MODERN TECHNIQUES TO IMPROVE

SURGE MARGIN IN AXIAL COMPRESSORS

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ABSTRACT

In modern military aircraft engines, the stable flight envelope is limited by the compressor surge which is an unsteady working condition affecting both compressor and receiving circuit (i.e. all engine components). This obliges the engine designers to provide for a "SURGE MARGIN" i.e. a limit of the working domain excluding this instability.

The choice of this stability margin must be made during preliminary design and design stages considering both global design parameters and fine costruction elements.

The modern design techniques must take into consideration not only the cnal geometry and the axial load distribution between compressor stages, but also the radial load distribution on each stage.

In addition to these global design parameters we discuss also in this paper the effect of fine construction elements such as nominal incidence, tip clearance, blade aspect ratio, stage solidity, casing treatement, bleeds and variable geometry.

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1. INTRODUCTION

The performance map of an axial compressor is classically represented on the plane "pressure ratio - mass flow". On each characteristic curve corresponding to a constant rotor speed there is a point (pressure ratio & mass flow) behind which local or global unsteady working conditions appear (Fig.1).
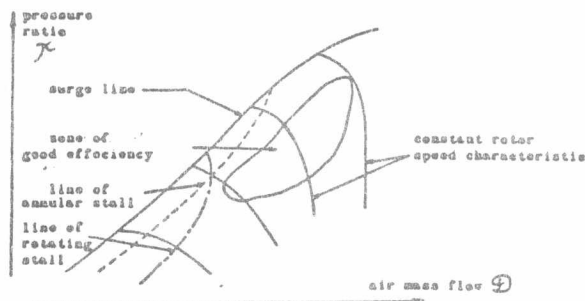

Fig.1. Schematic of a compressor map

These unsteady working conditions can be of four types: rotating stall, annulus stall, flutter or surge [1].

Surge is an unsteady working condition affecting both compressor and receiving circuit. It is characterized by low frequency great amplitude axial oscillations of mass flow.

The begining of a such phenomenon depends not only on the compressor itself but also on its environement. Nevertheless it can be launched by the presence of rotating stall.

The surge inacceptable working conditions, because of the serious dangers that it presents, oblige the engine designer to provide for a "SURGE MARGIN", i.e. a limit of the working domain excluding these instabilities.

The problem is critical all the more as the zone of best effeciency is situated near this surge limit. Taking an important safety margin means then excluding from the working domaine a part of the high effeciency zone.

In effect the modern engine has to satisfy a number of often conflicting requirements, such as, adequate stability, low fuel consumption, long life, low weight, and low cost. For example [2] stability margin could be achieved by matching the compression system below its peak pressure ratio and peak effeciency, but this would be in direct conflict with the low fuel consumption and the highest pressure ratio per stage requirements for low cost and weight.

To achieve an optimum performance compromise the engine is, therefore, designed to have just enough stability margin to function adequately.

The choice of this stability margin must be made during the preliminary design and design stages taking into consideration the precision of the surge line prediction method, its distance with respect to the design optimal point and the inlet flow heterogeneties.

2. SURGE MARGIN AND COMPRESSOR'S ENVIRONEMENT

As we have seen, the surge margin estimates the distance between the wor-
king line (defined generally as the geometic locus in the plane "pressure
ratio - mass flow" of the points of maximum effeciency) and the surge li-
ne (stable working limit). Two different methods may be used to define
this margin on a constant rotor speed characteristic:

- The first takes the distance between the nominal point and the
  surge point on the same constant rotor speed characteristic to
  express the surge margin in terms of the pressure ratio at const-
  ant reduced r.p.m.
- The second considers the distance between the nominal point and a
  point on the surge line representing the same mass flow as the
  nominal point to express the surge margin in terms of the pressu-
  re ratio at constant mass flow.

The surge phenomenon depends not only on the compressor but also on its
environement. The surge is a non steady working condition affecting the
compressor and its receiving circuit. Greitzer [3-4] has shown since 1976
the importance of the general circuit (up and down stream) in which the
compressor is installed on both surge and rotating stall phenomena.

On the other hand, it is known that the compressor surge line depends on
the heterogenities of the intake flow. This is the case, for example, in
the highiy inclined flight in which the air-intake causes radial and cir-
cumferencial distortions of the inlet velocity and the inlet pressure af-
fecting the surge line.

Mikolajczk and Pfeffer [2] showed that the required surge margin of a co-
mpression system has to make allowance for a number of effects including:

  (1) the expected levels of inlet distortions
  (2) the compressor tolerance to inlet distortions
  (3) Reynold's Number effects
  (4) engine-to-engine variations and tolerances
  (5) engine aging
  (6) excursions of compressor operating lines during transients

The same reference estimates that a modern compressor needs a surge marg-
in of about 20% expressed in terms of pressure ratio at constant flow re-
presenting the sum of the following requirements:

Distortion	5%
Tolerances	3%
Deteriorations	2%
Transients	10%

3. EFFECT OF DESIGN PARAMETERS ON SURGE MARGIN

As we have seen the surge margin is determined during preliminary design
and design stages. It is through the global and fine design elements th-
at we must try to find the compressor's optimization with respect to su-
rge.

3.1. Effect of Global Design Parameters

The surge phenomenon is generally initialized by the appearance of local
stalls or flow separations depending on the flow boundary conditions. All
actions leading to a retardation of these stalls and separations permit
to enlarge the engine stable working extent. Besides the profile optimiz-
ation methods we can state compressor canal geometry optimization and ax-

ial load distributions choices as surge margin determining elements.

3.1.1. Canal geometry

The compressor's canal geometry affects indirectly the surge margin through the velocity triangles. The effect the conical shape angle and the curvature of both inner and outer casings on the slowing-down coeffecient , the diffusion factor and other parameters directly related to the velocity triangles is obvious.

In practice the canal geometry is chosen so that we may find the same difficulties on each stage in order to obtain a well balanced nominal point.

The use of the so-called "stream line curvature method" can lead to this optimization by successive adjustements of the canal form.

3.1.2. Axial load distribution

In the case of a multi-stage transonic axial compressor, surge occurs only after reaching the subsonic flow condition in the successive compressor stages from down-stream to up-stream.

As long as the last stage remains sonic the up-stream stages will not be practically affected because of the supersonic blocking. Nevertheless, there is a perturbation transfer from down-stream to up-stream through the subsonic sections in the blades' roots and the annulus wall boundary layers.
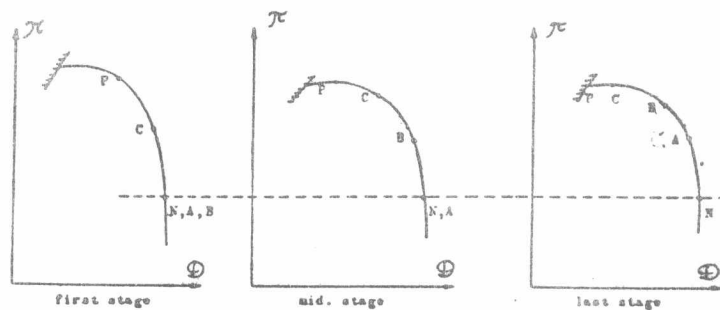

Fig.2. A three stages transonic compressor

Fig.2 represents an example of a compressor of three identically loaded stages at design point.

The last stage comes upon its surge limit whenever the two first stages have not yet consumed their surge margin. So we have a loss of the global margin and eventually surge occurence before reaching the maximum effeciency of the first stage.

3.1.3. Radial load distribution

Once the pressure ratio(or the load) per stage has been determined for a compressor, another degree of freedom remains for each stage, namely the radial distribution of its load. This distribution determines the blade turning angle distribution since the work realized by any blade section is proportional to the deviation that it realizes.

The radial load distribution must take into consideration:

- The mechanical constraints under the form of a mean compression level.
- The velocity triangle difficulties for each blade section (in respecting the design criteria such as zweifel criterion, diffusion factor, stators inlet Mach numbers...etc).

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- The secondary effects taken into consideration either by a local overloading in turning more the concerned sections or globaly in redistributing the additional work on the whole blade.

Increasing the pressure ratio over its design value at the same rotating speed changes considerably this radial load distribution in transonic compressors. The near root blade sections evolution differs from that of the near tip ones and the radial equilibrium changes:

- The near root sections, generally subsonic, suffer from an important slowing-down coeffecient, the boundary layers become difficult, the losses increase, the work changes slightly but the effeciency decreases considerably. If we try to increase the pressure ratio more and more, these subsonic blade sections can reach their critical loading (with critical angles of attack and aerodynamic stalls), and the compression work collapses.
- For the near tip sections, supersonic, the chocs remount the interblade canals towards up-stream (till throat) and hence occur in lower Mach number zones with less losses. A considerable amelioration is obtained in both compression and efficiency owing to the reduction of choc intensities and the increase of the profile turning (because losses diminuation is accompanied by deviation angles reduction.

The above analysis leads to the abandonment of the uniform radial distribution concept. We are oriented towards a negative or a positive radial pressure gradient choice.

For a negative radial pressure gradient at design point the tip sections are less loaded, less cambered, with small turning and weak chocs, leading to low profile loss coeffecients. The work done by these sections is small and the efficiencies are low. On the other hand, the root sections, more cambered, produce the inverse effect.

For a positive pressure gradient the situation is naturally inversed : the work done by root sections is small.

Increasing the pressure ratio at a constant rotating speed leads to a net amelioration of tip sections' work in the case of negative pressure gradient with losses caused by relatively weak choc waves. The work done by root sections remains practically constant in this case. For a positive pressure gradient a slight work amelioration is obtained through choc losses diminuation caused by the up-stream choc remount (at least till throat).

In the neighbourhood of the surge point and for a negative pressure gradient the blades becomes in better radial balance with a mass flow uniformly distributed between root and tip sections leading to a good surge margin. But for a positive pressure gradient the blade becomes desbalanced. The mass flow tends to pass in tip, and the secondary effects become important specialy on inner casing which is badly feeded.

The analytical and experimental study of El Bahi [5] confirmes the favourable effect of a rotor's negative pressure gradient on the compressor surge margin. An improvement ranging from 12.7% to 24.7% of the initial margin was realized in the studied case by optimizing the radial pressure gradient only on the second stage of a bi-stage transonic axial compressor. This improvement corresponds to an improvement ranging from 20% to

69% with respect to the case of a constant turning angle radial distribution with an efficiency penalty of 0.5% only.

3.2. Fine Construction Elements Leading to Surge Margin Amelioration

Among the other construction elements, nominal incidence, tip clearance, blades aspect ratio, stage solidity, casig treatement, bleeds and variable blade geometry can have, more or less a direct effect on compressor's surge margin.

3.2.1. Nominal incidence

A simple approach to ameliorate surge margin could be to match the compressor at its aerodynamic design point with the airfoils set at negative incidence. This would provide an increase in airfoil incidence range between the compressor operating line and its stability limit but would leak to an efficiency penalty along the compressor operating line.

A compressor matched toward negative incidences (choke) usually has its peak efficiency between the operating line and the stall line, and is always matched on a steep portion of the pressure rise characteristic. Since the steepness of a compressor pressure rise characteristic is usually indicative of the compressor's ability to attenuate inlet distortion [2], the compressor designer may deliberately match the first stage of a multi stage compressor toward choke in order to achieve an acceptable compromise between peak efficiency and distortion tolerance.

It became apparent very early in the history of compressor development that compressor endwalls, with the complex three dimensional boundary layers generated by blade - wall interactions, had less incidence range and pressure rise capability than the relatively two dimensional midspan regions. It was soon recognized that midspan airfoil sections could be designed close to minimum loss while the end wall sections remained at negative incidence.

3.2.2. Tip clearance

Rotor blade tip clearance and cantilevered stator root clearance play an important part in establishing loading capability of a compressor. The work of Smith [6] illustrates the loss of pressure rise capability and hence the loss of available stability margin with rotor tip clearance, and indicates the extreme importance of designing for minimum tip clearance (Fig.3). It is therefore not surprising that the modern engine designer uses extremely sophisticated techniques to maintain tight tip clearances.
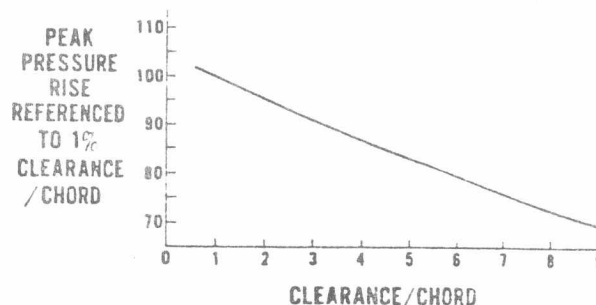

Fig.3. Effect of tip clearance on pressure rise

Engine transients are simulated on a computer in order to predict the running clearances accurately and the compressor materials are selected with appropriate thermal expansion coeffecient to maintain constant clearances during transients. Outer casings are constructed with abradable mat-

- The velocity triangle difficulties for each blade section (in respecting the design criteria such as zweifel criterion, diffusion factor, stators inlet Mach numbers...etc.)
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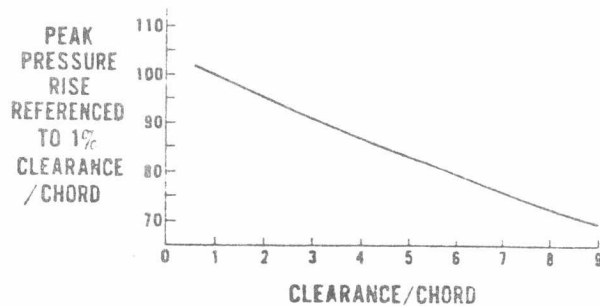

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erials and blades are sometimes machined in assembly to uniform length in order to obtain the precision of construction necessary to run with absolute minimum clearance.

3.2.3. Blades aspect ratio (h/c)

It has been shown [2] that long chords (corresponding to low aspect ratios) stabilize end wall boundary layers and hence increase the range between peak effeciency and stall. Some recent studies at the NASA illustrate the effect of low aspect ratios (0.7 to 1.5) on the global engine efficiency. The conclusion of these studies [7] is that the effect is complicated and depends on the blade loading (expressed in terms of the diffusion factor), and its solidity.

The results of [7] confirme those of 2 concerning the favourable effect of low aspect ratios on the surge margin for medium diffusion factors (D $\simeq$ 0.5) only.

3.2.4. Solidity (c/s)

The studies of [7] show that decreasing rotor solidity (corresponding to a less important number of blades per rotor) would have three effects at constant aspect ratio and diffusion factor :

- desplacement toward high flow coeffecients for the same pressure ratio
- surge margin amelioration
- rotor and stator maximum efficiencies amelioration.

3.2.5. Casing treatement

It has been recognized for some time that the presence of a treated casing rather than a normal smooth wall over a rotor tip could significantly increase the tip aerodynamic loading capability of the rotor and consequently the surge margin.

The use of casing treatement is particulary appropriate when rotor tip loading is setting the compressor stability limit. This always occurs when the fan is subjected to severe tip radial distorsion and it is for this reason the fan stages of turbofan engines have been the first to benefit from the application of casing treatement. Greitzer [8] showed that this treatement is very efficient whenever the compressor surge is caused by end wall boundary layer separation.

3.2.6. Bleeds

In many instances the threats to stable engine operation occur rather infrequently. For example , thermal ingestion from missile firings, thrust reversing, or rapid decelerations occur over only a small portion of total mission time, but requires large amount of stability margin.

In such situations it is desirable to provide temporary additional stability via bleeds and complex control systems. A mid-compressor bleed can be used to lower the loading of stages in front of the bleed location and thus provide additional stall margin as indicated in Fig.1.

This benefit is most pronounced at low engine rotating speeds where the front compressor stages are operating toward stall (Fig.5).

The bleed and its associated control system add weight and complexity to the engine. However, the improvement in compressor performance may offset these disadvantages.
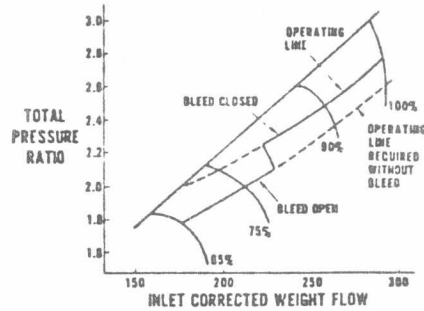
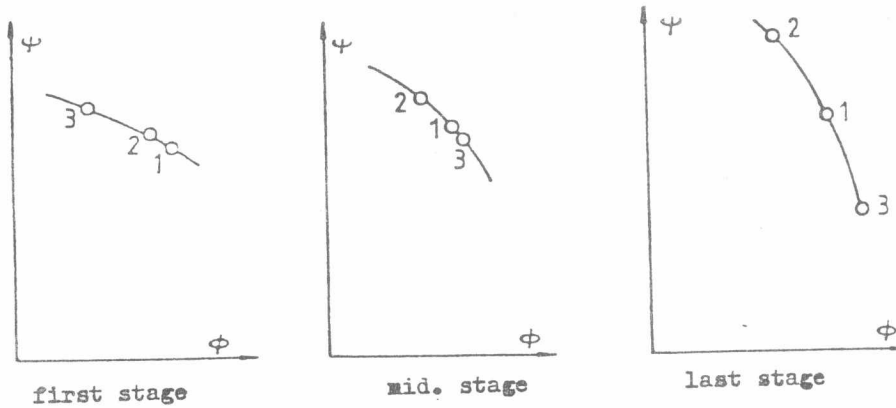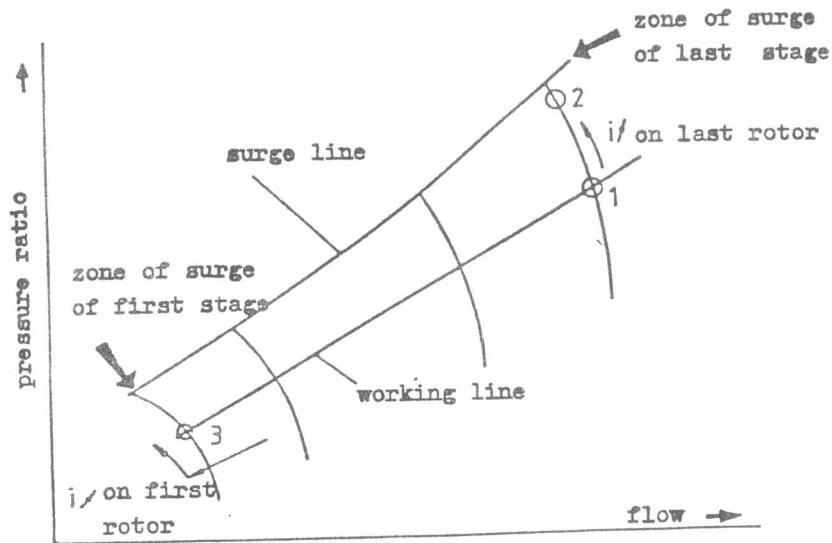Fig.4. Effect of bleed on a compressor map

Fig.5. Zones of surge of different compressor stages

3.2.7. Variable geometry

Another solution for the problem of the high loading of first stages at low speeds is to use variable stagger stators to provide flexibility for setting the desired stability margin at all compressor speeds. The front stage stators can be set at negative incidence at part speed so as to reduce the flow area and thereby reduce the front stage loadings, and then at full speed set at minimum or positive incidence to gain flow capacity and efficiency. The strong influence of using variable stator to modify the compressor map is shown in Fig.6.
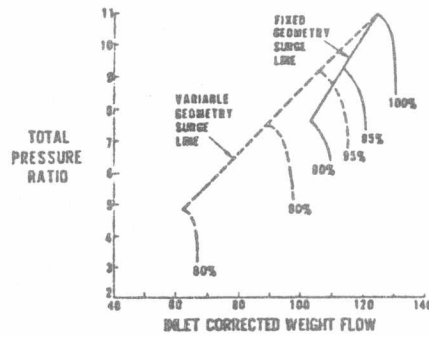
Fig.6. Effect of variable geometry on a compressor map

Fig.7 shows on the velocity triangles the function of a compressor stage rotating at a constant speed (U) when the axial velocity V_z (or the mass flow) decreases. In this case the relative flow angle with respect to rotor (β_1) increases and the stage moves toward surge. It can be useful modify the absolute flow angle (α) to conserve in off-design conditions an optimal incidence on the rotor blades. This is the case of variable stagger guide vanes or variable stagger stators in multi-stage compressor permitting the re-adaptation of the incidence angles.
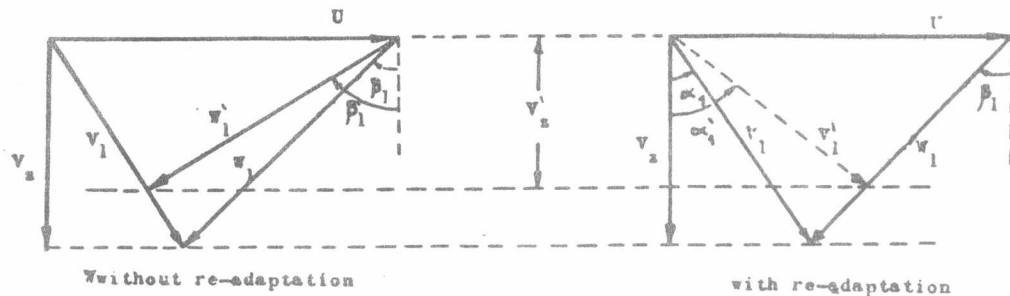

Fig.7. Effect of variable geometry on velocity triangles

4. CONCLUSION

In this paper we have discused many of the techniques used by engine designers to provide adequate stability within tolerable performance limits. The importance and criticity of the surge phenomenon were pointed out together with theoritical and practical methods to improve the surge margin of modern axial compressors. Whilst some of these methods involve trading stability for other performance parameters such as efficiency, weight or cost, others have negligible side effects. In this domain the radial pressure gradient optimization concept is very promising at least from a theoritical point of view

The need for accurate prediction of stability limits is providing an impetus for the development of two dimensional and three dimensional unsteady flow field calculations. The use of these sophisticated analytical techniques coupled with transient engine simulations should reduce stability margin requirements by 30 - 50% within the rest of this decade. The progress will be translated directly into improvements in engine fuel consumption, cost and weight.

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