RECENT ADVANCES IN THE PERFORMANCE OF HIGH BYPASS RATIO FANS

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Abstract

The single stage, high tip speed fan is one of the most important aerodynamic components in large gas turbine engines, producing 75% of the thrust at both take-off and cruise ratings. During the last 10 years the state of the art has advanced such that we have seen some remarkable improvements in the aeromechanical standard of transonic fan designs. The paper concentrates solely on the aerodynamic performance improvements achieved during this period, all of which have been incorporated into uprated versions of the RB 211 family of production engines. The direction in which fan technology is now proceeding is also discussed.

The presentation divides into three sections:-

1. Describes the special aerodynamic features of the high bypass ratio transonic fan, and presents the significant progress that has been made in fan performance since the original RB 211-22B design.
2. Compares aerodynamic performance data from model and engine size fans which illustrate the standard of test and analysis techniques required to ensure the improvements in fan technology are realised in full scale production engines.
3. Describes the advanced theoretical and experimental techniques now being employed to obtain a better understanding of the loss mechanisms within the high Mach number blading.

It is concluded that linking the classical axisymmetric design approach with the quasi-3D flow observations will result in more realistic theoretical modelling of the complex 3D flow field, enabling further improvements to be made in the performance of the high bypass ratio fan for future engine applications.

Introduction

The high tip speed fan must achieve the required pressure ratio in a single stage, and for large transport aircraft operating in the medium to high subsonic cruise range, fan bypass pressure ratios of 1.55 - 1.80 are required. These levels of pressure ratio can be obtained satisfactorily over the outer two-thirds of the span of modern single stage fans operating at tip speeds between 1350 - 1500 ft/sec. As the hub/tip ratio of the fan is kept low to provide high airflow per unit frontal area, it is necessary to limit the work at the hub to a level substantially lower than at the tip - referred to as graded work fans.

For some applications, such as the General Electric CF6-6 engine, it was decided to bring the hub energy level up to that of the tip by employing a quarter stage on the LP shaft behind the main fan rotor (Ref 1.). Alternatively, as demonstrated in all RB 211 engines, designing the 1st stage of the intermediate core compressor to accept the lower energy level of the fan hub, it is possible to obtain an efficient and stable fan design which satisfies the high overall core stream pressure ratio requirements of the engine cycle.

There are, however, potential aerodynamic problems associated with the design of graded work fans (Ref 2) which must be overcome to obtain the most efficient and stable engine compression system. These are best understood by considering some of the aerodynamic design features of the original RB 211-22B graded work fan design.

During the last 10 years, single stage fan technology has advanced to an extent where we have seen some very significant performance improvements in terms of specific airflow, levels of blade loading and bypass stage efficiency. Employing more efficient blade profiles in the transonic/supersonic Mach number regime has given efficiency improvements at the higher pressure ratio levels that go beyond those previously thought possible.

The classical aerodynamic design/development approach has proved particularly successful in the RB 211 series of fans, and is best illustrated by discussing the detailed fan performance analysis carried out at sea level and altitude cruise ratings on the fully instrumented RB 211-524 performance engine. This work led to a redesigned fan which was specifically tailored to meet the fan performance requirements of the RB 211-524 B4 engines installed in the Lockheed L1011-500 series aircraft.

The main effort is now directed towards further improvement in fan efficiency rather an increased loading. To obtain the necessary aerodynamic improvements will require a better understanding of the flow phenomena and a detailed description of the local blade row inefficiencies. Advanced measurement techniques such as laser holography and laser anemometry are now in use on 211 scale model fan rigs. The quantitative results will be used to improve the theoretical models of the complex 3D flow field, and supplement the classical axisymmetric design approach.

The paper covers all these aspects of fan technology, and is presented in three sections:

1. Describes the special aerodynamic design considerations of the high tip speed fan arrangement, and presents the significant progress which has been made since the original design of the RB 211-22B fan system in 1970.
2. Compares aerodynamic test data obtained from model rig and engine performance demonstrations which illustrate the current standard of technology of fan designs in committed engine projects.
3. Describes the advanced theoretical methods and measurement techniques being employed to further improve the efficiency of the high bypass ratio fans for future engine projects.

NOMENCLATURE

M	= mass flow	lb/sec
p	= static pressure	p.s.i.a.
P	= stagnation pressure	p.s.i.a.
T	= stagnation temperature	^{o}K
ΔT	= temperature rise	^{o}K
U	= blade speed	ft/sec
U_T	= fan tip speed	ft/sec
U_m	= mean hub blade speed	ft/sec
X_N	= net engine thrust	- Lb
R	= fan pressure ratio	
A	= fan annulus area	- in^2

$\frac{M\sqrt{T}}{P}$	= mass flow function
Qa	= non-dimensional mass flow $\frac{M\sqrt{T}}{AP}$
M_N	= relative blade inlet Mach number
N_D	= fan rotational design speed
s.f.c	= engine specific fuel consumption
W	= pressure loss coefficient = $\frac{\Delta P}{P - p}$
W_D	= datum pressure loss coefficient
η_a	= adiabatic efficiency
η_{DATUM}	= datum fan efficiency
η_{max}	= maximum fan efficiency
η_R	= fan rotor only efficiency

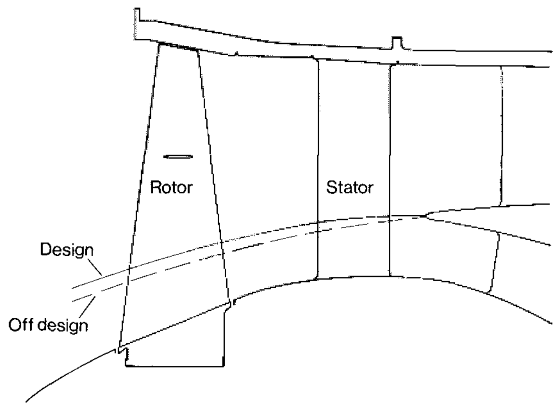

Fig.1 Fan with Remote Splitter

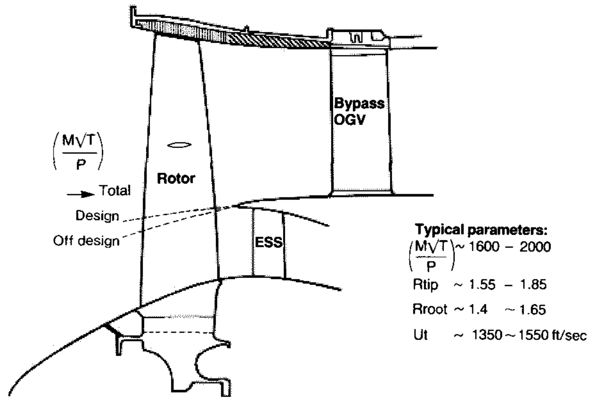

Fig.2 RB 211 Fan Arrangement

1. Aero Design Features of High Bypass Ratio Fans

The simplest type of fan arrangement is shown in Fig.1. Downstream of the stator vanes, where the flow direction is mainly axial, the static pressure is nearly constant from hub to tip. Therefore if the fan rotor has added less energy to the flow in the hub region than elsewhere, the lower total pressure near the hub will result in a lower dynamic pressure behind the stator near the hub. This leads to increased velocity diffusion through the stator, and could lead to a stalling condition at both design and off-design conditions. Due to the tendency for overloading near the stator inner diameter, only small spanwise variations of rotor work can be used with this type of fan arrangement with the consequence that either rather large hub/tip radius ratios or rather high tip speeds must be employed to accomplish the required duty.

In the RB 211 arrangement shown in Fig.2, the high loading difficulty at the inner end of the engine section stator (ESS) where the rotor work is low is partly overcome by the presence of the splitter. The geometry of the splitter is chosen to obtain an annulus area contraction through the hub stator which gives an acceptable bulk axial velocity diffusion across the stator vanes. The spanwise distribution of stator exit swirl angle is arranged to redistribute the static pressure gradient and give a more uniform axial velocity distribution into the core compressor. This feature of 211 fans also avoids high levels of diffusion factor on the hub sections of the core stator.

The bypass outlet guide vanes (OGVs) are placed well downstream of the main fan rotor for noise reasons and are required to discharge the bypass flow in the axial direction. The choice of mean duct entry Mach number is determined by considering the combined diffusion losses of the OGVs and bypass duct.

Factors Affecting Fan Design Configurations

When designing high performance turbomachinery the avoidance of boundary layer separation is a major goal. In the field of compressor design, the diffusion factor limits before boundary layer separation occurs are reasonably well understood, and the approach when designing a novel arrangement like a graded work fan is to shape the annulus walls, and to select the radial work distribution so that these limits are observed.

Early calculations using the axisymmetric flow model of Ref.3 to determine the meridional streamlines indicated that there were three locations in the 211 fan arrangement where careful design control was necessary to obtain acceptable diffusion rates:

1) the fan rotor section directly ahead of the close proximity splitter.
2) on the upper surface of the splitter and through the hub of the bypass OGV.
3) on the hub contour of the fan and through the inner sections of the engine section stator.

Other items of concern identified during early studies relate to the stability of both the bypass and core sections of the fan arrangement during off-design operation, recognising that the bypass ratio increases from 5:1 at cruise to 10:1 at flight idle.

The manner in which these items affected the RB 211-22B fan design are briefly discussed in the following sub-sections.

Overall Design Parameters of the RB 211-22 Fan

Engine thermodynamic cycle optimisation studies carried out to meet the thrust requirements of the initial RB 211-22B engine led to the selection of a by-pass ratio of 5.0 and a fan bypass stream pressure ratio of 1.62 for altitude cruise operation.

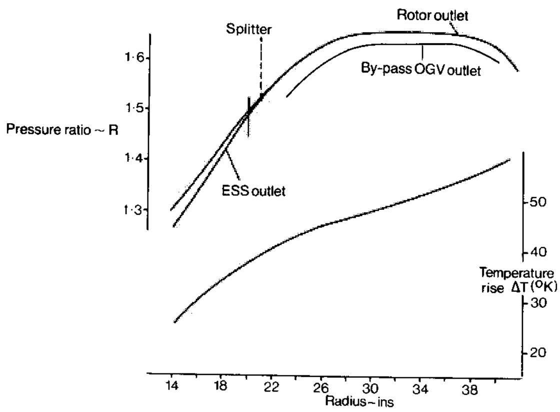

Fig.3 Variation of Radial Total Pressure and Temperature

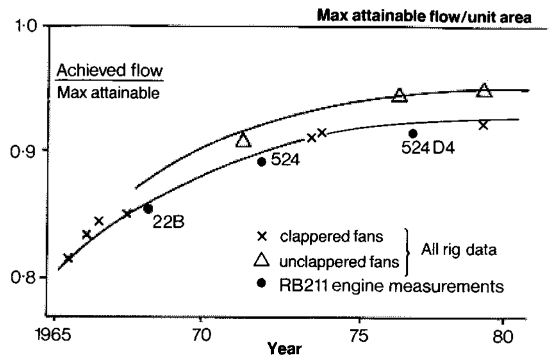

Fig.4 Flow Capacity Achievements

Aerodynamic preliminary design studies aimed at achieving a configuration that would match the cycle requirements, and have a good efficiency potential with adequate surge margin resulted in a fan specification with a nominal tip speed (corrected to 15°C) of 1500 ft/sec, an inlet hub/tip ratio of 0.3, and a specific airflow of 40 lb/sec/ft^2 annulus area.

Concurrent and interrelated aeromechanical studies yielded a fan rotor employing 33 titanium blades having a part-span clapper at 60% blade height.

The choice of these fan performance parameters gave a good basis for further thrust growth, since it was recognised that the 0.3 hub/tip ratio gave the potential of high fan flow/unit frontal area. Also at the nominal design speed the pressure ratio level of the fan bypass and core streams could be increased, without expensive changes to the LP turbine, when uprated t/o thrust levels were required.

Spanwise Work Distribution

The radial distribution of work for the -22B main fan rotor was influenced by the following three considerations; Firstly there was no point in employing a very high loading at the hub of the rotor or the E.S.S. since the following core compressor could handle the varying distribution of total pressure and the 1st stage of the intermediate compressor was designed to restore the required pressure ratio level of the cycle.

Also in the interests of high core efficiency and off-design stability during transient operation, it was desirable to keep a low Mach number and loading on the hub of the engine section stator.

Secondly, the loading at the splitter height was chosen as high as the rotor could take without separation in order to minimise the diffusion loading in the hub of the bypass OGVs. The logic here is the same as described in connection with Fig.1, low total pressure at the inner end of the stator leads to poor axial velocity and subsequent high diffusion loadings. The third consideration was that it would be best to let the work addition drop off somewhat at the outer wall. This reduces the aerodynamic loading on the rotor tip where stalling could initiate, and also gives slightly lower velocities on the outer wall of the bypass duct - reducing duct scrubbing losses.

The distribution of fan rotor and stage exit total pressure and temperature that was finally selected for the RB 211-22B is shown in Fig.3. The anticipated fall-off of total pressure in the annulus wall and splitter boundary layers is not shown; the displacement thickness of these boundary layers was accounted for in the design process, and the losses they cause were partially reflected in the design loss coefficient distributions.

Outlet Guide Vane Configurations

As mentioned previously, the two regions where the achievement of satisfactory diffusion levels presented the greatest challenge are associated with the outlet guide vanes; (1) the inner end of the bypass OGVs, and (2) the inner hub section of the engine section stators.

The selection of rotor work distribution and the choice of annulus wall curvature and area contractions enabled satisfactory designs to be accomplished. The radial distribution of residual swirl aft of the inner stators was also selected to limit the diffusion loadings at the hub of the engine stator, and present a more uniform axial velocity distribution into the following compressor. Employment of this approach led ultimately to a configuration for which the NASA Diffusion Factor of Ref.4 at the inner end of both bypass and core stators was held at 0.55 and for which the absolute Mach number onto the hub of the vanes was no greater than 0.75.

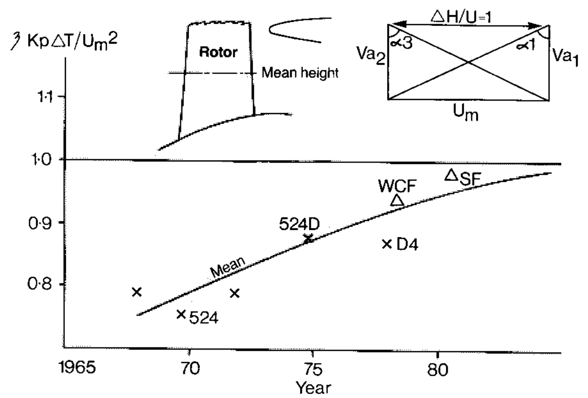

Fig.5 Fan Root Loadings

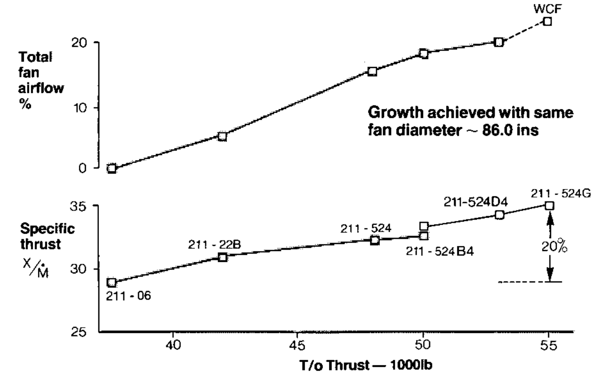

Fig.6 Variation of Specific Thrust and Fan Air Flow

The following sections describe some of the performance improvements that have been achieved during the last 10 years with uprated fan designs of similar geometry.

Improvements in Fan Performance

Since the design of the RB 211-22B fan, uprated versions of the engine have demanded improved levels of specific airflow at increased levels of pressure ratio and aerodynamic efficiency.

Fig.4 shows that to match the RB 211 engine requirements, higher levels of fan airflow/unit annulus area have been achieved by normal development and blade redesigns.

The airflow measurements presented in Fig.4 are obtained from model fan rigs at 0.4 scale operating at tip speeds equivalent to max climb rating of the engine. The corresponding engine airflow measurements are taken from performance tests carried out in the altitude cell at NGTE Pyestock.

The results show that since the original versions of the RB 211 fan design, the inlet mass flow/unit annulus area has increased by 10-12%. The highest levels of fan airflow have been demonstrated by low aspect ratio unclappered fans. In other words, the margin between achieved and maximum attainable airflow has been more than halved in the period considered.

Similarly, uprated engine thrusts have set the requirements for higher fan pressure ratios in both core and bypass streams. Fig.5 presents the increased levels of root loading, in terms of the work function kp $\Delta T/U^2$, that have been achieved since the early versions of the RB 211-22B fan. The 25% extra rotor work increases the absolute swirl and Mach number onto the hub stator. For uprated version of the RB 211 engine, the fan hub contour was changed and improved aerofoil sections designed for the inner stator to withstand the increased loading at all engine operating conditions.

The significance of increasing the maximum fan airflow per unit annulus area and the work level of the fan hub can best be appreciated by considering Fig.6. Since the original version of the RB 211 engine, supercharging the core compressors and rematching the engine cycle for the lower bypass ratios has contributed to the 20% increase in engine specific thrust. Uprated fan designs with an increased maximum airflow capacity of 10-12% at climb, has enabled the airflow at engine sea level take-off rating to be increased to a maximum of 20%

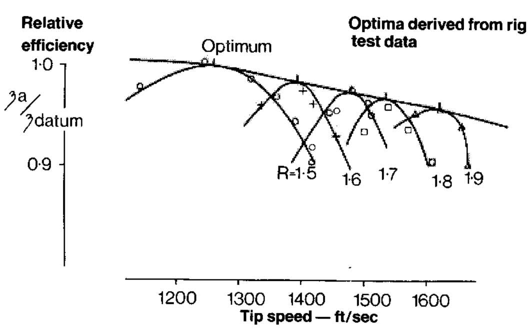

Fig.7 Variation of Rotor Efficiency with Tip Speed

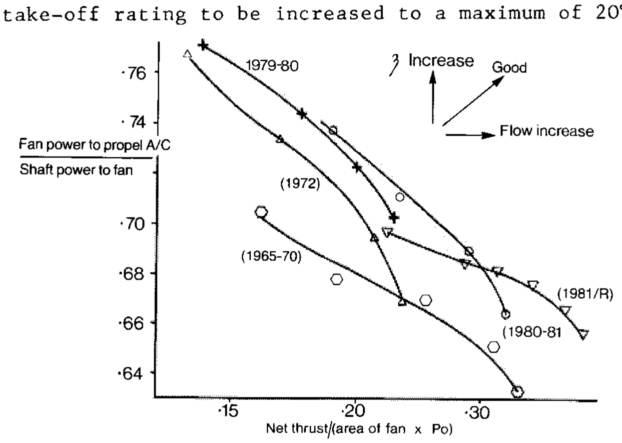

Fig.8 Progress in Fan Cruise Performance

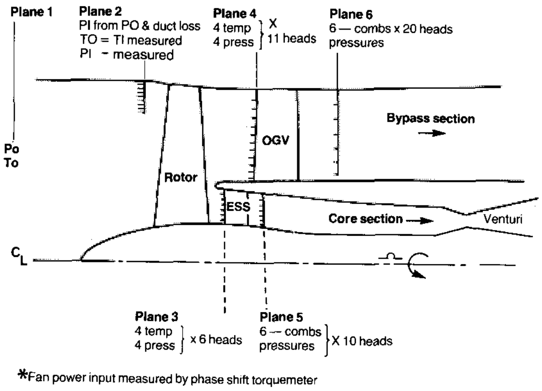

Fig.9 Model Fan Rig - Instrumentation

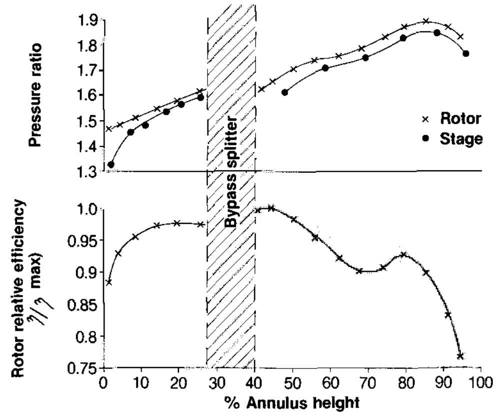

Fig.10 Rotor Exit Flow Conditions

for the RB 211-524D4 engine. The combined effect of these two performance parameters has produced a family of engines with take-off thrusts from 37500-53000 lb, maintaining the same overall pod diameter and engine length chosen for the original RB 211 design.

During the 10 year period, efficiency data have been obtained for graded work fan designs covering a range of tip speeds and blade loadings. Fig.7 presents the relative design speed rotor efficiency for a series of research fan designs and a number of fully developed RB 211 production fans. The performance demonstrations were carried out using representative 0.4 engine scale model rigs, and the efficiencies measured consistently using a shaft torquemeter to determine the overall temperature rise of the fan rotor blade.

The results show that for each level of design pressure ratio there is an optimum tip speed - where the blade profile losses and shock induced losses are a minimum for peak rotor efficiency. Similar efficiency correlations are used during initial engine performance studies to determine the choice of fan tip speed for the required duty. Equally important, from engine considerations the choice of fan tip speed will be influenced by the LP turbine characteristics such that, as in all RB 211 engine projects, the total adiabatic efficiency of the LP system will be optimised. For instance, after development the RB 211-22B fan was rematched to achieve the original 1.62 design pressure ratio at 1425 ft/sec tip speed consistent with the overall LP system requirements.

The trends shown in Fig.7 are for fan designs which have all followed the same general design approach and blade selection procedures employed at Rolls-Royce since 1970. It is recognised that different blading rules such as the choice of design incidence, blade throat area distribution and especially tip solidity will change the level of peak efficiency but the optimum tip speed for a required duty should be unaltered.

The relative levels of fan efficiency presented in Fig.7 for the higher tip speed designs show the progress made in fan technology during the period considered. Comparative fan test results prior to the start of the RB 211 fan performance programme in 1968 would have shown an 8% loss of fan efficiency for designs between 1250 and 1600 ft/sec tip speed. The more recent designs show only 4% efficiency loss for increasing levels of tip speed.

How has the changed situation been achieved ?

The main points are:

1. A better appreciation of the shock patterns at the tip of graded work fans, and the choice of blade geometry which reduces the shock strength at the higher tip Mach numbers.

2. Modified blade profiles and thinner blades to operate more efficiency at higher Mach numbers.

3. Computer throughflow models (Ref.5) which allows the physics of the process to be defined more accurately and still allows the equations of motion to be solved in a finite time. The streamline curvature, boundary layer growth, entropy gradients can be incorporated in the design of each blade row.

4. A better understanding of the effect of tip clearance and inlet velocity profile on the local efficiency of the fan rotor tip sections.

The understanding of the flow processes within the blade row are far from complete, but the results indicate that progress has been made.

How have the improvements in fan technology affected engine performance at altitude cruise ratings ? Complementary to the take-off thrust achievements shown in Fig.6 the improvements in fan efficiency and airflow capacity at typical cruise operation is illustrated in Fig.8. Avoiding the effects of bypass ratio, nozzle area etc. in the 3 shaft engine, a simplified efficiency of the fan can be expressed as the useful propulsive work done by the whole fan blade divided by the

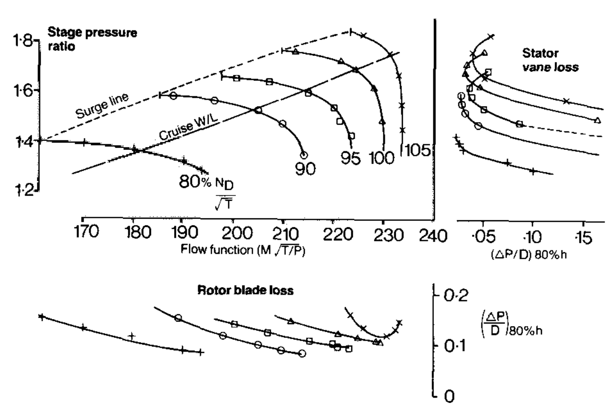

Fig.11 Fan Bypass Stage Characteristics

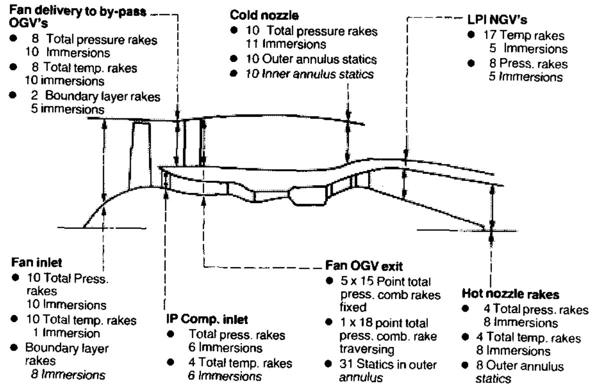

Fig.12 Engine 14007 LP System Instrumentation

shaft power supplied to the fan. Using the rig performance characteristics of several 211 single stage fans, and comparing performance along a typical 0.8 Mn, 35000 ft cruise operating line, the improved power output from the fan blade alone is evident.

It can be seen that over the last decade the fan has effectively been reduced in size for the same thrust by 40% and increased in efficiency by 4% at the same size.

In Section 2 of the paper, engine and rig performance data are compared which show the standard of test and analysis techniques required to ensure the improvements in fan performance demonstrated on model rigs are achieved in the full scale RB 211 production engine.

2. Performance Demonstrations

The RB 211 fan was initially built at 34" diameter for scale model rig testing. The scale model performance programme carried out in the Rolls-Royce Compressor Test Facility, Derby, during the spring and summer of 1969 proved that the fan concept was sound and demonstrated that the design method employed for graded work fans was basically satisfactory.

The model rigs have been continuously improved to fully represent the different engine fan arrangements with the correct simulation of the intake diffuser ahead of the fan tip and scale representation of the pylon and radial drive struts in the bypass duct. High accuracy instrumentation is installed in both bypass and core streams to establish aerodynamic performance at all radial heights of the fan/OGV arrangement.

A typical layout of the model rig instrumentation planes is shown in Fig.9.

Spanwise distribution of total pressure and efficiency measured at the fan rotor exit with the fan rig operating at design speed and with a back pressure close to the design value are shown in Fig.10. Each plotted point is based on the average of data taken from sensors mounted on the leading edge of the downstream stator vanes. The fixed LE instrumentation in the bypass stream down stream of the rotor blade consists of 4 pressure and 4 temperature vanes with 11 sensors spaced radially. Vanes are arranged at approx.90^o apart in the full OGV assembly. The instrumented vanes for pressure and temperature are mounted close together to minimise the effects of any gross circumferential variations when computing efficiency. Similarly, the fan core section is instrumented in the same way. The spanwise distribution of total pressure at exit from the outlet guide vanes is also shown for both streams in Fig.10. Each plotted point is based on the average of data taken from sensors mounted in a circumferential arc arrangement (combs). The bypass stream has 6 circumferential arc combs, mounted in a spiral array with 20 sensing heads each, and the core stream has 6 circumferential combs with 10 sensors each. The pressure combs at OGV exit span 2 or 3 vane passages allowing good definition of the total pressure loss in the bypass and core OGVs at all radial heights.

Radial traversing is employed at fan rotor exit and OGV inlet to augment the streamline measurements with the radial rakes and circumferential combs. Detailed measurements near the annulus walls and downstream of the part-span shroud can be obtained by radial traversing.

Blade Element Performance

During the test calibration of the scale model fan arrangement, the spatial coverage of the fixed instrumentation enables the performance of each individual blade row to be determined for all throttle settings from choke to stall on selected fan speed characteristics.

The pressure and temperature data are fed directly to the axisymmetric throughflow computer programs, enabling the blade element performance to be established over the entire fan compressor map.

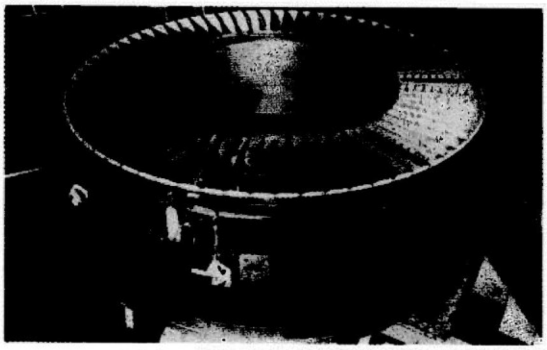

Fig.13 Engine 14007 Instrumented Fan OGV's

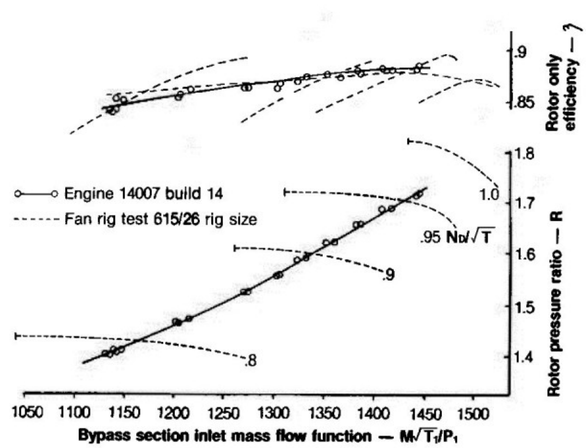

Fig.14 Fan Rotor Only Performance - Rig v Engine

A typical output of the analysis is shown in Fig.11. The overall performance characteristics are presented for the fan bypass stream, and the corresponding aerodynamic performance of the 80% blade height streamtube in the rotor blade and bypass OGV.

The results show that as the fan stream is throttled from choke to surge the rotor losses increase whilst the loss coefficients of the bypass OGV reduce and rise again next to surge. The sensitivity of the rotor and stator blade aerofoils to local air incidence and inlet Mach number can be determined from the throughflow analysis, and if required, development modifications employed to ensure that the peak efficiency of the fan design arrangement occurs along the cruise operating line of the high bypass ratio engine, shown in Fig.11.

A comparison of rig and engine performance data is used to illustrate that the fan performance improvements established on the 0.4 scale model rigs are realised in the full scale engine prior to production release.

Analysing all the available blade element data for the RB 211-524 standard production model fan blade, it was realised that by reprofiling the blade aerofoil shapes a significant improvement in fan efficiency could be achieved at typical engine cruise ratings. The challenge was to reduce the local air incidence levels on the high Mach number blading, whilst maintaining adequate flow capacity for the maximum engine climb ratings.

Comparative performance measurements were made in the full scale engine by employing high accuracy instrumentation in a -524 performance engine demonstrator shown in Figs.12 and 13. With similar instrumentation as the model rig a cross calibration was carried out using the same axisymmetric through flow analysis programs. Fig.14 shows that the measured data on the engine agrees closely with the 0.4 model performance and the radial distribution of rotor work and efficiency in Fig.15 demonstrates that the fan in the engine behaves the same as the fan rotor on the rig. There appears to be very little effect of size or Reynolds number on the performance of the high tip speed fan blade. The exercise was repeated for the improved -524 B4 fan and as shown in Fig.16 the improvement in part speed fan efficiency, originally suggested by model tests, was achieved at all typical cruise ratings.

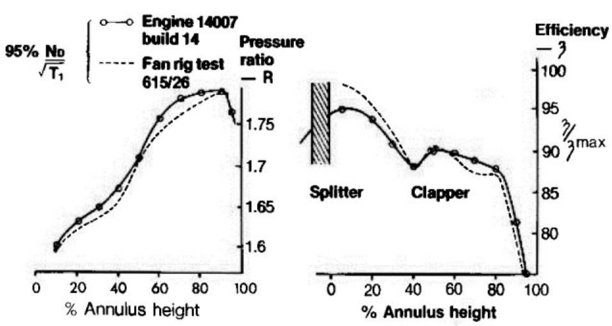

Fig.15 Fan Exit Pressure and Efficiency Profiles

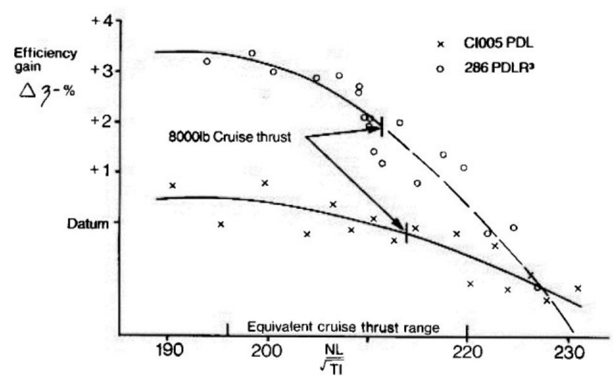

Fig.16 Comparison of Fan Rotor Efficiencies

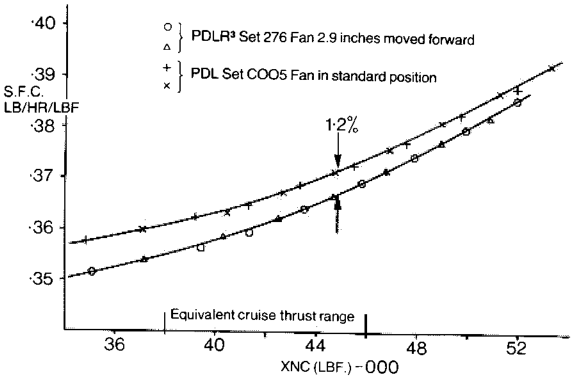

Fig.17 Comparison of Engine S.F.C.

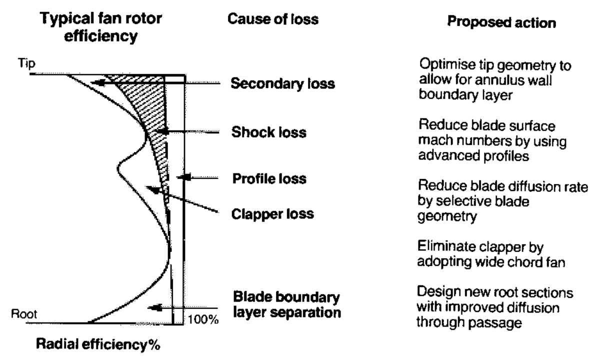

Fig.18 Breakdown of Rotor Inefficiencies

Fig.17 presents the corresponding improvement in engine s.f.c. for the redesigned fan blade and when the rotor was matched with an improved bypass OGV, the total fan package made a significant contribution to the improved s.f.c. level of the RB 211-524 B4 engine in the Lockheed L 1011-500 series aircraft.

Sources of Blade Aerofoil Loss

A large amount of blade element data has been made available from the various research and 211 model rig test programmes. These data form the basis of various blade aerofoil loss versus incidence correlations covering the whole Mach number range inherent in low hub/tip ratio fans. Design point efficiency prediction methods have been developed using the same test measurements and a typical loss breakdown for a design speed traverse is presented in Fig.18.

Unfortunately pressure and temperature measurements either side of a blade row only give the total loss occurring within the streamtube considered and the suggested efficiency breakdown relies heavily on the correlations of blade profile loss, shock induced loss, secondary flows and tip clearance loss. All the indications are that for a 1500 ft/sec tip speed condition the inefficiency caused by the presence of the shock waves in the high Mach number blading is approx. 4%.

Research fan programmes have been aimed at reducing levels of fan shock loss, and establishing comparative performance data for high Mach number blading. Several fan blades were designed at the same tip speed and loading, and the performance obtained for different blade geometry. Typical results are shown in Fig.19 for the streamline blade element at 85% annulus height. The results show that relative to the loss produced by a normal shock at inlet Mach number, some advanced blade profiles reduce the total blade element loss at the higher Mach number and incur a penalty at the lower tip speeds. The challenge is to select the low loss features of all the different blade profiles and maximise the efficiency of future 211 fan designs.

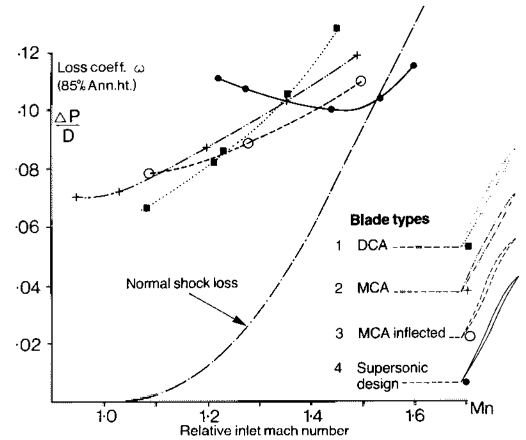

Fig.19 Effect of Blade Shape on Operating Range

It has been noted that a significant contribution to the overall rotor blade inefficiency is associated with the presence of the clapper (Fig.18). Whilst it is possible, by suitable design of aerofoil and clapper, to mitigate the effect, the most satisfactory approach is to delete the clapper from the fan. Deletion of the clapper removes its fundamental aerodynamic loss and the shock-induced interactive losses on the surrounding aerofoils, giving a larger efficiency gain than purely from the removal of the clapper.

The mechanical implication of the clapperless fan is that it must be designed at lower aspect ratio to achieve satisfactory vibration characteristics and avoid stall flutter conditions. This results in a reduced number of blades of fabricated titanium construction giving improved F.O.D. integrity and a lighter fan assembly.

The major benefits of the wide chord clapperless fan blade are:-

1. improved engine fuel consumption (1-1½%)which comes from the typical fan efficiency improvement shown in Fig.20.
2. better thrust growth potential since the fan is not limited at the very high levels of air-

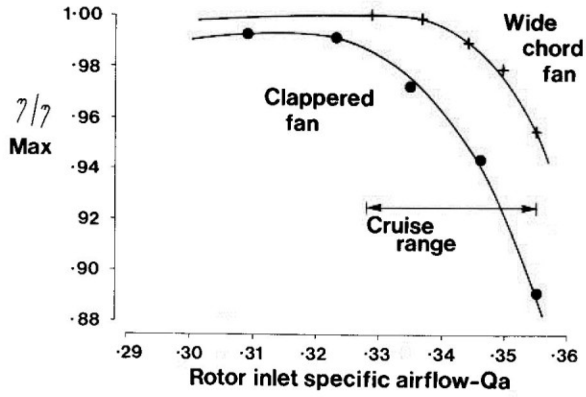

Fig.20 Performance Gain with Wide Chord Fan

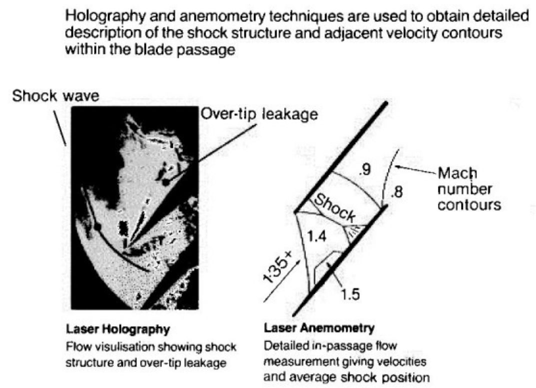

Fig.21 Flow Visualisation - Rotating Fans

flow by supersonic flutter constraints present in some clappered fan assemblies.

It has been recognised for some time that to further reduce blade element losses, the Fan Designer would need a better description of the flow processes and loss mechanisms within the blade passage.

Advanced theoretical and measurement techniques have been developed to give the better description of the flow field. Several aspects of the blade-to-blade results are discussed in the following section.

3. Recent Advances in Measurement Techniques and Flow Calculations

Until the last 2 or 3 years the only method available for improving fan designs was to select specific aerodynamic blading parameters such as blade design incidence, capture area ratio and blade throat area distribution etc. The fan was designed and the test and analysis procedures described previously used to establish that the changes in blade design produced the required result. Combined with the classical axisymmetric design approach, these design techniques have proved particularly successful for the RB 211 fan variants, although the relationship between high fan speed performance and design remains rather tenuous.

The advent of laser holography, laser anemometry and improved blade-to-blade flow calculations has opened up the prospect of giving the Fan Design Engineer what he wants - a better description of the loss mechanisms and inefficiencies within the fan blade.

Laser Holography

Laser holography techniques described in Ref.6 have been in use on model fan test rigs for 2 to 3 years. The holography technique used for high tip speed fans is to pulse the laser twice within a small time interval. The hologram then has fringes determined by the difference in density along the light path between the two shots. Since shock waves create density discontinuities then between the two laser pulses, the most visible changes are caused by the physical rotation of the shock in the blade passage. In principle the entire shock structure can be visualised, but due to practical difficulties, especially with clappered fan blades, only the outboard portion of the fan blade flow can be viewed. Fig.21 presents the flow pattern at the extreme tip of a typical model fan blade operating at 1400 ft/sec tip speed. The holograph shows the bow shock at the front of the passage and details of the overtip leakage flow against the annulus wall.

With varying back pressure and lower inlet Mach number the position and strength of the passage shock will vary. Fig.22 presents the holography results for various points on the fan compressor map using the observations for an early RB 211-524 production fan blade. The tip shock structure can clearly be seen as a function of the bypass throttle setting and blade speed. The passage shock is pushed forward along the entrance region of the passage as the fan is throttled to surge, and similarly the shock wave is expelled as the tip speed is reduced and LE incidence increased.

The tip clearance vortex can also be seen as a blurred line running from the leading edge of the blade to mid-passage. If the fan section has the shock well back in the passage (swallowed), then this vortex also starts further back along the blade, apparently springing from the suction surface coincident with the position where the shock wave intersects the pressure surface. The radial effect of this overtip leakage is small and confined to the tip section (Fig.21).

More detailed measurements of the shock wave/boundary layer interaction near the tip are presented in Fig.23. Visualisation of this flow phenomenon shows that the lambda foot is large and covers about ¼ of the blade to blade gap. The results also show the interaction of the shock wave and the end wall boundary layer. When the shock is more nearly axial, along the engine axis, then the interaction

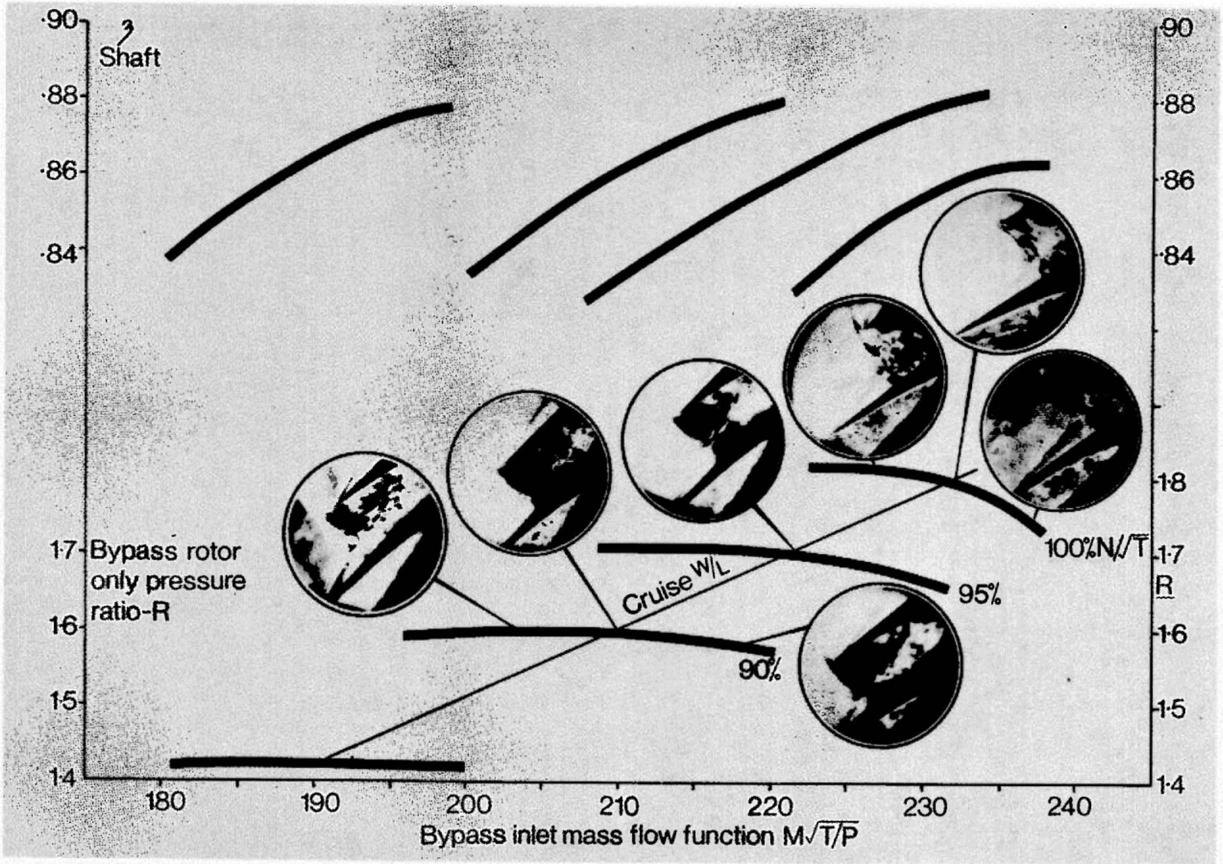

Fig.22 Rotor Tip Shock System Holograms

is small since the shock wave is parallel to the absolute flow and hence in general all parts of the flow have more or less the same Mach number normal to the shock. There will be no regions of reversed flow for the low back pressure case.

When the blade passage shock moves away from the engine axis (axial) there is a variation of Mach number normal to the shock so a branched system is created between shock and annulus wall boundary layer.

The important lesson from the holography results is that for fan designs of today's tip speeds and pressure ratio, the shock waves are principally at the entrance to the passage, and are actually spilt over much of the operating range of the fan compression system.

From the point of view of improving aerodynamics performance and reducing noise at take-off and landing, means are required to control the shock wave position and prevent excessive flow spillage.

Laser Anemometry

This is described in Ref.6. The laser anemometry technique used is the 2-spot system, where laser beams are focued at 2 spots of known distance apart and the mean time for a gas particle to travel between the two spots is measured. The flow direction is determined by rotating the two spots until the particles cross both beams. The time and angle are first converted into absolute and then relative velocities.

Regular measurement of inter-blade row flows is made with laser anemometry, and typical results at the tip of the same fan blade used for holography is also shown in Fig.21. The most significant feature of the results is the closeness of the blades to choking downstream of the shock, although using the conventional choke margin rules the blade passage should have adequate margin. The difference between design throat area calculations and the actual flow distribution is thought to be due to a combination of blade boundary layer blockage and additional radial streamline movement. The streamline movements relative to design may be partly due to the swept shock waves and the radial equilibrium conditions within the blade row. This clearly identifies two features of the flow that earlier fan designs ignored, although normal performance development may well have corrected for these omissions.

The laser anemometry results can also be used to calibrate and improve the theoretical flow models. Fig.25 presents laser anemometry results for a high tip speed unclappered fan with the shock position identified and regions of constant Mach number within the blade passage described at several radial heights. The results show that the shock front is more radial at the tip and more swept

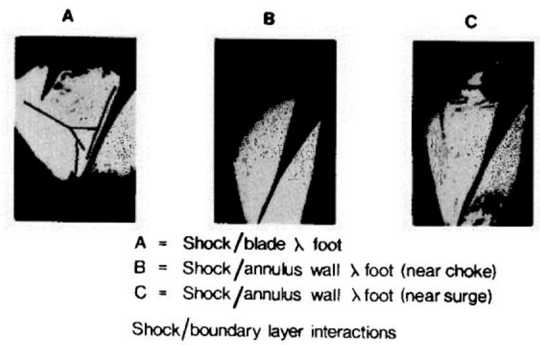

Fig.23 Shock/Boundary Layer Interactions

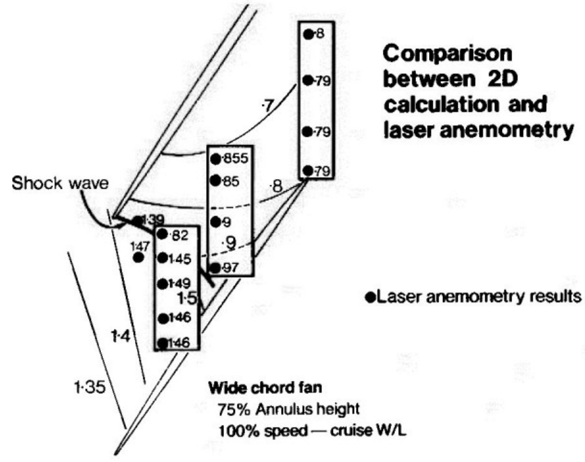

Fig.24 Comparison of 2D Calculation and Laser Anemometry

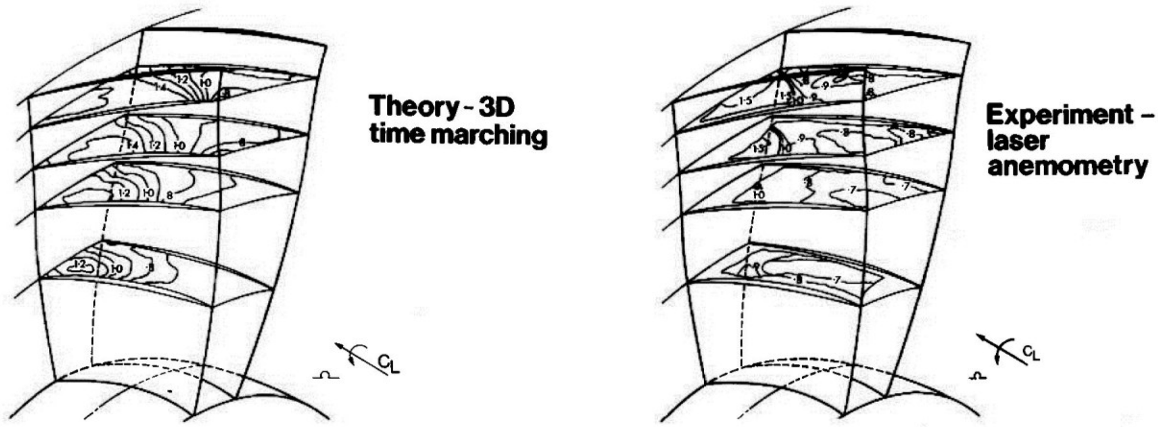

Fig.25 Comparison of 3D Calculation and Laser Anemometry

or inlined further down the blade. The strong message for the fan designer is that in the blade passages near the fan tip, the flow reaccelerates after the shock and ends up apparently near the choking limit.

Blade-to-Blade Calculations

Both two and three dimensional calculations are made of the flow between fan blades, the methods currently in use are due to Denton (Ref.7). The method solves the unsteady Euler Equations,the solution sought being the final steady state flow solution. For the 2-D calculation a set of axi-symmetric streamtubes are created from the through-flow analysis programs generating the boundary conditions in which the calculation is performed. The methods in common use are inviscid, although it is recognised that downstream of the passage shock wave a full description of the flow field still requires development. The accuracy of the 2-D or quasi 3-D solution can be seen in Fig.24 where a comparison is made of one section of a fan blade at ¾ span. The shock is adequately described, but the exit conditions are incorrect and the reacceleration of the flow to the passage throat is not identified in the calculation.

Various improved methods including boundary layer growth are under development and correct these errors giving a more realistic modelling of the flow.

Fully three-dimensional calculations of the flow within the fan blade are presented in Fig.25 for comparison with the laser anemometry results presented at different blade heights. The calculation is performed within a 3-D channel bounded by the blade surfaces and the hub and tip annulus walls only. The theoretical shock position is somewhat further back in the blade passage compared to the actual laser measurements. Once again the absence of viscosity in the basic method is probably the cause of the difference. Increasing the back pressure in the calculation moves the shock waves to the position seen on test.

The 3-D calculation, however, does show that the shock must be radial at the extreme tip, and relative to a normal shock at the local inlet Mach number, the passage shock becomes stronger and more inclined further down the blade. Both theory and measurement confirm that the fan operates as a 3-D device and not as a series of stacked 2-D sections.

Future Developments

Viscosity or its effect must be included in the calculations. It is already included in development versions of the 2-D or quasi 3-D flow model and under development for the full three dimensional flow calculation. The overall objective is to test fans numerically before exposing them to the air.

The development of laser holography should be principally in the direction of increasing the field of vision to encompass all shock wave features and the radial depth of the hologram. The development of laser anemometry is controlled by the rate at which data can be acquired and analysed. The rate is already an order of magnitude faster than when the technique was first demonstrated.

The overall effect of the current generation of advanced measurement and calculation techniques is to demonstrate that the fan blade behaves as a 3-D device, and by exception shows the powerful effect of viscosity on the flow. To date the use of these more powerful techniques has only been marginal on the development of current engine fan performance, since they were not proven when existing production fan designs were frozen. The new engine fan designs have used laser measurements in conjunction with the conventional instrumentation techniques and performance development is continuing. Future fan designs will rely heavily on these techniques and advanced clapperless fans are currently being fine-tuned with the help of both calculations and laser measurements.

It is concluded that linking the classical axisymmetric design approach with the quasi 3-D flow observations will lead to 2-3% further improvement in fan efficiency - and a reduction in the cost of engine fan development by designing 'right first time' and minimising the number of full scale fan performance tests before production commitment.

ACKNOWLEDGEMENTS

The author wishes to express his thanks to the Directors of Rolls Royce Ltd., for their permission to publish this paper, and to the Ministry of Defence M.O.D.(PE) who supported the aerodynamic research work.

Special thanks to colleagues in the Fan Technology Department for their technical assistance in preparing this paper.

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