Advanced Compressor Technology – Key Success Factor for Competitiveness in modern Aero Engines

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Abstract

Since day one of the gas turbine engine the compressor was the key component for the success and always required high development efforts and costs.

The importance of the compressors for modern high bypass ratio engines is demonstrated by the fact that 50–60% of the engine length and up to 40% of the manufacturing costs are covered by the compression system. The advances achieved allow engines to operate with core engine thermal efficiencies in the 50% area and propulsive efficiencies approaching 80%.

Integrally bladed rotors permit blade speeds significantly above conventional rotors and hence stage pressure ratios of >1,8.

A core compressor study with independent variation of the number of stages and aspect ratios from 0,6 to 2 gives insight into the relative weight, manufacturing and direct operating costs.

Nomenclature

$AR = \frac{h}{c}$	-	Mean aspect ratio
c	mm	chord length
C_F	N	centrifugal force
c_o	m/s	flight velocity
c_p	kg J/kg K	specific heat at constant pressure
$\Delta\ DOC$	%	change in direct operating cost
F_N	N	net thrust
g	mm	axial gap between blade rows
H	kg J/kg	specific compressor work
Hu	kg J/kg	lower heating value
h	mm	mean blade height
Ma	-	Mach Number
s	mm	blade spacing
T_i	K	stage inlet temperature
u_M	m/s	blade speed at mean radius
W	kg/s	weight flow
W_f	kg/s	fuel flow
z	-	number of stages
h_s	-	isentr.effic.
h_{th}^*	-	thermal effic. based on isentropic gas power
h_P	-	propulsion efficiency

k	-	ratio of specific heats
r	kg/m^3	density
s	kN/m^2	yield strength
$y = \frac{2H}{u_M^2}$	-	work coeffic.

Thermal efficiency:

$$h_{th}^{*} = \frac{W_{48} \cdot (His - c_o^2/2)}{W_f \cdot Hu}$$

His is the isentropic expansion work to ambient pressure starting from an intermediate position within the LP turbine which covers the compression work for the core stream on the LP shaft, i.e. fan core stream + booster.

Propulsion efficiency:

$$h_P = \frac{2}{1 + \frac{F_N + W_2 \cdot c_o}{W_8 \cdot c_o}}$$

Introduction

Since the beginning of the development of the modern gas turbine engine the compression system is a decisive key component demanding big development efforts on especially expensive test beds.

It was the lack of compressor technology which prevented an earlier appearance of the gas turbine engine, which in fact is the last heat engine coming to fruition as late as 1937.

The development of the efficient light weight axial compressor with high circumferen-tial speeds required knowledge of cascade aerodynamics which were not available for a long time.

The fact, that the gas turbine engine first paved its way in aircraft propulsion is due to its extremely high power concentration per volume – about 15 times the value of a passenger car engine.

The compression system of modern aero engines

The gas turbine engine has gone a long successful way since its first flight 64 years ago and matured in several steps from the straight jet to the modern high bypass ratio engines.

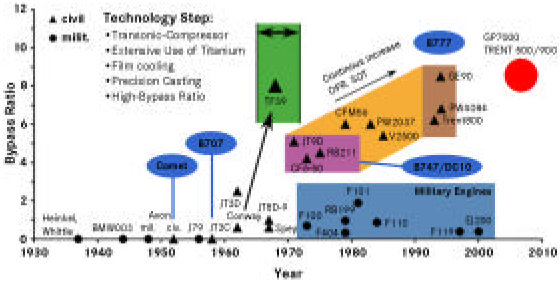


Fig.1 Evolution to the energy efficient engine

As shown on Fig 1, a big jump in technology was realised with the development of the military high bypass ratio TF39 for the Galaxy transport aircraft. This engine introduced a whole bunch of advances and in particular the transonic compressor, which first time allowed compression ratios in the 25–30 range, boosting thermal efficiency into attractive regimes.

The importance of the usual 2- and 3 shaft compression systems of modern engines is described in Fig. 2.

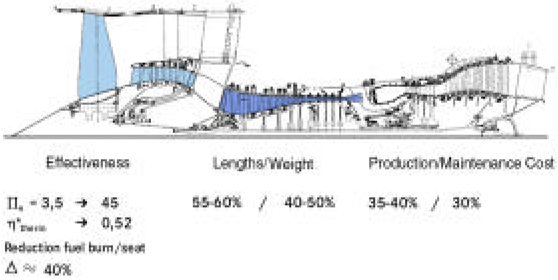


Fig.2 The compression system - key technology for aero engines

Transonic compression systems achieve pressure ratios up to 45 and – together with high turbine inlet temperatures allow thermal efficiencies during cruise of over 50%. This reduced the fuel consumption per passenger seat by 40% compared to the first jet liners, whose engines were of military origin.

The compressor section represents still 50–60% of the engine length, 40–50% of the weight, 35–40% of the manufacturing and 30% of the maintenance costs.

In spite of all the advances achieved in the analysis of aerodynamics and vibrations, the development of modern gas turbine engines is still very much affected and sometimes dominated by the compression system.

The difficulty to describe analytically the very complicated physics of the 3D flows in multistage transonic compressors and to predict the response of all important blade resonances is still somewhat beyond our abilities, also because the requirements increase continuously. However the quality of the available simulation tools improves steadily.

The requirements for the compression system

The catalogue of requirements is long and stringent. As shown on Fig 3, it reflects the three main success criteria for civil aviation:

- Safety
- High power concentration, i.e. low weight
- Affordable cost

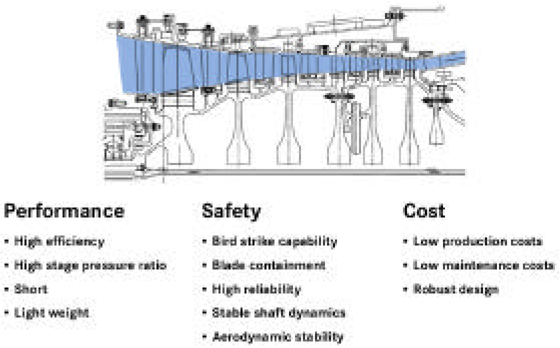


Fig.3 Requirements for the compression system

The development of the high speed transonic compressor permits a level of propulsion effectiveness which was never achieved before.

However as a result of the high blade speeds the safety requirements as bird strike resistance, blade containment etc are more difficult to fulfil.

Engine reliability is a decisive criterion for present days dense air traffic. The enormous standard achieved is best demonstrated by the big number of twin engine aircraft crossing the oceans safely day by day.

The development history of the aero engine compressor

The gas turbine originated from aircraft propulsion and hence was always driven towards the lean axial compressor of high specific flow per frontal area.

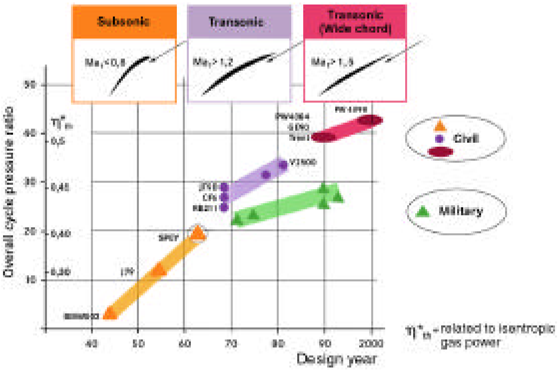


Fig.4 Evolution of the turbo compressor for aero engines

As shown on Fig 4, up to the 1960’s the subsonic multi-stage compressor with tip Mach numbers < 0,8 limited the achievable compression ratios to about 17 within the shaft dynamic constraints.

The 1968–73 generation of military and civil engines made extensive use of the narrow chord transonic compressor technology, the development of which started as early as 1952 at NACA in the US. This application of transonic compressors to real engines doubled the average temperature rise per stage from 21 to 42 K.

The next step followed with the introduction of the wide chord compressor with blade aspect ratios around 1 and stage temperature rises of 60–75 K.

The increase of the specific energy addition and hence the stage pressure ratio depends on two basic parameters:

- Mach number of blade speed
- Flow deflection in the rotor, i.e. the work coefficient

Fig 5 illustrates this relation and the split into the two parameters, where the Mach number is replaced by the simplified proportional blade speed $u/\sqrt{T}$.

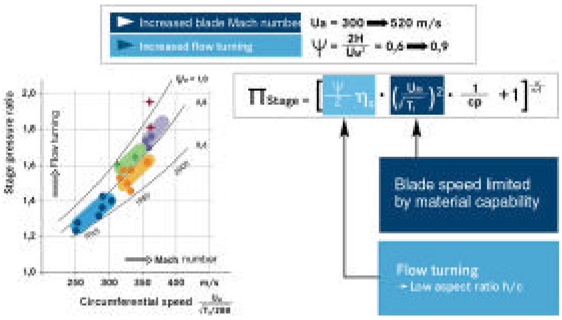


Fig.5 Parameters affecting stage pressure ratio

Both elements of the equation increased strongly over the years, first the blade speed and later the work coefficient as a result of the wide chord design philosophy.

The ability to handle transonic cascade flows efficiently improved continuously since the late 1950’s and resulted in a steadily deeper understanding of the physics.

As shown on Fig 6, the low solidity and highly cambered subsonic cascades with high suction surface curvature produced unacceptably high shock losses and a rapid fall off in stage efficiency.

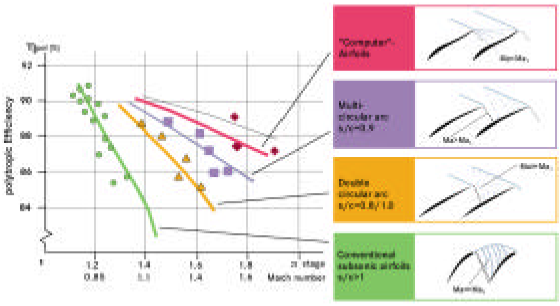


Fig.6 Efficient control of high speed flow

The way to lower shock losses led through a much decreased suction surface curvature in double circular are profiles and increased solidity – both measures limit the supersonic expansion ahead of the shock and hence the shock intensity and the inherent losses.

The empirical development to the multi-circular arc profile with further reduced or even zero curvature in the forward – supersonic - part of the profile enabled the design of remarkable compressors with Mach numbers up to 1,6 until the analytical codes became available and allowed much better optimisation, also in form of profiles with external compression.

The second path to high stage pressure ratios with increased pressure coefficient became viable through the application of lower blade aspect ratios, i.e. the 'wide chord' design. Although the empirical observation that low aspect ratio compressors show superior flow range goes back to before 1960 there were few attempts to make use of this possibility in western engines. Only as late as 1985 a new generation of military engines applied the wide chord design philosophy (EJ200, F119) and reduced the number of compressor stages to 8–9 instead of 12–13 of the older generation engines.

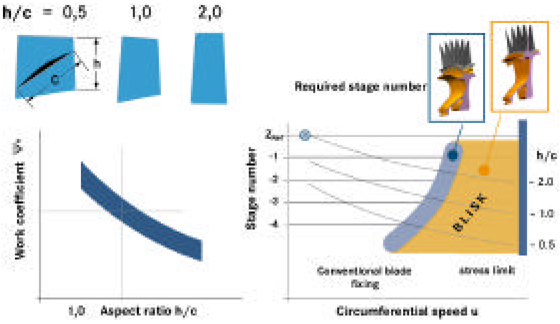


Fig.7 Effect of aspect ratio on pressure coefficient

Fig.7 shows a statistical analysis of the maximum pressure coefficients achieved by 19 compressors, correlated as a function of aspect ratio. The superiority of the low aspect ratio blading is impressive; it can safe several stages when combined with the high blade speeds possible by use of integrally bladed rotors. However equally clear are the problems and limits which the wide chord design provides to the mechanical stress situation due to its high weight and the resulting forces in the disk – blade root fixing and in the disk itself.

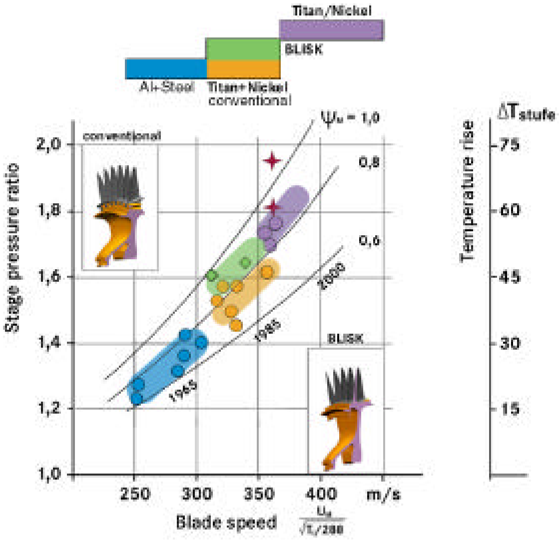


Fig.8 Achievable stage pressure ratio and required circumferential speed

Fig 8 shows the present state of the art and the limits of conventional and integrally bladed rotors with our present standard metallic materials, namely Titanium and Nickel alloys. These materials have a very big data base behind them and are very safe due to their benign failure behaviour with a big elongation to rupture.

Advantages and disadvantages of low aspect ratios

The low aspect ratio blading provided the compressor aerodynamicist with a new degree of freedom to achieve increased blade loading levels and allowed him to design for high stage pressure ratios with good aerodynamic stability in clean and distorted inlet flow. The superior aerodynamics of the low aspect ratio design is connected to the flow in the corner between the blades and the side walls and the reduced axial pressure gradient along the side walls. The details are still not really well understood but the 3D Navier-Stokes codes will help to provide more insight.

As shown on Fig.9 low aspect ratios result in more robust blades with better erosion resistance.

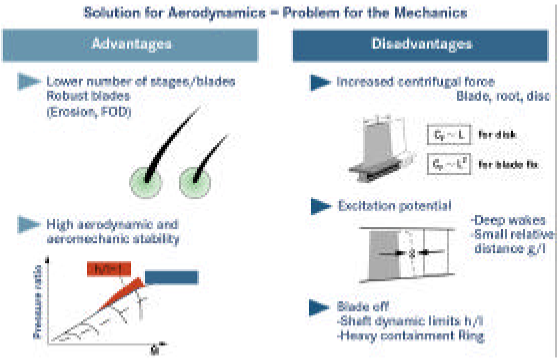


Fig.9 Advantages and disadvantages of low aspect ratios

However there is the serious disadvantage in the big weight which increases the load to the disk and the root fixing and produces a much higher unbalance in case of a blade loss. This may easily result in quite significant increases in overall mass for a given duty.

The high energy input of modern wide chord blades results in large wakes which incorporate a heavy excitation potential for the upstream and downstream cascades. This is amplified by the aerodynamically smaller axial gaps to the neighbour cascades as wake decay scales with chord. Weight and shaft dynamics usually do not allow for increased axial gaps.

In addition low aspect ratio blades contain a number of plate type vibration modes which were not so important in narrow chord designs. The possible ways out to avoid critical blade resonances have narrowed down considerably in high speed low aspect ratio designs and the potential coupling of vibrational excitations over several stages is increased. The heavy containment rings required for wide chord machines are an additional weight penalty to be carried.

Competence in compressor design

Design and analysis

Competence in compressor design requires many capabilities. Besides a solid knowledge in aerodynamics, vibrational and shaft dynamics, stress and thermal analysis a long experience and good capability in the manufacture of high speed rotors, blisks, rub strips etc is required. Specific knowledge in metallurgy and a well established experimental capability is needed to test and analyse compressors down to detailed inter-stage data out of the rotating system to understand the aerodynamic and vibrational behaviour.

Modern compressors use the material properties to their limits. The freedom for the designer to find a solution which is acceptable to all disciplines is small and the number of iterative steps grows.

Rub Systems

A topic of continuous difficulty is the blade/casing rub system. The blade materials need to be protected against thermal over stressing in case of a rub which usually is done by coating the casings with abradable material. The problem with these abradables is their temperature and erosion limitation, resulting in harder coatings which eventually became abrasive.

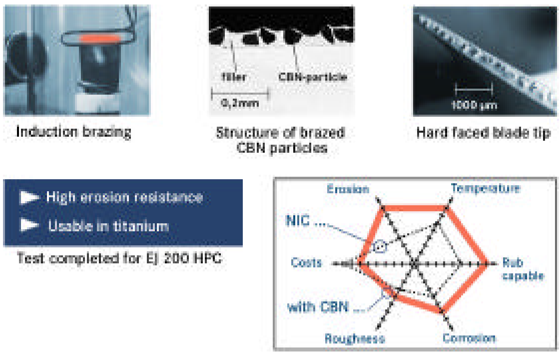


Fig.10 Rub system
rotor blade - casing

As illustrated on Fig.10 MTU developed a hard faced rub system for titanium blades which brazes cubic boron nitride particles on to the blade tip as cutting material. The system runs against a metal filled silicate coating. It is quite stable and provides good performance retention as well as protection to the blade.

Manufacturing methods

The high speed low aspect ratio rotors with high solidity require the integrally bladed rotor to overcome basic mechanical problems in the blade root fixing. What originally appeared as a specific element of military engines for thrust/weight ratio reasons has become accepted in the civil field as well.

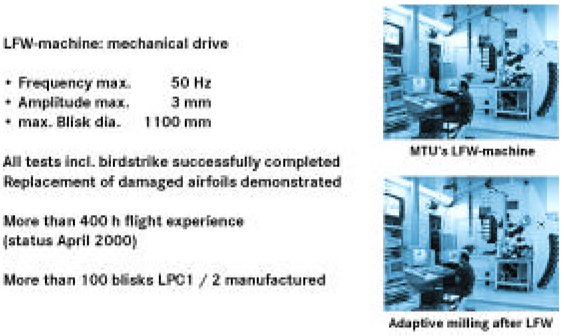
If the designers follow the market requirements to reduce the high complexity of present engines with up to 26 turbo machinery stages they will have to follow the path of the military engines which reduced the stage numbers by a third in the EJ 200 and F119 engines.

Fig.11 EJ 200 All blisk fan with
linear friction welded
rotor

MTU moved to integrally bladed rotors quite early and covers today all currently used manufacturing methods, depending on the size of the rotor:

- High speed milling
- Electro-Chemical machining
- Linear friction welding

Fig. 11 shows the EJ200 All Blisk Fan which is the first linear friction welded rotor in production, manufactured on the machine shown on Fig 12.



LFW-machine: mechanical drive

- Frequency max. 50 Hz
- Amplitude max. 3 mm
- max. Blisk dia. 1100 mm

All tests incl. birdstrike successfully completed
Replacement of damaged airfoils demonstrated

More than 400 h flight experience
(status April 2000)

More than 100 blisks LPC1 / 2 manufactured

Fig.12 Linear friction welding machine at MTU

Materials and advanced designs

At present most parts of an engine are fabricated from titanium and nickel based alloys. However the continuous increase of circumferential speed has placed us close to the limits of these materials and the blisk design will probably help only for some limited time. In fact the engine designers look for new materials which allow them more freedom as the introduction of titanium gave them 30 years ago.

New materials like Titanium Aluminides (TiAl) and Metal Matrix Composites (MMC) with high strength SiC fibres promise many good characteristics but lack the big elongation to rupture of the metallic materials. As illustrated in Fig 13 the new materials are in fact much more brittle with elongations to rupture of only around 1%. This makes them difficult for use in engines with their stringent safety requirements. It will take big efforts to design rotors from these materials which are safe, reliable and predictable.

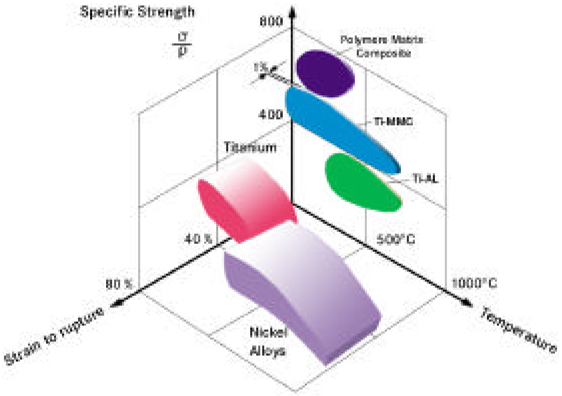


Fig.13 Comparison of todays and future materials

MTU sees good chances for TiAl compressor casings and works on a MMC compressor rotor as a bladed ring. The lighter weight and the lack of disks allow for good shaft dynamics at even higher blade speeds.

Costs and the effects of compressor design

The compression system is a large and expensive part of the engine, both in development and production. It is therefore important to know how it affects the costs of ownership.

Effect of main engine cost elements on airline profits

As shown on Fig 14 the engine related direct operating costs for short and long range aircraft are about 28% and 34% respectively at present fuel prices. The by far biggest part of this is represented by the fuel consumption, namely about 18,5% and 25%, followed by the next important fraction, the maintenance/material costs which accumulate to 7,5% and 5% respectively.

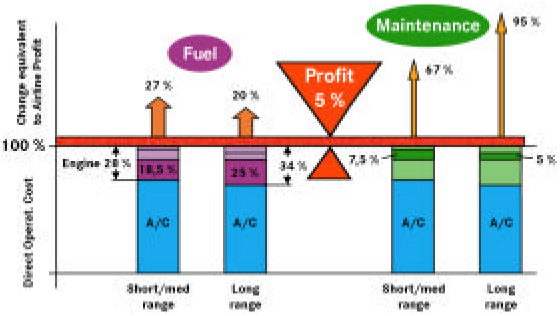


Fig.14 Effect of main engine cost elements on airline profit

A well operated airline may gain 5% profit margin of the direct operating costs in good times. It takes only a 25 % increase in fuel price to eliminate this margin if competition does not allow to transfer the extra cost to the passengers, as it happened 7 years ago. Fuel price is very sensitive, i.e. it is very difficult or impossible to beat fuel efficiency by reduced maintenance, especially in future when oil prices will inevitably grow.

Compressor development costs

The development costs of a new high pressure compressor are around $150–200 Million and thus too high to be carried purely on company money.

As all companies do, MTU uses technology from publicly funded military and civil programs – full engine development programs and rig compressors.

MTU’s technology stems from the RB199 and EJ200 fighter engine programs and over 20 military and civil research compressors with pressure ratios ranging from 2.5 to 16 and Mach number levels from 0,8–1,7.

Production and Life Cycle Cost

In order to evaluate the relative production and life cycle cost of a core engine compressor with about 11:1 pressure ratio, a parametric study was carried out varying the number of stages from 5 – 8 and the mean aspect ratio from 0.6 to 2 at constant mean diameter and at constant surge margin.

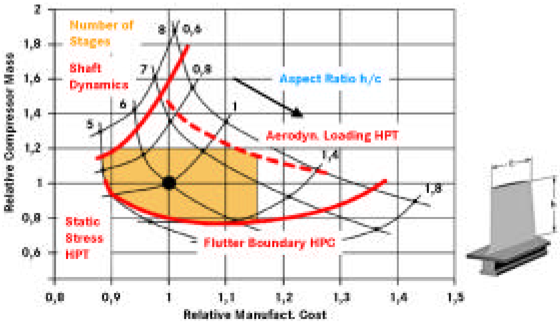


Fig.15 Core compressor study - relative mass and manufacturing cost

Fig 15 summarises the results of this investigation, showing the relative weight vs relative manufacturing costs with aspect ratio and stage numbers as independant parameters. Limits for shaft dynamics, first stage flutter and aerodynamic as well as mechanical loading limits of the single stage high pressure turbine are drawn in to show the area of realistic compressors. The marked area shows the technically most interesting regime.

Fig 16 illustrates the results on life cycle cost for a short/medium and a long range application. The results show that generally the higher aspect ratio compressors show higher direct operating costs due to their higher number of airfoils. At present fuel prices there is little difference between the 5 – 8 stage compressor variants at aspect ratios < 1.2 for the short/medium range application and virtually zero difference between 5 and 6 stages. This will change very significantly towards all higher efficiency variants as fuel prices increase in future. For the long range mission the higher efficiency variants with 7 and 8 stages are already better off but up to about 1.3 aspect ratio the differences are not big enough to force a decision; it will have to be made on other criteria like ease of aircraft installation and aircraft specific requirements.

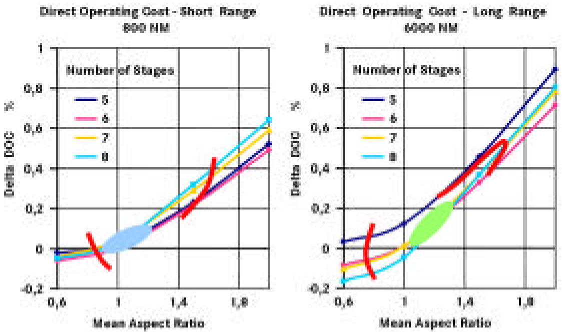


Fig.16 Effect of compressor design on direct operating cost

The general impression however is, that the low aspect ratio compressor wins on DOC due to its lower manufacturing cost and the high efficiency it can achieve.

Compressor development – an integrative challenge

Aero engine compressors always represented the very leading edge of the state of the art in turbo compressor technology. Compressor development at the limit of available knowledge is a highly integrative challenge to all involved.

As a result of the steadily increasing blade speeds, stress levels are approaching the limits of the classic materials which in turn requires a very intense interactive effort between aerodynamicists, mechanical designers and stress specialists as well as the manufacturing experts to find satisfactory and generally acceptable solutions for performance, life and cost.

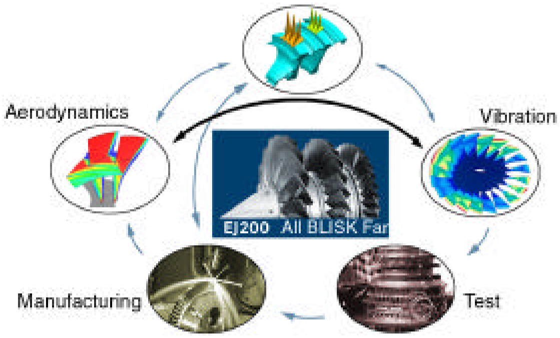


Fig.17 Compressor design - an integrative challenge

As shown on Fig. 17 it is especially between the aerodynamicists and the vibration experts where most iterations are needed.

Only most intense co-operation from the very beginning of a design between all disciplines will lead to the best product.

Compressor development was a big challenge and a key technology since day one – it will remain so.