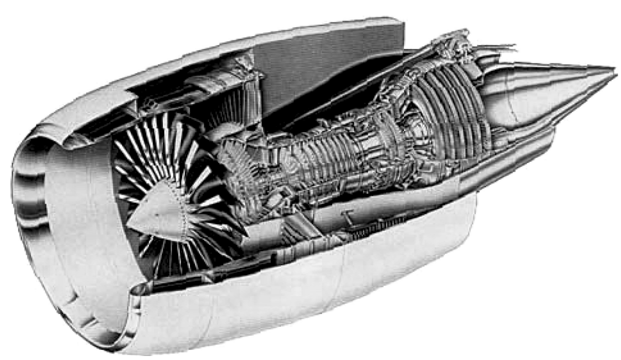

Fig. 53 GE90 engine for B-777.

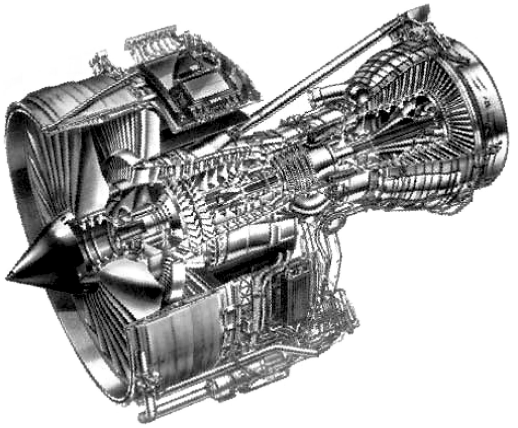

Fig. 54 Rolls-Royce Trent 800 for the B-777.

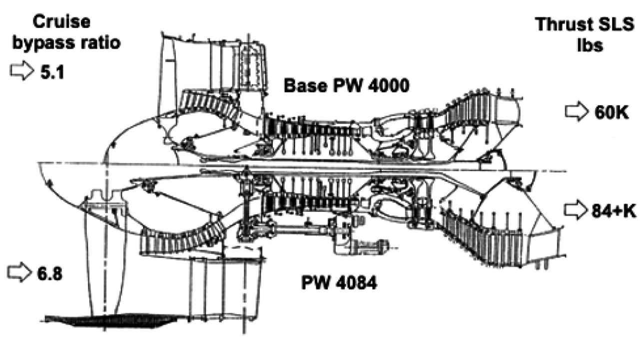

Fig. 55 Thrust growth with common core.[8]

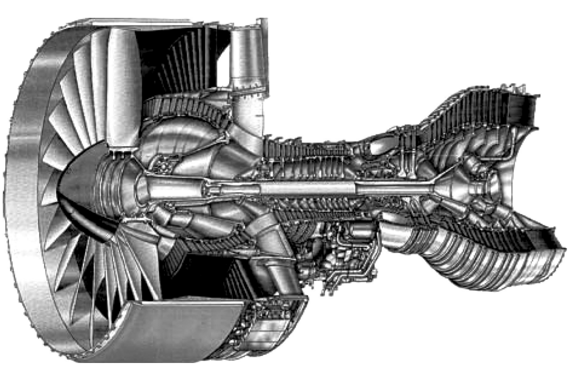

Fig. 56 PW 4084 engine for B-777.

After a considerable time passed, P&W and Rolls-Royce teamed to develop the V2500 turbofan to compete with the CFM56. This engine was late to the marketplace due to Rolls-Royce compressor problems. In 1994, the chairman of UTC rejected funding certification of the V2500 on the B-737 aircraft, leaving the bulk of the business to the GE/SNECMA partnership.

By the early 1990s Boeing launched the B-777, a new large 300–400 passenger twin engine aircraft, to cover more efficiently both regional and international markets. The requirement for the three engine manufacturers was to provide more fuel-efficient engines with a wide thrust range, including a takeoff thrust of 84,000 lb, lower noise, low exhaust emissions, and high reliability at introduction. The designers set out to increase both thermal and propulsive efficiency, with a design that would meet requirements for improved noise and emissions, reliability/durability/maintainability/weight/cost and timing. Both GE and Rolls-Royce engineers were funded to design all-new propulsion systems targeted to minimize the number of turbomachinery stages for reduced cost. The GE90 engine had 22 stages of turbomachinery on two spools and used a compressor derived from the 1970's NASA demonstrator.[3] This compressor design gave GE a good start because it had demonstrated a pressure ratio of over 20 in only 10 stages with high efficiency, an unprecedented success for the aero design team. The GE design team also developed a 123-in.-diameter composite fan (the world's first) with a cruise bypass ratio of 8.4 (Fig. 53).

During this period, Rolls-Royce developed the Trent, an efficient three-spool engine with 22 stages of turbomachinery a 110-in.-diam fan at lower bypass ratio (Fig. 54).

GE and Rolls-Royce (RR) were already working on these new large engines when P&W realized that they needed to enter the race. P&W engineers were challenged to build an 84,000-lb thrust engine for two-thirds the cost of a new engine.[8] The 1990 PW4000 core engine compressor was efficient, with the highest exit corrected flow in the industry, and, therefore, could produce more power for a given turbine temperature. This base engine compressor with a relatively low 10.25 pressure ratio allowed higher supercharging without exceeding the T_3 discharge temperature limit. By the addition of a fan and six booster stages driven by a seven-stage low-pressure turbine, the base engine with a thrust of 60,000 lb increased to 84,000-plus lb (Figs. 55 and 56). The overall engine pressure ratio increased to 36, and core upgrades were made that were compatible with current fleet engines that pleased customers.

A new, higher-strength nickel superalloy material had been developed earlier and was used for the uncooled turbine drive shaft to accommodate the large increase in fan torque. A new, hollow shroudless titanium fan blade was introduced, and turbine technology was transitioned from the higher-temperature military engines. The PW4084 derivative engine had 27 stages of turbomachinery, which was five more than the competition; however, it was first into the marketplace, gaining wide acceptance and sales.

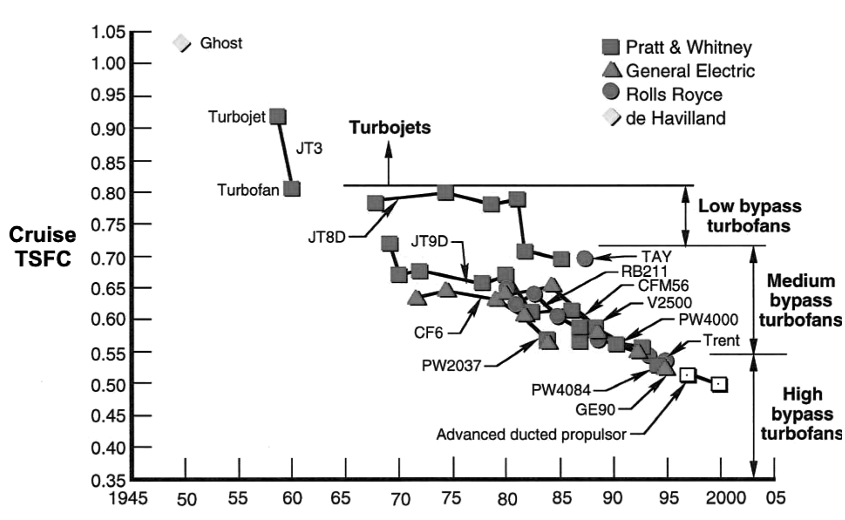

Fig. 57 Transport engine performance.

The P&W management worked closely with engineering and understood the risk that this engine, with five additional stages, would likely have a higher production cost than the competition. The strategy was to win and retain a major share of the market by using, if necessary, a portion of the development funds saved by choosing a derivative. This strategy was initially successful, and the PW4084 captured a major share of the market. However, in 1993, a new P&W management did not subscribe to the PW4084 marketing plan strategy for the commercial business. This action resulted in the loss of a major 1995 worldwide competition, causing a serious loss in PW4084 sales.

Transport aircraft have been dependant on the evolution of the jet engine, which has seen a 50% reduction in thrust-specific fuel consumption (TSFC) during the past 50 years (Fig. 57). The reduction in TSFC can be observed by engine type categorized into turbojets, beginning with the Ghost 50, followed by low, medium, and high bypass ratio turbofans. The lower exhaust velocity of the higher bypass turbofans more closely match the subsonic flight speed of transport aircraft, resulting in higher propulsive efficiency. Higher fan bypass ratios require a small core engine with enough power to drive the fan bypass flow. Without increasing core airflow, which would reduce the engine bypass ratio, higher core power requires increased turbine temperature.

Although there is still room to increase turbine temperature to the levels used in the newer fighter engines, increased NO_x emissions at higher combustion temperatures becomes a challenge. The GE 90 growth engine (Fig. 58), introduced in 2003, is the newest and also largest of the high bypass turbofans rated at 115,000 lb of thrust at sea-level static. Notable features include a larger 128-in.-diam composite fan blade (an industry first) combining both rearward sweep at the midspan and forward sweep at the tip. To drive this larger fan with 11% higher flow than the GE90-94B, the last stage was dropped from the compressor to increase the core flow and power. An increase of the core power with airflow decreased the cruise bypass ratio from 8.4 to 7.1. GE reports that higher-efficiency components derived using 3-D analysis and the addition of a booster stage offset the loss in propulsive efficiency.

The relationship between core engine thermal efficiency and propulsive efficiency determines the overall engine efficiency and subsonic TSFC for turbojet and turbofan engine classes[9] (Fig. 59).

The final frontier for subsonic flight with turbofans is the center of the cross-hatched area at 50% overall efficiency, representing a thermal efficiency of 62% and a propulsive efficiency of 80%. At this performance level, an overall efficiency of 50% is 39% higher than the newest high bypass turbofans in service. Core thermal efficiencies of 62% require much higher cycle pressure ratios. The small circle just below the cross-hatched area represents a propeller-type propulsor with a bypass ratio of 50. A unique GE counter-rotating aft unducted fan (UDF) driven by engine exhaust gases and a P&W gear-driven front fan have demonstrated propulsive efficiencies of 80%. The GE UDF, driven by exhaust gases, was a particularly creative concept because the stator vanes also rotated counterclockwise to the rotor. These whirling windmills, similar to propellers, have very high bypass ratios and higher fuel efficiency but have not been accepted because of noise, safety, and installation issues. The red circle located at 40% overall efficiency represents a 10% reduction in TSFC, which was approached in a 1992 demonstration with a ducted fan at 12 bypass ratio (Fig. 60).

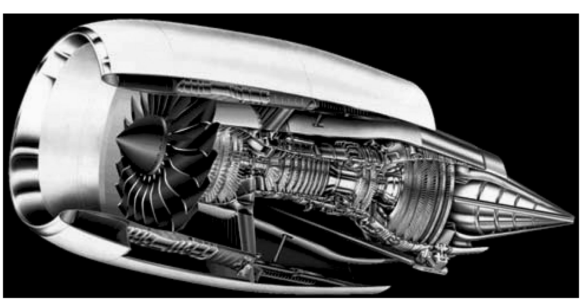

Fig. 58 GE 90-115B growth engine (courtesy of GE).

Fig. 60 Advanced ducted propulsor engine (courtesy of P&W).

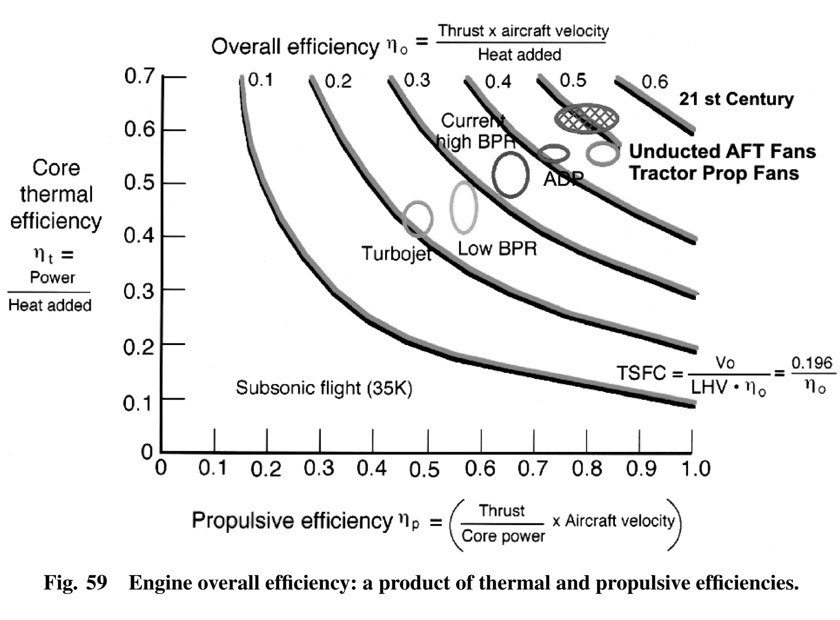

Fig. 59 Engine overall efficiency: a product of thermal and propulsive efficiencies.

The higher bypass ratio fan was driven by a high-speed, low-pressure turbine through a gear drive with a 3.7 reduction ratio (Fig. 61). A gear-driven turbofan is more complex than the standard direct drive engine, but offers the potential for achieving higher propulsive efficiency.

A gear drive requires at least 98% efficiency, but allows the low-pressure turbomachinery to operate at higher rotational speeds, reducing the number of stages. The gear-driven turbofan configuration offers the possibility to increase overall efficiency without an increase in core thermal efficiency. However, because these very high bypass ratio engines produce less thrust per pound of airflow, they require a larger diameter to produce the same thrust as a direct-drive turbofan. The high bypass ratios use lower fan pressure ratios and likely require a variable fan exhaust nozzle in combination with the thrust reverser.

Transport aircraft are required to meet noise regulations in airport locations based on the engine perceived noise in decibels (EPN-db) over normal ambient conditions. Progress in noise reduction for nonafterburning turbofan engines has been aided primarily by increasing the fan bypass ratio (Fig. 62). Internal sound suppression panels in the fan duct with multiple degrees of acoustic treatment and positioning the fan stator aft of the rotor by one and one-half blade chords was introduced in the late 1960s and has remained effective.[10]

The recently retired supersonic Concorde powered by four Rolls-Royce Bristol Siddeley 593 afterburning turbojet engines has been restricted from many of the world's airports. This impressive aircraft, with a Mach 2 cruise speed, was in service since 1968. NASA and industry studies of the next generation of commercial supersonic airplanes resulted in technology programs to reduce both airport noise and NO_x emissions at altitude. Some progress was made in the reduction of exhaust noise by the use of blow-in door ejector nozzles. NASA and industry technology efforts to reduce NO_x during cruise at 60,000 ft, where the ozone concentration is highest, have not been encouraging. However, the primary barrier in developing a supersonic airliner has been economics issues due to the high development cost in the range of $20-plus billion.

People

It is important to realize that people make things happen and that the evolutionary successes were brought about by many individuals in government, industry, and academia. Sometimes, the inventor of new innovative ideas found it difficult to gain acceptance and the support needed for development. A successful designer must have a high degree of engineering knowledge, persistence, and marketing skills to sell his or her ideas. Observations on understanding the often-generalized role of engineers and managers is offered for our current and new generation of pioneers (Fig. 63).

The gas-turbine is the world's most complex product because it involves every scientific and engineering discipline. History has shown that successful corporations are those that have the engineers and managers working together in concert, using insight and foresight, to resolve differences in perspective.

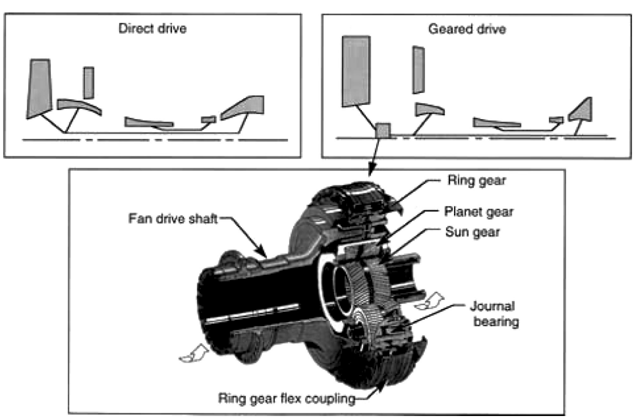

Fig. 61 Planet gear drive for high bypass fan demonstrator.

Priority Generalizations

Design Engineers	Managers
- loyal to the product	- loyal to the company
-product superiority	- product differentiation
- knowledge based design	- rules based design
- individualistically oriented	- team oriented

Neither realizes:

"You don't know what you don't know"

Fig. 63 Engineer and manager generalizations.

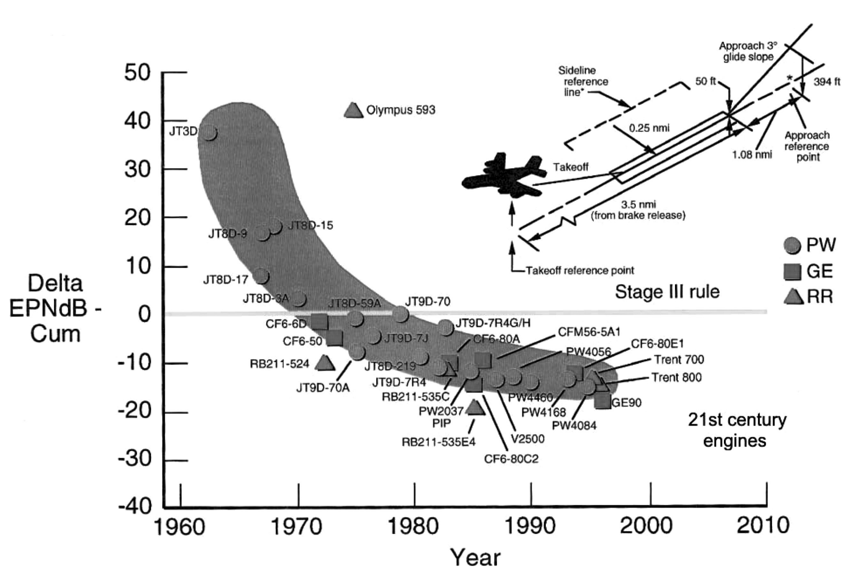

Fig. 62 Progress in aircraft engine noise reduction.

Summary

During the past 50 years, it took many thousands of engineers and considerable resources to bring the jet engine to its current state of evolution. Overall engine efficiency (thermal x propulsive) for transports has increased from 20% for turbojets to 36% for high bypass turbofans. Lessons learned have repeatedly shown that technology should lead the commitment to meet program milestones and achieve the lowest development cost. However, many technologies were developed out of necessity, in the heat of battle, during major development initiatives suggesting that insight, motivation, ability, and perseverance are also key ingredients for success.

The advanced turbine engine gas generator initiative (ATEGG) began in the early 1960s and was included in the IHPTET program in the 1980s. These ATEGG and IHPTET government programs provided a benchmark in leadership for engine technology development during the past 40 years, accounting for much of the technology evolution presented.

In the case of the supersonic transport, issues of economics, NO_x emissions, and airport noise have not been resolved. A national initiative would be required to overcome these remaining technical barriers to realize the supersonic travel that was Frank Whittle's dream when he proclaimed, "the future will be more exciting than the past."

Technologies presented have been transitioned to marine propulsion, industrial engines for power generation, rocket engine turbomachinery, and ramjets. Looking forward, candidate technologies that designers need for the future include the following: 1) higher temperature and strength, nonoxidizing superalloys; 2) very low NOx combustors for aero engines; 3) high-temperature, high-strength nonburning titanium or other lightweight alloys; 4) ductile composites with increased strain range; and 5) a national initiative to improve the performance for both subsonic and supersonic aero engines.

Ultimately, such technologies need transition into products to advance the state of art while providing superiority in defense and a positive aeronautics balance of trade for the United States. The technology evolution story makes a convincing argument that design is the creative art form of analysis and the source of competitive advantage.

References

[1] Midland Air Museum, Coventry Airport, Baginton, Warwickshire, England, U.K.

[2] Deutsches Museum, Bonn, Meisterwerke, Germany.

[3] Univ. of Southampton, Highfield, Southampton, England, U.K.

[4] U.S. Air Force Museum, Dayton, OH.

[5] U.S. Air Force Museum, Dayton, OH.

[6] Koff, B. L., "Designing for Durability in Fighter Engines," *International Journal of Turbo and Jet Engines*, Vol. 1, 1984, pp. 209–222.

[7] Rosen, C., Koff, B. L., and Hartmann, M., "Airbreathing Propulsion Component Technologies," *AIAA Journal*, Vol. 18, No. 6, 1980.

[8] Koff, B. L., "PW4084 Thrust Growth with Commonality," Tenth International Symposium on Airbreathing Engines, ISABE and AIAA, Sept. 1991.

[9] Koff, B. L., "Spanning the Globe with Jet Propulsion," AIAA Paper 91-2987, April 1991.

[10] *The History of Aircraft Gas Turbine Development in the U.S.*, Peter, St. J., International Gas Turbine Inst., Atlanta, GA, 1999.