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Advanced Technologies Impact on Compressor Design and Development: A Perspective

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ADVANCED TECHNOLOGIES IMPACT ON COMPRESSOR DESIGN AND DEVELOPMENT: A PERSPECTIVE

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ABSTRACT

A historical perspective of the impact of advanced technologies on compression system design and development for aircraft gas turbine applications is presented. A bright view of the future is projected in which further advancements in compression system technologies will be made. These advancements will have a significant impact on the ability to meet the ever-more-demanding requirements being imposed on the propulsion system for advanced aircraft. Examples are presented of advanced compression system concepts now being studied. The status and potential impact of transitioning from an empirically derived design system to a computationally oriented system are highlighted. A current NASA Lewis Research Center program to enhance this transitioning is described.

PREFACE

Receiving the 1989 Cliff Garrett Turbomachinery Award is a great honor for me, since I have spent most of my 32-year career in turbomachinery-related research. I treasure my associations and friendships developed over the years within the extensive turbomachinery community. And, without the benefit derived from these relationships, receiving this award would not have been possible. I thank each of you, with a special thanks to my co-workers at the NASA Lewis Research Center.

Upon joining NASA, I was initially involved in research on liquid hydrogen pumps for rocket engines. I soon moved on to the compressor world, conducting and managing research on both axial and centrifugal compressors for the Brayton-cycle space power systems. Following the space power research activities, I moved on to pursue fan and compressor research in support of aircraft gas turbine engines. Most recently I have been involved in the overall management of the propeller and turbine research activities along with the compressor research activities ongoing at NASA Lewis Research Center.

I have thoroughly enjoyed my career in the high-technology turbomachinery field and have found it rewarding. I would like to take this opportunity to encourage new engineering graduates to consider the turbomachinery field as a career. There are many opportunities to excel with much yet to be accomplished technically as more demanding requirements are imposed to meet advanced aircraft mission needs.

And as for the preparation of any paper or lecture, I have found that the author or presenter benefits as much as the listener or reader. This one will be no exception. In the spirit of this award, that is, to honor Cliff Garrett and to promote presentations of SAE papers on turbomachinery, I gratefully accept this opportunity.

INTRODUCTION

The compression systems in today's aircraft gas turbine engines are high-performing, highly sophisticated devices. They are the result of the extensive research and development that has taken place since the design and development of the first aircraft gas turbines by Sir Frank Whittle of England and Dr. Hans von Ohain of Germany, working independently during the late 1930's and early 1940's. For those not up to date on aviation history these engines in the form of turbojets were actually conceived by these great aviation pioneers during the early to middle 1930's. These first developments culminated in flight tests on August 27, 1939, in Germany and on May 15, 1941, in England. Since then the collective efforts of governments, industries, and universities worldwide have made the extensive technological contributions that allow the high levels of performance now being achieved. After the initial developments and flight testing of the turbojet engines, the United States took on a lead role in furthering the development of the gas turbine engine in its various forms for a wide range of aircraft applications. In recent years, with nearly half a century of research and development behind us, some have felt we were reaching a technological plateau. I will take this opportunity to challenge and dispel that thought for those who feel that to be the case. I believe that significant technological advances are yet to be made. In some cases these advances, if achieved, could be considered as breaking technological barriers, with potential payoffs making them revolutionary in nature. I realize that most people view the advances in turbomachinery technology as evolutionary, and to a large extent they have been.

Coupled with the potential to significantly impact the present state of compressor technology are the strong needs and unique opportunities to do so. Historically, the technology has been driven by a need for enhanced capabilities to meet the requirements of advanced aircraft missions. As we approach the 21st century, advanced aircraft missions with demanding propulsion system requirements are being considered for both subsonic and supersonic flight. Examples are ultra-high-bypass-ratio turbofan engines for advanced subsonic transports and advanced supersonic propulsion systems for long-range supersonic transports.

The purpose of this paper is first to briefly review the most significant technological advances to date and then, with history as a background, to project a bright view of the future that reflects the need and plans to further advance compression system technology. In tracing the history of compressor development for aircraft propulsion systems, selected examples are noted to help portray the advances in technologies that have taken place. The examples chosen were limited to those available in the open literature. Many different examples exist, but out of the need for brevity, only a few were chosen. If I have left out your favorite examples, I apologize.

In looking to the future I discuss some concepts now being studied at the Lewis Research Center that I believe have the potential for significantly impacting the design and performance of future compressor and engine systems for selected aircraft applications. One such concept being studied for supersonic flight is the processing of the flow supersonically through the fan stage at supersonic flight speeds, thus reducing the need for a long, heavy supersonic inlet.

The status of transitioning from an empirically derived design system to a computationally oriented system is highlighted and includes an experimental program being pursued at the Lewis Research Center to enhance this process. And last, I present a view of the future in which I believe there will be a strengthening of the synergisms between design systems and designers of the various turbomachinery components, whether it be pumps or turbines for rocket engines; compressors, fans, propellers, or turbines for air-breathing engines; axial, mixed flow, or radial; single stage or multistage; large or small.

The coalescence of several technologies has normally resulted in the largest overall advancements in the various components of the gas turbine engine. This has been especially true for the compressor. However, this paper emphasizes the advances in compressor aerodynamics, both steady and unsteady, since the major improvements in compressor performance have come about through advances in the aerodynamic design. In comparison, the turbine advancements have come about primarily from the application of advanced high-temperature materials coupled with advances in turbine cooling technology.

HISTORICAL PERSPECTIVE

A historical perspective on the design and development of compressors for aircraft gas turbine engines must start with the Whittle and von Ohain engines. It is interesting to note that Whittle's 1930 patent disclosure (Fig. 1) showed two axial-flow compressor stages followed by a centrifugal stage, a configuration common in the smaller gas turbine engines of today. However, in his 1939 patent disclosure (Fig. 2) he showed a double-entry, single-stage centrifugal. His W-1 and W-2 engines incorporated the double-entry concept. A cutaway view of a General Electric early prototype turbojet based on Whittle's W-2 design is shown in Fig. 3. The double-entry impeller can be seen in the figure. Tests of Whittle's W-1 engine began in 1941, and on May 15, 1941, he brought Britain into the jet age when his W-1 engine propelled the Gloster E.28/39 aircraft on its first flight.

Dr. von Ohain designed and built three different engines, culminating in the He.S.3B engine, which on August 27, 1939, propelled the He 178 aircraft on the world's first jet-propelled flight. This engine incorporated a single-stage centrifugal. Two years later he developed the He.S.8A engine for the He 280 aircraft, the first jet fighter aircraft. A cutaway view of this engine is shown in Fig. 4. The compressor consisted of a single axial-flow rotor followed by a centrifugal stage. Later, he

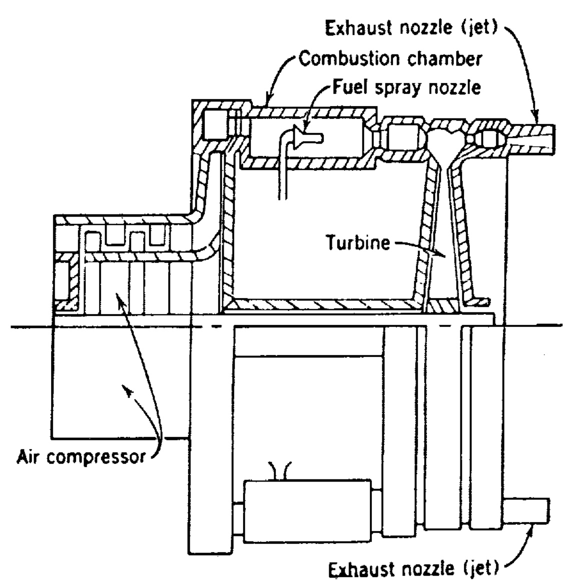


Figure 1. – 1930 Patent disclosure by Whittle.

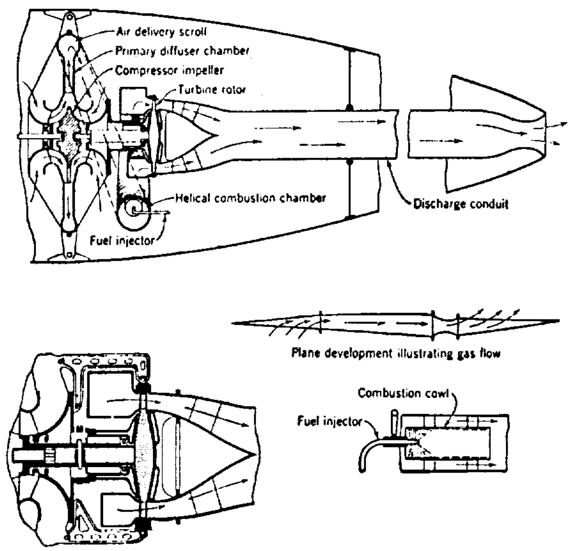


Figure 2. - 1939 Patent disclosure by Whittle.

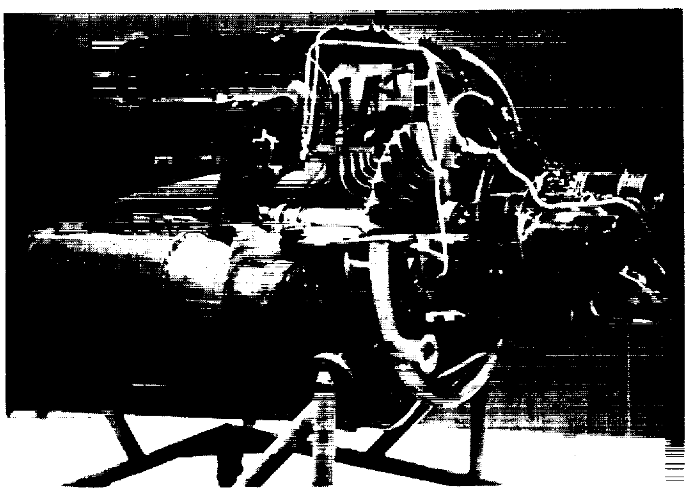


Figure 3. - General Electric prototype turbojet based on Whittle's design.

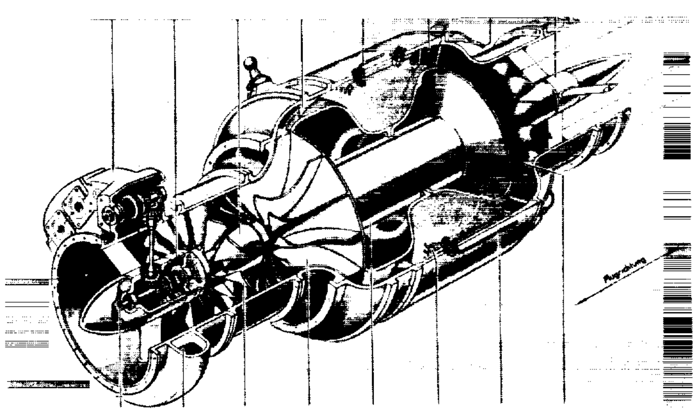


Figure 4. - Cutaway view of He.S.8A turbojet engine.

developed the He.S.011 axial-flow engine, which at the end of World War II was considered the world's most powerful jet engine.

An excellent in-depth review of the development of the gas turbine engine for aircraft applications is given by Dr. von Ohain (1).* Included is a review of the compression system developments. The pressure ratios for these early engines were in the 3 to 4 range. Early use of the centrifugal stages capitalized on the technology developed from turbo-supercharger research for aircraft piston engine applications. The double-entry centrifugal concept allowed for increased airflow per unit frontal area over a single-entry design. This increased airflow of course was desired to minimize engine-related drag as a means of meeting the ever-increasing demand for greater aircraft speed. The addition of axial-flow stages ahead of a centrifugal stage not only increases the overall pressure ratio, but it also provides an increase in airflow per unit frontal area. The single axial-flow rotor (no stator) in von Ohain's He.S.8A engine was probably incorporated to allow for the use of axial-flow blade shapes in the inducer section of the compression system. The axial-flow blading provides greater aerodynamic design control over the blade span than can be achieved in an integral inducer-impeller configuration. This could have permitted higher inlet flow Mach numbers and thus increased flow per unit annulus area. The incorporation of an axial-flow rotor also permits reducing the aerodynamic loading in the impeller inlet region, which could have provided aerodynamic performance benefits. The incorporation of an axial-flow rotor without a stator would not significantly increase the overall pressure ratio other than that which would result from any improvement in overall compression system efficiency.

The potential benefits of an all-axial-flow compressor as a means for maximizing airflow per unit frontal area were apparent to the early engine designers. However, the technology base for designing multistage axial-flow compressors to achieve the desired overall pressure ratios with an acceptable number of stages and overall efficiency was just emerging in the middle to late 1930's, when the prototypes of the first turbojet engines were designed. Earlier attempts to develop multistage axial-flow compressors left much to be desired in aerodynamic performance. A large number of stages were required to achieve the pressure-producing capabilities of a single centrifugal stage; efficiencies tended to be low; and the stable flow range was marginal. The turbine, however, with its favorable pressure gradient, was achieving good aerodynamic performance in both axial and radial designs. At that time it was recognized that the critical technologies for the turbine were advanced materials and turbine cooling. This remains basically true today as mentioned earlier. It is interesting to note that Whittle included the possibility of using silica-ceramic turbine blades in his 1930 patent disclosure. Now, nearly 60 years later, exten-

*Numbers in parentheses designate references at end of paper.

sive research is being pursued to actually realize the potential benefits of uncooled ceramics in the hot-section components of gas turbine engines being developed for automobiles as well as those being developed for aircraft applications. The subject of turbine materials was amply covered by Dr. Blake Wallace in the 1988 Cliff Garrett Award Lecture (2). For the compressor, with its adverse pressure gradient, aerodynamic performance was the critical technology in the early development of compressors for aircraft gas turbine applications, and it also remains true today. This is not to say that other technologies related to turbomachinery are unimportant. And as noted earlier, in general it is the merging of the various advances in technologies that has resulted in the greatest performance gains.

An important earlier application of compressor and turbine technology was ground power generation. A historical review of activities to develop compressors and turbines for ground power application prior to the development of the first aircraft gas turbine is given in Ref. 3.

One of the first persons to explore the use of axial-flow turbomachinery for aircraft gas turbines was Herbert Wagner of Germany. He started his studies in 1934. Neither von Ohain nor Whittle was aware of Wagner's activities at the time. In the late 1930's Helmut Schelp of Germany pursued the axial-flow turbomachinery concept, and in the early 1940's Anselm Franz, also of Germany, led the development of the first all-axial-flow turbojet, the Jumo 004. The axial-flow compressor for this turbojet is described later.

Since the early 1940's, with the evolution of the jet age, a strong continued need for improved performance has pushed the technology of the gas turbine engine for advanced aircraft applications. This need has been driven primarily by the highly competitive nature of the lucrative aircraft market and the desire of nations to gain or maintain military air superiority. The early research focused on component and systems technologies for turbojets. Following shortly was an emphasis on turboshaft engines for propeller-driven aircraft. Later the emphasis turned to turbofan engines and smaller turboshaft engines. After the development of the first turbojet engines, axial-flow turbomachinery tended to be the predominant type used in the larger engines with the smaller engines utilizing axial/centrifugal and staged centrifugal arrangements.

Extensive research was conducted on the compression systems of these various engines to pursue the technology that would permit designing for higher overall pressure ratios, increased stage pressure ratio and thus fewer stages, higher efficiency, lighter weight, and improved stable operating range. The ability to avoid compressor stall and surge and the aeroelastic phenomena of blade flutter and forced response were critical to extending engine life.

To provide a perspective on the trends in research and the technologies pursued in the compression system area over the years, the work at NACA/NASA is highlighted. This work is considered reflective of the work pursued within other government agencies and in industry and the universities within the United States and worldwide.

COMPRESSOR RESEARCH AT NACA/NASA - The initial compressor research by the National Advisory Committee for Aeronautics (NACA), predecessor to the National Aeronautics and Space Administration (NASA), was directed toward applying turbosuperchargers to reciprocating aircraft engines. The work commenced in the mid-1930's and continued into the 1940's. The early work was conducted at the Langley Memorial Aeronautical Laboratory of NACA, Langley Field, Virginia. In addition to the centrifugal compressor research for superchargers, extensive cascade tests were begun in the late 1930's to provide advanced blade shapes for axial compressors. The responsibility for continuing the compressor research was transferred to the newly formed Aircraft Engine Research Laboratory of NACA located at Cleveland, Ohio (now the Lewis Research Center of NASA) in the early 1940's. At the Aircraft Engine Research Laboratory, extensive research was conducted in order to improve overall performance and to understand the various aspects of the unsteady flows encountered in compression systems for superchargers and aircraft gas turbine engines. Both centrifugal- and axial-flow stages were included in these studies. Much of the unsteady-flow research effort was concentrated on understanding compressor stall and surge and associated aeroelastic effects. This work was continued into the mid-1950's. During this time Lieblein developed the aerodynamic loading parameter referred to as "D-factor," and Hartmann developed his shock loss model for transonic compressors. The results of the axial-flow compressor research effort at NACA prior to 1956 are summarized in Ref. 4. The work of others published in the open literature prior to 1956 is also noted in this reference.

With the forming of the National Aeronautics and Space Administration in 1958, NASA turned its attention toward rocket propulsion research, and a concerted effort was made at the Lewis Research Center to study the flows in pumping machinery with emphasis on cavitating flows and associated system dynamics.

During the 1960's NASA Lewis renewed its effort to further advance the technology of aircraft propulsion systems. The compressor research was initially focused on improving design-point performance. Advanced transonic blade shapes such as the multiple circular arc (MCA) were conceived to permit operation at higher rotative speeds. Much of the effort was directed toward fan stages. Investigations were conducted to study the effects of inlet distortion on stall margin. Casing treatment was conceived and developed as a means for improving stall margin.

In the early 1970's the research effort was directed toward advancing the technology for core or high-pressure compressors. Higher aerodynamic loading was investigated along with lower-aspect-ratio blading. Improved blade shapes to accommodate the endwall flows were studied. Computational methods were implemented in the design process for subsonic blading. Research programs to quiet the engines through improved modeling of the noise generated within the turbomachinery were pursued.

The results of the aircraft-propulsion-related research effort at Lewis from the mid-1960's through the 1970's are summarized in Refs. 5 to 8.

As the overall technology of the aircraft and engine systems was being advanced through application of the research conducted in the 1960's and early 1970's, problems associated with unsteady flows were being encountered with increasing frequency in engine development and flight qualification programs. Solving the problems associated with unsteady flow phenomena, such as the excessive loss of stall margin with inlet distorted flows, blade flutter within the operating range, and forced-response blade excitations causing premature failures, began to consume more and more of the total development cost of new engine systems. In an attempt to reverse this trend, NASA, the Department of Defense (DOD), academia, and U.S. industry joined forces with the professional societies and the NATO community through AGARD to organize specialists conferences, panels, and working groups to treat various aspects of the unsteady-flow problem (9 to 14).

In the mid-1970's it was generally accepted by those working in the turbomachinery field that further advances in the technology of aircraft turbomachinery systems would depend largely on being able to more accurately model and properly account for the unsteady flows and their effects in the design process. This concern was highlighted by Platzer in his technical evaluation report for the AGARD conference on unsteady phenomena in turbomachinery (12, pp. ix-xvi) and by Mikolajczak in his paper titled "The Practical Importance of Unsteady Flow" (12, pp. 1-1 to 1-12).

Because of these concerns NASA and DOD jointly began a program to address aeroelastic effects with a focus on flutter research. Also, during this time increased emphasis was given to advancing the technology of compressors for small engines. Fundamental experiments for modeling and code verification were pursued.

In the late 1970's increased emphasis was given to fundamental experiments and the study of highly loaded, high-tip-speed multistage compressors. Work was continued on the aeroelastic effects.

The research from the early 1980's to date has been focused on obtaining detailed time-dependent measurements of the internal flow within the blade passages of high-speed axial and centrifugal stages as well as within large low-speed machines, on validating advanced computational methods, and on continuing the flutter and forced vibration research. A program to improve the understanding of stall and surge for high-speed, high-prèssure-ratio multistage compressors was begun, and advanced analytical concepts are in various stages of being developed.

Additional in-depth treatments of compression system research and development for aircraft applications can be found in Refs. 15 to 17. In the following discussions selected examples are used to highlight axial and centrifugal compressor design and development to date.

AXIAL-FLOW COMPRESSOR EXAMPLES - A cross section of the first turbojet that used an axial-flow compressor is shown in Fig. 5. The engine was designated the Jumo 004. The compressor was composed of eight stages, produced a pressure ratio of 3, and achieved an adiabatic efficiency of 78 percent (polytropic efficiency of 81 percent). The design tip speed is unknown but was probably under 305 m/sec (1000 ft/sec). The stage reaction for this machine was 100 percent (i.e., all of the static pressure rise occurred in the rotor blades). A free-vortex design was used as was typical of the early German design approach. The high-reaction design inherently resulted in lower efficiency because of the associated high relative Mach numbers for the rotor blades. The airfoil blade shapes available at that time were not optimum for these relatively high Mach numbers. Dr. von Ohain in his design of the axial compressor for his He.S.011 engine incorporated symmetrical diagrams and thus 50 percent reaction in an attempt to reduce rotor blade relative Mach numbers and increase the efficiency. It was not until the development of high-Mach-number blade shapes with thin leading edges, such as the double and multiple circular arc shapes, that the general design approach for multistage axial-flow compressors moved back toward higher reaction vector diagrams.

An early Westinghouse turbojet engine with an axial-flow compressor is shown in Fig. 6. The compressor had 10 stages and was designed to produce a pressure ratio of 4.4 at a tip speed of 335 m/sec (1100 ft/sec). At this speed, inlet guide vanes were needed to impart a swirl in the direction of rotation and thus keep the relative velocity to the rotor subsonic in order to minimize the losses and maximize efficiency.

The first United States turboprop engine was the General Electric TG-100. It had 14 stages and produced a pressure ratio of 5.1 at a

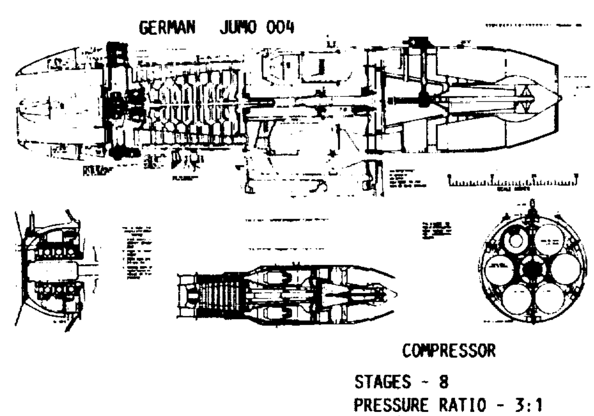


Figure 5. - First turbojet using axial-flow compressor.

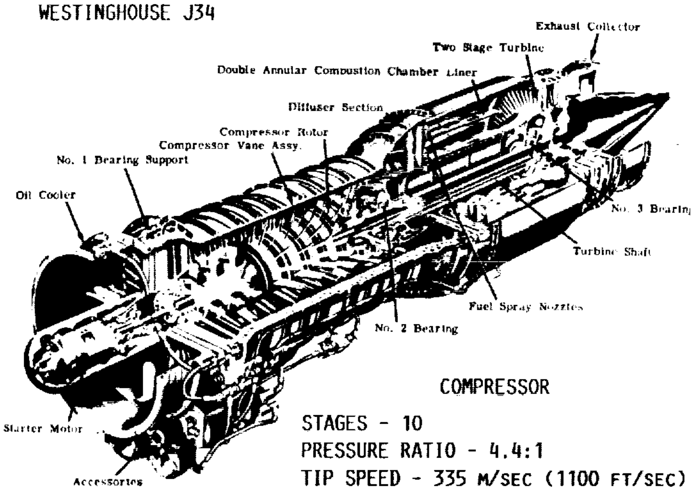


Figure 6. - Early United States axial turbojet.

tip speed of 285 m/sec (935 ft/sec). The compressor rotor for this engine is shown in Fig. 7.

The multistage compressor that first demonstrated the feasibility of operating at transonic flow conditions is shown in Fig. 8. It is the NACA eight-stage, axial-flow compressor designed for a pressure ratio of 10.3 and an inlet tip speed of 356 m/sec (1168 ft/sec). The work was conducted at the NACA Lewis Flight Propulsion Laboratory (now NASA Lewis Research Center) during the early to middle 1950's. This pioneering research provided a major breakthrough in technology for multistage compressor design and performance and has contributed significantly to the high performance of present-day turbine engines.

Only the first two stages of the NACA eight-stage compressor operated with transonic flow. The last six stages operated subsonically. This was achieved through employing high reaction (nonsymmetrical velocity diagrams) in the first two stages and 50 percent reaction (symmetrical diagrams) in the last six stages.

The compressor produced an overall pressure ratio of 10.2 (average stage pressure ratio of

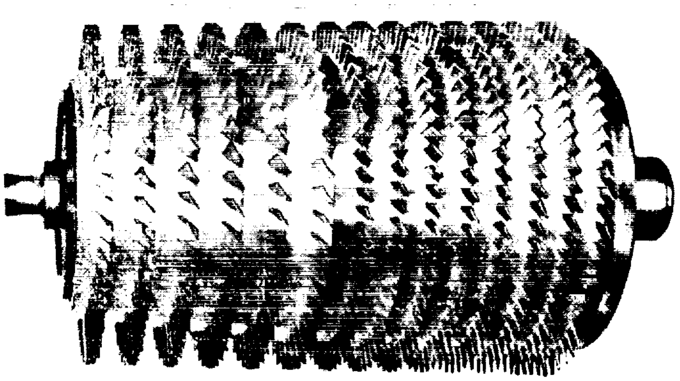


Figure 7. - Compressor rotor for General Electric TG-100 turboprop engine.

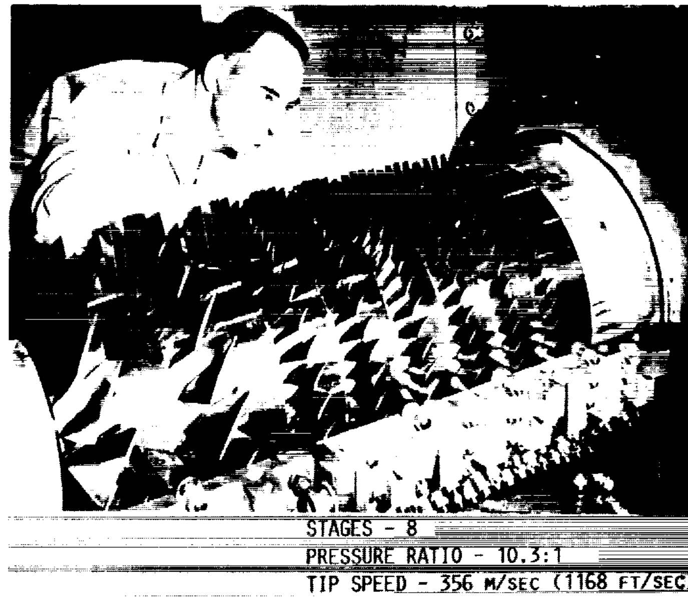


Figure 8. - NACA eight-stage axial-flow transonic compressor.

1.34) at an adiabatic efficiency of 83 percent (polytropic efficiency of 87.5 percent) and a flow of about 146 kg/(sec·m^2) (30 lb/(sec·ft^2)) of frontal area. These values represent about an 11 percent increase in average stage pressure ratio and about a 20 percent increase in flow per unit frontal area over those obtained with subsonic stages. The measured performance map for the eight-stage compressor is shown in Fig. 9. Extensive research was conducted on this compressor. This extensive data base fostered numerous transonic compressor designs for aircraft applications.

The Energy Efficient Engine (E^3) Program was begun by NASA in the late 1970's and extended for approximately 5 years. The objective was to demonstrate significant reductions in fuel consumption for advanced high-bypass-ratio turbofan engines. Two contracts were awarded, one to Pratt & Whitney and one to General Electric, (1) to design, build, and test the components separately and (2) to test an integrated-core low spool (ICLS).

A cross section of the Pratt & Whitney engine is shown in Fig. 10. The high-pressure compressor had 10 stages and was designed to produce a pressure ratio of 14 at a tip speed of 379 m/sec (1243 ft/sec) (18). The compressor achieved the design pressure ratio of 14 at design speed and demonstrated an adiabatic efficiency of 85.7 percent (polytropic efficiency of 89.8 percent).

A cross section of the General Electric engine is shown in Fig. 11. The high-pressure compressor for this engine was designed to achieve a pressure ratio of 23 in 10 stages at a design inlet tip speed of 456 m/sec (1495 ft/sec) (19). Design weight flow and pressure ratio were exceeded at design speed. The measured peak adiabatic efficiency at design speed was 84.5 percent (polytropic efficiency of 89.5 percent). This is considered excellent for such a high overall pressure ratio.

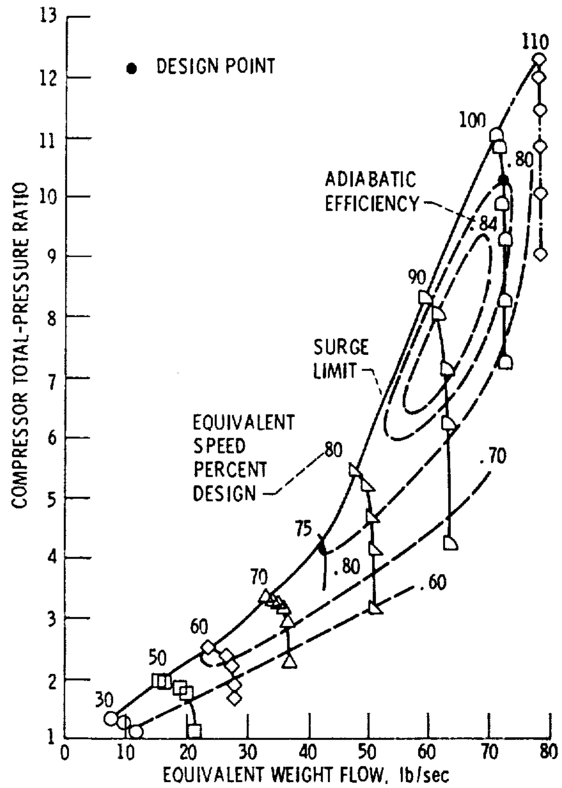
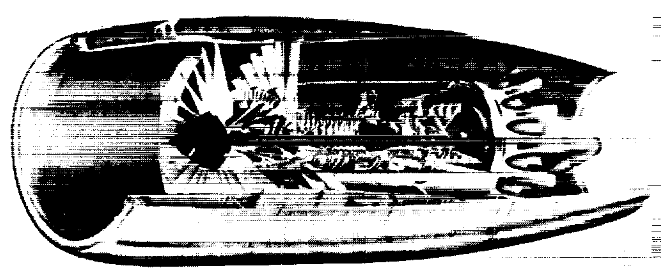
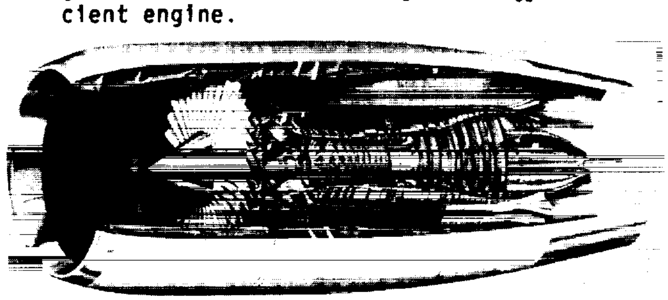


Figure 9. - Overall performance of modified eight-stage axial-flow research compressor.



H.P. COMPRESSOR
STAGES - 10
PRESSURE RATIO - 14:1
TIP SPEED - 379 M/SEC (1243 FT/SEC)

Figure 10. - Pratt & Whitney's energy efficient engine.



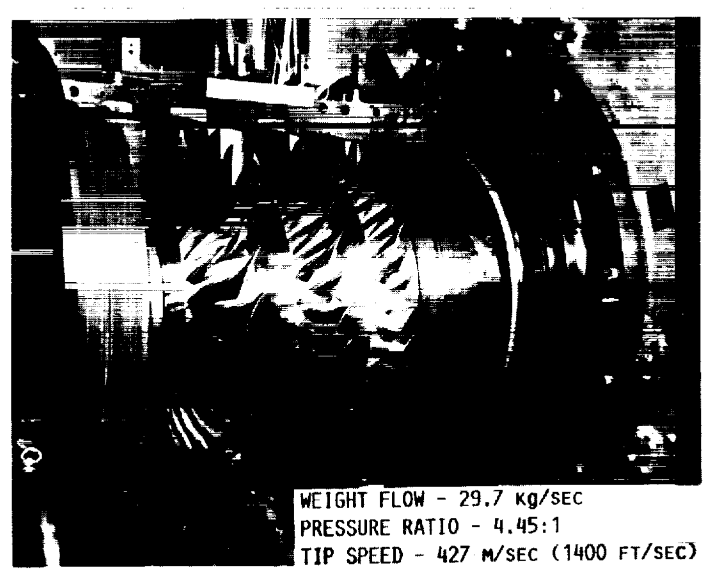
H.P. COMPRESSOR
STAGES - 10
PRESSURE RATIO - 23:1
TIP SPEED - 456 M/SEC (1495 FT/SEC)

Figure 11. - General Electric's energy efficient engine.

Projected polytropic efficiencies for both E^3 compressors exceeded 90 percent when the measured performance was corrected for Reynolds number differences between test conditions and engine operating conditions and the estimated loss in performance associated with the extensive intrusive instrumentation was accounted for.

The E^3 compressors reflect modern-day technology and represent a significant advance over the earlier compressors in terms of overall pressure ratios, tip speeds, and the number of stages required to achieve a given pressure ratio.

A NASA research compressor is shown in Fig. 12. It represents a modern-day core compressor inlet stage group. It was designed to produce a pressure ratio of 4.45 in three stages at an inlet tip speed of 427 m/sec (1400 ft/sec) (20). The compressor achieved the design pressure ratio at the design-speed peak efficiency of 86.5 percent (polytropic efficiency of 88.4 percent). The part-speed adiabatic efficiency reached 90 percent. The compressor was originally designed for relatively high blade aspect ratios for the aerodynamic loading levels that we were trying to achieve, and the demonstrated performance was low. The design was modified to incorporate low-aspect-ratio blading. Changes were also made in the design loss correlations, deviation angle correlations, blockage allowances, axial velocity diffusion through the blading, and stage loading distribution. These changes were made based on rapidly emerging data being obtained in experiments on high-tip-speed, highly loaded, single stages. The changes resulted in tremendous improvement in performance. It is thought that lowering the blade aspect ratio probably had the greatest impact on performance. The effect of aspect ratio on the performance of highly loaded compressor stages was addressed by Dr. Arthur Wennerstrom in his 1986 Cliff Garrett Turbomachinery Award Lecture (21). A more complete background on the development of this compressor can be found in Ref. 17.



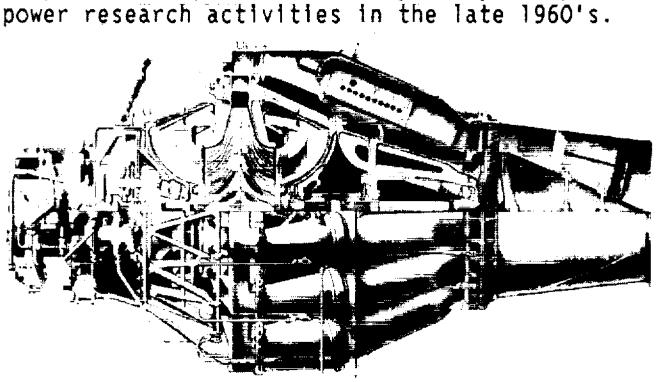
WEIGHT FLOW - 29.7 KG/SEC
PRESSURE RATIO - 4.45:1
TIP SPEED - 427 M/SEC (1400 FT/SEC)

Figure 12. - NASA research compressor.

In summary, we have seen the impact of technology on the design and development of multistage axial compressors in comparing the first axial-flow turbojet, which had eight compressor stages and produced a pressure ratio of 3, with the General Electric E^3 compressor, which achieved a pressure ratio of 23 in 10 stages, and a NASA research compressor, which achieved a pressure ratio of about 4.5 in three stages. At the same time the polytropic efficiency increased from about 80 percent to 90 percent. The higher stage pressure ratio with improved overall efficiency came about primarily from the development of highly efficient transonic blade shapes such as the multiple circular arc (MCA) and the controlled diffusion (CD) airfoils, low-aspect-ratio blading, and higher wheel speeds, along with the benefit of an extensive empirical data base derived from a multitude of single- and multistage experiments. The later designs also benefited from the application of computational fluid mechanics in the design process, such as that required to design CD or similar airfoils.

CENTRIFUGAL COMPRESSOR EXAMPLES - As noted earlier, the first turbojet engines, designed by Whittle and von Ohain, employed centrifugal compressor stages. The first turbojet engine to propel a United States aircraft was built by General Electric and was based on the Whittle design. A later version of the General Electric basic design (Fig. 13) was designated the I-40. This engine is also known as the J33. It produced a pressure ratio of 4.1 with an impeller tip speed of 494 m/sec (1620 ft/sec).

The Garrett TPE 331-14 turboprop engine (Fig. 14) is a typical small engine configuration. It employs a two-stage centrifugal compressor and produces an overall pressure ratio of 8.0. Its design flow is on the order of 2.7 kg/sec (6 lb/sec). An earlier version of this engine employed radial-bladed impellers. With advances in structural design capability the engine was modified to accommodate a swept-back design with improved efficiency. NASA studied a scaled version of the second stage of this compressor as part of the Brayton-cycle space power research activities in the late 1960's.



COMPRESSOR

STAGES - 1
PRESSURE RATIO - 4.1:1
TIP SPEED - 493 M/SEC (1620 FT/SEC)

Figure 13. - Cutaway view of General Electric's I-40 turbojet engine.

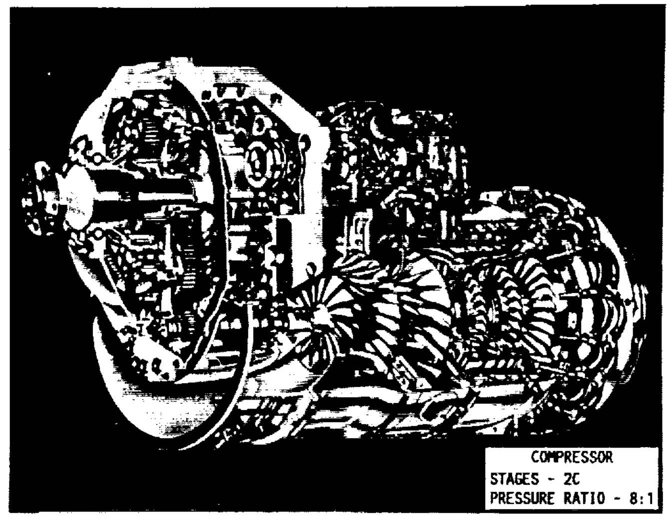


Figure 14. - Garrett TPE 331-14 turboprop engine.

The General Electric T-700 engine is shown in Fig. 15. It employs five axial stages followed by a centrifugal compressor and achieves an overall pressure ratio of 17 with an axial compressor inlet tip speed of 457 m/sec (1500 ft/sec). The design flow for this engine is 4.5 kg/sec (10 lb/sec).

A cutaway view of the T800-LHT-800 developed by the Light Helicopter Turbine Engine Company, a partnership between Garrett Engine Division of the Allied-Signal Aerospace Company and the Allison Gas Turbine Division of the General Motors Corporation, is shown in Fig. 16. It employs a two-stage centrifugal compressor that represents the state-of-the-art technology for small engines. The design values for this engine are not yet available in the open literature.

In summary, the impact of technology on centrifugal compressors is apparent in comparing the earlier designs with those incorporating the latest technology. The increase in overall pressure ratio has came about (1) through increases in impeller tip speed, thus increasing the stage pressure ratio, and (2) by going to staged arrangements. In the smaller flow sizes the two-stage centrifugal arrangement is often utilized,

Figure 15. - General Electric T-700 turboshaft engine.

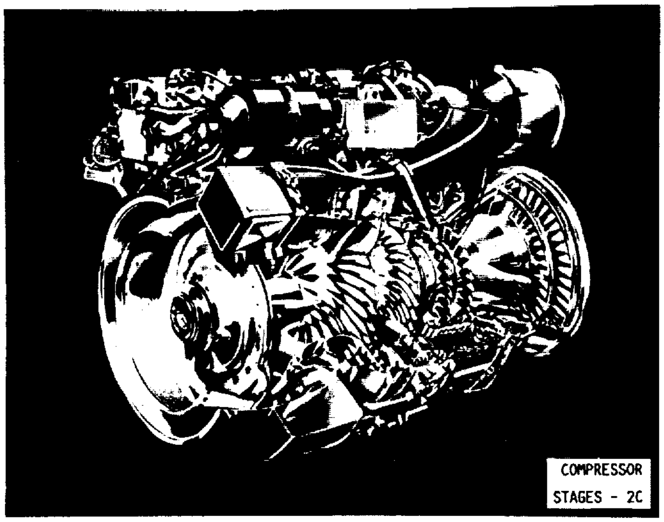


Figure 16. - Light Helicopter Turbine Engine Co. T800-LHT-800 turboshaft engine.

and in the larger flow sizes axial-flow stages preceding a centrifugal tends to be the preferred configuation. These trends came about through consideration for maximizing the overall efficiency as well as the overall pressure ratio. Significant improvements in polytropic efficiency, on the order of 5 to 7 percentage points, have accompanied the increase in overall pressure ratio. The efficiency improvements to a large extent are attributed to high degrees of sweepback in the impeller blading and to improvements in the vaned diffusers. As the sweep is increased, more of the static pressure rise occurs in the impeller and this lowers the Mach number entering the diffuser vanes. It was the advances in structural analysis coupled with the development of higher strength materials that have allowed going to the high degrees of sweep. Significant increases in stable operating range have accompanied the changes in sweep, with the peak efficiencies occurring well away from the stall/surge line. Both staged centrifugals and axial/centrifugal staging arrangements are prevalent in the smaller engines of today.

A VIEW OF THE FUTURE

In looking to the future, two examples of advanced concepts will be discussed, followed by a discussion on transitioning from an empirically derived design system to a computational oriented system. One of the examples reflects on propulsion needs for subsonic transport aircraft and the other involves supersonic throughflow fans for supersonic aircraft.

ULTRA-HIGH-BYPASS (UHB) TURBOFAN ENGINES - NASA is presently conducting studies with the industry to look at advanced propulsion concepts for UHB engines for subsonic transports. Achieving the maximum benefit of going to ultrahigh bypass ratios (BPR>10) requires increasing the cycle pressure ratio. Engines for overall cycle

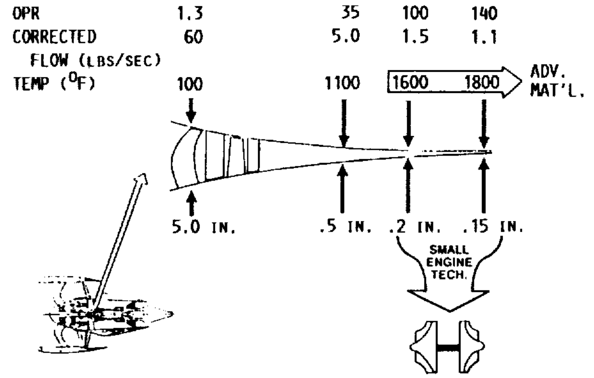


Figure 17. - Impact of very high core overall pressure ratios.

pressure ratios as high as 100 with fan pressure ratios of about 1.3 and bypass ratios as high as 20 are being studied. Figure 17 depicts the problem in going to high overall pressure ratios. At an overall pressure ratio of 35, about where we are at today, the corrected flow is 2.27 kg/sec (5 lb/sec), assuming a core inlet corrected flow of 27.3 kg/sec (60 lb/sec). This results in a blade height of about 1.27 cm (0.5 in.) when assuming an inlet core compressor diameter of 50.8 cm (20 in.) and a hub-tip ratio of 0.5. Assuming constant axial velocity through the compressor and further compression of the flow to achieve an overall pressure ratio of 100 results in a blade height of approximately 0.5 cm (0.2 in.). The corresponding corrected flow is 0.68 kg/sec (1.5 lb/sec). The general rule today is to try to keep the exit blade height on multistage axial compressors greater than 1.25 cm (0.5 in.) for performance and manufacturing reasons.

The corrected flows above an overall pressure ratio of 35 are typical for small engines, where centrifugal stages tend to dominate because of their higher performance potential in the smaller flow sizes. NASA is presently studying configurations to maximize performance for the high overall pressure ratios. One concept, shown in Fig. 18, depicts ducting the flow to an off-shaft, nonconcentric high spool that could be optimized for the small flows to include the use of a centrifugal stage. In UHB engines the fan

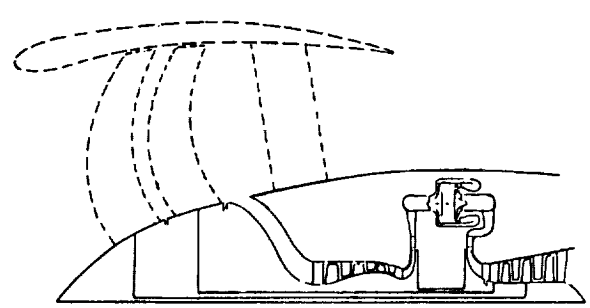


Figure 18. - Advanced concept for ultra-high-bypass-ratio turbofan engines.

dominates the frontal area and there is plenty of room within the core engine cowl to package the nonconcentric high spool. Multiple parallel spools are being considered for ease of packaging. This concept is not new. It has been considered and studied by Teledyne CAE for small engines. In addition to potential performance gains it could also provide for quick removal of the hot-section components. The concept could capitalize on small engine technology research. Even the secondary power technology needs could in part be addressed by pursuing a strong research program focused on the off-shaft UHB engine concept. The importance of secondary power and its technology needs were addressed by Colin Rodgers in his 1987 Cliff Garrett Award Lecture (22). The development of these concepts can benefit from the application of computational fluid mechanics, more effective cooling technology, and high-temperature materials research. If the technology to accomplish this goal is put in place, approximately a threefold increase in cycle pressure ratio will result.

SUPERSONIC THROUGHFLOW FANS - Increased need for more efficient long-range supersonic flight has revived interest in the supersonic throughflow fan as a possible component for advanced high-speed propulsion systems. A fan that can operate with supersonic inlet axial Mach numbers would reduce the inlet losses incurred in diffusing the flow from supersonic to subsonic at the fan face. In addition, the size and weight of an all-supersonic inlet will be substantially lower than those of a conventional inlet. However, the data base for components of this type is practically nonexistent. Therefore, in order to furnish the required information for assessing the potential for this type of fan, the NASA Lewis Research Center began a program in 1986 to design, analyze, build, and test a fan stage capable of operating with supersonic axial velocities from inlet to exit. The objectives are to demonstrate the feasibility and potential of supersonic throughflow fans, to gain a fundamental understanding of the flow physics associated with such systems, and to develop an experimental data base for design and analysis code validation. We believe that the successful demonstration of fans operating with supersonic throughflow velocities could be the impetus to revolutionizing the design and performance of future high-speed aircraft propulsion systems.

Ferri, in 1956, was the first to point out the potential advantages of supersonic inflow compression systems (23). In 1961, Savage, Boxer, and Erwin (24) studied the starting characteristics in transitioning to supersonic inflow. Under U.S. Air Force sponsorship in 1967 (25 and 26), General Applied Science Laboratory (GASL), with Detroit Diesel Allison (DDA) as a subcontractor, and United Technologies Research Center (UTRC) conducted design studies and proposed turbojet engine concepts incorporating supersonic throughflow compressors. Also in 1967, Boxer proposed a high-bypass-ratio turbofan engine/ramjet combination with a variable-pitch supersonic inflow compressor (27). In 1975, Breugelmans conducted the most thorough supersonic throughflow fan experiment to date (28). In 1978, Franciscus presented the results of his first analysis (29) showing significant payoffs of supersonic throughflow fan engines for supersonic cruise aircraft. His later studies continue to support the benefits he showed in his first study (30 to 32).

In moving into the supersonic throughflow regime, where the data base is essentially nonexistent, applying computational methods was considered paramount in executing the design of the supersonic throughflow fan and its associated test facility. It was felt that the application of the codes would greatly enhance the quality of the experiment. Extensive effort was therefore put forth to modify an array of codes to perform design and analysis functions. These codes were then used extensively in the design process.

Figure 19 depicts the NASA supersonic throughflow fan, the facility inlet needed to accelerate the flow to supersonic velocities at the fan face, and the diffuser needed downstream to decelerate the supersonic flow leaving the fan to subsonic conditions. The design fan-face Mach number is 2.0 and the exit Mach number is 2.9. The fan was designed with a constant annulus area to minimize three-dimensional effects in the initial design. The design pressure ratio and tip speed were selected to be representative of those required of a turbofan engine fan operating at supersonic cruise conditions.

An axisymmetric design code was used in the design of the fan to obtain initial blade shapes. The quasi-three-dimensional, thin-shear-layer Navier-Stokes code was used to analyze the design. The design was then adjusted by using the design codes, and the process was repeated until the desired loading distributions and wave patterns were achieved. The Mach number contours for the rotor and stator show that the waves off the leading edge are contained within the bladed passage (Fig. 20). Also, the expansion waves off the suction surface tend to cancel the compression waves off the pressure-surface leading edge, thus reducing the pressure gradient along the suction surface. The strength of the expansion and compression waves at the trailing edge was minimized by controlling the loading near the trailing edge.

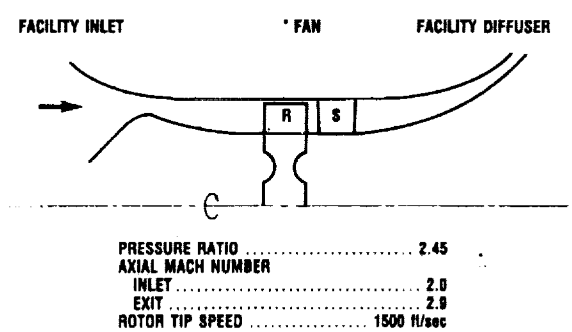


Figure 19. - Supersonic throughflow fan experiment.

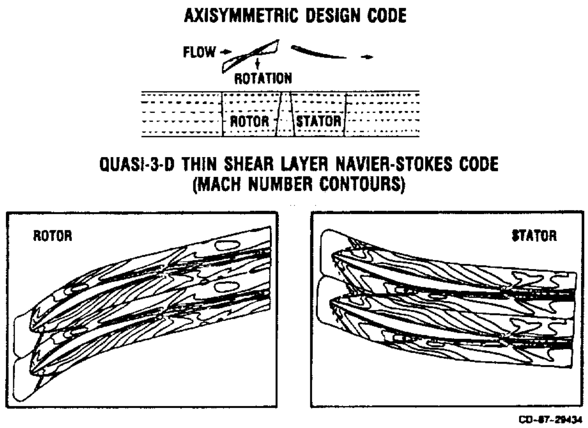


Figure 20. - Supersonic throughflow fan design procedure.

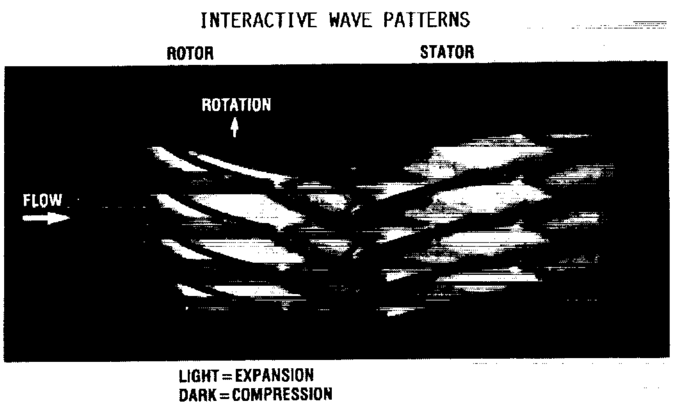


Figure 21. - Three-dimensional unsteady Euler code results.

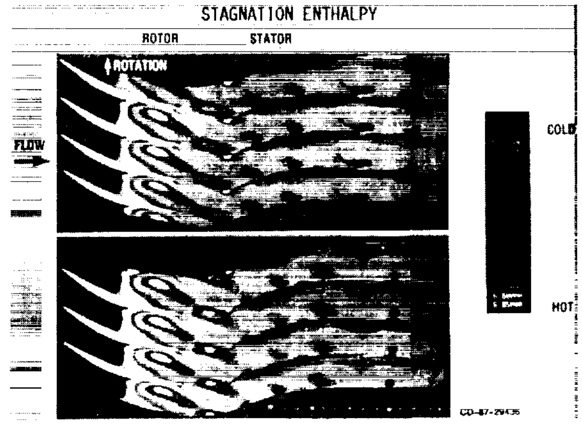


Figure 22. - Supersonic throughflow fan rotor-stator interaction.

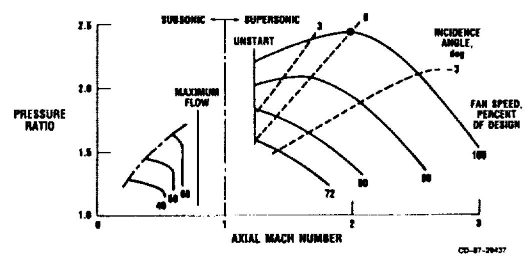


Figure 23. - Predicted fan performance map.

A three-dimensional unsteady Euler code was used to study the rotor-stator flow-field interactions with supersonic throughflow (Fig. 21). Computer graphics were used to obtain the interactive wave patterns for a given index of the rotor relative to the stator. The picture can be thought of as a schlieren photograph, with the light patterns being expansion waves and the dark patterns, compression waves.

The effect of the time-dependent flow fields behind the rotor on the stator flow field can best be seen in Fig. 22. This figure shows the stagnation enthalpy, and thus temperature, for two different indexes of the rotor blades relative to the stators. The interactive wave patterns within and exiting the rotor result in a time-dependent flow field entering the stator. This unsteady flow field relative to the stator appears to result in cyclic movement fore and aft of the stator leading-edge compression wave, which emanates from the pressure surface. The wave motion is nonlinear, with more energy being added when the shock moves forward than is subtracted when the shock moves rearward. Analyses are being continued so that we may fully understand this phenomenon. The cyclic nature of the local temperature is apparent from the difference in the magnitudes of the local white (highest temperature) regions. However, the unsteady aspects of the flow field can best be seen from a motion picture generated from the three-dimensional unsteady Euler code analysis.

The predicted performance map for the supersonic throughflow fan shown in Fig. 23 was derived by using a combination of codes, including the off-design axisymmetric code and the quasi-three-dimensional viscous code. Presenting the performance as a function of inlet axial Mach number results in a performance map similar to subsonic/transonic fan maps.

A layout of the supersonic throughflow fan test package is shown in Fig. 24. The variable-inlet nozzle and the variable downstream diffuser will be used to provide control over the fan-face Mach number and the diffusion of the supersonic fan exit velocities to subsonic conditions entering the exhaust system. Boundary layer bleed capability is provided at the inlet to the fan and the diffuser. The package is now being installed in the test facility and is scheduled for testing late in 1989.

A more detailed discussion of the supersonic throughflow fan design, the facility inlet, and the downstream diffuser is given in Refs. 33 and 34. The codes used in the design and analysis process are described in Refs. 35 to 43. A summary of the design along with a discussion of a unique supersonic throughflow engine concept is presented in Ref. 44.

In summary, we have made extensive use of

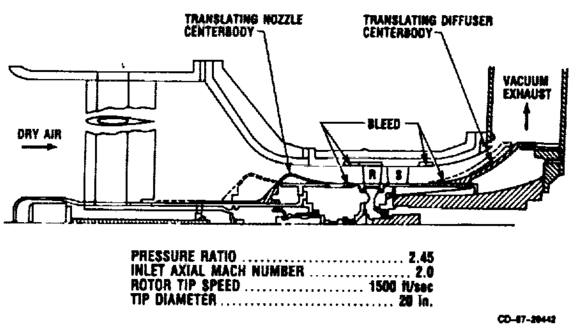


Figure 24. - Supersonic throughflow fan test package.

computational methods in designing the supersonic throughflow fan experiment. We feel this will greatly enhance the quality of the experiment and will contribute heavily to its success.

COMPUTATIONAL FLUID MECHANICS - Computational fluid mechanics is a key technology for further advancing the design and development of compressors for future aircraft propulsion systems. With the rapid increase in computer power over the last several years, coupled with the development of advanced computational methods, significant progress has been made in applying computational fluid mechanics to the study of advanced turbomachinery concepts. An example of this has been the use of advanced codes to design the previously discussed supersonic throughflow fan experiment.

Paralleling the rapid advancements in computer power and computational methods are the advancements in nonintrusive measurements of the internal flow fields using laser techniques. Figure 25 depicts a laser anemometer probing the flow field of a fan stage. A comparison between measured and computed results is also shown. Relatively good agreement was achieved, thereby providing a degree of validation for the computational method, a three-dimensional Euler code. This ability to validate the computational methods is key to applying them with confidence in the design process.

Past compressor design systems and those employed even today are largely based on empirical correlations. The objective of the NASA compressor research program is to enhance the transitioning from the empirically derived design system to a computationally oriented system. The program, depicted in Fig. 26, includes the development of advanced analytical methods in parallel with obtaining detailed flow-field measurements in both axial and centrifugal stages. Experiments are being conducted in large low-speed compressors to get more detailed flow-field measurements in viscous and endwall regions that are too small to measure in high-speed machines. Experiments in high-speed machines complement the low-speed experiments by focusing on the study of compressibility effects that are absent in the low-speed environment. A view of the NASA large low-speed centrifugal compressor is shown in Fig. 27 along with the results obtained from a quasi-three-dimensional thin-shear-layer viscous analysis. A 4.53-kg/sec (10-lb/sec), 4:1 pressure ratio centrifugal will be used to study compressibility effects. In parallel to the centrifugal program, which is aimed at obtaining detailed flow-field measurements within that type of geometry, is a similar program NASA is pursuing in multistage axial compressors. A large low-speed axial is now being fabricated for use in low-speed experiments, and a scaled version of the NASA research compressor shown in Fig. 12 is being fabricated to obtain detailed multistage measurements of internal flow in the high-speed environment.

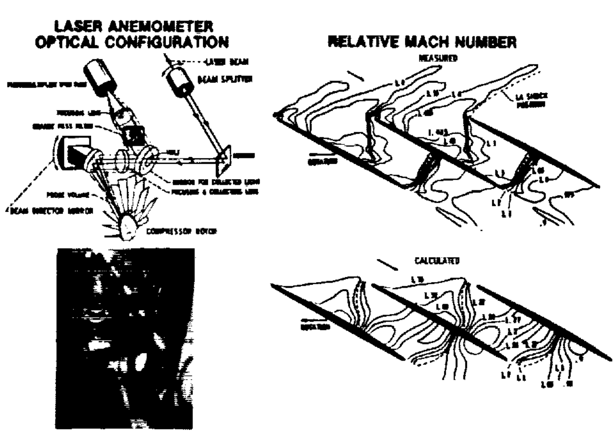


Figure 25. - Advanced methods for compressor flow research.

Figure 26. - NASA compressor technology thrusts.

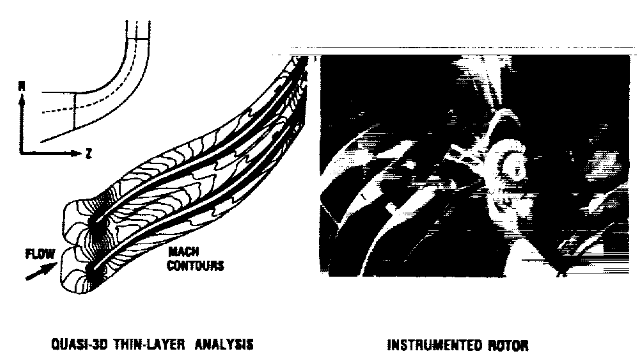


Figure 27. - NASA large low-speed centrifugal compressor.

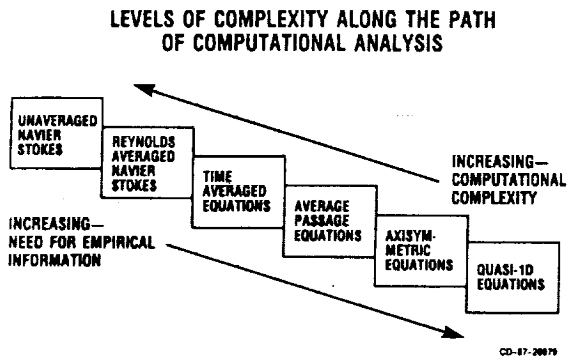


Figure 28. - Design/analysis code hierarchy.

One of the most complicated flow fields to model is that within a multistage axial-flow compressor. The hierarchy of design and analysis codes that can be brought to bear on this problem is shown in Fig. 28. Moving from right to left in the hierarchy results in increasingly complex numerical models that rigorously account for flow physical phenomena and therefore require increased computer resources to generate numerical flowfield predictions. Conversely, moving from left to right in the hierarchy requires the increased use of empiricism, which in turn requires extensive data bases. Present design systems in general have as their basis the axisymmetric throughflow equations.

A recent approach conceived and structured to address the multistage axial-flow compressor modeling problem is the average-passage technique being developed by Dr. John Adamczyk (45). As shown in Fig. 28, Adamczyk's technique represents a move toward less empiricism. A model for closing the inviscid form of the average-passage equation system is given in Ref. 38. This work has the potential of bringing the compressor design system a step closer to a fully viscous design system. Adamczyk's code has been applied to analyzing the flow in fans, a turboprop, a rocket engine fuel pump, and a multistage turbine (38, 39, 46, and 47).

Another code being applied in the analysis of a range of turbomachinery geometries is the quasi-three-dimensional thin-shear-layer Navier-Stokes code developed by Dr. Rod Chima (48). This code has now been extended to full three-dimensional capability and used in the analysis of the supersonic throughflow fan. It has also been used to analyze the flow field within a rocket engine turbine.

The ultimate goal is to be able to compute the performance of advanced concepts in the design phase, to include the calculation of size-related effects in order to assess potential, and to reduce risk in the experimental validation. We have a long way to go, but we have made a good start. The intent of the NASA program is to enhance the transition from the highly empirical axisymmetric design system to a more computationally oriented system.

I believe that the transition to a computationally oriented design system will result in a tightening of the bonds between the various types of turbomachinery designers, thus providing additional inherent advantages. The codes being developed are capable of handling a range of turbomachinery geometries including propellers as shown by examples presented in Refs. 29 to 48. A summary of the Advanced Turboprop Project managed by the Lewis Research Center is given in Ref. 49. A reflection of the impact of computational fluid mechanics on the advancement of propeller technology can be gained from this reference.

SUMMARY

In summary, advancing technologies over the past 50 years have had a significant impact on compressor design and development for aircraft gas turbine engine applications. Looking to the future, computational methods applied to the study of unique concepts will allow for even more advancements, some of which could be revolutionary in nature because of their potential impact on performance.

ACKNOWLEDGEMENT

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16. Abstract

A historical perspective of the impact of advanced technologies on compression system design and development for aircraft gas turbine applications is presented. A bright view of the future is projected in which further advancements in compression system technologies will be made. These advancements will have a significant impact on the ability to meet the ever-more-demanding requirements being imposed on the propulsion system for advanced aircraft. Examples are presented of advanced compression system concepts now being studied. The status and potential impact of transitioning from am empirically derived design system to a computationally oriented system are highlighted. A current NASA Lewis Research Center program to enhance this transitioning is described.

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