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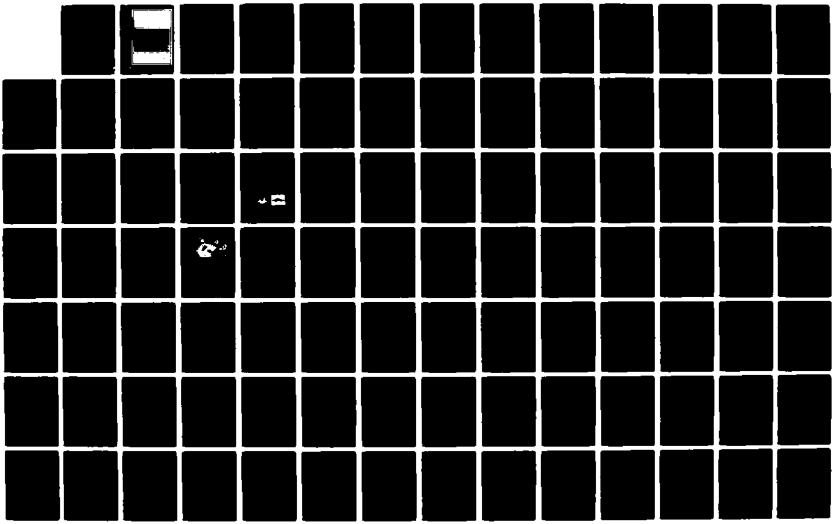

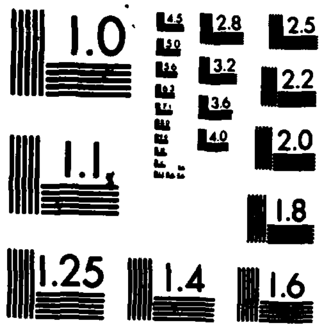

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Engine Handling

NORTH ATLANTIC TREATY ORGANIZATION

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.324

ENGINE HANDLING

Papers presented at the 60th Symposium of the AGARD Propulsion and Energetics Panel, held at the Golden Coast Hotel, Marathon, Attika, Greece, on 11–14 October 1982.

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TECHNICAL EVALUATION REPORT

by

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SUMMARY

Experience has proven that "engine handling" represents a quality criterion equally important as that of, for instance, general performance or mechanical reliability of the powerplant. It was therefore the (successfully achieved) goal of this meeting to bring together experienced pilots, engineers from the engine industry, aircraft test engineers, researchers, validation specialists, maintenance experts and, last but not least, government officials in order to cover as many aspects as possible of this very wide field.

Although some of the papers and discussions dealt with engine handling on the ground, as experienced by the service crews, the main emphasis of the meeting concerned handling as a response to throttle changes.

A plea was made by the pilots for as fast as possible thrust changes to throttle commands without having to worry about possible malfunctioning or a load of special instructions as is the case with some currently in-service engines.

On the engineering side, the following four qualities were defined as essential for satisfactory handling:

- adequate compressor surge margins with good recovery characteristics from stall,
- avoidance of excessive component clearance changes during engine transients,
- smooth and consistent afterburner light up and light across characteristics,
- control systems that can cope with problems resulting from inadequacies in the above areas.

The papers presented highlighted the fact that much effort has been devoted to handling in particular to increase the understanding of the recoverable/unrecoverable stalls mainly caused by afterburner misbehaviour. Another area represents the improved control of transient clearance changes and the definition of the resulting effect on component/engine performance. Perhaps one of the most interesting aspects of this meeting was the fact that at least some of the severe handling development problems as experienced with one particular type of engine have not been experienced at all with another engine because of some fundamental design differences in the basic engine and control system layout. Instead, this engine has given rise to some development problems in completely different areas not heard of before.

A great value of this meeting can therefore also be seen in the very open discussion of not only the commonly shared problem areas but also of the very special problems particular to the different design characteristics.

1. INTRODUCTION

"Engine Handling" represents a quality criterion of equal importance as for instance overall performance or mechanical reliability of the powerplant. It comprises a large number of individual aspects which can be grouped under the two main headings, i.e.

- engine handling in flight as experienced by the pilot
- engine handling on the ground as experienced by the service and maintenance crews.

The meeting dealt almost exclusively with the former subject and in particular concentrated on the "engine response to throttle change complex" being the most critical and difficult area.

As widespread as the disciplines involved in engine handling were the professions of the attendees of this meeting. The approximately 150 officially registered participants in the meeting consisted of experienced test pilots, service pilots, engineers from the engine industry, aircraft test engineers, researchers, validation specialists, maintenance experts and government officials of the respective departments.

The technical part of the meeting was opened with a keynote address by a former chief test pilot. The meeting itself consisted of six sessions with the aim by the organisers to keep a reasonable balance between general handling requirements and present experience with combat aircraft engine handling on the one hand and the analysis and improvement concepts in the field of transient component and interaction effects, engine modelling and control concepts on the other.

The meeting was organised and scheduled in a way that after each of the 31 presentations a lively discussion could take place. The meeting was concluded with a roundtable discussion on a range of topics previously defined by the audience.

2. CONTENT OF THE MEETING

2.1 Introduction

In his keynote address J.H.Pollitt (ex-chief pilot and consultant to Rolls Royce UK) made a plea for better engine handling with modern engines in terms of faster and more direct thrust (power) response to throttle changes, more reliability in getting the required performance change in a consistent manner and less work load for the pilot when performing a required power setting change. Ideally, the pilot should not be required to watch gauges during transients nor should he have to carry out complicated and time consuming manoeuvres when starting an engine. Reference was made both to earlier piston engines and also to modern helicopter engines where a rapid power response was/is inherent. For certain flight manoeuvres this rapid engine response is absolutely essential but even in less important areas a slow response rate would distract the pilot's attention from his main role as a fighter pilot for a longer time. A fast response and reliable engine handling would also be a good basis for future applications where the powerplant may become a more integral part of the total flight controls.

2.2 General Considerations of Design

Bauerfeind (MTU Munich) offered a general definition of good engine handling as:

"achieving a desired state with a minimum of manual effort in the shortest possible time without any undue safety risk".

This definition should hold true for any one of the major handling aspects. When concentrating on the most critical aspects, i.e. thrust changes as a response to throttle changes, four areas are usually responsible for good handling, i.e.:

- adequate compressor surge margins with good recovery characteristics from stall
- avoidance of excessive component clearance changes during engine transients
- smooth and consistent afterburner light up and light across characteristics
- control systems that can cope with problems resulting from inadequacies in the above areas.

The paper also discussed briefly two new control concepts. The first one is an open loop afterburner control which gives very fast response times (typically 1 sec from min to max afterburning), also avoiding any stability problems, whilst being insensitive to all types of irregularly delayed afterburner light across. These delayed light acrosses typically occur on military engines with larger bypass ratios but no forced mixing of the cold bypass air with the hot core flow. This control concept is used on the RB199 engine in Tornado.

The second concept is "Mode Control" aimed at reducing the need for large surge margins under normal operational conditions just to cater for an extreme situation lasting only for a few seconds. The modes of operation are derived from aircraft computer data and the throttle position respectively and determine the amount of trimming of the basic control laws as well as the opening of compressor bleed valves under extreme conditions. This concept will shortly be built into a new digital engine control.

B.L.Koff (Pratt & Whitney US) dealt with a large number of engine handling-related development problems encountered during the past 25 years with the main emphasis on the following issues:

- transient radial clearance changes can lead to a considerable loss of surge margin after accels and to rubs on decels – design measures to control clearances are essential
- a fair amount of problems can be introduced by the variable compressor stators (hysteresis, leakage, scheduling errors due to thermal effects, etc.)
- the most severe transient condition is the hot reslam or "Bodie" where a still hot engine accelerates from idle along an elevated running line in the compressor map in combination with a lower surge line due to a thermal heating mismatch
- since surges cannot be completely eliminated with most engines the blades should be designed to have sufficient life in the surge mode
- a "bowed rotor" occurring after engine shut down due to the hot air concentrating at the top can cause vibration problems during start up; an oil damped bearing can reduce vibration amplitudes by up to 50%

– airstarts can considerably improve the in-flight restart capability over normal windmilling starting.

The second part of the paper was addressed to design criteria to improve the LCF life of the hot components by:

– introduction of combustor designs with the liner shell structure shielded from the hot gas path with thermally free panels

– optimization of the shape of the gas temperature vs time curve with digital controls in order to reduce the transient strain levels in the turbine

– introduction of shaped dovetail slot bottoms and discs without bolt holes through the disc web in order to limit the turbine disc rim stress.

2.3 Experience with Combat Aircraft Engine Handling

E.J.Bull (A&AEE UK) discussed typical criteria to assess the quality of engine handling and operating procedures for Service use. From the experience with a number of engines during the last decade the following conclusions were drawn:

– engines have become more prone to surge and handling problems

– flight conditions have also become more severe

– full authority electronic controls can cope better with handling problems as they occur during development.

H.Fichter and E.Hienz (MBB Ge) described a specific handling problem as encountered with the RB199 engine when installed behind the left hand Tornado intake. The problem is caused by a counter-rotating swirl affecting the characteristic of the fan which does not feature any inlet guide vanes. Since the lack of inlet guide vanes has proven advantageous in many respects, instead of adding inlet guide vanes to the engine a modification was developed for the intake in the form of a simple fence type flow straightener. Comprehensive model and flight tests have proven the full effectiveness of this modification which is production standard now and allows the Tornado to reach the full specified angle of attack.

W.Koschel (Technical University Aachen Ge) reported on a LEADS 200 recording system introduced into an F104G wing of the German Airforce to assist the maintenance personnel in engine condition monitoring and fault diagnosis. It was concluded that cycle counting (slightly refined and using more parameters) is a valuable tool in determining LCF-life.

M.Rougevin-Baville (EMAA/BPM, Fr) made a strong point that for air combat the pilot requires a fast responding engine that does not need any special attention during transients or manoeuvres. Particular attention was drawn to the static throttle lever-thrust relationship which should avoid sudden slope changes.

G.Gothan (BWB Official Flight Test Center, Ge) highlighted two engine development problems during the official Tornado trials and shows the solution.

R.R.Hastings (National Defence Headquarters, Ottawa, Ca) addressed first of all some typical handling problems as experienced with the J85 engine in two different aircraft. The main reasons for the stalls were:

– afterburner light-ups

– Reynolds-Number effects at altitude

– transient errors in the schedules of the compressor variables

– inlet distortion

– errors in the compressor inlet temperature signal.

The situation could be improved by using anti-icing air to lower the running line under extreme conditions prior to afterburner selections and by increasing the bay ventilation to avoid false temperature signals. The second complex addressed was the determination of engine life. It was found that pilot surveys, although better than nothing, are not sufficient and cycle counters will be fitted to 15% of the total fleet engines.

J.T.Bakker (Royal Netherlands Airforce) gave a very comprehensive survey of handling problems and restrictions that can occur with modern fighter engines today. The handling problems of the F100 engine as installed in the F16 fighter after two to three years' service related mainly to the afterburner control. It was stated that the problem had been brought under control by:

– turning off the fuel supply to the outer spray nozzles of the afterburner at altitude

– observing certain restrictions in various zones of the flight envelope

– additional pilot' instructions on how to handle the throttle in case of a stall which is not automatically recovered by the control system.

On the maintenance side the modular F100 engine replacing the J79 brought a change from preventative to inspection maintenance. In spite of the advantages of the modular concept it was claimed that "an advantage in total workload is not noticeable" when compared with the J79 engine.

K.Piehl (KHD-Luftfahrttechnik GmbH, Ge) contributed handling experience with the APU as installed in Tornado. He showed test data where regular compressor washing can avoid any lasting performance deterioration. Cold start performance on the other hand was found to improve considerably by introducing additional swirl type starter nozzles in the combustion chamber.

2.4 Thermal Transient Effects on Component Characteristics

P.F.Neal (Rolls Royce Ltd, UK) pointed out that a full deflection analysis of the engine under all operational conditions is essential. More attention should be paid towards asymmetric deflections in future. Simple passive tip clearance control and the choice of low expansion material in the casings is sufficient to achieve radial tip clearances of 0.015 to 0.020 in.on the RB211 engine.

J.P.Lagrange (SNECMA, Fr) presented a methodical survey of the various types of seals and the criteria to be met with each type. He stressed the importance of proper definition and layout of the seals during the design stage so that only "fine tuning" of the seals is required during the development phase of the engine.

K.Trappmann (MTU Munich, Ge) discussed the principal methods of passive and active clearance control systems. Two examples actually relating to design solutions in the turbine areas were given.

P.Pilidis (University of Glasgow, UK) presented a simple model of predicting transient tip clearance changes. Disc hub, diaphragm, disc rim, blade and casing can be represented by equivalent elements with very simple shapes for easy handling of the model.

2.5 Aerothermodynamic Interactions and Modelling in Engine Handling – I. Compressor Systems

J.P.K.Vleghert (NLR, Ne) discussed both the low corrected speed stall where the forward stages become overloaded and the more dangerous high corrected speed stall with the loading shifting to the rear. While the first phenomenon can be encountered during the starting phase, including windmill- and airstarts in flight, the second type of stall can be caused by several different reasons, e.g.: blade deterioration, intake flow distortion, acceleration dry, afterburner transients in particular with bypass engines, control system tolerances/wear etc. The point was made that stalls and surges encountered in flight are usually difficult to reproduce on the ground, and the development of better analysis tools is desirable.

R.G.Hercock (Rolls Royce Ltd, UK) reported on distortion assessments within still manageable proportions. His paper was in many respects an update and extension of the information given in AGARD LS 72. Main topics of the paper were the measurement of flow distortion, tolerance of an engine pressure distortion, mathematical models for solving inlet distortion questions and flow distortion other than total pressure.

J.Huard's (Onera, Fr) paper dealt with the basic research on the transmission of an inlet distortion through a single stage axial flow compressor using a low cost test facility.

N.R.L.Maccallum (University of Glasgow, UK) had studied the effects of heat transfer in the characteristics of an axial-flow compressor by two different methods. It is predicted that in a particular altitude case up to 25% surge margin may be lost transiently.

S.Baghdadi (General Motors Corp., US) presented several applications of a mathematical model for compression system stability (including pressure and temperature distortion, pressure pulsations, hot gas ingestion etc.) emphasising the importance of the accuracy with which airfoil loss and turning characteristics are represented in the analytical model.

W.G.Steenken (GE, US) described extensions to a basic "Aerodynamic Stability Program" to handle with more precision both the influence of flow distortion from the aircraft intake as well as the effect of an afterburner control mismatch through the outer fan into the core compressors. The distortion effects are handled by a circumferential flow-redistribution parallel-compressor model while a radial flow-redistribution model derives the optimum splitter location with respect to the magnitude of an afterburner pulse being transmitted to the high-pressure compressor.

2.6 Aerothermodynamic Interactions and Modelling in Engine Handling – II. Engine Transient Behaviour

H.I.H.Saravanamuttoo (Carleton University, Ca) gave an overview of how an effective model of transient engine performance was being built up. He concluded that the quality of the model depends on the following four criteria, flexibility, credibility, availability, and reliability. In the discussion it was pointed out that the simulation of start up performance is usually difficult due to non-availability of component characteristics in this region.

R.Rick (Technische Universität München, Ge) presented a computer program for simulations of the steady state and transient performance of the various types of gas turbine engines. As an example the simulation of a helicopter engine operating close to the ground was given.

R.A.Onions (NGTE, UK) presented modelling techniques currently used at NGTE for transient simulation work and highlighted rather large discrepancies between predictions and actual engine results in the initial part of a very fast slam acceleration. He showed that a momentary loss of combustion efficiency would account for them. In the discussion, the relative importance of this effect and heat transfer effects was queried.

G.T.Patterson (Sverdrup Technology Inc., US) dealt with the experimental evaluation of stall characteristics of mixed flow turbofan engines with afterburners. With this type of engine considerable stability problems have been met in the US. Within the research work described stalls have been obtained by nozzle closures, fuel pulses and throttle transients. Additional dynamic instrumentation for temperature, pressure, burner flame detection, stall detection and mass flow measurement have been used. The test results have been produced in digital form. Software programs for digital filtering and flexible display options are important. Analysis of the stall mechanisms involved require techniques built around the fast Fourier transform.

H.D.Stetson (Pratt & Whitney, US) also dealt with the stall problem of the turbofan-afterburner engine. The problem is caused by non-consistent afterburner transient behaviour mainly in the upper left hand corner of the flight envelope with the standard type of afterburner control. Although the majority of afterburner-caused problems will result in a self-clearing stall some of the stalls may be of the non-recovery type. Based on a criterion presented by Greitzer in an ASME-paper in 1975 the author claimed that it is now possible to configure new engines practically free from non-recoverable stalls.

2.7 Control System Concept for Advanced Engine Handling

C.Barrouil (Onera, Fr) offered an acceleration control for a two-shaft engine based on a mathematical model. A Fortran computer program computes the ideal trajectories allowing a direct comparison at any instant of time between the actual and the ideal transient performance.

M.J.Porter (NGTE, UK) presented an afterburner closed loop control system concept of fairly high authority which he claims is also suitable for very fast transients in either direction. The system also features automatic buzz-avoidance operating on the scheduled fuel flow.

C.A.Hoelzer (Grumman Aerospace Corp., US) reported on a system analysis with respect to the incorporation of a "Full Authority Digital Electronic Control – FADEC" into an existing turbofan engine (TF30)-fighter aircraft (F14) configuration. The advantages claimed for the various areas are impressive and mainly due to:–

- establishing the weak points of a particular powerplant installation over many years of service
- analysis and understanding of the problems observed
- definition of necessary control software changes to cure particular problems
- relatively easy implementation of the necessary control software changes by employing digital controls.

3. CONCLUSIONS AND RECOMMENDATIONS

The well attended meeting with some 30 papers presented has proven the great interest in engine handling topics. The presentations and the discussions highlighted the fact that a very large number of completely different engineering disciplines are involved in and contribute to the quality of engine handling. While naturally the majority of problems presented and discussed are common to all engine manufacturers a substantial number of the problems are not shared and are obviously related to the respective individual design concepts.

As to be expected from a meeting of this nature a number of topics were not covered in very great depth or were even not covered at all. The following examples may highlight this.

(a) Apart from the pleas made by the pilots for generally shorter handling times and a more direct power response no definite requirements were specified at all; aircraft companies, institutes, pilots should make an attempt to define the handling requirements with respect to transient times of future military engines more precisely; the best transient times today are approximately 4 secs from idle to military, 1 sec from min to max afterburner, 2 secs from military to max afterburner and 5 secs from idle to max afterburner; what are reasonable and technically justifiable targets for the next generation of fighter engines?

(b) No detailed discussion took place on items like acceptable/nonacceptable throttle dead bands, desirable throttle arrangements, etc. Also the amount of future integration of the powerplant with its controls into the overall aircraft control concept needs clarification; this could have an impact on the design of the new generation of digital engine controls.

(c) Engine component life recording was only touched upon; it is still not clear whether future military engines should generally be equipped with individual life recorders; the number of parameters to be handled and with what precision and the logistics required on the ground are also yet to be defined.

(d) Although the most important physical effects governing engine transients are well established and very comprehensive computer models including these effects are available in industry there appears to be a need for simplified models for easy handling with a more transparent approximation of these physical effects.

(e) A number of US papers dealt in detail with the complex mechanisms of non-recoverable compressor stalls mainly caused by a misbehaving afterburner. Another paper defined the fairly severe operational limitations due to these problems with the afterburner control on an actual modern engine. The stalls are usually caused by an irregularly delayed light up/light across in the afterburner when operated behind a bypass engine without any forced mixing. However, none of the papers really dealt with the problems at the source, i.e. a suitable control system philosophy approach to cope with this problem and/or improvements to the afterburner itself.

In general, the meeting provided a worthwhile platform for the exchange of ideas and information in the very wide field of engine handling. It is recommended that a follow-on meeting should be held in about 2 to 3 years' time. In the meantime, however, it could prove useful if the relevant engine handling aspects were considered in other meetings that may take place on specialist subjects such as:

- control systems
- combustion
- compressors
- cockpit layout
- engine component life recording/health monitoring systems.

Any relevant information resulting from such meetings should be fed into the next meeting on engine handling.

KEYNOTE ADDRESS

by

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It is an honour and a privilege to have been asked to give this keynote address. I am not sure that I meet the job specification, as the original requirement was for a serving Air Force General. I therefore dread to think how far down the line your Committee had to go before they found me. However, the original idea of representing the views of the ultimate customer remains valid because of my role for the past 25 years, as a test pilot. Throughout the world, industry test pilots are in the same position. They are paid by the companies, but spend almost their whole working lives criticising the product. Any test pilot would jump at the opportunity of telling such a large group of senior engineers that they have all got it wrong in some way or other, without them having the chance of an immediate reply.

This Symposium is devoted to engine handling. The definition of good engine handling is difficult as the requirement differs from aircraft to aircraft and even with the same aircraft as the role changes.

As an example of this role change, I would like to consider the engine handling requirements of the Red Arrows aerobatic team. The extreme cases are the leader and the outer wing man.

Leader's Task

(1) Smooth and accurate flying to avoid any whiplash effect down the line.

(2) Great finesse.

(3) Sense of position relative to the display line and the crowd.

(4) Restrict the rpm range and rate to allow the other team members the opportunity to achieve relative position changes and compensate for errors.

Wingman's Task

(1) Maintain relative position. The problem is partly due to errors generated down the line and partly due to the fact that he is flying a slightly different flight path from the leader.

(2) Position changes require rapid thrust changes and maximum and minimum throttle positions are frequently achieved.

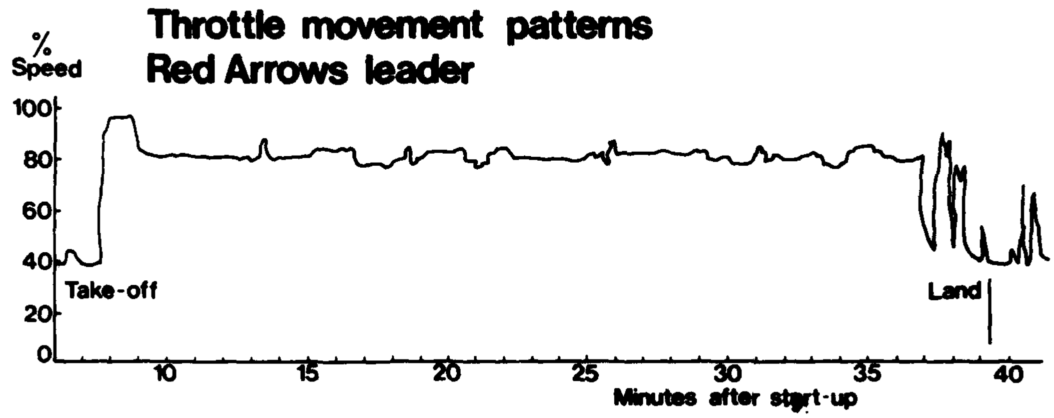
The figure overleaf, shows that the leader uses only a small part of the engine envelope, whilst the wingman uses all of it. It is almost impossible to give the wingman too much response, or thrust range. Instantaneous response to avoid error build up is every wingman's dream.

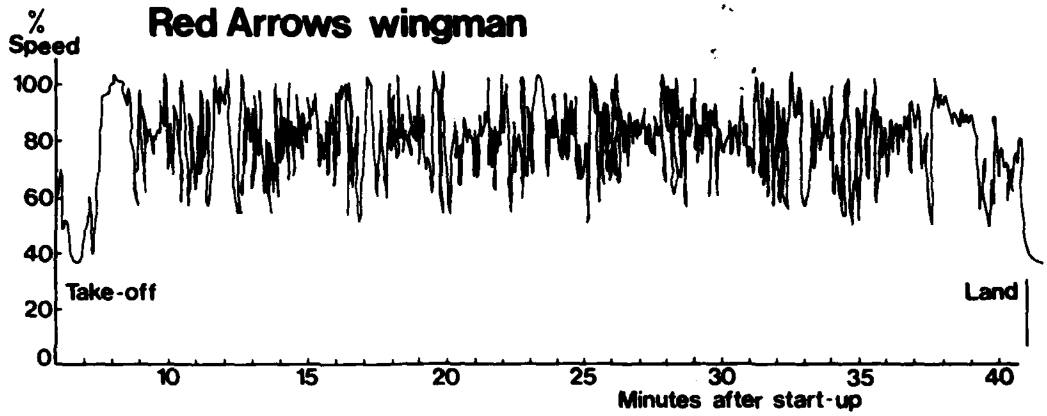
What do we mean by "good engine handling?" In general, from a pilot's point of view it means:–

1. Surge free
2. Rapid response
3. Carefree operation with auto limiting
4. No pilot monitoring requirement, allowing "eyes out of the cockpit" operation.

In many ways Items 1. and 2. conflict in that, for a given performance requirement, surge is a function of the overfuelling margin and surge can therefore be avoided by increasing the accel. time.

It is interesting to note how pilots have come to accept the present situation. When given a blank sheet of paper, not one of them would suggest that full power should be available some 5 seconds after selection. Historically, at the end of the piston engine era, engine response was almost instantaneous. The jet engine provided such an enormous performance advantage that pilots would accept the very poor engine handling. Many experienced fighter pilots had difficulty mastering the problems of jet engine monitoring and very long response times.





These days, operational requirements experts have defined minimum handling standards which inevitably are based on past experience. These requirements also tend to be secondary in so far as prime contractual targets are still payload/ range, time to height, SEP, weight, sfc, instantaneous turn rate, etc.

The engine manufacturer therefore has, more or less, a well defined engine from which he has to extract good engine handling. It is not surprising, therefore, that we are defining engine handling requirements for the year 2000 which are worse, in terms of response, than those available in 1945.

Items 3 and 4 are really a way of saying reduce pilot workload. The modern single seat fighter has a very high pilot workload. In some ways, the aircraft have become easier to fly, but the operational aspects such as weapons, navigation, ECM, etc. have increased pilot workload to almost the limit. The whole subject of cockpit workload justifies a symposium in its own right and has proved to be almost impossible to quantify. One thing is certain, and that is that each fighter pilot will tell you that *he* does not have a workload problem. The whole subject produces responses which one might expect from a virility test. There are several aircraft which are known to be difficult to fly and consequently the pilots of these aircraft wear their squadron badges with pride because they have been selected (or creamed off) and, therefore, must be amongst the best pilots.

The training systems throughout the world are orientated around the solution of difficult tasks and only rarely do the instructors suggest that the task should be made easier. I can remember that as an Air Force instructor I argued that a new trainer aircraft was too easy to fly, relative to the current operational type. It never occurred to me that the fighter workload should be reduced to that of the trainer.

What can we do? I suggest that our aim should be to reduce the pilot workload from the engine operation to as near as possible zero and suggest a couple of examples:

(1) Engine starting should be completely automatic, involving no pilot monitoring. At present this engine monitoring time could be better spent on combat equipment and represents a significant proportion of the total "scramble" time.

(2) Fully automatic control of all limits and modes – the use of analogue or digital control systems should reduce pilot workload. There is, however, a trap in that these electronic systems allow the engineers to display failures which were not feasible on hydromechanical systems. This increases workload. At all costs we must avoid warning lights and drills for subsystem failures. These electronic systems should either operate normally or when the red light comes on the pilot knows that all drills, switching and options have been exhausted. The days of wall to wall engine dials, lights and switches should be numbered.

Present generation fighters do not have an engine handling problem during air combat as the pilot uses full throttle for the whole sequence. During the initial climb at high "g" the aircraft loses speed and on the way down speed is regained into the optimum manoeuvre envelope. Future combat aircraft will have higher thrust/weight ratios and lower induced drag so that, for the first time, the pilot will need to throttle back during air combat, with the consequent need to reaccelerate the engine. The pilot who has the better engine response will have the advantage.

It is now entirely feasible to couple the engine control and response into a total energy management system. The problem is that of engine response times. It is clear that anticipating pilot commands by up to five seconds is impossible.

The traditional options between thrust, sfc and response times are hard facts of life and better engine handling will have a price. I suggest that the traditional solution to this problem is open to serious question for the next generation of fighters.

My message is a simple one. The future fighter engine will be just a part of an integral attack system which, for the foreseeable future, will have a pilot in the loop. This man/machine relationship may well turn out to be the most vulnerable part of the concept. We must give the pilot every opportunity to do those tasks which he can do better than the machine.

It has been said that the human computer is already available throughout the world and requires only unskilled labour for its production. However, it requires considerable expensive training to transform this computer into a pilot and during this period it tends to acquire opinions. This is just one of them.

SOME GENERAL TOPICS IN THE FIELD OF ENGINE HANDLING

by

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SUMMARY

A general definition of "engine handling" can probably be given as "achieving a desired state with a minimum of manual effort in the shortest possible time without any undue safety risks". The "desired state" may comprise such general items as combat readiness or a more specific item like a certain performance level.

Engine transient operations as a response to throttle changes represent one of the most critical handling aspects with respect to a potential danger of stalling a compressor under adverse conditions and/or wearing out components. These detrimental effects may have a large influence on the life cycle costs of a weapon system as well as on its effectiveness.

Based on good engine/afterburner thermodynamic models for both steady state and transient performance the use of advanced control concepts in conjunction with digital controls will lead towards optimised transients and a more even distribution of safety margins.

With the modern digital control boxes there is also a trend and a potential available to include and integrate more and more functions and features directed at fault diagnosis and/ or component life accounting.

1. DEFINITION OF THE TERM "ENGINE HANDLING"

For the aero engine engineer the term "engine handling" comprises a number of different items all of which, however, are related to the actual operation of the propulsion unit. "Engine handling" therefore defines the operational quality of an engine either relative to its own technical specification or relative to other comparable engines or simply just in absolute terms and numbers.

The handling quality of an engine can affect the life cycle costs of a weapon system in a significant way, it will certainly affect directly the effectiveness of a weapon system in combat and last but not least can be responsible for the safety record of the system. The pilots usually judge the quality of an engine by its handling characeristics in flight. This judgement together with that of the ground crew is very important to the engine manufacturer since it can also become an important aspect in a campaign to sell his engine to other potential customers. Advertising campaigns in general aviation magazines stress this point and highlight the value of good engine handling quality.

In most cases a very general definition of good handling quality can probably be given as "achieving a desired state with a minimum of manual effort in the shortest possible time without any undue safety risks." The "desired state" may comprise such general items as combat readiness or a more specific item like a certain performance level.

2. OPERATIONAL HANDLING REQUIREMENTS

The operational requirements may concern two areas, i.e.:

- combat readiness
- in flight performance

Below, some typical examples will be given for both groups.

2.1 COMBAT READINESS

The degree of combat readiness is an important cost factor since it largely dictates the number of units of a weapon system required for a given task. It is mainly confined to "handling actions" on the ground and includes all those jobs necessary to clear the aircraft for the next flight.

These jobs may comprise the replacement of a complete engine, the in field repair of an engine by a module change, the replacement of a control system or just a series of routine functional checks or the tracing and elimination of a failure that may have occured during the last mission. Also the amount of engine setting up and the need of regular trimming required may be important factors. Over the past, say, 20 years a lot of engineering effort has gone into design concepts which will considerably ease the handling of the powerplant on the ground. Changing an engine in the Tornado for instance is only a matter of a couple of hours. Most modern engines are built to a modular concept making any necessary replacement of an engine component a comparatively easy task.

However, the engine handling quality on the ground also depends largely on the quality of information available to the ground crew. This information may come from a number of sources but a definite trend towards automated analysis procedures may be observed. The new digital controls with their potential of extra calculation and memory capacity offer very attractive opportunities to ease the tasks of the ground crews. Overall it can be stated that the refinements of modern designs in this area have already led to a considerable improvement in the engine handling on the ground in spite of the fact that the powerplant with its accessories etc. has also become much more complex.

Nevertheless the handling of the engine on the ground is an area where further improvements appear necessary and also very cost effective.

2.2 IN FLIGHT PERFORMANCE CRITERIA

The general handling quality of a typical fighter engine in flight depends mainly on how the engine can cope with the criteria listed below. However, care has to be taken here since in many cases the engine handling quality may depend to a very substantial degree on the various airframe features such as type and design quality of the intake, the air- and power offtake system, the arrangement of the guns and missiles relative to the air intake, the quality of the low pressure fuel system etc.
An engine performing completely satisfactorily in one airframe with which it may have been developed in parallel could give problems when installed somewhere else. It has been recognised at least since the introduction of the high bypass ratio military fighter engines that the interfaces between engine and airframe have to be observed very carefully. However, the engine manufacturer in most cases has a vital interest to design his engine such that its dependance on installation features is minimized in order to keep the market for the engine as open as possible. Independent of all these considerations a pilot flying one of these modern aircraft will judge the handling quality of the powerplant by the amount and severety of problems it may give under the following conditions:

- high angle of attack
- air and power offtakes including emergency requirements
- transients as a response to throttle changes
- hot/cold day climates
- firing of guns and missiles
- starting and restarting in flight
- use of the thrust reverser (if available)
- bird strike.

A very important criterion is the handling time to execute a performance rating change as demanded by the throttle. There are very obvious cases where a couple of seconds slower response may even already make the difference between life and death. A typical example is a necessary overshoot after an attempt to land on a short runway perhaps with the aid of a thrust reverser.

Another very important requirement and quality criterion is the desirable avoidance of restrictions with respect to throttle handling. The engine manufacturer should not have to concern pilots with detailed requirements on how to handle the throttle when selecting afterburner for instance or to have to watch cockpit gauges indicating engine parameters before further advancing the throttle beyond a certain point. This represents a distraction of the pilot from his essential mission tasks and lowers his confidence in the powerplant of his aircraft.

3. TRANSLATION OF THE PILOT'S OPERATIONAL HANDLING REQUIREMENTS INTO ENGINE PERFORMANCE REQUIREMENTS

Good engine handling qualities in the air mean avoidance of engine problems under the various conditions as listed under 2.2. What are the potential problems an engine can be threatened with? The most common ones known to every control system engineer are listed below:

- shaft overspeeds
- turbine overheating
- overpressures
- excessive aerodynamic speeds
- compressor stall and surge
- blade flutter
- rubbing out of seals / damage of coatings

- engine/afterburner flame outs
- afterburner buzz (low frequency in longitudinal mode)
- afterburner screech (high frequency in transversal mode)

Apart from the obvious target of avoiding the above problems it is equally important to design an engine such that

- in case of a surge, it should be self-clearing after removal of the cause

and

- in case of one of the more likely mechanical failures occurring the engine should have fail safe characteristics.

4. ENGINE TRANSIENTS AS A RESPONSE TO PILOT'S THROTTLE CHANGES

One of the most demanding handling aspects is the response of an engine to pilot's *throttle changes*. With a typical two/three-shaft military bypass engine equipped with an afterburner (which from a control management point of view represents the most demanding engine configuration) the following four distinct modes of transient operation have to be dealt with:

a. transients in the non-afterburning (dry) mode in both directions between idle and maximum dry rating

b. transients in the afterburning mode in both directions between minimum and maximum afterburning with the engine at constant max dry rating

c. transients in the afterburning mode in both directions between maximum dry rating and maximum afterburning with the engine at constant max dry rating

d. transients between idle and maximum afterburning in both directions with both the dry and afterburning modes overlapping in order to reduce the transient times.

The underlying mechanisms for these four modes of transient operation are very different and can be described as follows:

to a:

A transient condition either towards more or less thrust is initiated by an over - or underfuelling respectively relative to the steady state condition. As a result the H.P. compressor running line level is raised or lowered respectively (Fig.1a). The most important criteria to be watched are H.P. compressor surge and/or turbine overtemperature during an acceleration, and a flame out (and to a lesser degree blade flutter) during a deceleration.

Typical effects on the fan running line are minimal (Fig.1b) but the fan speed seems to have a tendency to lead on a transient relative to its steady state relationship.

In the case of a three-shaft engine like the RB199 the effect on the I.P. compressor running line during a transient depends (amongst other factors) on the distribution of the polar moments of inertia between the three shafts. As an example such an I.P. - compressor running point shows only a rather small displacement during transients in either direction relative to its steady state running line (Fig.1c).

to b:

Afterburner transients are controlled by two parameters, i.e. afterburner fuel flow and propulsion nozzle throat area. Thermodynamically the resulting afterburner temperature rise must be matched to the nozzle throat area so as to minimize the effect of the afterburner on the engine. This effect is usually restricted to the position of the fan running point with negligible effects on the core engine compressor(s). The most important criteria to be watched are therefore fan surge and/or afterburner flame out, and in addition with some fans, fan blade flutter. It has been found that a fixed relationship between temperature rise (degree of afterburning) and propulsion nozzle area gives consistent fan running lines over the entire flight envelope. Typical running line displacements are shown in fig.2.

to c:

In addition to mode b a light up of the afterburner has to be performed. This usually consists of

- filling up of the empty fuel manifolds
- preopening the propulsion nozzle area
- lighting the fuel.

On the way down the afterburner has to be cancelled and the fuel manifolds emptied.

to d:

This mode is usually the most difficult one since the afterburner has to be lit well before the engine has reached its maximum rating. This entails in the critical afterburner light up phase lower pressures and afterburner entry temperatures than with mode c which may lead to potential problems in the upper left hand corner of the flight envelope.

Again similar to b and c any mismatch between the size of the nozzle area and the afterburner temperature rise will adversely affect the fan operating point, and in this particular case even the fan running line since the engine speed level is still changing while the afterburner boost is increasing towards its maximum.

5. DESIGNING FOR GOOD TRANSIENT ENGINE PERFORMANCE

Over the past 10 to 15 years it has become obvious that in particular the military multispool bypass engine with afterburner requires a lot of attention during the design and development stages to achieve a satisfactory handling performance. The potential problems are usually greater with increasing bypass ratio. In addition the three spool arrangement requires the application of certain special design criteria during the design stage to make full use of its good handling potential. However, it has been demonstrated that when designed carefully and when employing the most suitable and effective control systems these modern engines can have excellent handling characteristics and the severe problems as experienced with some engines in the past can be avoided.

5.1 ADEQUATE AND STABLE SURGE MARGINS

One of the most complex issues when designing a new engine is that of the required fan and compressor surge margins. Unfortunately no fixed rules exist for defining the required surge margins. There are operational engines which need 30% and more while others handle perfectly with about 5%. Since surge margin is in essence the difference between the position of the surge line and the operating line any transient changes of these two lines in combination define the minimum steady state surge margin required for troublefree handling. Some typical influences on the movement of both lines are:

a. Influence on fan and compressor surge lines

   - distortion from the aircraft intake during the various flight manoeuvres
   - aerodynamic interaction effects between the various compressors in a multispool arrangement
   - interstage service bleeds
   - variation of tip clearances due to changing metal temperature levels (steady state and transient)
   - Reynolds Number effects
   - deterioration of blade quality
     - foreign object damage
     - dirt and surface roughness
   - deterioration of tip clearances and seals.

b. Influence on operating lines

   FAN:

   - depends almost exclusively on quality of afterburner control (amount of steady state and transient mismatch between nozzle area and afterburner temperature rise)

   I.P.-COMPRESSOR (in case of a three spool arrangement):

   - rate of accelerations and decelerations (only secondary effect)
   - power off take from HP shaft
   - service bleed

H.P.-COMPRESSOR:

- rate of accelerations and decelerations
- power off take
- service bleed (interstage and from behind compressor)
- engine deterioration

Experience has shown that fan surges may cause severe mechanical damage to the engine and therefore it is very important to eliminate them by good design.

5.2 AVOIDANCE OF TRANSIENT RUBBING BETWEEN ROTATING AND STATIC PARTS

Any change of the rotational speed of the spools is normally accompanied by a change of the gas temperature and pressure levels in the various components. Depending on the heat transfer and the material and dimensions of the various component parts their material temperature will follow the gas temperature changes with varying time lags. Since a material temperature change will always lead to a dimensional change of the component part, it is probable that during transients a rotating part will grow into a static part or vice versa. This leads to a rub of the two neighbouring parts with usually detrimental effects on performance , handling margins or even to destruction.
The critical areas in an engine are therefore the many seals including the tip seals of the compressor and turbine blades.

Measures to avoid severe problems usually concentrate on

- design of seals
- use of material (coatings etc.)
- temperature response matching

Both for civil and military engines this is a very important area which is receiving increasing attention from the engine designers. With quite a number of engines the acceptable acceleration and deceleration rate limits of the spools are dictated by the transient thermal mismatch of the components with its corresponding results rather than by the available surge margins. The worst condition from this point of view is usually an engine shut down from full rating by closing the fuel shut off cock. With most engines also a warm up period at or near idle speed is required/recommended.

5.3 GOOD AFTERBURNER LIGHT UP AND FLAME PROPAGATION PERFORMANCE

The afterburner behind a bypass engine represents a potential source of problems. It burns large fuel flows when compared with the actual engine fuel flow rates and raises the exhaust gas temperature to levels well above 2000 K. The energy flows involved are enormous. While the potential problems are still relatively easily manageable with low bypass ratio engines and even more so with straight jet engines theses potential problems usually increase significantly with the larger bypass ratios above say 0.6. While the earlier low bypass ratio engines normally employed a gas stream mixer arrangement (like the Rolls Royce Spey in the F4) the modern high bypass ratio engines (like the F100 or the RB199) have no such arrangement mainly in order to save structural weight and length. This means that a large portion of the fuel has to be burnt in the cold stream at comparatively low pressure levels. This entails potential problems when lighting the fuel on afterburner selection and when lighting across the individual zones during an "acceleration" of the afterburner towards maximum boost. The response of the temperature rise to fuel flow changes is considerably less definite than in the engine combustion chamber which can lead to severe problems. Depending on the type of control concept used the potential problems can range from a nuisance to a definite flight safety hazard.

More design and development effort seems to be required to improve the fuel light up performance in the cold stream with modern afterburners.

5.4 OPTIMAL CONTROL CONCEPTS

The main requirements for a good control concept are first the general avoidance of problems, i.e. to keep the engine within its safe limits. However, in case this is not fully achieved in an exceptional situation the system should have a self recovering characteristic without having to concern the pilot with complicated instructions.

The most difficult task is usually the avoidance of fan and compressor surges respectively. In spite of a lot of invested effort in this area no practical concept is available for the direct control of surge margin and hence surge prevention by taking the necessary action quickly enough when approaching the stall point. The basic problem still appears to be the sensing of the critical condition and the lack of time available to make the necessary response.

Therefore all the present control concepts tackle the problem by just controlling the operating point or line in the compressor characteristic. The operating point is normally defined by a pressure ratio at a given nondimensional mass flow. Since both quantities are difficult to measure, other parameters have to be used instead. Many possibilities exist and every engine manufacturer seems to have his own favourate candidates.

Typical examples are:

a. For controlling the fan operating point with afterburning

CLOSED LOOP:

- turbine pressure ratio vs nondimensional speed
- nondimensional N_{LP} vs N_{HP} speed
- fan exit Mach Number

OPEN LOOP:

- scheduling afterburner temperature rise vs nozzle throat area

b. For controlling the HP compressor operating point during engine transients

CLOSED LOOP:

- N_H/P_{t1} vs nondimensional speed
- turbine temperature/ compressor entry temperature vs nondim. speed
- exit Mach Number
- pressure ratio vs nondimensional speed

OPEN LOOP:

- Nondimensional engine fuel flow vs pressure ratio or vs nondimensional speed

The overwhelming majority of control system engineers has a distinct preference for the closed loop control because of its apparent accuracy potential. However since closed loop controls usually have to control derived parameters which could also be adversly affected by a number of so called secondary effects like air and power off takes, Reynolds Number levels, engine deterioration etc. a substantial portion of this accuracy potential may be lost. In addition high authority closed loop systems have their dynamic limitations when it comes to very rapid response requirements. They are usually not "clever" enough to detect a problem, i.e. a deviation from what they have been designed for. This can lead to severe problems for instance in the case of irregularly delayed afterburner light up's or a stall during the acceleration of the engine spools. The latter can also be particularly severe when employing the otherwise very attractive N dot - acceleration control. In the case of even a mild stall during an acceleration (for instance because of an excessive power off take at the same time) the rate of the spool speed increase drops and the control system tries to regain the prescribed level by opening the fuel tap which drives the compressor deeper into surge instead of recovering by reducing the fuel flow. Only some additional features like a fast responding pyrometer sensing the surface temperature of the turbine blades and designed with full authority over the N dot acceleration control can cut the fuel flow to an acceptable level.

On the other hand closed loop controls are very attractive when it comes to speed-, temperature- and pressure control. But already with the afterburner buzz and screech control the argument has been reopened of whether it is better to rely on a sensor with a closed control loop on afterburner fuel, or to design the fuel schedule such that the dangerous region of buzz and screech can be safely avoided. In any case an "optimal" control concept depends on a number of factors, the most important ones being:

- type of engine
- in house available background experience
- control system hardware available (hydromechanical, electronic analogue/ digital etc.)

5.5 MODE CONTROL

The surge prevention control concepts whether closed or open loop as briefly discussed under 5.3 have the following shortcomings:

- they cannot sense any deterioration of the surge line and therefore have to be based on the worst case
- with most concepts the so called secondary effects, which are not measured or corrected for, can play an important role and their respective effects must be allowed for

Both reasons increase the static surge margin which must be provided, thereby sacrificing steady state performance and requiring more compressor stages. Mode control is a concept presently under development at MTU (Ref.1) which aims at:

- establishing the actual requirements of the engine at any given instance of time under any flight and handling condition
- executing the necessary measures by trimming the two fuel flows, nozzle area, bleed valves etc. to cope with any momentary situation.

Fig.3 illustrates the principle arrangement of mode control. Mode control consists effectively of two steps. Firstly, the working conditions and the requirement of the pilot from the engine at any instance of time have to be established. This defines the mode of operation. The information required comes from two sources namely the aircraft computer and the pilot's lever position and its rate of change respectively. Secondly, the established mode of operation has to be translated into control actions to trim the engine variables such as to produce the most suitable and adequate engine performance at any given instance of time. This could mean a requirement for an increased surge margin in the compressor(s) because of a severe aircraft manouvre resulting in a distorted flow into the engine perhaps accompanied by a rapid engine acceleration. Under these conditions it is acceptable to sacrifice thrust and SFC for a short period of time until the engine operation returns to a more normal status. Another example for a typical mode of operation is a cruise condition where a minimum of surge margin is permissible, maximum thrust not required but the lowest possible fuel consumption essential. For the first example an opening of the bleed valves and of the propulsion nozzle to offload the compressors is indicated while for the second example a closing down of the propulsion nozzle to pick up fan efficiency at a slightly higher pressure ratio than normal could be the right measure.

Fig.4a shows -as an example- a set of parameters which allows the computation of a number of modes (I to VI) characterising the severity of the working conditions of the engine in terms of distortion (type and magnitude), Reynolds-Number, $N/\sqrt{\theta}$ - level, service power offtake, service air offtake. Fig.4b explains -again as an example- how the requirement of the pilot on the engine at any instance of time can be directly derived from the pilot's lever itself. Here the essential operational criteria have been listed for a typical military engine with afterburning leading to modes 1 to 9.

It is now necessary to combine modes I to VI with modes 1 to 9 such that the required control actions can be determined. This could be carried out in a matrix as shown in Fig.5. The particular matrix shown produces 54 different parameters each one defining a particular control action of a certain magnitude. It is obvious that the number of parameters $x_{\nu-\mu}$ should be in a reasonable relationship with the number of control variables available on a particular engine. If the latter are somewhat limited then a number of different $x_{\nu-\mu}$ parameters may define identical control actions.

6. THE POTENTIAL OF DIGITAL CONTROL FOR FUTURE ENGINE APPLICATIONS

The introduction of the full authority analogue electronic control systems for instance on the Olympus engine in the Concorde and on the RB199 in Tornado represent a major step forward in modern engine control techniques. The next obvious step is the introduction of the digital electronic controller. It opens a large number of additional possibilities but will also require considerable attention with respect to very fast transients. Here the typical sampling and dead times of present digital controls require special measures in order not to degrade the control quality. The next generation of faster computer CPU's will, however, reduce these potential problems. On the other hand its potential for storing complex informating and managing much more complex functions offers a chance for a higher overall quality of control in terms of achieving all of the essential handling requirements with less engine performance margins being wasted.

Another important aspect is the ability to memorise events during flight operation. This opens a great opportunity for carrying out different types of functional checks, analysing them and identifying the necessary actions for the pilot in flight and/or the ground crew when servicing the engine. A further possibility is life and health monitoring of the engine components.

While BITE (Built in test equipment) to self check the electronic control system prior to take-off and FIM (Fault identification module) to identify control system faults in flight appear to be fully accepted features for digital control systems, the philosophy on how to proceed with the other monitoring systems is not yet clear.

Nevertheless the digital controller offers for the first time the chance to cope in one box with the full spectrum of all the handling features, i.e. combining the in flight performance criteria with the combat readiness aspects affecting the efficiency of the engine service on the ground.

Ref.1:
Bauerfeind K.: Mode Control - A Flexible Control Concept for Military Aircraft Engines - AGARD Propulsion and Energetics Panel 54th Meeting 1./2. October 1979

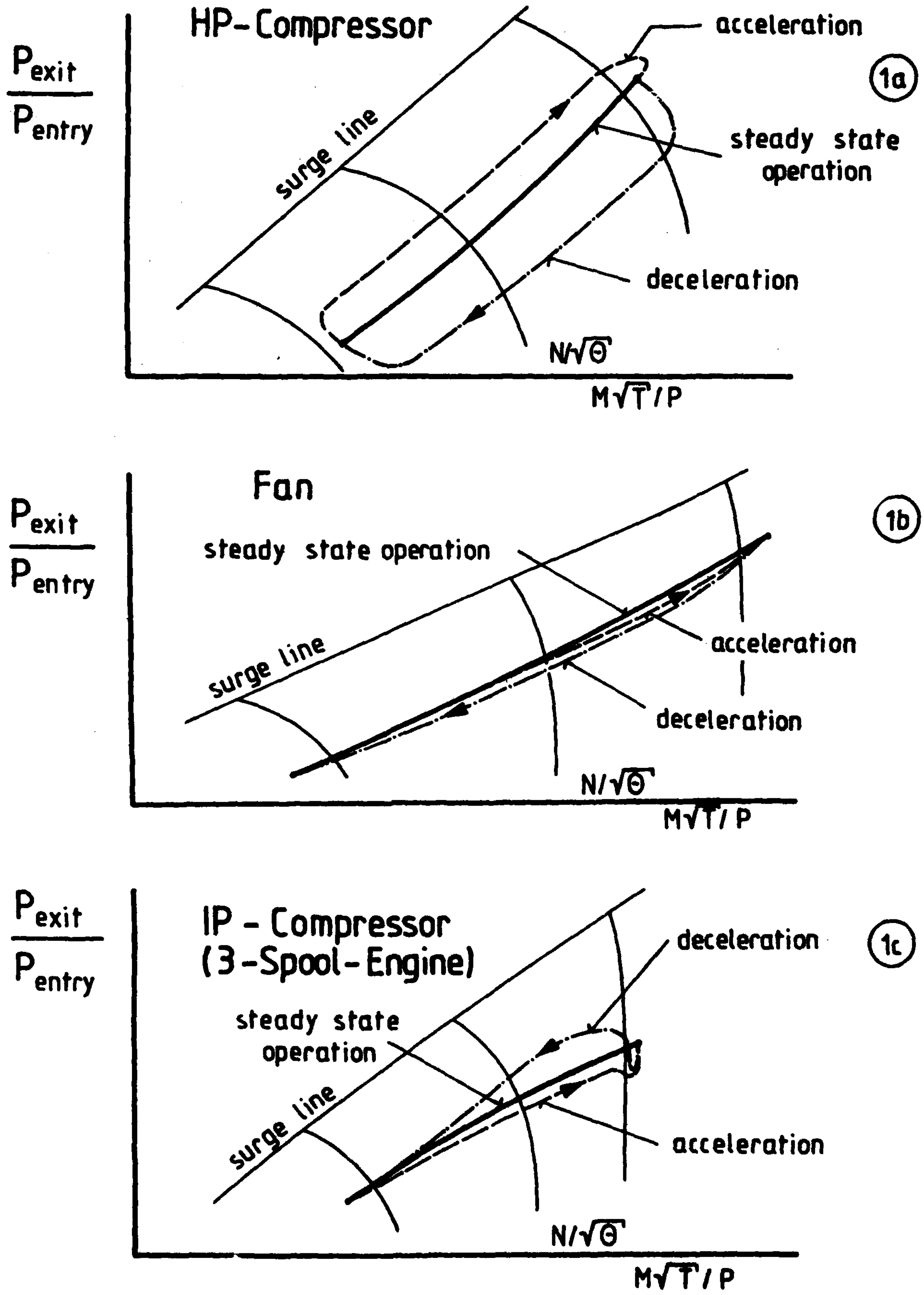
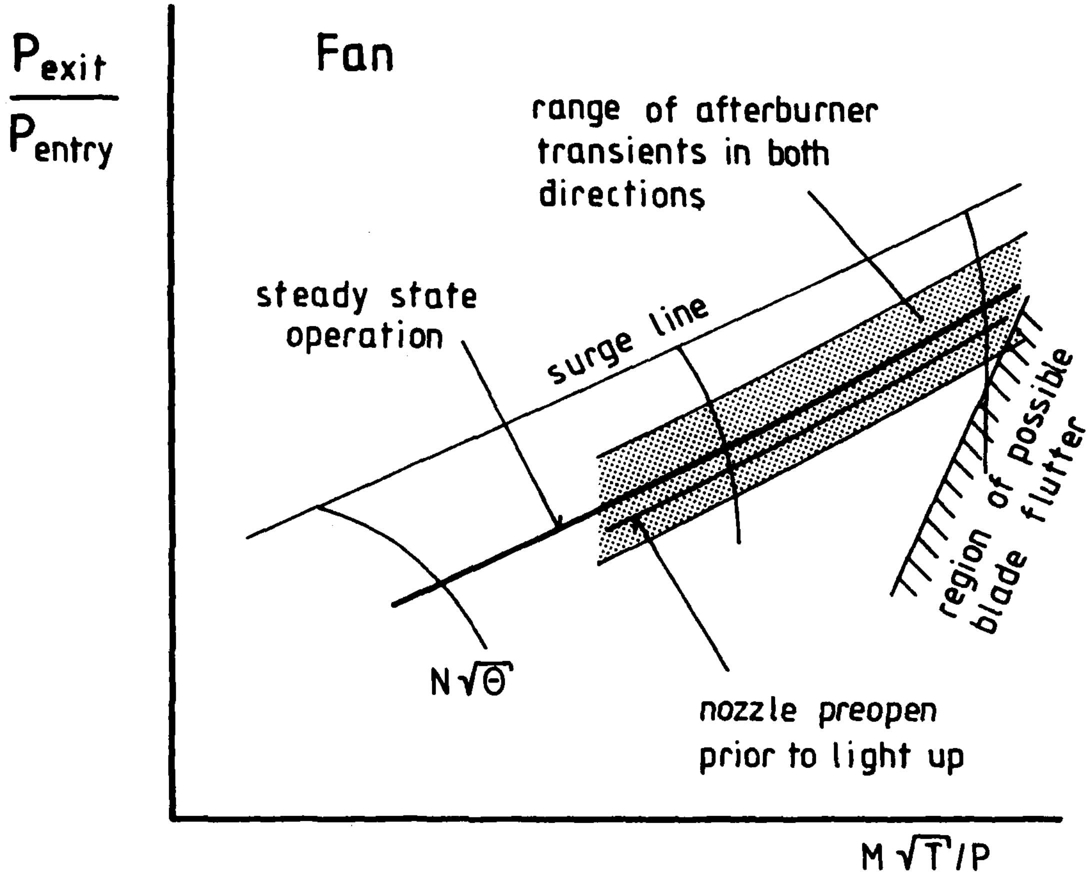


Fig.1 Typical steady state and transient operating lines for dry operation



Note: There is only a small second order movement of the core compressor working point

Fig.2 Typical steady state and transient operating points for afterburner operation

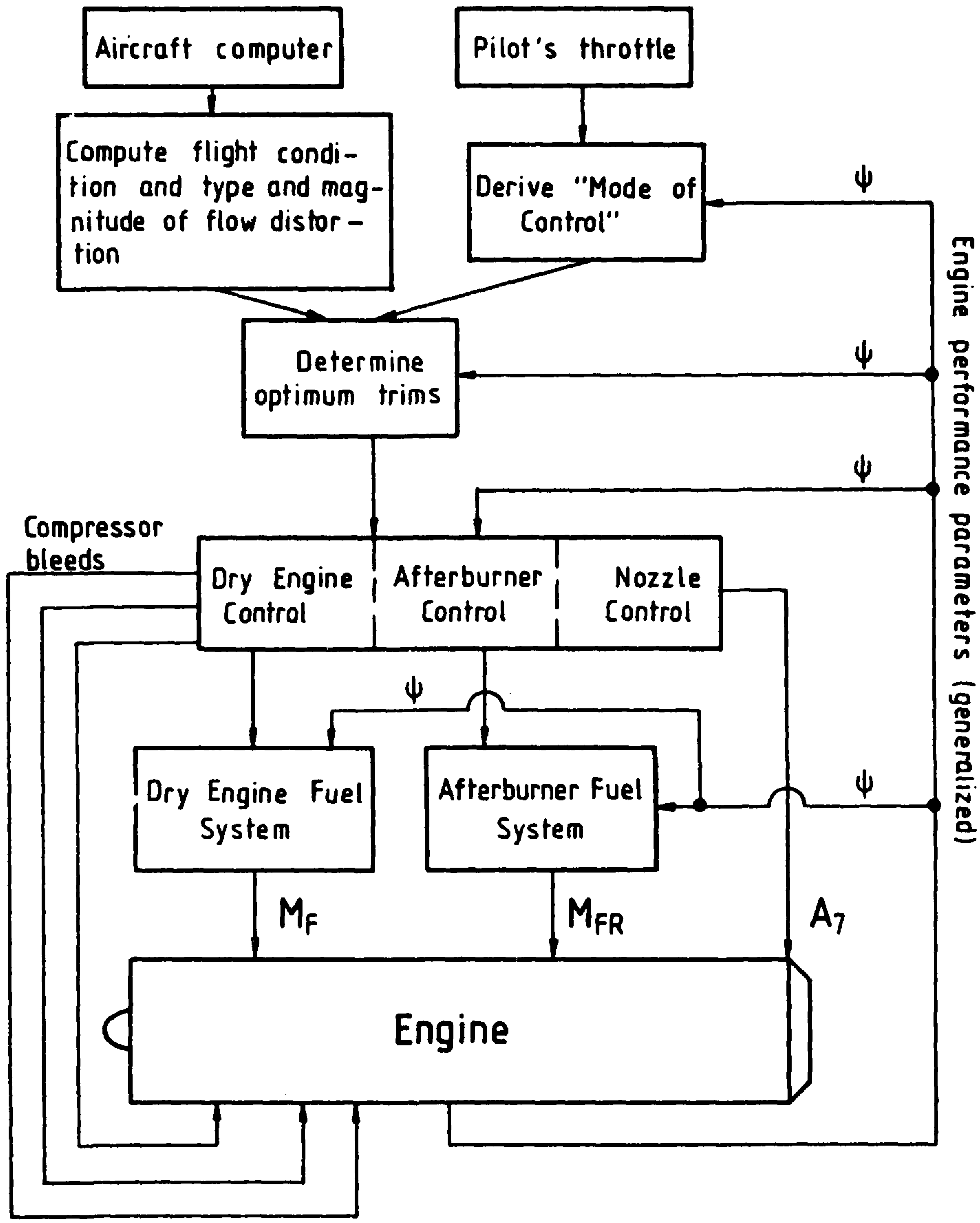


Fig.3 Principal arrangement of mode control

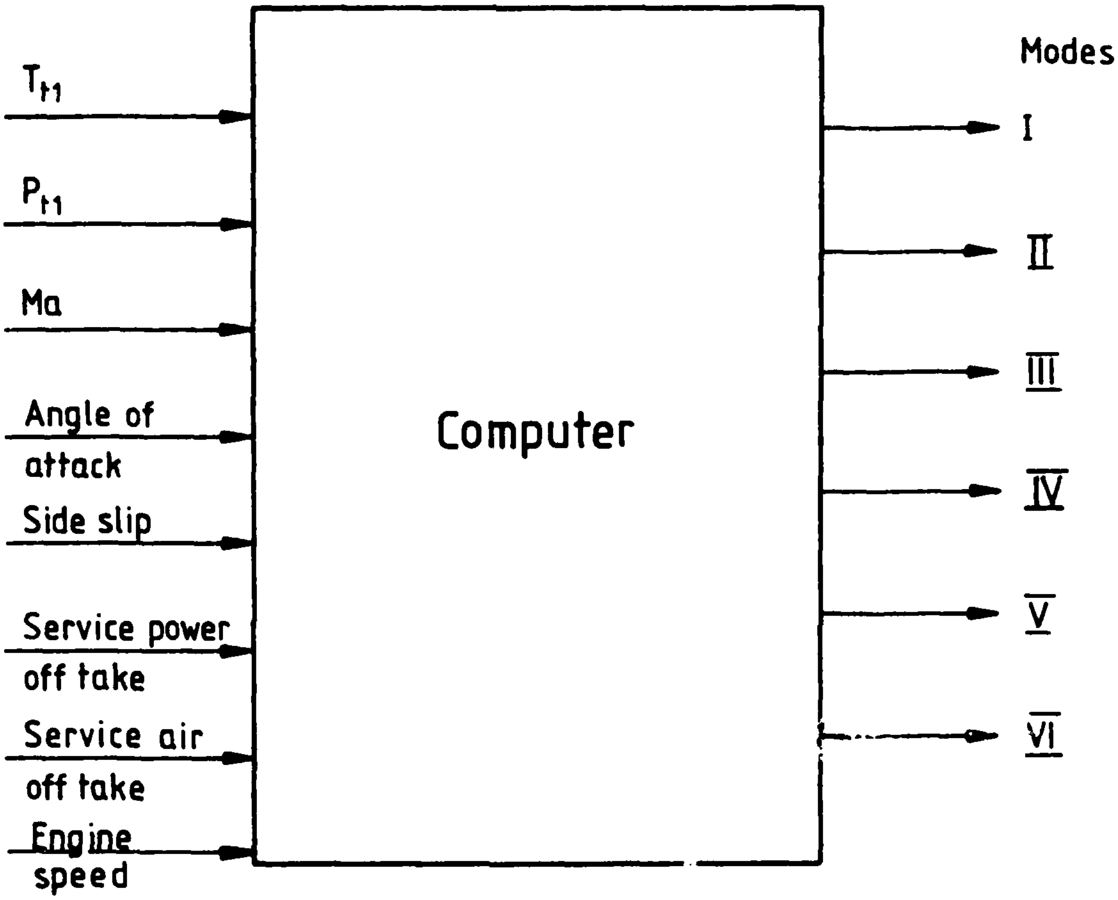


Fig.4(a) Typical fighter engine modes of operation
Modes based on aircraft computer data

Nº	Mode of Operation	Source of Signal for Selection of the Mode	Essential Requirements
1	Idle	$\alpha = \alpha_{idle}$	lowest thrust
2	Cruise	$\alpha_{max\ dry} > \alpha > \alpha_{idle}$	lowest installed SFC
3	Max Dry	$\alpha = \alpha_{max\ dry}$	highest max dry thrust
4	Max Afterburner	$\alpha = \alpha_{max\ AB}$	highest max AB thrust
5	Dry Acceleration	$\alpha_{idle} < \alpha < \alpha_{max\ dry}$ and $(\dot{\alpha} > a$ or $dN_H / dt > b)$	sufficient surge margins
6	Dry Deceleration	$\alpha_{idle} < \alpha < \alpha_{max\ dry}$ and $(\dot{\alpha} < a$ or $dN_H / dt < c)$	sufficient surge margins
7	Afterburner Acceleration	$\alpha_{min\ AB} < \alpha < \alpha_{max\ AB}$ and $(\dot{\alpha} > a$ or $dA_7/.dt > d)$	sufficient surge margins correct fuelling
8	Afterburner Deceleration	$\alpha_{min\ AB} < \alpha < \alpha_{max\ AB}$ and $(\dot{\alpha} < a$ or $dA_7/dt < e)$	sufficient surge margins correct fuelling
9	Slam from Dry into Afterburner	$\alpha > \alpha_{min\ AB}$ and $(A_{7max\ dry} < A_7 < A_{7min\ AB}$ or $dA_7/dt > d)$	sufficient surge margins correct fuelling

Fig.4(b) Typical fighter engine modes of operation
Modes based on pilot's throttle α

Mode of operation from aircraft computer data	Mode of operation from pilot's throttle								
	1	2	3	4	5	6	7	8	9
I	X_{1-I}	X_{2-I}	X_{3-I}	.	.	.	.	.	.
II	X_{1-II}	X_{2-II}	X_{3-II}	.	.	.	.	.	.
III	X_{1-III}	X_{2-III}	X_{3-III}	.	.	.	.	.	.
IV	.	.	.	.	.	.	.	.	.
V	.	.	.	.	.	.	.	.	.
VI	.	.	.	.	.	.	.	.	.

X_{1-I} ...... X_{9-VI} define the control actions in terms of trims, operation of air bleed valves etc. Depending on how many control variables are available, different values of $X_{v-\mu}$ may define the same control actions.

Fig.5 Possible control matrix arrangement to determine required control action

DISCUSSION

D.P. Davidson, UK

As far as I am aware, the RB199 engine is the only one in service with an optical pyrometer with such a high level of authority. How important do you regard it in achieving the dynamic performance of the Tornado aircraft?

Author's Reply

This pyrometer had been included to achieve a precise turbine temperature control. Because of its fast response it allowed us later in the development programme to include $\dot{N}_H$-acceleration and -deceleration control without the danger of turbine over-heating in case of a stall during an accel since the pyrometer is fast enough to cut the fuel. This $\dot{N}_H$-control has a number of attractions, one being a consistent accel/decel performance without requiring any trimming during service inspections to maintain transient times.

DESIGNING FOR FIGHTER ENGINE TRANSIENTS

by

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ABSTRACT

Operational experience with high performance fighter engines has highlighted the effects of thermal transients on performance, durability, weight and cost as an increasingly important consideration in engine design. Transient thermals within the engine, resulting from rapid throttle movements, cause changes in radial and axial clearances, variable stator vane position, rotor balance, hot part stresses and aerodynamic matching of components. The effect of these transients causing differential thermal expansion and aerodynamic mismatching of components must be accounted for in the design to avoid maintenance action resulting from abnormal engine operation such as stall, high fuel usage and excessive vibration. The component design involving configuration, materials selection, structural analysis and producibility is strongly influenced by the thermal response of rotors, cases, seals, sensors and hot section parts.

The designer's challenge is to accurately predict and provide configurations that minimize the influence of transient thermal changes within the engine.

INTRODUCTION

The "lessons learned" during the past 25 years have shown that the traditional design tradeoffs in performance, durability, weight and cost are greatly influenced by engine usage where cyclic transient operation is the major factor. For fighter aircraft, such as shown in Figure 1, rapid engine thrust response is required for takeoffs, landing waveoffs, formation flying, touch-and-go landings, refueling and air combat. Providing fast engine response to throttle movements imposes severe thermal transients, particularly on those components with surfaces exposed to the flowpath. These thermal transients cause temperature gradients, differential thermal expansion and aerodynamic heating. Performance, operability and parts life are directly impacted by these effects. In the absence of oxidation or foreign object damage, the part life is a function of the cyclic stress variation created by mechanical and thermal loading. The overall performance, including operability, depends on minimizing the axial and radial clearance changes between rotating and stationary components. It is important to minimize the effect of these clearance changes on engine performance, operation and deterioration in order to reduce the cost of ownership and improve operational readiness.

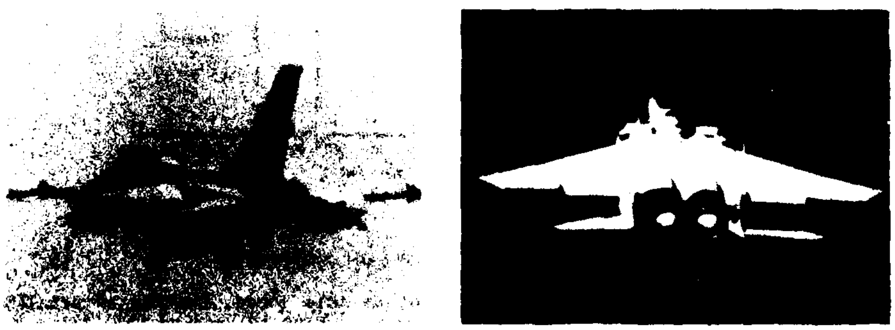

FIGURE 1
FIGHTER AIRCRAFT TRANSIENT OPERATION - AN ENGINE DESIGNER'S CHALLENGE

THE THERMAL TRANSIENT - A DESIGN CHALLENGE

Fighter engine designers must take into account the effects of thermal transients in both the design of the components and how they operate as a system. Major transients and their effect on engine design resulting from aircraft maneuvers will be discussed with primary attention to the turbofan gas generator.

The development of powerful computers and related software has enabled improved analytical and experimental design techniques to more accurately model and predict the effects of transients. Prediction and verification of transient thermal response, aerodynamic mismatch of components, internal cavity pressures and structural dynamic effects are essential to the design and development process after dealing with the better understood steady-state analysis.

Figure 2 shows engine throttle position as a function of time for a typical fighter engine mission involving a "cold" to "hot" thermal cycle and a minimum to maximum speed mechanical cycle. Acceleration and deceleration between idle and military (MIL) power and higher (high rotor speed) normally defines the thermal-mechanical loading absorbed by the engine turbomachinery.

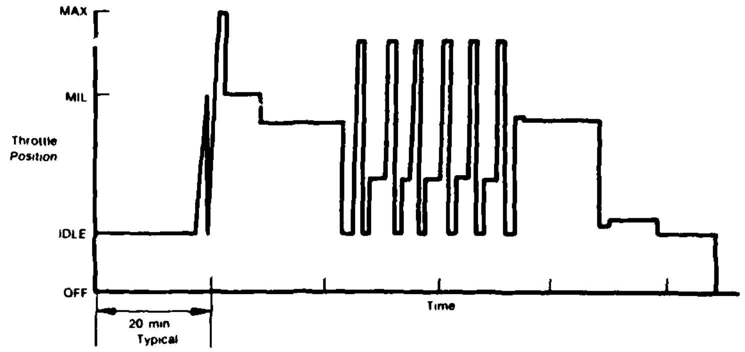

FIGURE 2
TYPICAL FIGHTER AIR COMBAT DUTY CYCLE

Throttle movements to maximum power involve afterburner operations which imposes back pressure fluctuations on the turbomachinery with negligible change in rotor speed. In the case of an afterburning turbofan engine transient, ignition initiated pressure waves are transmitted through the fan duct, causing a momentary increase in fan operating line and reduction in stall margin.

The largest heat transfer effects due to throttle movements occur in the high spool, or core components, since these parts operate at the highest cycle temperature and pressure.

COMPRESSOR TRANSIENT OPERATION

Figure 3 illustrates the relative radial excursion of a high pressure compressor rotor and stator during an accel and decel transient.

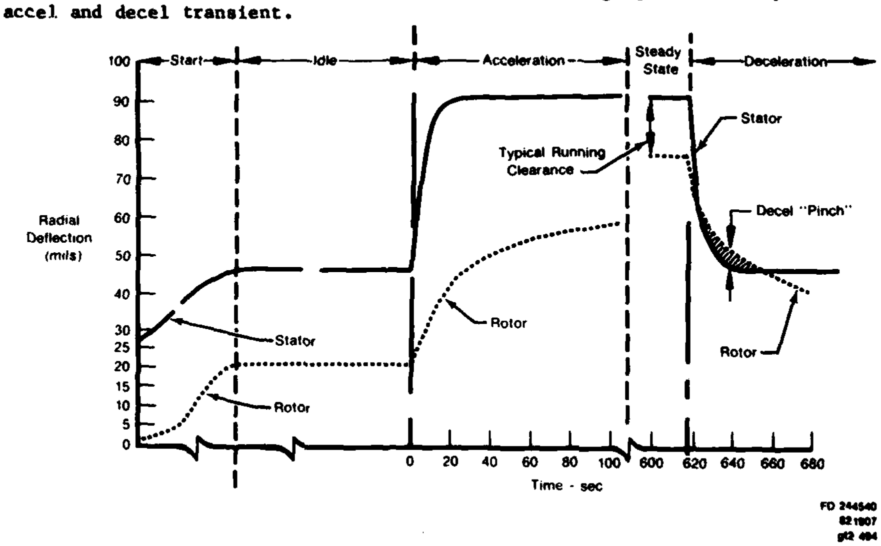

FIGURE 3
TRANSIENT RADIAL CLEARANCES FOR HIGH COMPRESSOR

Initially, the rotor and stator are relatively cool at idle speed, showing a radial clearance. During an accel to full speed, the radial clearance increases, primarily because the stator has a lower thermal lag and heats up faster than the rotor disks, which have a small surface area exposed to the hot flowpath. The increase in blade radial clearance during the accel reduces the compressor stall margin. However, for variable geometry compressors, the effect of increased clearances is offset by a hysteresis lag of the variable stators in the closed direction, tending to increase stall margin during this transient. This is the rare case where hysteresis is beneficial. A more ideal situation is to have the control system schedule the stators.

The radial clearances close as the rotor disks heat to stabilized operating temperature. At full speed, after thermal stabilization, the radial clearances are tighter than at buildup, primarily due to the rotational stress growth of the disks. The running clearances at stabilized operating conditions are a major factor in component efficiency and overall engine specific fuel consumption.

A sudden decel from stabilized maximum speed conditions can cause a rub as shown by the shaded area in Figure 3. The rub is caused by the faster thermal response and cooldown of the stator combined with a slower cooling of the rotor. A compressor with shrouded stators and abradable seals can absorb a rub on the order of 10 - 15 mils without damage to the seal labyrinths. If the rub takes place 360°

instead of locally, the operating clearance at full speed is increased after the first decel transient and the performance is degraded accordingly. However, the lowest radial clearance at full speed is always achieved with a configuration capable of rubbing away abradable material and "running in".

The combination of designing a controlled thermal lag in the stator, while simultaneously heating and cooling the rotor structure, provides a powerful measure of clearance control. Reduction in operating radial clearances results in higher overall performance and lower operating cost, due to increased stall margin and less energy required to drive the compressor.

Figure 4 illustrates a modern fighter engine compressor with clearance control features such as abradable coatings at the blade tips, "run-in" honeycomb stator shrouds, an internally vented and cooled rotor and thermally isolated rotor tip seals and stator vane assemblies.

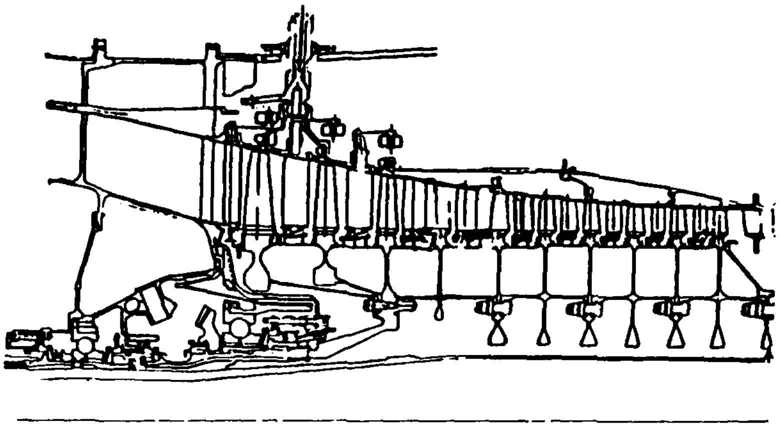

FIGURE 4
FIGHTER ENGINE COMPRESSOR CROSS-SECTION

Figure 5 illustrates the compressor variable stator mechanical drive system used to schedule the stator vanes. This mechanism is coupled to the engine control system for position scheduling.

The vanes, usually scheduled by the control as a function of corrected compressor speed, transition from a closed down position at idle to a more open position at high speed. The large number of parts shown in Figure 5 requires a tight-fit design, often using precision type tolerances to minimize the hysteresis lag, or "lost motion", in either direction during accel and decel throttle movements. The variable vane stages are linked together and "proportionalized" by the mechanical drive mechanism, which provides the required vane travel and rate of change as a function of corrected rotor speed. Minimizing the hysteresis in variable geometry mechanisms and linkages, as well as in the control system and its sensors, becomes increasingly important when the "lost motion" and thermal effects combine during transients to affect stall margin.

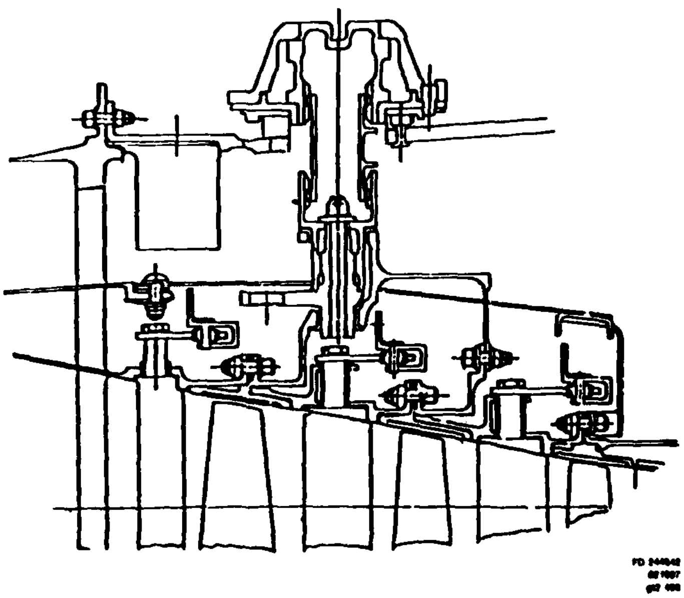

FIGURE 5
COMPRESSOR VARIABLE STATORS

THE "BODIE" TRANSIENT

During high power operation the core engine parts absorb and store considerable heat energy from the gas stream. During deceleration, that stored energy is released back into the flowpath. This heat transfer to the flowpath gas stream affects the compression system performance characteristics by reducing the gas density, especially in the rear stages, where the heat transfer rate is greatest. Reducing the gas density in the rear stages increases the local axial velocity to blade speed ratio (Cx/U), which decreases their pressure rise capability and loads up the mid to forward compressor stages closer to their stall limits. This "thermal heating/mismatch" reduces the overall stall pressure ratio limit of the compressor with a resulting loss in compressor stability margin.

A "Bodie" (power lever movement MIL-IDLE-MIL) shown in Figure 6 is among the most severe system transients, combining both mechanical and thermal effects. The sudden decel from MIL to IDLE throttle setting quickly followed by a burst back to MIL power results in:

a. the casing and rotor turbomachinery essentially staying in the hot condition during the entire transient since there is little time for them to cool,

b. the hot blades, vanes, disk rims and stator parts transferring heat back to the gas path in the compressor *rear stages*, reducing their relative corrected speeds, causing aerodynamic mismatch and backpressure of the middle compressor stages, resulting in both lowering the stall line and raising the operating line,

c. the stall line being further lowered, unless biased, as a result of the variable stators tracking off schedule during the decel by lagging in the open or high speed position,

and

d. an overall reduction in transient stall margin due to the combination of a lower stall line with a higher operating line.

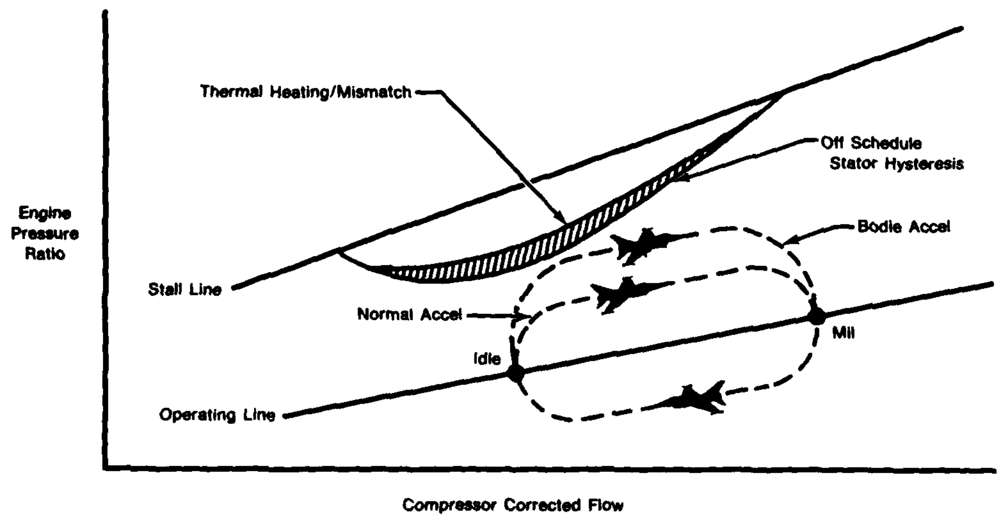

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FIGURE 6
THE "BODIE" TRANSIENT-ADVERSE STACKING OF
MECHANICAL AND THERMAL EFFECTS

A design bias in the control system can be incorporated that places the stators in the closed rather than open position during the Bodie accel, to offset the hysteresis effect of the variable geometry system. This corrective bias eliminates the major reduction in the stall line pressure ratio (shaded area in Figure 6) due to operating with the stators off schedule. Without this offsetting benefit, a serious reduction in the accel speed and transient thrust response would be required to prevent stall for a given aerodynamic configuration.

The Bodie transient illustrates the importance in the design process in understanding adverse mechanical and thermal effects to provide sufficient part speed operability margin.

DESIGNING FOR STALL TOLERANCE

The design tradeoffs involved in selecting the compressor configuration must also account for the effects of compressor stall. Stalls can occur during component or control system malfunctions or during an adverse stackup of engine-aircraft operating conditions beyond the operability margin provided. As a result, the compression system must be designed to accommodate stalls without a significant loss in blade fatigue life or other events requiring maintenance action.

Figure 7 shows the generalized relationship of the vibratory stress during stall and rotor aspect ratio. For low aspect ratio, large chord blading, the maximum single amplitude stress during stall approximates the endurance limit or a stress ratio (σ / σ_e) of unity for either titanium or superalloy materials.

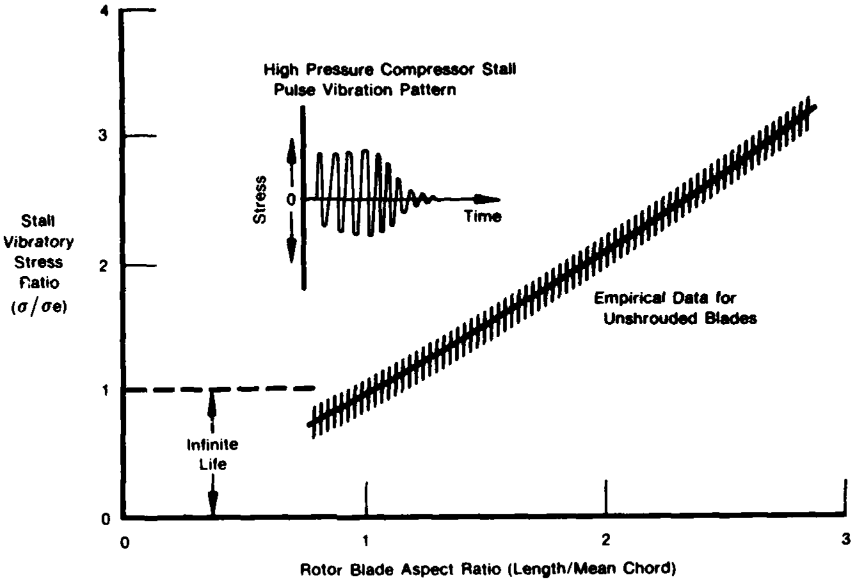

FIGURE 7
GENERALIZED RELATIONSHIP OF STALL STRESS TO ROTOR BLADE ASPECT RATIO

The vibratory stress resulting from compressor stall is significantly higher for high aspect ratio blades. For blading aspect ratios above 2.5, the vibratory stress can exceed the 10^7 endurance limit (σ_e) by 3:1 for a high pressure ratio compressor. At this stress level, the stall pulses that can be absorbed without failure are significantly reduced. For low aspect ratio blading, the magnitude of the stall stress is reduced significantly, providing high tolerance to stall without inducing fatigue cracks. Figure 8 illustrates the advantage of using nickel superalloy blading over titanium when the stall stress exceeds the endurance limit.

When the stall stresses are in the high amplitude fatigue region and also at temperatures above 800°F, nickel superalloy blading offers improved durability over current titanium alloys.

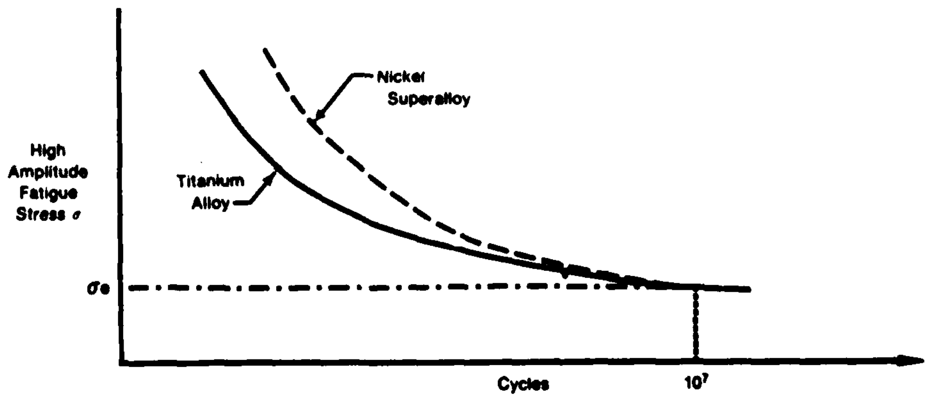

FIGURE 8
HIGH AMPLITUDE FATIGUE STRESS VS CYCLES FOR TITANIUM AND SUPERALLOY BLADING

Empirical data rather than analytical modeling are used to predict the required rotor blade aspect ratio to adequately absorb stall stresses. Experience has shown that when the airfoil root section uncorrected gas bending stress is below 35,000 psi, the stall stresses will be acceptable for current materials. The uncorrected gas bending stress is created by gas pressure bending the airfoil as a beam with no correction for the centrifugal restoring moment.

Blade aspect ratio and material selection are significant factors in providing the capability to absorb stall stresses resulting from transient operation.

Low aspect ratio blades also have the advantage of having significantly higher margin in avoiding self-excited vibration or flutter. Flutter vibration is caused by wide variations in airfoil incidence angles aggravated by engine inlet distortion due to aircraft maneuvers or high Mach number and low compressor corrected rotor speed operation. The increased torsional and flexural strength of the low aspect ratio airfoils accommodate a wide range of airfoil incidence angles without encountering classical beam flutter.

If a stall occurs, both the air volume downstream of the compressor and the engine control system can play an important role in providing the system dynamics leading to a self-recovering characteristic for the compression system. Recent developments have been successful in understanding compression system design parameters that influence transient performance in the stalled condition. These findings will contribute to providing future engine systems that are free of non-recoverable stalls.

THE COMPRESSION SYSTEM STABILITY AUDIT

A convenient representation of adverse factors affecting the net remaining compressor stall margin is shown in Figure 9 as a function of pressure ratio and airflow.

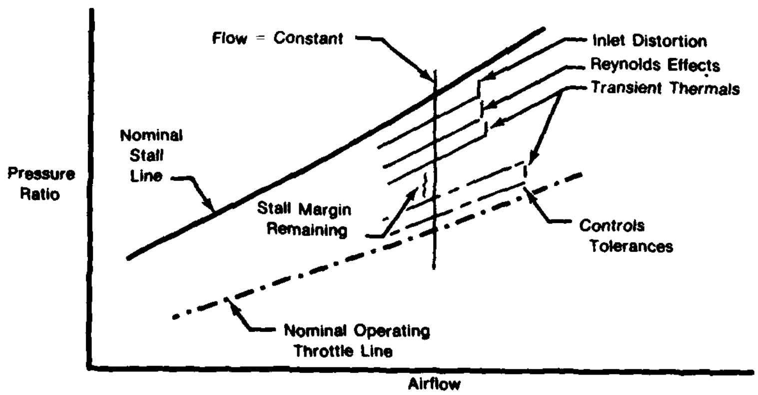

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FIGURE 9
COMPRESSOR STABILITY AUDIT

Adding the stall margin degradation due to transient thermals to the effects of inlet distortion, Reynolds effects and hardware tolerances illustrates the severe requirements placed on the compression system design. The stability audit is used in the design process to identify those areas in the fighter aircraft's operational envelope which require control scheduling or matching changes in order for the engine to remain stall free.

ENGINE STARTING

A modern fighter engine must be able to start reliably, consistently and without a complicated cockpit procedure which hampers the pilot's effectiveness. Ground starts, especially in an alert or "scramble" mode must be quick and dependable. Airstarts, at any time, must be reliable and rapid for safety and survivability. The following sections discuss some of the major thermal effects that the engine designer must consider in meeting system operability requirements.

THE "BOWED" ROTOR

Occasionally, fighter engines can experience a unique problem during ground starting resulting from a "bowed" rotor A "bowed" rotor occurs when an engine undergoes a shutdown and there is either insufficient ram effect to cause windmilling or lockup is caused by temporarily tight seals. The heat within the engine causes a chimney effect where the hot air rises and stratifies in the top section of the rotor spool. The top half of the stationary rotor beam undergoes a larger axial thermal expansion than the lower half since the metal is at a higher temperature. This induces a curve or bow in the rotor. As time passes, the magnitude of the bow reaches a peak and the continued engine cool down results in thermal equilibrium, with the rotor resuming its original shape.

Although the bowed shape of the rotor is very slight, the unbalance during a subsequent start can cause significant whirl vibration while passing through a critical speed range. This usually results in significant blade and seal rubs opening the clearances and deteriorating performance. Alternatives in avoiding a bowed rotor start include slowly motoring the rotor for a while after shutdown to prevent thermal distortion and/or incorporating an oil damped bearing. Figure 10 shows the bowed rotor vibratory deflection characteristic relative to the stator as a function of rotor speed for both solid mounted and oil damped bearing configurations.

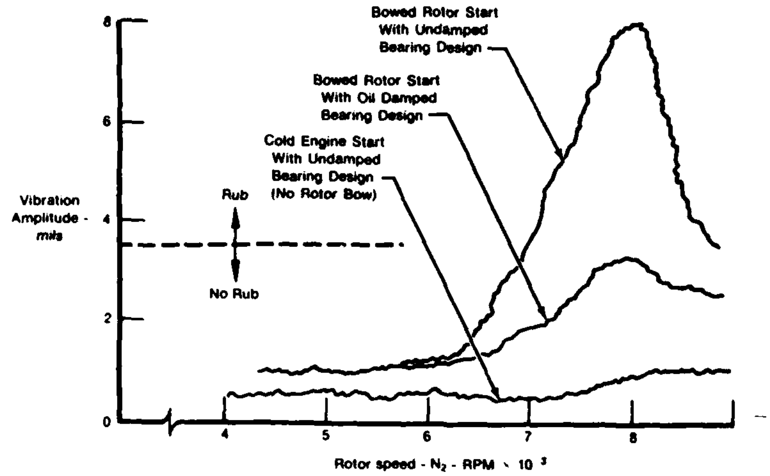

FIGURE 10
REDUCING BOWED ROTOR WHIRL VIBRATION WITH DAMPED BEARING

During a start, the oil damped bearing reduces the rotor whirl deflection passing through the critical speed range by approximately 50 percent. The reduced deflection is within the available rotor-stator clearance, avoiding a clearance rub-out and associated loss in performance.

AIRSTARTING TRANSIENTS

All engines must be capable of an emergency restart following an in-flight shutdown. As a result, airstarting is a critically important procedure and thermal transients during this event can impact both operability and aircraft survivability.

Windmill airstarting is a procedure, following an engine shutdown, in which the aircraft glides at an airspeed sufficient to sustain restarting rotor speed from the ram effect. In the past, windmill starting has been a popular method of airstarting for multi-engine applications and where there is adequate time to cool the engine to minimize thermal effects. For single engine applications, diving and associated altitude loss is required to achieve the correct rotor speed.

The high accessory loads of modern fighters, being driven from the smaller high pressure spool, has resulted in less surplus engine energy and an undesirable windmilling airstart envelope. Spooldown airstarting, however, is a procedure that offers a practical and satisfactory alternative to windmill airstarting. After the engine is shut off, this procedure involves initiating the airstart during the spooldown period and, as shown in Figure 11, reduces the need for high airspeeds to maintain sufficient rotor speed and significantly improves the airstarting envelope and restart time.

The transient characteristics of a spooldown airstart are shown in Figure 12.

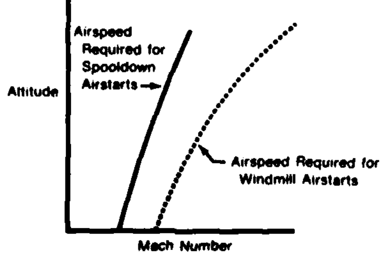

FIGURE 11
COMPARISON OF AIRSPEEDS REQUIRED
FOR WINDMILL AND SPOOLDOWN AIRSTARTING

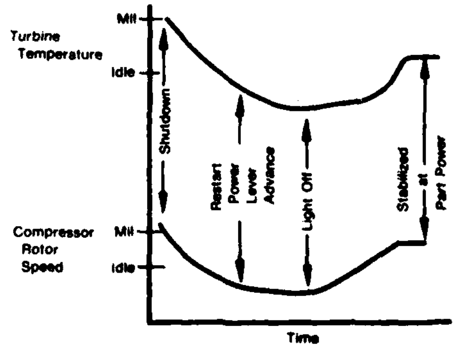

FIGURE 12
SPOOLDOWN AIRSTART CHARACTERISTIC

The benefits of spooldown airstarting relative to windmill airstarting are; (1) longer glide time, since spooldown starts can be made at lower airspeeds and (2) shorter total engine recovery time, since the start is initiated shortly after shutdown. However, this procedure also results in having to account for larger transient thermals during spooldown. After engine shutdown from high power, the compressor metal parts are still hot. During spooldown, heat is transferred to the air as it passes through the compressor and, as discussed earlier, an aerodynamic mismatch occurs between the front and rear stages. When the start is initiated, the rear stages are hot, transferring heat to the gas path, and causing a decrease in stall margin. Additional reduction in starting stall margin is caused by a further increase in operating line due to improved combustion efficiency with a hot engine. This results in having to provide a lower fuel schedule to avoid a stall.

The spooldown stall limit is transient in nature; that is, it deteriorates and then increases to the more stabilized windmill level as a function of time. This characteristic is similar to the Bodie stall line except for being in the start region below idle. Several variables such as altitude, Mach number, and engine conditions before shutdown can influence the initial stall line level as well as the rate at which the available stall margin changes with time. The required fuel scheduling is generally defined through engine testing which evaluates these phenomena on an operational basis.

Advanced control systems allow flexibility to provide consistent and reliable starts, thus minimizing potential stall or hung starts. Fuel flow is scheduled by a closed loop system allowing a completely automatic start with no pilot action required other than power lever movement.

HOT SECTION THERMAL TRANSIENTS

The hot section (combustor and turbine) of a fighter engine undergoes rapid and frequent thermal transients due to power lever cycling. Studies have shown that fighters with high thrust-to-weight ratios have more full throttle cycles per flight hour than fighters with low thrust-to-weight. Since modern fighters have high thrust-to-weight ratios for acceleration and maneuverability, the engine must also be designed with high thrust-to-weight and rapid response. The high hot section gas temperatures and frequent cold to hot cyclic operation presents a tough design challenge.

During the decel transient a minimum effective fuel-air ratio is required to assure continued combustion. Flame extinction or lean blow out occurs when the fuel-air ratio is reduced below a value that results in the heat transfer rate out of the flame zone being greater than the heat release rate. Figure 13 illustrates the transient operation and relationship between fuel-air ratio and throttle setting.

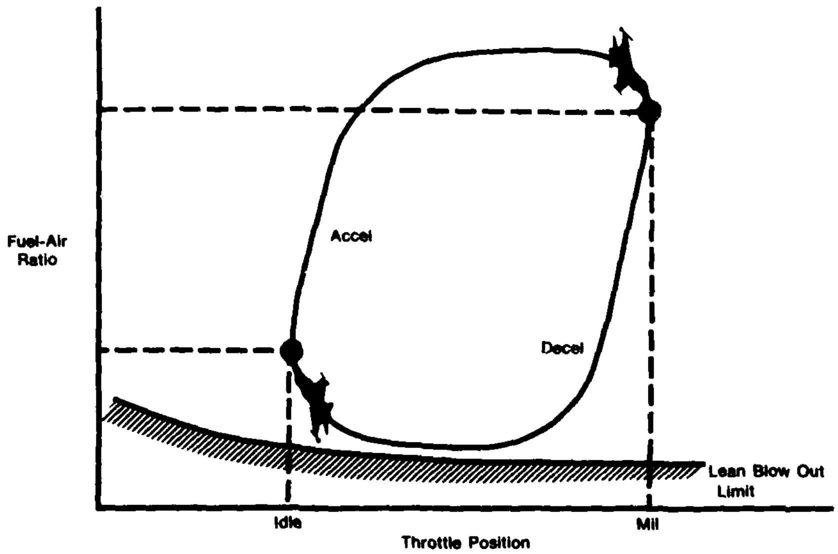

FIGURE 13
TRANSIENT RELATIONSHIP BETWEEN FUEL-AIR RATIO AND THROTTLE POSITION

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Additionally, the combustor's operating characteristics must encompass stable combustion from ignition at low pressures of one-third of an atmosphere to high ram conditions with pressures of 40 atmospheres. Also, gas temperatures increase from low temperatures up to 2800°F from cold starts to maximum power. The combustor must be designed to accommodate this wide operational range while still providing high reliability and durability, low pattern factor and no visible exhaust smoke. The turbine must be capable of operating in this hostile environment for long periods of time before requiring inspection. These factors make the fighter engine's hot section the most demanding engineering challenge faced by industry.

COMBUSTOR TRANSIENT DESIGN

A primary operational transient requirement for the combustor is to have adequate lean blow out margin. When the engine throttle is chopped, the fuel flow to the combustor is dramatically reduced. This significantly lowers the fuel-air ratio and causes the engine to decelerate.

Accelerating the engine from idle requires a significant increase in energy and is achieved by increasing the fuel-air ratio. Combustors do not normally limit acceleration performance because they generally have considerably more heat release capability than the engine needs. Deceleration, however, results in fuel-air ratios approaching the lean blowout limit of the combustor.

To prevent lean blowout, the minimum fuel flow must be set above the stability limit during a decel. However, while the minimum fuel flow must be high enough to prevent lean blowout, it must also be a low enough value to permit rapid deceleration and reduction of engine thrust. The combustor and engine system must be designed for compatibility with fighter engine deceleration time requirements.

Combustor liners are primarily life-limited by low cycle fatigue (LCF) failures. The LCF results from strains induced by temperature differences within the structure. Figure 14 shows the effect liner geometry can have on the peak temperature differences in the structure.

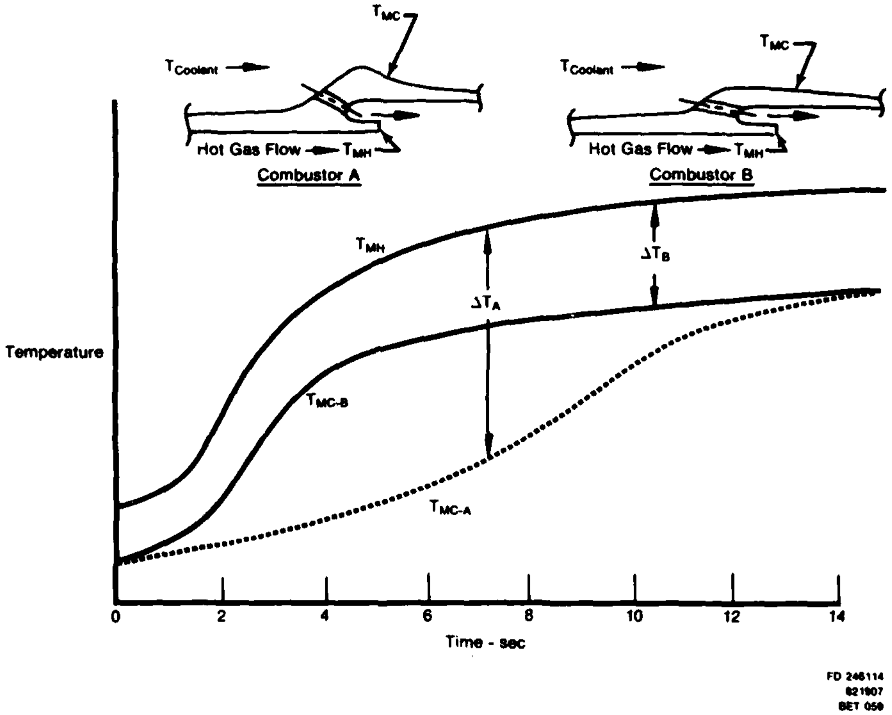

FIGURE 14
COMBUSTOR LINER TEMPERATURE GRADIENTS DURING RAPID ACCEL TRANSIENT

Design A is a configuration where the maximum temperature difference (ΔTA) between the hot and cooler structural locations occurs during the transient from idle to MIL power. This is because the mass of the "nugget" forming the structural shell of the liner takes substantially longer to heat than does the hot louver lip exposed to the combustion process. Design B has less mass in the cooler region of the "nugget" and results in a lower temperature gradient during the transient and with the peak temperature difference (ΔTB) occurring at MIL power. As a result, design configuration B is less sensitive to thermal transients and, provided the liner has adequate buckling margin to absorb the negative pressure loading, will exhibit a higher cyclic life.

Determination of temperature characteristics requires good transient and heat transfer models of the engine and of the combustor structure. Structural and cooling design improvements will continue to play an increasingly important role in offsetting the effects of transient operation on combustor durability. Combustor designs with the liner shell structure shielded from the hot gas path with thermally free panels will be used in future higher temperature engines. These designs will also require less liner cooling air so that more air is available for both dilution and the combustion process. At combustor temperatures approaching 3000°F and above, these low liner cooling configurations also favor the turbine since the pitch line temperature can be lower for the same average gas temperature.

TURBINE TEMPERATURE TRANSIENTS

The high temperature turbine transient environment is the most severe that occurs in all of the fighter engine components. A snap accel from IDLE to MIL power requires a low rotor polar moment of inertia, the capability to absorb a peak temperature overshoot, and a design configuration that resists low cycle fatigue (LCF). Figure 15 illustrates variations in thrust and combustor exit temperature (CET) for moderate and fast accels from IDLE to MIL power.

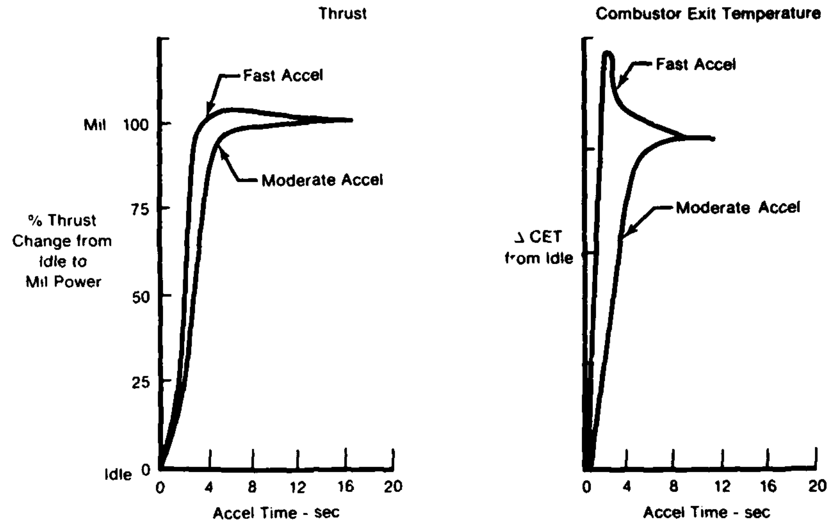

FIGURE 15
THRUST AND COMBUSTOR EXIT TEMPERATURE FOR MODERATE AND FAST ACCELS

The fast accel results in a significant temperature overshoot required to produce a desired increase in thrust within a given time requirement. The overshoot in temperature is the result of scheduling the acceleration fuel flow at a high level while core engine rotor speed is below its steady-state, military power level. The additional energy in the hot gas stream expanding through the turbine provides the excess turbine power required to give a faster acceleration. The thermal overshoot during a snap accel increases the temperature differential and total strain range absorbed by the vanes, platforms, shrouds and blades.

Figure 16 shows the variation in turbine airfoil leading edge total strain range versus metal temperature for fast, moderate and slow accels.

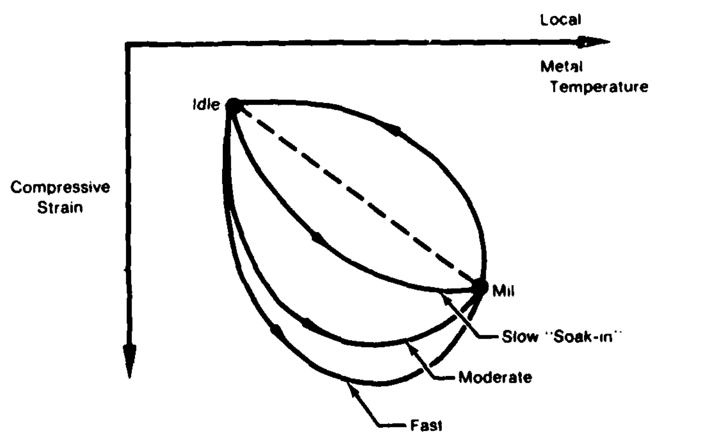

FIGURE 16
RELATIONSHIP OF TURBINE AIRFOIL LEADING EDGE STRAIN vs METAL TEMPERATURE FOR FAST, MODERATE AND SLOW ACCELS

The slow "soak-in" accel, shown in Figure 16, represents the lowest strain range that can occur in accelerating from IDLE to MIL power. Note the significant increase in strain for the moderate accel and even larger increase for the fast accel relative to the slow "soak-in" path. The total strain range and rate of engine cyclic usage determines the low cycle fatigue life and the eventual wear out mode for the part.

As a result of this experience, research and development was conducted to develop means of controlling both the rate and path of the transient. Figure 17 shows how control of the combustor exit gas temperature during an accel can be improved by using a digital electronic engine control (DEEC) instead of the standard hydromechanical unit.

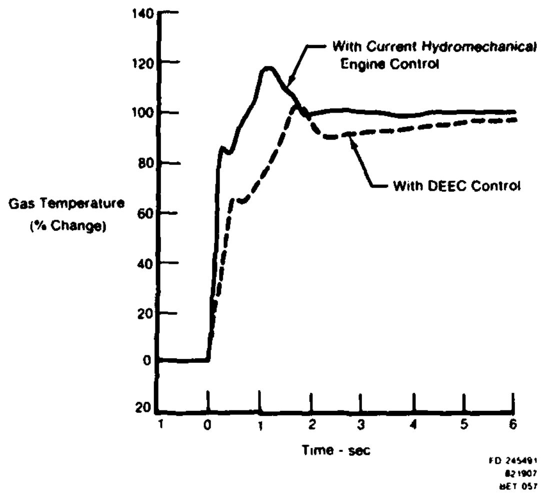

FIGURE 17
CONTROLLING THE COMBUSTOR EXIT TEMPERATURE TRANSIENT TO IMPROVE TURBINE LIFE

The DEEC allows precise management of the fuel/air combustion mixture during the accel. The control of the accel path, by using scheduled downtrim as the gas temperature approaches the steady state value, is the most effective way to minimize the adverse effect of the accel transient on the airfoil strain range. Both the reduction in peak gas temperature during the transient and lower temperature/time slope within the path significantly improves turbine durability and life.

TURBINE ROTOR TRANSIENT RESPONSE

Turbine rotors are subjected to a wide range of temperature variations during decel and accel transients in addition to mechanical rotational stresses. Disk rim and bore cooling are required to both minimize weight and prevent material overtemperature and potential creep rupture failure. However, the cooling essential to steady state operation is often detrimental during transients where large strains are induced with unfavorable impact on disk cyclic life. Figure 18 shows an air cooled turbine disk subjected to decel and accel transients as a function of throttle position between MIL (full speed) and IDLE.

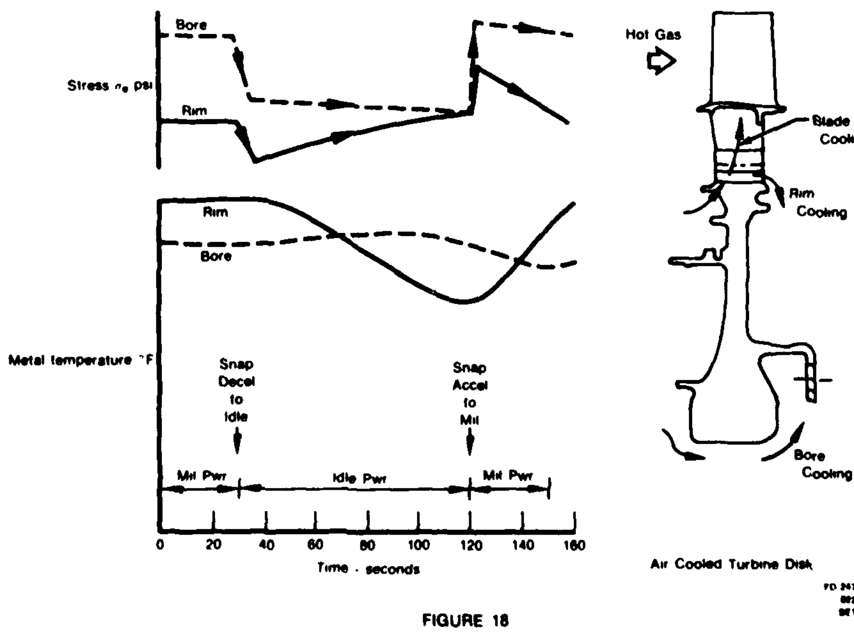

FIGURE 18
TURBINE DISK TEMPERATURES AND STRESSES DURING DECEL AND ACCEL TRANSIENTS

The operating transient shown describes a rapid decel from military power (MIL) to IDLE, a ninety second dwell and a snap accel back to MIL power. The disk rim and bore metal temperatures and stresses are plotted as a function of time. During the transient, the bore metal temperature undergoes a relatively small change due to its larger mass and remote location from the flowpath. The disk rim however, is subjected to a large swing in temperature due to its low thermal mass, proximity to the flowpath and efficient cooling system. The resulting stresses, illustrated, show that the rim transient stresses are significantly higher than at the steady state conditions. During the transient, the bore stress change is primarily due to the rotational centrifugal forces and it is only slightly higher than at steady state conditions. However, after the dwell at idle and shortly before the accel, the rim reaches essentially the maximum steady state stress level. This is because after the dwell at idle, the rim is cooler than the bore and undergoes a forced expansion which induces a high tensile stress although the rotor is at low speed. During the accel, the centrifugal stresses become additive and this results in a substantial increase in rim stress over the steady state value.

Because of the high level of turbine rim stress that must be accommodated during transient operation, it is desirable to minimize stress concentrations in these high stress locations. This can be achieved with shaped dovetail slot bottoms and by using a disk configuration as shown in Figure 18 without cooling or bolt holes through the disk web.

Traditionally, the turbine has been the most life limiting and highest cost component in the engine. As a result, considerable research and development has been directed towards improving both the mechanical and thermodynamic design configuration as well as the materials and manufacturing processes. Figure 19 shows a modern turbine designed to meet mechanical and thermal transient requirements that have been identified by the lessons learned during the past 25 years.

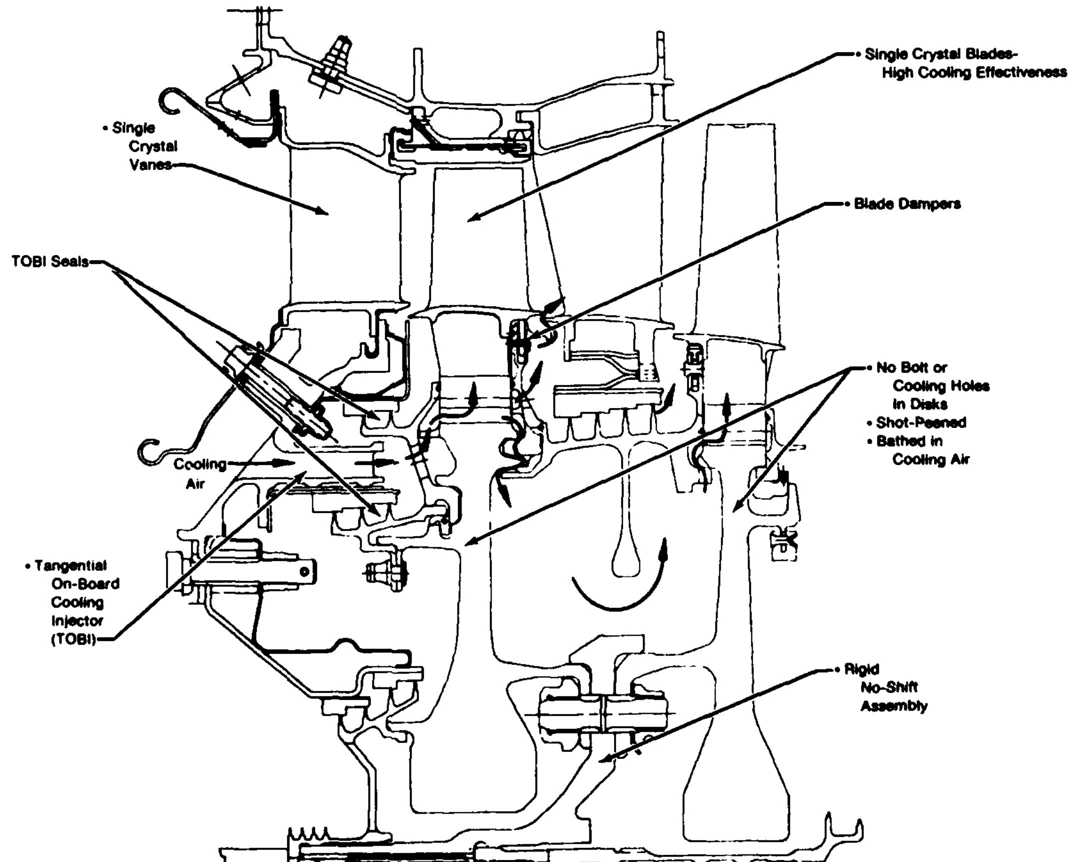

FIGURE 19
MODERN FIGHTER ENGINE TURBINE
APPLYING 25 YEARS OF "LESSONS LEARNED"

Design features to accommodate thermal and mechanical transients include:

a. no bolt or cooling holes in the disk webs to eliminate life limiting critical stress concentrations,

b. shot-peening of high stress critical rotating parts to avoid crack initiation from material or manufacturing surface flaws,

c. bathing the disk rims in compressor cooling air to reduce metal temperature and thermal stresses,

d. a tangential on-board cooling injector (TOBI) system to reduce the relative rotor cooling air temperature,

e. a blade-disk cooling system that is not sensitive to operational changes in seal clearance and is therefore resistant to deterioration,

f. two stages for high efficiency with low polar moment of inertia for fast throttle response,

g. high strength crack-growth resistant materials for disks,

h. single crystal blades with high cooling effectiveness and dampers to meet life requirements,

and

i. a rigid no-shift assembly for balance retention.

More than any other component, progress in turbine design has been paced by materials and processes. The use of these advanced materials and processes has resulted in higher temperature operation with less cooling air and at higher speeds. Continued efforts to economically process these blade, vane, disk and shroud materials are essential to minimize and control product cost.

CONTROLS MANAGE ENGINE TRANSIENTS

The engine control manages how fast the engine responds to throttle input. The capability of the control system including the computational device, sensors, actuators and fuel management is important to accommodate thermal transients.

The time proven hydromechanical control system has an outstanding record of achievement in the control of gas turbine variable vane geometry, exhaust nozzle area, bleed schedules, fuel flow and selection of augmentation delivery zones. However, during the past few years, a shift away from the traditional hydromechanical control toward the digital electronic control has begun. This is primarily because the use of electronics offers simplicity and lower cost, while providing additional control perfection, precision and capability.

Removal of the computational hydromechanical cams and linkages and replacement by an electronic digital computer program (software) reduces hysteresis while providing additional functions. Elimination of mechanical stackup and wearout tolerances allows the electronic control to precisiely operate the engine both closer to the component and system limits and with greater operational control margin.

The electronic digital computer also has an overwhelming logic and computational advantage over its hydromechanical counterpart, yielding additional capacity to perform other control functions. Electronic sensors with a higher degree of precision provide data that allows the electronic digital control to more accurately compute air flow in the augmentor core and bypass duct air streams. This results in a more precise fuel delivery, which is required to maintain a desired fuel-air ratio, while minimizing ignition pulses and eliminating the undesireable problems of augmentor combustion blow-out and instabilities.

With the additional computational power, the electronic control can accommodate various faults within the control system by first identifying the fault point and then modifying the control mode as required, allowing continued operation without exceeding component or system limits.

The ability of the computer program to identify a control system fault point allows it to communicate the fault point to an external fault reporting device. This can provide maintenance personnel with a quick and simple analysis of problems within the control and engine system.

Advanced fighter engine controls must perform all the supervisory and monitoring functions necessary to manage engine systems in the fighter aircraft combat environment. The control system must schedule fuel flow precisely from starting through maximum low altitude ram conditions. It must schedule the engine's variable compressor geometry with a precision that provides maximum performance with adequate stall margin. It must meter augmentor fuel flow and simultaneously schedule nozzle area to maintain adequate fan stall margin and augmentor blowout margin. Also, it must protect the engine from combustor blowout and compressor stall during accelerations and decelerations. The control system is the component that ultimately provides the fighter engine with the inputs and restrictions that allows the fighter aircraft to operate at the maximum extremes of the flight envelope.

"MAKING IT HAPPEN"

Understanding the operational and life limiting characteristics of the core components is essential for successful integration into the total engine system. Figure 20 shows a fully instrumented gas generator on test to evaluate the operability and potential life limiting characteristics for a new hot section.

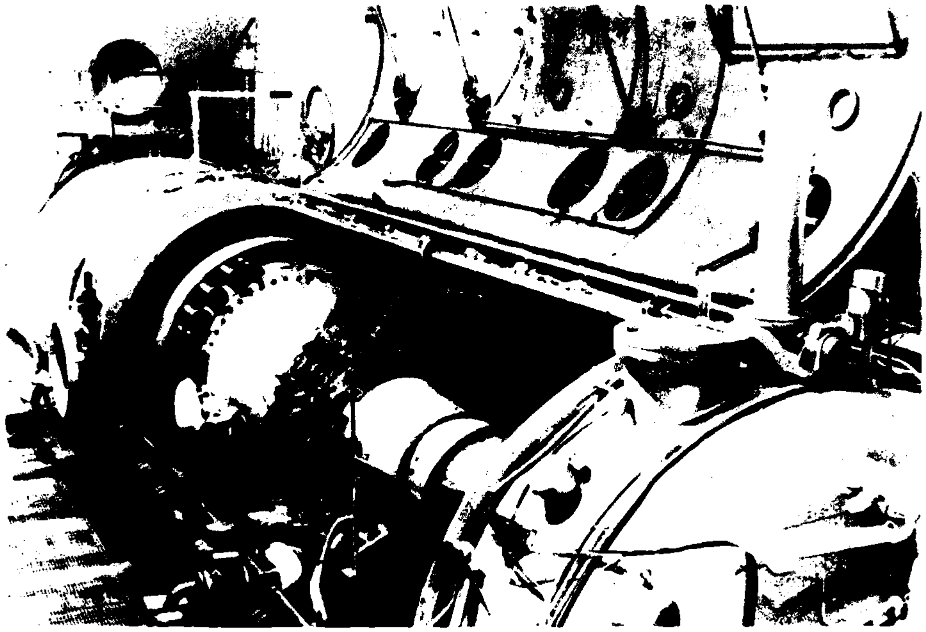

FIGURE 20
RAM TESTING THE INSTRUMENTED CORE ENGINE

FD 244540
822307
GT2879

These instrumented tests measure internal pressures, temperatures and vibratory stress characteristics for both steady state and transient operation. The inlet to the core engine is both pressurized and heated to simulate the fan supercharging effect. "Getting the facts" is an integral part of both defining and understanding potential problems in order to "make it happen".

SUMMARY

This paper has discussed some of the major considerations that must be addressed in designing fo fighter engine transients with particular emphasis on the gas generator. Creative application of a wide range of engineering disciplines can be used to overcome problems identified during the past 25 years and forge ahead. The many lessons learned in the design and development of fighter aircraft engines have also identified the technical barriers that will be tackled and breached in the future. The rate of this progress will be determined by the effort expended. A continuing research and development activity has the potential of providing further significant improvements in fighter engine performance, durability, weight and life cycle cost.

DISCUSSION

Ph. Ramette, Fr.

(1) In the curve which gives stall vibratory stress ratio versus rotor blade aspect ratio is it maximum stress or mean stress (like RMS) which is plotted?

(2) For turbine blades, are you working on cyclic stresses with ceramic blades?

Author's Reply

(1) The life limiting stress shown is the maximum value on the airfoil.

(2) Industry is considering ceramic turbine blades and other applications for this material. Cyclic transients induce stresses that limit the use of current generation of ceramics in high stress components subject to thermal shock.

THE CRITERIA USED FOR ASSESSING ACCEPTABLE ENGINE HANDLING ON UK MILITARY AIRCRAFT

by

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SUMMARY

This paper reviews the criteria used by A&AEE to assess the acceptability of engine handling qualities and operating procedures for Service use.

Criteria discussed include those used to assess engine behaviour and response during starting, relighting, throttle movement (in both dry power and reheat range), changes in flight condition, and changes in intake conditions caused by gunfire and weapon delivery. The factors to be considered in establishing these criteria are multifarious and it is not possible to generalise because of the differences posed by different engine installations and operational roles. The paper therefore, considers the qualities that were found necessary during recent trials to meet the required levels of safety and operational effectiveness.

Finally, the paper discusses the application of these criteria in the definition of engine operating limits and procedures for Service use.

1 INTRODUCTION

The procurement of military aircraft for the UK armed forces is the responsibility of the Procurement Executive of the Ministry of Defence (MOD(PE)). In this context the procurement process encompasses, research, development and production. From the point of view of the UK operator, the culmination of this process is the issue of the CA Release for the aircraft which permits the aircraft to be used by the operator.

This is defined in Reference 1, Chapter 308 as:

> 'Notification by the Controller of Aircraft (CA) to the Service Department that a new type or mark of aircraft or aircraft weapon system has been developed to the stage where it is suitable for use by Service aircrew'.

It is the CA Release which formally states all the limitations to be observed by the aircrew in operating the aircraft. The CA Release therefore contains all the engine operating limitations pertinent to the operation of the aircraft.

The Aeroplane and Armament Experimental Establishment (A&AEE) at Boscombe Down is the MOD(PE) organisation responsible for the assessment of military aircraft and for making CA Release recommendations. Amongst these responsibilities, it is responsible for conducting engine handling assessments and in making recommendations for engine operating limits and procedures.

In discharging this responsibility, A&AEE must decide the method and scope of the required assessment, and the manner in which the results are to be incorporated in defining the recommendations for the CA Release. In making these decisions, A&AEE must take many considerations into account so that engines complying with the criteria used possess the standard of handling qualities and performance characteristics necessary to meet the required level of safety and operational effectiveness.

This paper is concerned with the criteria used by A&AEE in reaching these objectives. Obviously the handling qualities and performance characteristics exhibited by an engine are dependent on its own design features, the design of the engine installation, and on such factors as the prevailing flight condition, and piloting technique used, etc. In assessing engine behaviour there are many other factors (eg the operational role of the aircraft and the consequences of malfunction) that may have a significant influence on the criteria used. Consequently it is not possible to establish sets of criteria, which when universally applied to all engine installations ensure that the aircraft possesses the required level of safety and operational effectiveness. Hence, while the criteria of acceptability commonly applied are discussed in this paper, alternative criteria may be applied in particular instances to enable each installation to be assessed on its own merit.

This paper is limited to those topics relevant to the clearance of turbojet engines fitted to fixed wing aircraft as it would be impracticable to deal with all types of engine installations in the one paper. Many topics which are specific to particular aircraft (eg vectored thrust on the Harrier) are omitted as this paper is intended to give an overall feel for the criteria used by A&AEE in assessing engine handling.

2 GENERAL CONSIDERATIONS ON THE CA RELEASE TASK

The task facing A&AEE is to assess the suitability for Service use of the handling qualities and performance characteristics exhibited by the engine. A&AEE must also verify that the observed characteristics satisfy the general requirements for UK military aircraft expressed in D Eng RD 2300 (Reference 2) and in AvP 970 Volume 1, MEMO 51A (Reference 3) and the specific requirements for the particular engine installation expressed in the engine or airframe specification. It is contended that the primary task is to assess the observed characteristics qualitatively against the intended Service role and to form an independent judgement on their acceptability (whether or not they satisfy the specified requirements).

From the results obtained from the assessment, A&AEE must determine the engine operating limits, the piloting technique suitable for Service use, and the need for any prior modification which must be embodied before the aircraft or engine is used in any particular role.

The task is not one of engine development as the assessment is made in the light of the development work already carried out by the Contractor. Thus due account is taken of all evidence submitted by the Contractor, including his recommendations in respect of Service operating limits and techniques.

The evidence available from the Contractor's flight trials is of particular value if the test method is independent of piloting technique (eg slam acceleration tests for surge). For this reason, the collection of such data is often delegated to the Contractor or a joint trial is undertaken with the Contractor. While such data may not be subject to A&AEE 'spot checks', A&AEE always makes an independent assessment of those aspects involving piloting technique, or a subjective judgement, to ensure that the circumstances of Service usage are properly taken into account. Where the evidence from A&AEE CA Release trials does not corroborate the Contractor's evidence, the A&AEE data is taken as overriding.

3 CA RELEASE FLIGHT TRIALS

The scope of the engine trials required by A&AEE before making CA Release recommendations is decided by the Establishment on the basis of the use that may be made of the Contractor's data, the need for confirmatory tests, and the number of tests which because a subjective judgement is required must be carried out by an A&AEE pilot. The opportunity for conducting these trials is often constrained by programme dictates including airframe and engine availability. Also it may be necessary to phase the clearance programme for the engine to match the clearance of the airframe and its role equipment (eg guns, flight refuelling probes, etc). Nevertheless in agreeing to a flight trials programme, A&AEE endeavours to ensure that sufficient data has been obtained at the completion of each stage of the engine trial for the CA Release recommendations to be made with confidence.

In selecting the trials airframe, a 'sampling' approach is used so that such aspects as the variation between individual aircraft in the fleet due to tolerance in manufacture and maintenance, and the engine deterioration in Service down to the minimum acceptable standard prior to overhaul, are taken into account. For this reason, A&AEE will usually insist that the CA Release trials are conducted on a different airframe with different engines than that used by the Contractor during development trials. Deterioration is less easy to allow for (see Para 6) and it is usually only possible to test what is considered to be a 'representative' engine on the basis of standard air test results. When the tests are to be conducted on a single-engined aircraft, or a mult-engined aircraft with known differences between particular installations, A&AEE may also insist that the CA Release trials include confirmatory tests on a second airframe.

The trials airframe is assessed to be fully representative of the intended Service standard in the absence of any defined differences (eg the presence or absence of a modification). The same applies to the engine, and in addition the Engine Contractor is required to declare whether any adjustments have been made to the engine control system which might make the engine unrepresentative of the Service standard. Where tolerances on settings are permitted which affect engine handling (eg bleed valves and variable inlet guide vanes), tests are made at the upper and lower limits of these tolerances. The same also applies if the tests involve fuel with a permitted range of characteristics which may affect engine handling.

The trials must result in CA Release recommendations applicable to all reasonably predictable operational circumstances. However as the choice of possible combinations of flight and engine condition is extremely wide, a rationale must be established for choosing the flight conditions. As noted in Para 2, the need is to complement, as well as to corroborate the data obtained by the Contractor without unnecessary duplication of effort. Thus, the test conditions are chosen after considering the conditions likely to be experienced in Service, the data available from the Contractor, and past experience of similar aircraft or trials. As the CA Release trials are necessarily limited, the philosophy adopted is usually to test those conditions which are potentially the most critical consistent with flight safety considerations. It should be noted that in choosing the 'worst likely' conditions, it is necessary to disregard those conditions which are considered to be extremely remote (eg improbable combinations of flight condition, throttle setting and power off-take).

CA Release trials are normally conducted within the range of conditions previously demonstrated in flight by the Contractor. Flight limitations for the trial are therefore promulgated by A&AEE on the basis of the envelope previously explored. An exception to this procedure is made for certain trials (eg climatic, shipborne, etc) when the Contractor does not have previous experience. On these occasions A&AEE will promulgate flight limitations for the trial taking into account past experience to extrapolate the test data. Extra care is also taken to ensure that the tests are made in a cautious and progressive manner.

4 GENERAL OBSERVATIONS ON ACCEPTANCE CRITERIA IN RESPECT OF THE PILOTING TASK

The acceptability of the piloting task in using the engine is largely governed by the ease with which the pilot can follow the specific engine operating instructions. The acceptability of particular handling qualities may in turn be judged against the strength or weakness of the engine controls, instruments and failure warning devices. Both of these aspects concerned with the piloting task are therefore of central importance in the assessment of engine handling qualities.

4.1 ENGINE OPERATING INSTRUCTIONS

All A&AEE assessments of the piloting task are made using as far as possible the precise operating procedures recommended by the Contractor. To this end, all normal operating procedures (eg starting and handling), and where possible emergency drills (which would follow engine malfunction), are assessed by observing to the letter the procedures proposed for the Service pilot in the draft Flight Reference Cards (FRC) prepared by the Contractor

In making the assessment, A&AEE give careful consideration to the operational role of the aircraft, the likely cumulative workload on the pilot caused by interaction with other tasks, and the likely experience of the future Service pilot (eg is the aircraft destined to be flown by a student pilot). It is self-evident that the proposed operating instructions must be unambiguous, logical and easy to follow. Where the procedure is not precise, or a tolerance permitted for a particular operation, A&AEE will assess the most adverse conditions likely to be experienced in Service. The objective is to ensure that within the envelope recommended for Service use, likely variations in the operating procedures that might be introduced by the Service pilot do not adversely affect engine behaviour. If the proposed procedure is considered unsatisfactory, then A&AEE may recommend an alternative procedure. Where the clearance is conditional on a particular procedure being followed, then this will be stated when making the CA Release recommendations.

In assessing emergency drills, A&AEE also consider the time required by the pilot to recognise that a malfunction has occurred, and the consequence of the malfunction if missed. The recognition of the malfunction by either sight, sound, or occasionally even smell, will depend on whether the failure is made known to the pilot as a progressive change away from the normal condition (eg a rise in engine temperature on a temperature gauge) or as a discrete indication that a specified value (eg a maximum temperature) has been exceeded. In the latter case, the pilot is usually warned by a designated indicator or warning device. In general terms, the assessment will consider whether the strength of the warning is sufficient (eg it is likely to be missed by the pilot), bearing in mind the likely state of alertness of the pilot. This will depend on the degree of attention required by the piloting task which, in turn will depend on the flight condition (eg the proximity to the ground) and any distractions due to other tasks (eg updating of navigation systems). The assessment will also consider whether the warning will promote instinctively the appropriate pilot reaction.

Typical pilot reaction times are suggested below (Reference 4).

STATE OF ALERTNESS	RECOGNITION TIME	DECISION TIME (sec)	REACTION TIME (sec)	OVERALL PILOT RESPONSE TIME AFTER RECOGNITION (sec)
Active	Dependent	0 (Instinctive)	½	½ to 1½
	on type of	1 (Considered)	½	
Passive	warning	2	½	2½

Thus it can be seen that even after the recognition of a malfunction the pilot may require some considerable time to take appropriate action. This aspect is dealt with further in Para 6 in discussing the definition of limitations.

4.2 ENGINE CONTROLS AND INSTRUMENTS

The basic layout and form of the engine controls and instruments are normally established early in the development programme of the aircraft and first assessed by A&AEE pilots during the review of the cockpit mock-up and during previews of the initial aircraft. Subsequent development problems may give rise to cockpit changes or result in a different emphasis being placed on a particular control or instrument. Design changes, particularly when production components are embodied may also significantly change the pilot's task. For these reasons, A&AEE insist on assessing the intended Service standard of cockpit before CA Release. The assessment is made with due regard to the types of clothing and equipment that may be worn by the Service aircrew and the likely range of physical dimensions of the aircrew.

The purpose of the assessment is to ensure that all engine controls and instruments satisfy the general requirements for UK military aircraft and are suitable for their intended use. The general requirements are laid down in Chapter 110 (Crew Station - General Requirements) and in Chapter 112 (Pilots' Cockpit - Controls and Instruments) of AvP 970 Volume 1 (Reference 5). Amendments to Chapter 112 which are particular to engine controls and instruments have been promulgated in MEMO 71A (Numbering of Engine Controls and Instruments), MEMO 88A (Engine Controls and Switches) and MEMO 90A (Engine Instruments) of Reference 3. Broadly these requirements are aimed at ensuring that the mode of operation of controls is consistent with established practice and that the displays and instruments conform to a standard layout and presentation. Instruments must be legible unless all visibilities and lighting conditions and free of ambiguities.

Clearly it would be impracticable to attempt to describe in this paper all the qualitative criteria that would be used by the pilot to form an opinion on the acceptability of engine controls and instruments, except to observe that all controls and instruments must aid the piloting task. In particular if displays are to be used in observing the limitations defined in the CA Release the limiting values must be clearly visible. Controls must function correctly under all circumstances and must be easy to operate when carrying out the recommended operating procedures.

Chapter 110 and 112 of Reference 5 also specify the type of signal that shall be used to inform the pilot of engine failure or malfunction. In brief, a red warning indicator, coupled to red flashing attention lights and audio warning shall be used for emergencies which are likely to prove catastrophic unless the pilot takes immediate remedial action. Amber caution lights, with resettable attention light shall be used for emergencies which do not involve the risk of immediate catastrophe and although requiring attention do not require immediate action.

In assessing the acceptability of these emergency signals for CA Release, the criteria used is that the signal must be of the correct type (ie be appropriate for the emergency) and have sufficient strength to alert the pilot bearing in mind the likely pilot reaction time and the consequence if the emergency condition were missed. It is also self-evident, that the pilot must not be falsely informed of emergencies which have not occurred, or be provided with warning or cautionary signals which he is instructed to ignore (ie low oil pressure under negative 'g' conditions).

5 CRITERIA USED TO ASSESS ENGINE HANDLING

The criteria commonly used to assess the acceptability of observed engine handling characteristics are listed below. As noted in the introduction to this paper, the uniqueness of most engine installations and the differences caused by operational roles, makes it impossible to establish a set of criteria which can be universally applied to all installations. Thus, while every effort has been made to make the criteria listed as general as possible, particular engine installations may require alternative criteria.

5.1 ENGINE STARTING

The UK military specification which covers the design and manufacture of gas turbine engines (D Eng RD 2300 Reference 2) lays down in Chapter 8 the general requirements for engine starting which normally must be met. Further requirements are specified in MEMO 51A (Reference 3) and for the particular installation in the engine and airframe specification. A&AEE in addition to ensuring that the observed characteristics satisfy these requirements, assess the acceptability for Service use on the basis of the following criteria:

a The engine shall not exceed any operating limit (or give the pilot reason to believe that the limiting value will be exceeded) during successful starts within the cleared envelope.

b The proposed starting procedure should be simple (ie except for fuel cock control, the operation of only one control should be necessary to start the engine (MEMO 51A also refers)). Simultaneous operation of controls or the precise timing of a control movement to coincide with a particular RPM should not be required.

c The starting procedure should not require adaptation to suit different ambient conditions or specific gravity of the fuel (ie one throttle setting should be sufficient to effect all starts).

d The pilot should be provided with a clear indication of engine light-up (normally not more than 15 s from opening the throttle).

e The acceleration to idle should be smooth and progressive (ie free of stagnation). Automatic devices to control the acceleration should not result in sudden changes in indicated engine parameters.

f The procedure to abort the start should be simple (ie the straight forward movement of a single control).

5.2 ENGINE RELIGHTING IN FLIGHT

All of the criteria discussed above for engine starting also apply to engine relighting in flight, but greater emphasis is placed for obvious reasons on the reliability of the relight and the simplicity of the procedures. Few UK military aircraft, with the exception of the multi-engined long range patrol aircraft (eg Nimrod), are operated to procedures which permit the deliberate shutdown of an engine in flight, except that it is during pilot conversion training and air tests (prohibited on single-engined aircraft) or for emergency reasons. The result is that the majority of service pilots only experience relights on the rare occasions (about 1 per 3000 flying hours) when it is necessary to relight an engine following flame-out, surge etc. Some unfamiliarity with the relight procedure must therefore be expected, as must the additional work load on the pilot caused by the emergency and the undoubted 'pressure' to achieve a relight. It should also be recognised that unlike ground starts, when a failure to start is likely to be no more than an operational nuisance, a failure to relight in flight may have an immediate effect on flight safey.

5.2.1 IMMEDIATE RELIGHTS

The aircraft specification normally requires that it should be possible to carry out a relight immediately after engine flame-out anywhere in the aircraft flight envelope. It could be argued on the basis that the current generation of engines do not flame out without good reason, that it would be better to attempt a cold relight, rather than attempt 'instinctively' an immediate relight on run down. Nevertheless there are conditions (multiple flame-out at any altitude or on single-engined aircraft flame-out at low altitude) when the ability to relight an engine immediately may be crucial. A&AEE in assessing the immediate relight capability uses the following criteria against this requirement:

a The relight procedure must offer a high probability of a successful relight within the cleared envelope.

b The procedure employed should make use of the 'instinctive' reaction of the pilot to flame-out and allow for a degree of pilot mishandling.

c Failures to relight should not result in a condition more hazardous (eg silient stall) than the original flame-out condition.

d Aborted relights should not prejudice later relight attempts once the correct flight conditions for a cold relight have been achieved.

5.2.2 COLD RELIGHTS

Cold relights (ie relights made from windmilling RPM, with or without starter assistance) are normally required by the aircraft specification at altitudes up to the engine-out ceiling and at airspeeds down to the single engined recovery speed. On single-engined aircraft it is usually a mandatory requirement to be able to relight the engine at the gliding speed for the minimum rate of descent or maximum range. In conducting the assessment, A&AEE are concerned with verifying that the engine satisfies these requirements and the procedure used is acceptable for Service use. The criteria used (in addition to those used during ground starting) are:

a The relight procedure must offer a very high probability of a successful relight within the cleared cold relight envelope.

b The procedure employed must be simple (ideally it should be based on the ground starting procedure to assist familiarity with the techniques).

c The procedure should not compromise other pilot actions that may be necessary to recover the aircraft.

d Repeat attempts must not be prejudiced by failures to relight. If starter assistance is used, aborted attempts should not prejudice later windmilling attempts once appropriate flight conditions have been achieved.

5.3 ENGINE OPERATION IN FLIGHT

The engine is required to operate satisfactorily within the flight envelope defined in the aircraft and engine specification, or if this is not possible because of aircraft handling problems, to the boundaries of the envelope that can be cleared from considerations of aircraft handling. It is seldom possible to clear the full extent of this envelope without restricting engine operation in the more extreme regions of the envelope. The reduction in compressor surge margin with decreasing Reynolds No and with throttle handling (Figure 1) is almost certain to require restrictions on the use of the throttle at low total pressure/high altitude conditions. At the more extreme conditions it may also be necessary to restrict the engine to constant throttle operation (at idle or above), or even to restrict the cleared flight envelope so that the aircraft is not flown outside the boundaries that can be cleared from engine considerations.

The evaluation of appropriate 'throttle handling' and 'constant throttle' envelopes must be made in the light of the intended role of the aircraft. This is best illustrated by considering the different requirements stemming from two typical operation roles, the trainer (Figure 2) and the strike aircraft (Figure 3). It can be seen from these figures that whereas the training role requires an engine extremely tolerant of spins, post stall gyrations and high altitude/low speed aerobatics (including recoveries from the vertical and combat turn reversals), this is not the case with the strike aircraft, which is only likely to approach such conditions if the aircraft is flown outside its normal operating envelope. It follows that far greater importance is attached to ensuring satisfactory operation of the engine at extreme flight conditions on the trainer than on the strike aircraft. However, despite these differences in emphasis caused by the operational role of the aircraft, the basic criteria of acceptability remain predominately the same.

5.3.1 'CONSTANT THROTTLE' OPERATION

Since no clearance can be recommended for flight outside the envelope providing satisfactory 'constant throttle' operation, it is necessary to ensure that the engine can tolerate the most adverse conditions anticipated in the intervening region between the 'throttle handling' boundary and the full flight envelope. The assessment is therefore primarily concerned with those areas where throttle handling tests have shown an inadequate surge margin. Also assessed throughout the full flight envelope is the ease with which desired engine power settings may be selected and maintained, and the operation of control devices intended to maintain the engine conditions selected or to prevent the exceedance of prescribed limits.

The criteria of acceptability are:

a The engine must not surge, flame-out, or suffer from instabilities or malfunctions requiring corrective action by the pilot, in any part of the flight envelope up to the maximum permitted normal acceleration or incidence.

b The reheat must burn without buzz or instabilities requiring corrective pilot action within the declared reheat burning envelope (Note: Reheat extinction is permitted beyond the declared reheat altitude limit, provided the engine automatically reverts to a satisfactory dry engine condition).

c The engine control system must prevent the engine exceeding prescribed max RPM and max temperature limits throughout the flight envelope.

d The engine control system must maintain flight idle RPM without flame-out throughout the flight envelope (including stalling and spinning if appropriate).

e The engine control system must maintain the desired setting without creep or unacceptable fluctuations in thrust (eg thrust pulsing).

5.3.2 THROTTLE HANDLING

The assessment of throttle handling is aimed at ensuring that the engine can be handled safely (which may in turn depend on the use of a recommended handling procedure), and in ensuring that the engine response is satisfactory for the operational role of the aircraft. In assessing the safety aspects of throttle handling the objective is to define the flight envelope limits within which the engine can be handled without exceeding prescribed limits and within which there are no limits on throttle handling. The assessment of engine response is primarily made qualitatively against specific tasks (ie flight refuelling, formation flying, overshoot, etc), but also quantitatively to verify that the engine specification requirements are met.

The criteria used to assess 'throttle handling' are:

a The engine must not surge, flame out or suffer from instabilities, stagnation or malfunction requiring corrective action by the pilot as a result of moving the throttle (at any rate or in any sequence of movements), in the dry or reheat range, in any part of the flight envelope up to the maximum permitted normal acceleration or incidence.

b Reheat selection by moving the throttle at any rate from the dry range into the reheat range or reheat cancellation by moving the throttle from the reheat range into the dry range must not result in surge, flame-out, or malfunction requiring pilot corrective action below the declared reheat light up/cancellation boundary. (Note: Self clearing 'pop' surges during the transition from dry power into reheat or from reheat into dry power may be acceptable below this boundary).

c The engine control system must prevent the engine exceeding prescribed max RPM and max temperature limits during transients in any part of the flight envelope.

d The engine control system must prevent flame-out when flight idle RPM is selected (by slow or rapid throttle movements) in any part of flight envelope.

e The engine response must be satisfactory for approach and overshoot (eg the engine must have acceptable acceleration from approach RPM, the thrust step between the max dry and min reheat throttle positions must not prevent single-engine approaches, etc).

f The thrust change with throttle movement must be satisfactory for close formation flying (including flight refuelling if appropriate) and the general operational role of the aircraft.

5.3.3 ENGINE BEHAVIOUR DURING GUNFIRING AND WEAPON DELIVERY

The effect on engine behaviour of firing any weapon which may influence the conditions at the engine intake is assessed for all practical firing conditions which lie within the proposed flight envelope. Ideally, the assessment should be made at fixed throttle settings throughout the practical RPM range of the engine, and during relevant throttle acceleration and deceleration, which lie within the required weapon firing envelope. Whilst this may be possible with guns, the cost of guided weapons preclude such comprehensive coverage. It is therefore of fundamental importance to establish before the assessment the firing envelope required by the operator, and the criteria that will be used during the assessment (see MEMO 93A Reference 3 for general flight test requirements applicable to gun installations).

The criteria of acceptability as far as engine operation is concerned are:

a The ingestion of blast from gun gasses (or rocket motor exhaust) must not cause the engine to surge, flame-out, or have a lasting effect on engine operation requiring corrective pilot action within the required weapon firing envelope. (Note: Air-to-air clearances should not be limited to less than the maximum normal acceleration or incidence permitted for the aircraft).

b The operation of surge alleviation devices (ie bleeds, fuel dip, etc) must not interfere with the target 'tracking' or the ability of the aircraft to engage in combat.

c The effect of vibration caused by gunfiring must not adversely affect the operation of automatic engine control systems.

6 INTERPRETATION OF TRIAL RESULTS FOR CA RELEASE

The objective of the CA Release assessments undertaken by A&AEE is to enable judgements to be formed on the need for engine limitations (which may in turn be dependent on the adoption of a particular piloting technique and engine build standard) before the engine enters Service. The need is established after identifying regions of unacceptable engine behaviour using the criteria described. The task is then to define limitations which represent the limiting condition within which satisfactory operation should be achieved in Service. That is to say the task is to define suitable limitations which will ensure a very high probability (rather than an absolute guarantee) of satisfactory operation on all engines built to the specified standard.

To achieve this objective, it is necessary to define the limitations in a form that can be observed by the pilot, including where appropriate, suitable margins to prevent unsatisfactory operation. It is also necessary to ensure that all engines built to the specified build and modification standard, possess the required level of handling qualities in Service for satisfactory operation within the declared limits.

6.1 DEFINITION OF LIMITATIONS FOR SERVICE USE

The engine handling limitations recommended by A&AEE are normally made in accordance with Leaflet 900/1 of Reference 5. This states that limitations for Service use should be so chosen that they:

'a can be reached as frequently as the pilot is likely to need, and without risk of damage of any kind, and

b can be exceeded occasionally by a small margin without disastrous consequences.'

The first criteria covers the normal use of the aircraft, the second being intended to safeguard the aircraft against inadvertent transgressions, and to cover operational contingencies.

On rare occasions, in order to permit the full exploitation of the aircraft capabilities, the limitations may be promulgated on a 'two tier' basis. In addition to 'Normal Operating' limitations, which are intended to satisfy the above criteria, 'Never Exceed' limitations are defined because the consequential effect of exceeding the limit is either hazardous or unknown.

In formating recommendations in respect of engine limitations, there must be sufficient evidence from the trial to show that the limit defines the condition within which the particular aspect of engine operation was completely satisfactory. In particular, the tests must be sufficiently rigorous for there to be confidence that the worst combination of conditions encountered during the trial (within the proposed limits) represent the worst conditions that will be experienced in Service. Also, there must be evidence to show that small excursions beyond the proposed limits do not result in a direct hazard to flight safety. No general rule can be laid down for the margin of safety that is used. Clearly, it will depend on the type of limitation, the probability of the limitation being exceeded, and the consequences if it is exceeded (type of malfunction and the required pilot reaction time to take timely corrective action).

To permit the full exploitation of the engines capabilities, the proposed limit should accurately describe the limiting conditions found during the CA Release trials. In reality this is seldom possible. For the limits to be observed they must be expressed in a form that can be readily appreciated by the pilot. This requires that the limit must be framed in terms of the indications of the primary flight instruments (eg airspeed, Mach number, altitude, normal acceleration and on some aircraft incidence). Alternatively the onset of natural or artificial warnings may be used (eg the onset of pre-stall buffet in lieu of an incidence limit).

The obvious outcome of this process is that the proposed limits will inevitably be more restrictive than the limiting conditions found during the CA Release trials (Figure 4). This may in turn cause a reappraisal of the margins originally thought necessary on flight safety grounds.

Finally, it may be necessary to simplify or rationalise the proposed limit to ease the complexity of the piloting task in monitoring the limit. When this process compromises the operational capabilities of the aircraft the final judgement on the form cf the limitation is made in close collaboration with the user. Alternatively A&AEE may propose limits, but leave with the user the final decision on how the Release limitations are to be framed.

6.2 LIMITATIONS AND THE NEED TO ENSURE APPROPRIATE ENGINE STANDARDS

To achieve satisfactory operation within the proposed limits, it is necessary to ensure that all engines accepted into service have the required handling qualities, and that this level is maintained throughout the life of the engine.

The proposed limits are normally tied to a minimum build and modification standard as a first step to ensuring that all engines in Service possess handling qualities at least as good as those tested. However in all probability, the CA Release assessment will be made with engines at the beginning of their service lives. It is therefore probable that insufficient data will be available about the effect of production tolerances, repair schemes, age deterioration and other factors affecting the handling characteristics during the life of the engine (Figure 5). There is therefore a problem in interpreting the results from trials if the stated aim of formulating limits within which satisfactory operation should be achieved, is to be met for the service life of the engine.

A solution to this problem would be to build into the limit an allowance to cover the worst possible configuration. This solution is seldom adopted in practice, however, because of the obvious difficulty in deciding the correct allowance, and because such a solution would unnecessarily penalise most engines.

The preferred solution is to use a 'surge margin' test to reject unsatisfactory engines. Ideally the test will combine the use of gauze tests on the bench prior to installation with specified air test requirements to be carried out after installation and periodically thereafter. The air test is the vital component in this solution as it enables the surge margin to be checked against a declared standard on installation and periodically thereafter. Thus it can be used to monitor age deterioration and changes that might be introduced during the life of the engine (including module changes) which otherwise would go unchecked. However if the air test is to be effective the conditions chosen must be as close as practical/safety considerations will permit to the operating limits. Any deviation away from the limiting conditions will permit inferior engines to enter service.

A&AEE will therefore normally tie the proposed limit with the requirement that all engines must satisfy a particular air test if there is doubt about the representativeness of the engines tested. The user is of course free to choose a less rigorous condition or accept engines on a concession basis, but this may invalidate the boundaries of satisfactory operation presented in the CA Release.

7 CONCLUDING REMARKS

The past decade has seen significant advances towards the goal of higher operational effectiveness through lower fuel consumption, higher thrust and lower weight engines. In general, these advances have not diminished the engine assessment task described in this paper. Engines have tended to become more prone to surge or handling problems in the pursuit of higher effectiveness, and at the same time, the flight conditions required from the aircraft have tended to become more severe. On the credit side the advent of electric instead of hydro mechanical control systems has resulted in systems better able to cope with handling problems as they occur.

It is reasonable to expect this process to continue. Certainly the advent of aircraft manoeuvre demand and stability augmentation systems will make greater demands on the engine, but this is likely to be balanced by the greater authority offered by digital engine control systems. In the long term, the possibilities of automatic correction of engine handling problems and 'built-in' protection against limit exceedance offer the exciting prospect of freeing the pilot from most of the problems associated with observing operating limits, and will permit the aircraft to be flown to its full capabilities.

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4 J K Appleford, "Performance Division Clearance Philosophies for Fixed Wing Aircraft", 5 December 1978, A&AEE Note 2111

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ACKNOWLEDGEMENTS

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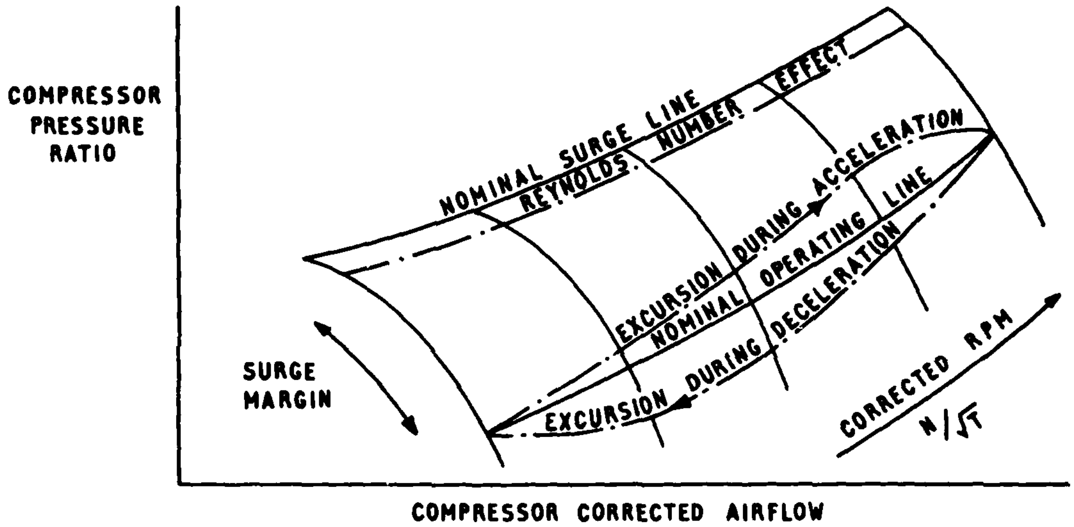

FIG.1 TYPICAL COMPRESSOR PERFORMANCE MAP.

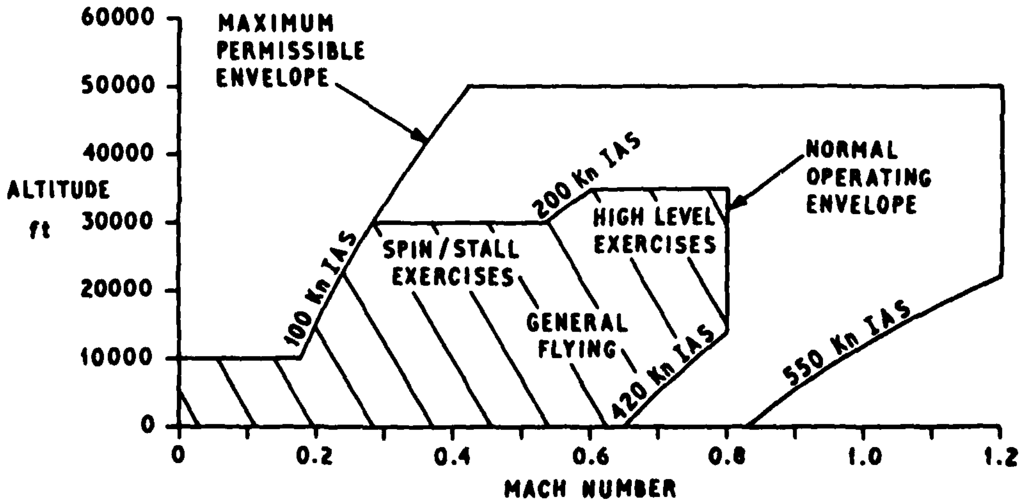

FIG.2 TRAINER AIRCRAFT – FLIGHT ENVELOPE.

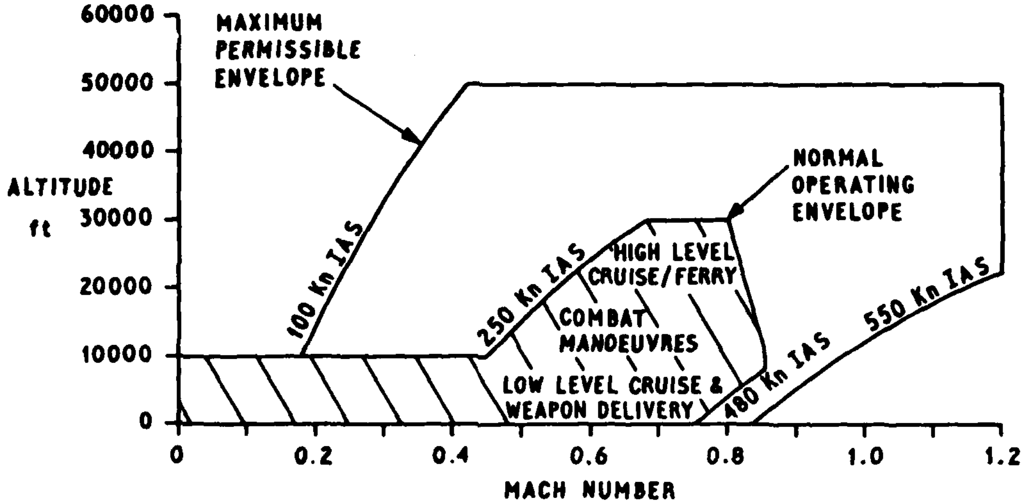

FIG. 3 STRIKE AIRCRAFT – FLIGHT ENVELOPE.

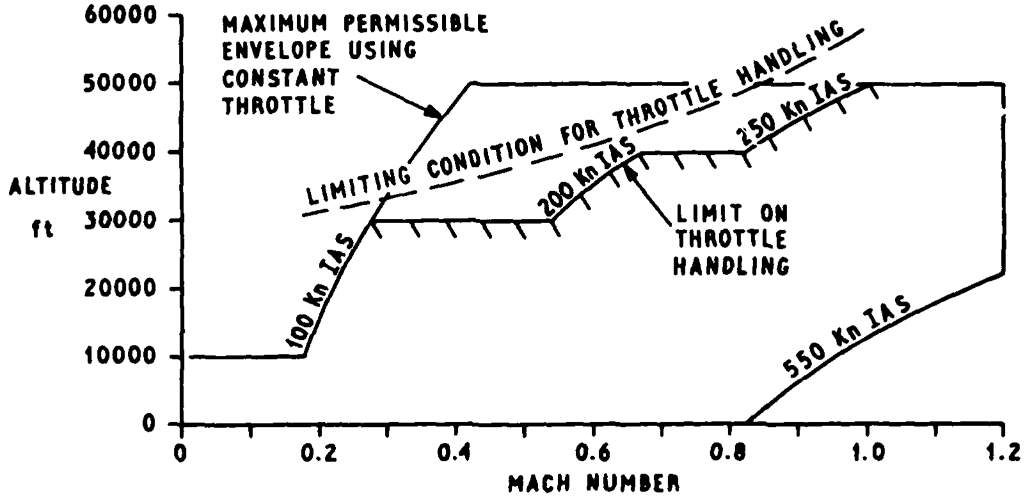

FIG. 4 DEFINITION OF FLIGHT ENVELOPE LIMITATIONS.

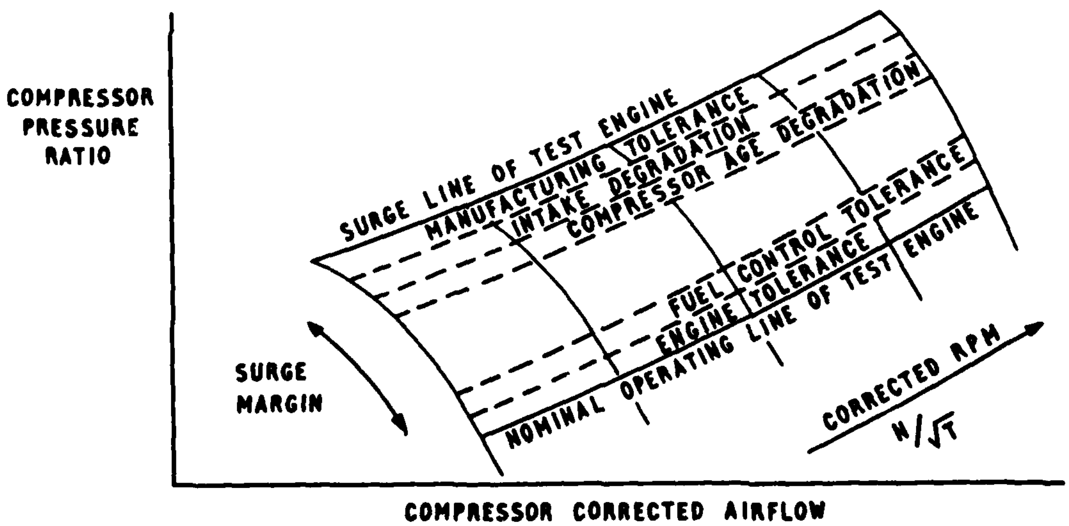

FIG. 5 CUMULATIVE REPRESENTATION OF DEGRADING FACTORS ON COMPRESSOR PERFORMANCE MAP.

A LOOK AT ENGINE HANDLING ASPECTS FROM AIRCRAFT FLIGHT TEST EXPERIENCE; WITH AN OUTLOOK FOR FUTURE REQUIREMENTS

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SUMMARY

The question of airframe/engine compatibility with regard to engine handling under high incidence and various power off-takes assumed special significane, as the engine (RB 199 in Tornado) was gradually approaching its production configuration.

The effort at localizing and quantifying the somewath lower than predicted compatibility between airframe and engine involves two separate approaches:

1. Full-scale (development aircraft) trials
2. Model tests

During the course of flight testing, various palliatives designed to improve engine handling were successfully implemented. These modifications were developed by means of model trials and later verified in flight tests, ascertaining safe and full handling capability even at the corner points of operational conditions.

Future requirements, as viewed for advanced projects, will have to cover an extended handling envelope. High manoeuvrability may put some additional strains on aircraft engine integration.

INTRODUCTION

It is the intention of this paper to report on the integration of the RB 199/34R engine into the MRCA-TORNADO airframe (Fig. 1).

This paper deals with two major subjects which are interrelated:

- engine/airframe compatibility and handling characteristics of the installed engine
- development of modifications to improve compatibility and handling.

It is selfevident that the problems described herein were encountered early during flight testing and have been resolved in the course of program development.

The model tests established the aerodynamic reasons for the airframe/engine incompatibility and aided in confirming the efficiency of the palliatives.

Furthermore some thoughts as to future requirements and future projects are being presented.

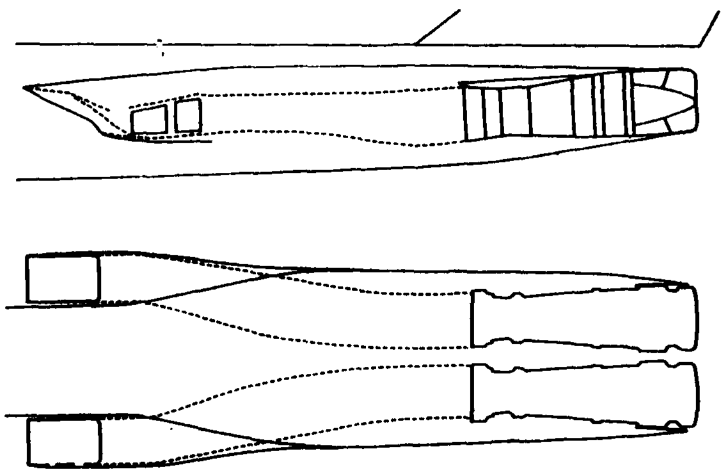


Fig. 1: TORNADO INTAKE WITH RB 199 ENGINE

NOMENCLATURE

Ma	=	Mach number
TT1	=	ambient total temperature
$\frac{M\sqrt{T}}{P}$ ≙ WAT	=	corrected engine massflow
δ_2	=	intake rampe angle
TBT	=	turbine blade temperature
TIT ≙ SOT	=	turbine inlet total temperature
SOT/θ	=	corrected turbine inlet temperature
$N/\sqrt{\theta}$	=	corrected engine speed
N_1 ≙ N_L	=	fan speed
N_3 ≙ N_H	=	HP compressor speed
$N_L/\sqrt{\theta}$	=	corrected fan speed
$N_H/\sqrt{\theta}$	=	corrected HP compressor speed
ALPHCU ≙	=	aircraft incidence
POT	=	power off-take
$\frac{KW}{\delta\sqrt{\theta}}$	=	corrected power off-take
RN INDEX	=	Reynolds number INDEX
p1 - p2	=	pressure gradient
	=	swirl angle
T	=	thrust
D	=	aircraft drag
δ	=	total pressure / 101.325
θ	=	total temperature / 288

1. DESCRIPTION OF CRITICAL CASES

1.1 The sea level static case

During all ground installation running a difference in engine characteristics between the left and the right hand nacelle existed due to directional rotation. The result is that engines installed in the left hand nacelle are more surge prone during ground operation especially after the engine has aged (deterioration).

1.2 The case of

- low altitudes
- medium speeds
- high angle of attack and constant reheated power settings (Max Reheat or Combat)

Handling problems have been experienced with engines installed in the left hand nacelle during high incidence flight manoeuvres.

Flow separation along the lower lip at high incidence produces a counterrotating swirl which changes the low pressure compressor characteristic and ultimately upsets the core engine.
In order to overcome the problem at least initially special reheat tuning instructions were defined for engines installed in the left hand side in order to improve the incidence capability at Max Reheat conditions: reheat fuelling was reduced and the nozzle area was increased relative to the pass-off setting. The pass-off values were retained for engines installed in the right hand nacelle. The fuel reduction lowers the reheated fan working line and thus gives more incidence capability in the critical speed/altitude range.

1.3 At intermediate to high altitudes

- medium speeds
- high angle of attack and various power off-take loads
- engine handling between Idle and Max Dry

In this context special attention has to be paid to the power off-take capability at high and medium altitudes and low TT_1 especially under high angle of attack. Power off-take induced surges have been experienced in both nacelles but mostly in the left hand side. Core compressor running lines are raised by high non dimensional power off-takes and the compressor surge lines are dropping due to low Reynolds-numbers.

1.4 The supersonic surge problem

This became evident during flight test evaluation. The non dimensional operating point of the engine reduces with TT_1, i. e. $N/\sqrt{\theta}$ and airflow $\frac{M\sqrt{T}}{P}$ are reduced and the intake ramp closes - δ_2 (ramp angle) increases.

Detailed analysis of supersonic flights revealed that the engine surges if the corrected fan speed falls below a certain level or δ_2 rises above a certain level (right hand only). This means that a surge free envelope is significantly affected by engine rating and ambient temperature.

Various palliatives, designed to improve intake-engine compatibility, and effective within the above-described critical operational region, as a minimum requirement, were fitted to several models and extensively surveyed in more than one wind tunnel. The range of fixes investigated included duct and cowl fences, duct guide vanes, a honey comb flow straightener, and a cowl slot. All of these, except for one, were rejected because they did not satisfy the prerequisites of practicability and/or cost effectiveness, although some were quite effective.

2. FLIGHT TEST RESULTS

During all engine ground running it became evident that there is a difference in certain engine characteristics between a right hand and left hand installed engine (see Fig. 2).

In the right hand nacelle the fan is deloaded compared to the left hand installation. Fig. 2 shows the corrected fan speed vs. the corrected high pressure spool speed and there is a clear indication of a 2 % difference between left and right.

The turbine inlet temperatur/high pressure spool speed relationship (SOT/ vs. N_H/) is not affected by the installation. This is in line with observations from many engines which have run in both left and right nacelles. Engines installed in the left hand nacelle are more surge prone during ground operation i. e. surges can come about under this operational condition if the engine installed is either in a "deteriorated" and/or "cold" (warm up surges) state.

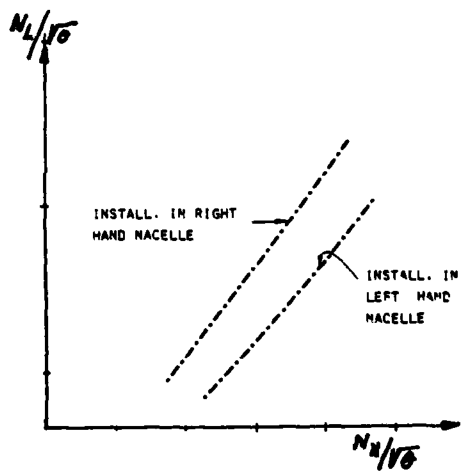


Fig. 2 EFFECT OF LEFT AND RIGHT HAND INSTALL.

The second problem, which has been experienced with engines installed in the left hand nacelle, was during high incidence flight manoeuvres at Max Reheat power setting at low altitude and medium speed.

Variations in fan speed were observed during increase in aircraft incidence. The fan speed in the left nacelle is progressively reduced, and the fan speed in the right nacelle is progressively increased as incidence increases (see Fig. 3, 4 and 5).

This eventually results in a surge at medium angles of attack.

Reheat fuelling was reduced and the nozzle area was increased relative to the pass-off setting during engine acceptance test. The pass-off values were retained for engines to be installed in the right hand nacelle. The fuel reduction lowered the reheated fan working line i. e. by implication deloading the fan line and thus giving more incidence capability in the critical speed/altitude range. On the other hand the reduction in reheat fuel reduced the maximum reheat thrust marginally.

A further problem at intermediate to high altitudes/ medium speed/high angles of attack and various power off-take loads within the dry engine handling regime appeared during flight test evaluation. Power off-take induced surges occur in both nacelles but mostly in the left hand side.

During engine testing in a high altitude test facility good correlation was seen between surge onset with power off-take and Reynolds number for different levels of distortions (see Fig. 6).

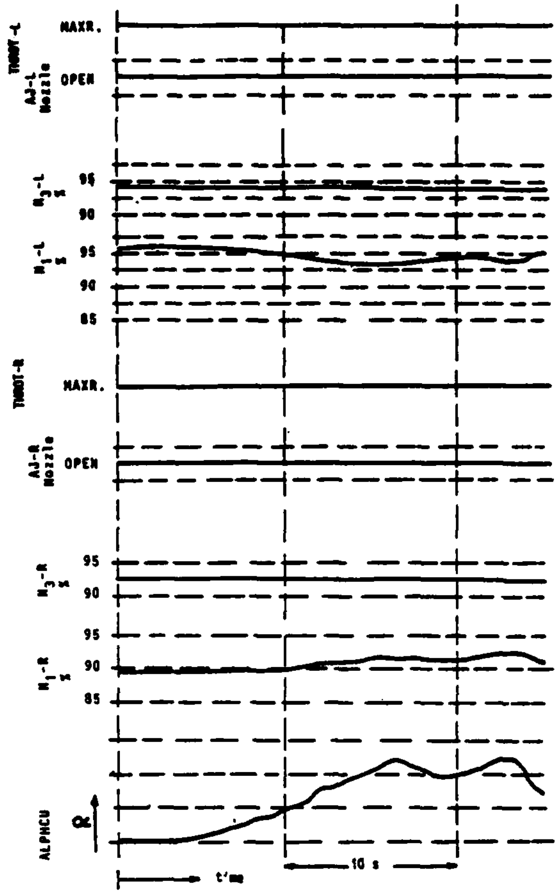


Fig. 3 TIME HISTORY OF A HIGH INCIDENCE MANOEUVRE AT MAX.REHEAT POWER SETTING

At supersonic speeds the compatibility problem existed in the right hand intake.

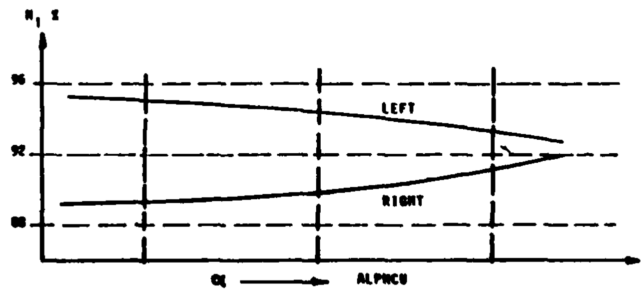


Fig. 4 FAN SPEED VARIATION WITH INCIDENCE

Plots of corrected fan speed $(N_1/\sqrt{\theta})$ versus Mach number produced an apparent correlation in terms of a surge boundary (s. Fig. 7). This form of plot takes into account the variation in ambient temperature (altitude and ISA-cond.) and variation in engine rating.

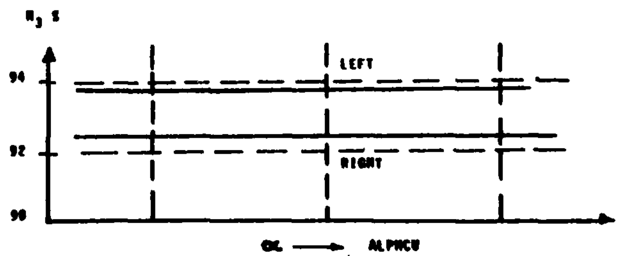


Fig. 5 NH BEHAVIOUR DURING AIRCRAFT INCIDENCE

Levels of fan speed as given in Fig. 7 are sensitive not only to engine rating (different levels of turbine inlet temperature) but also to the reheat fuel system setting and engine bleed flows. Changes in fan speed can also occur due to engine and fuel system deterioration. Therefore, a certain band of $N_L/\sqrt{\theta}$ against TT_1 for an engine must be considered.

Flight test experience reveales, that an $N_L/\sqrt{\theta}$ of 75 % is a critical value which is indicative of a possible surge onset. It appears that the parameter $N_L/\sqrt{\theta}$, besides being an engine related parameter was apparently also indicative of the quality of the airstream in the intake, where $N_L/\sqrt{\theta}$ = 75 % appeared to be a threshold value.

By crossplotting the threshold values on to a Mach/altitude graph, one can establish surge free operating boundaries in the supersonic area.

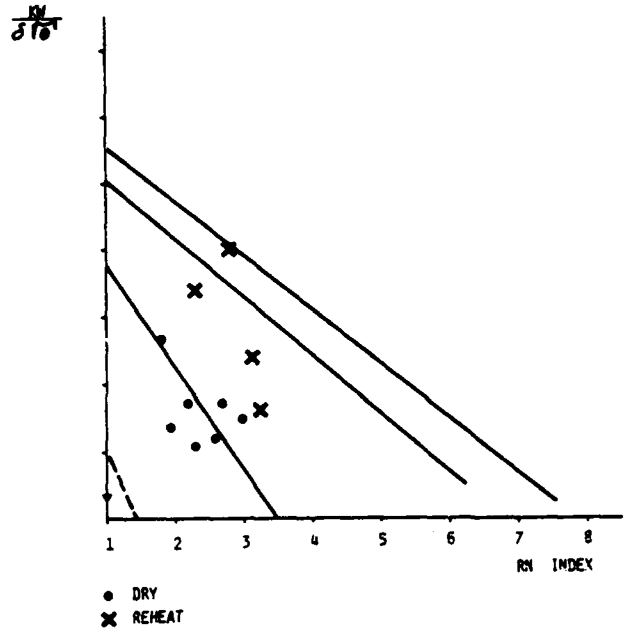


Fig. 6 NONDIMENSIONAL POWER OFF-TAKE VS. REYNOLDS NUMBER INDEX

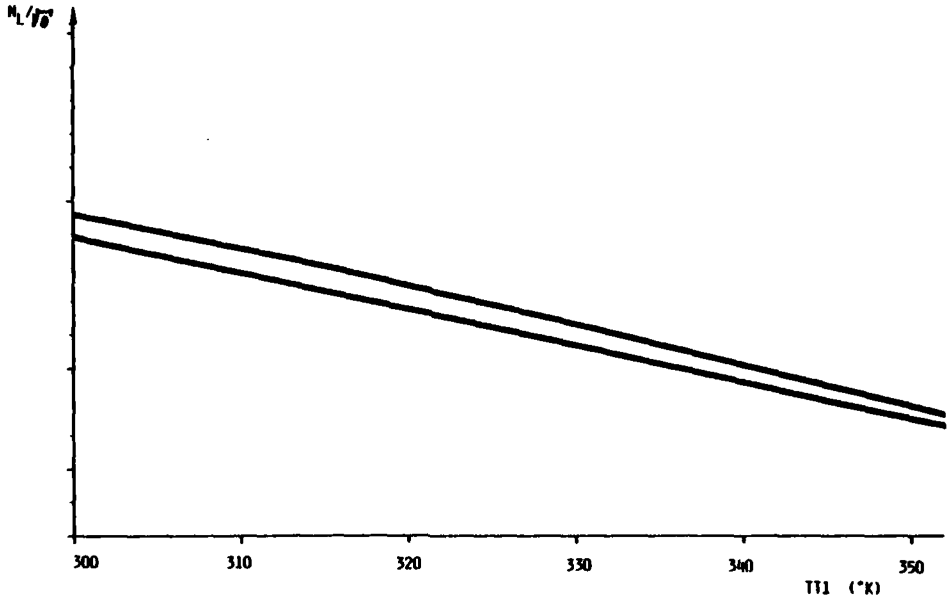


Fig. 7 CORRECTED FAN SPEED VS. TOTAL TEMPERATURE

3. SPECIAL MODEL INVESTIGATIONS INTO THE CHARACTERISTICS OF THE INTAKE FLOW

Investigation by model testing revealed that two distinct flow phenomena, intensifying with increasing angles of attack, combine to reduce the intake engine compatibility and eventually, especially under power off-take conditions, cause engine surges:

1. Intake flow swirl caused by

   flow separation form the intake cowl lip (subsonic) and
   flow separation from the third ramp (high supersonic)

2. Intake flow total pressure distortion (steady state and dynamic)

A closer lock at the intake and its location with respect to the rest of the airframe shows that the flow field, as seen by an installed engine, is the sum of the flow fields around/through two separate complex geometries with their own distinct aerodynamic characteristics:

- The flow field around the front fuselage
- The flow field through the intake duct.

The geometry of the intake is, in simplified terms, made up of a rectangular, nearly square, S-shaped duct transitioning into a circular, slightly contracting cross section at the engine-air frame interface.
The forward opening of the duct is bounded, left and right, by flat, cut-back (swept back) side plates, at the top by a wedge, and at the bottom by a cambered cowl lip.

This basic geometry of the intake duct is modified through the addition of two, inward opening auxiliary air intake doors, located outboard near the front opening of the duct, two flat, moveable ramps, adjoining the top wedge and separated by a slot to allow for the exhaust, or possible induction, of air and a so-called corner fillet in the upper inboard corner of the duct, below the second moveable ramp, extending partway towards the engine face.

There exist, as outlined above, several distinct operational conditions which affect intake - engine compatibility to such an extent that engine surges may occur:

3.1 The sea level static case:

That part of the total airstream which passes across the cowl lip, having a fairly small leading edge radius (mandatory for supersonic inlets) and also cambered upward, produces local flow separation. This was established by full scale flow visualization and model total pressure measurements.

Additionally the S-bend of a duct causes secondary flow in bends.

This imparts a velocity component to the airstream which is directed inboard, i.e. a swirl, rotating conterclockwise in the left hand intake, will be generated. This, in turn, loads the fan which is rotating clockwise. On the right hand side a corotating swirl is produced which effects a deloading of the engine.

As a diagnostic test the 8 arm intake rake was installed in the left hand nacelle. This rake was initially used for intake steady state pressure measurements and was to serve now as a flow straightener in front of the left engine to compensate the counter-rotating swirl (s. Fig. 8).

The observed tendency is in line with the expected effect:
At the same high pressure spool speed (N_H %) the fan speed (N_L %) was increased by 1 %, i. e. the fan was deloaded by a certain degree. The relationship - turbine inlet temperature ($^{TIT}/\theta$) against high pressure spool speed ($N_H/\sqrt{\theta}$) stayed unchanged.

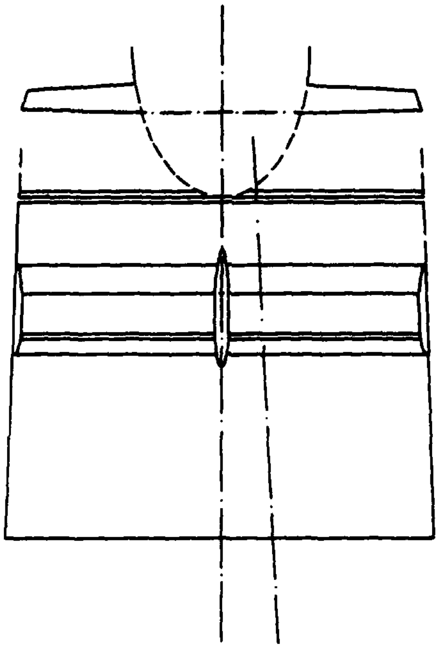


Fig. 8 TOTAL PRESSURE INTAKE RAKE USED AS A FLOW STRAIGHTENER

3.2 In the case of low altitudes/high angle of attack and constant reheated power setting and also under high angles of attack and various power off-take loads the airstream reaching the engine has a completely different quality:

The total air approaching the intake, operating at a fairly high angle of attack with respect to the free stream, will consist of free stream air plus air flowing along the side of the fuselage as well as air flowing along the bottom surface and across the bottom edge of the fuselage. This means that there exists the typical vortex generated by the forebody

In addition to that at high incidences separation at the cowl occurs as shown by model flow visualization and total pressure measurements. Again, the separated flow, with its reduced energy content, will pass through the S-bend of the duct and a counter clockwise rotating swirl will be sent up. One has to note, that at these forward speeds the auxiliary air intake doors are closed and that no additional air is being induced.

Normally the reheat system is set up in such a way as to obtain maximum reheated thrust. Intake flow bulk swirl, intake flow (total) pressure distortion and intake flow turbulence, functions of aircraft angle of attack, will combine to erode the engine surge margin.

Everything said about the aerodynamic situation of the airframe/intake combination for the previous case applies equally as well to the case of higher altitude with power off-take (upper/left hand corner of the flight envelope).

The positive effect of the intake rake as a flow straightener, which has been seen previously, improved also the overall handling qualities of the engine under the above mentioned flight conditions.

The progressive intake swirl effect during build up of incidence was reduced and handling improvements were gained from this swirl reduction especially during incidence build-up at constant reheated power.

3.3 The supersonic case

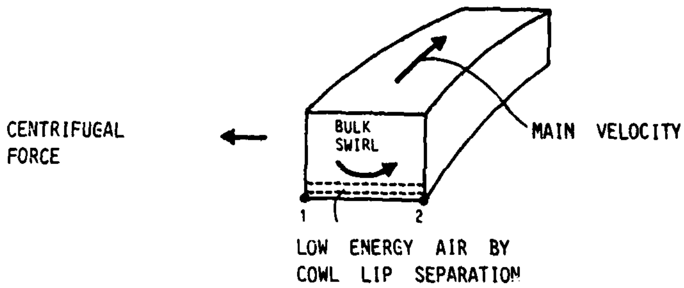
At supersonic speeds compatibility problems may occur in the right hand intake, when the intake ramp is at high δ_2 - values (i.e. high Mach numbers and high temperatures).

4. DESCRIPTION OF FLOW PHENOMENA

As already mentioned, the total air approaching the intake, operating at high angles of attack, will consist of free stream air plus air flowing along the side of the fuselage as well as air flowing along the bottom edge of the fuselage.

In addition, when the intake is operating under high angles of attack flow separation along the cowl lip will be triggered and this separated flow, with its reduced energy content, will pass through the S-bend of the duct.

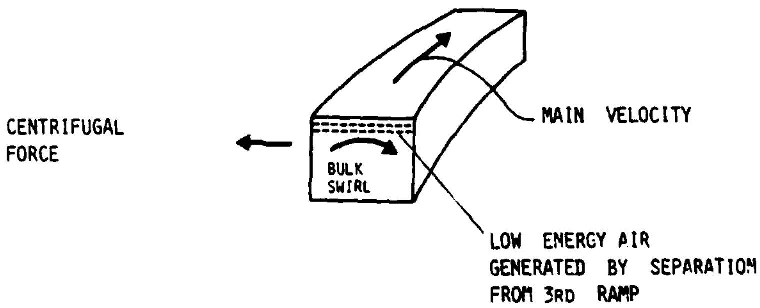
If one considers a rectangular bent duct with air passing through, the pressure gradient built up in the main stream will be of such a form, that the static pressure along the outside wall of the bend is higher than that along the inner wall because



of the centrifugal forces, i. e.

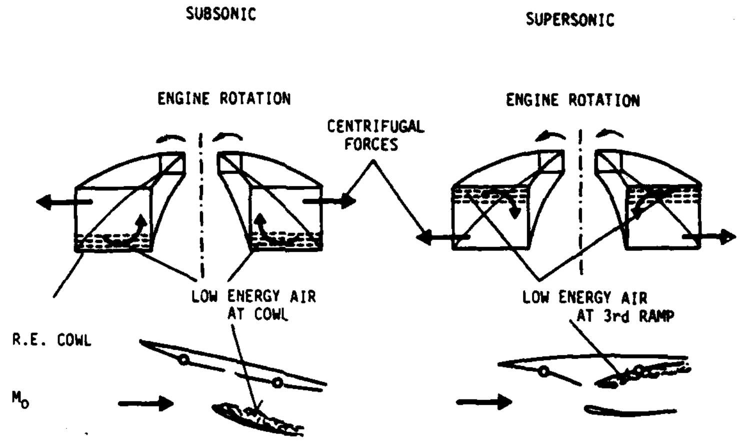
$$P1 > P2$$

With a boundary layer or a region of low energy air the local centrifugal pressure gradient is insufficient to work against the pressure difference $P_1 - P_2$. Thus, a flow direction from 1 to 2, will be generated which eventually results in a swirl in the duct.



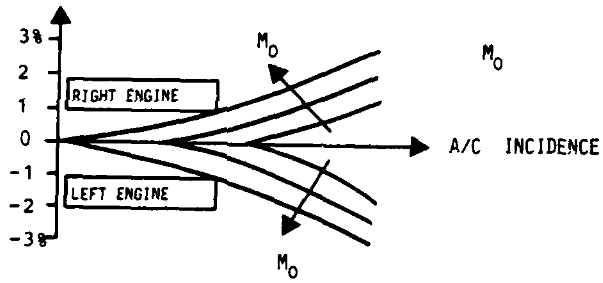
It has to be noted, that separation on the 3rd ramp of the intake at high supersonic speed will, by this theory, generate a swirl in the opposite direction.

SUMMARY PICTURE



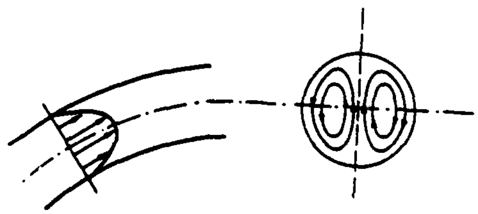
It must be reiterated, that the bulk swirl is counterrotating to the fan rotation in the port intake and corotating to the fan rotation in the starboard intake (subsonic flight at high incidence). This is in agreement with the results found during flight testing, where during incidence build-up a deloading of the fan in the R.H. intake, occurred, whereas the fan in the L.H. intake was loaded and therefore more surge prone under certain conditions.

ΔN_L = CHANGE IN FAN RPM DUE TO A/C INCIDENCE, $M_0 < 1$



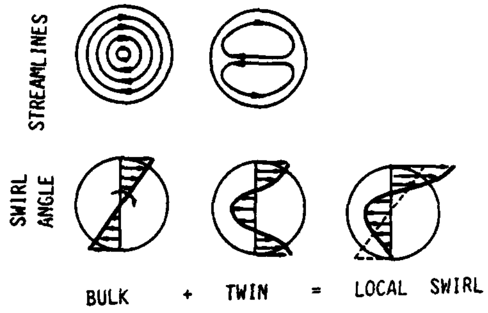
Another flow phenomenon, which obviously contributes to the above mentioned engine behaviour, is the secondary flow (twin swirl) in a curved pipe.

The fluid particles in the center of a bent duct will be forced under the action of centrifugal forces, to move toward the outside of the duct compared to those particles which are located in the region closer to the wall, i. e. the lower velocity region. Because the high energy particles move toward the outside of the bend, they displace the particles with the lower stagnation pressure. Since the cross section of the duct is closed and continuity conditions must be maintained, the low energy particles will move along the wall of the duct toward the inside of the bend. This motion, set up by the outward movement of the particles with the higher stagnation pressure and the simultaneous movement of the lower stagnation pressure particles along the wall of the duct toward the inside of the bend, constitutes the secondary flow.



Superimposing bulk swirl on the twin swirl components results in local flow angles, which vary between 0° and max at the duct wall.

This was measured by model tests and indicates clearly that during one revolution a fan blade is exposed to flow angles which vary locally.

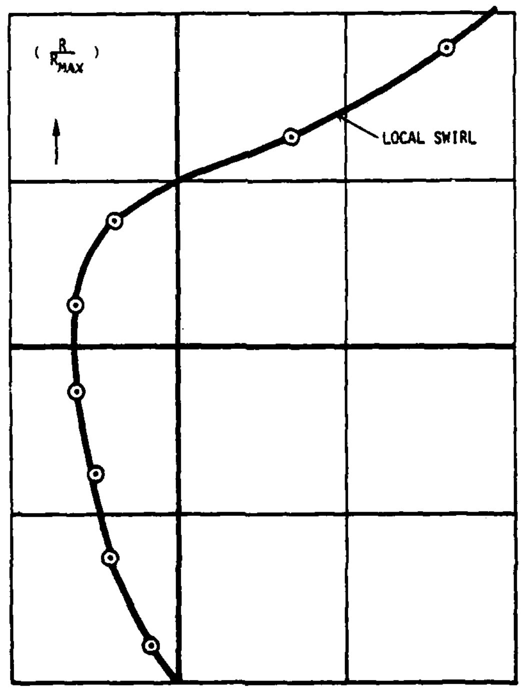


INTAKE MODEL RESULTS [1]

The effects observed and confirmed by analyses of experimental results are typical for intakes of this type (F-14; F-15).

SUBSONIC SWIRL

RADIAL DISTRIBUTION [1]



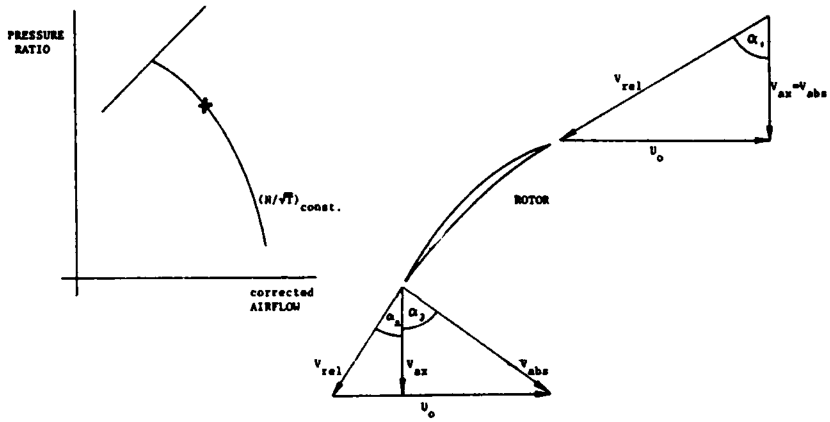
5. THE EFFECT OF INLET SWIRL VARIATION ON STAGE PERFORMANCE

Any compressor without inlet guide vanes has a natural tendency to demand swirl free inlet flow. The fundamental velocity triangles without swirl on a rotor blade are described in Fig. 9
The running point in the simplified compressor map is indicated by a cross. The appropriate rotor inlet and rotor outlet velocity triangles are represented by solid lines Fig. 10 . Also the effect of corotating and contrarotating swirl relative to the rotation of the fan with respect to stage performance and velocity triangles are shown in Fig.10 assuming constant airflow and constant fan R.P.M.

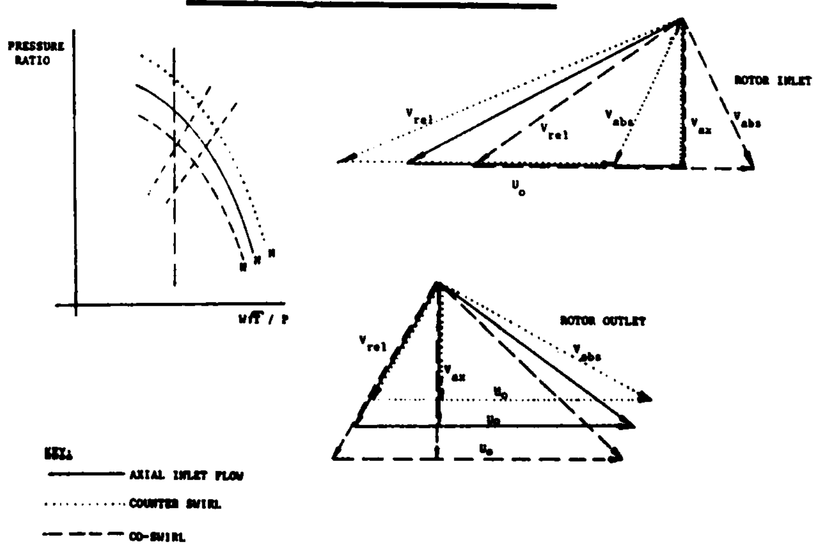
A counterrotating swirl (dotted lines) increases the load on the blades; as a consequence, the blade's leading edge has to operate at a higher angle of attack. Thus the pressure ratio increases at constant R.P.M. and airflow. In fact, the equilibrium between turbine and fan will cause the spool to reduce its R.P.M. somewhat. Thus the running point will ultimately shift to a slightly higher pressure ratio at a lower airflow, i. e. to a point on a distinct R.P.M. line below the dotted one. The result is a reduced surge margin.

If the same engine is exposed to a corotating swirl, the fan will be unloaded (dashed lines). Besides the effect that the fan is running faster, the above described problems do not appear because the flow conditions across the fan are being influenced in a positive manner as long as the decreasing angle of attack at the fan blade's leading edge is kept within reasonable limits. The fan pressure ratio decreases and R.P.M. and airflow increase.

Fig.9 BASIC ROTOR VECTOR TRIANGLES (COMPRESSOR)



EFFECT OF INLET SWIRL VARIATION ON STAGE PERFORMANCE



6. THE EFFECT OF POWER OFF-TAKE

It was shown that several distinct flow phenomena, intensifying with increasing angle of attack, combined to reduce intake/engine compatibility and eventually caused engine surges.

A factor of great importance is the power off-take (POT) capability of the engine especially in the left hand corner of the flight envelope, where the Reynolds number has a significant effect on the compressor map. If one considers the worst case with respect to POT, i. e. the maximum demand phase, placing the entire load on one engine, then the engine showed, in combination with the above described flow phenomena, a reduced power off-take capability.

It has to be pointed out that there are many discrepancies between the power off-take capability of an engine as demonstrated in an high altitude test facility (ATF) and as demonstrated in an aircraft. These differences are due to engine deterioration and shortcomings in the available test facilities:

- The simulation of intake distortion is difficult; the use of gauzes in the ATF will result in non-representative levels of DC 60 and distortion patterns

- Intake swirl is not simulated in the ATF

- It is more difficult to simulate in the ATF transient power off-take loads as they prevail in the aircraft (wing sweep etc.).

Fig. 11 shows the effect of a sudden POT demand on an engine and the loss of surge margin of the compressors.

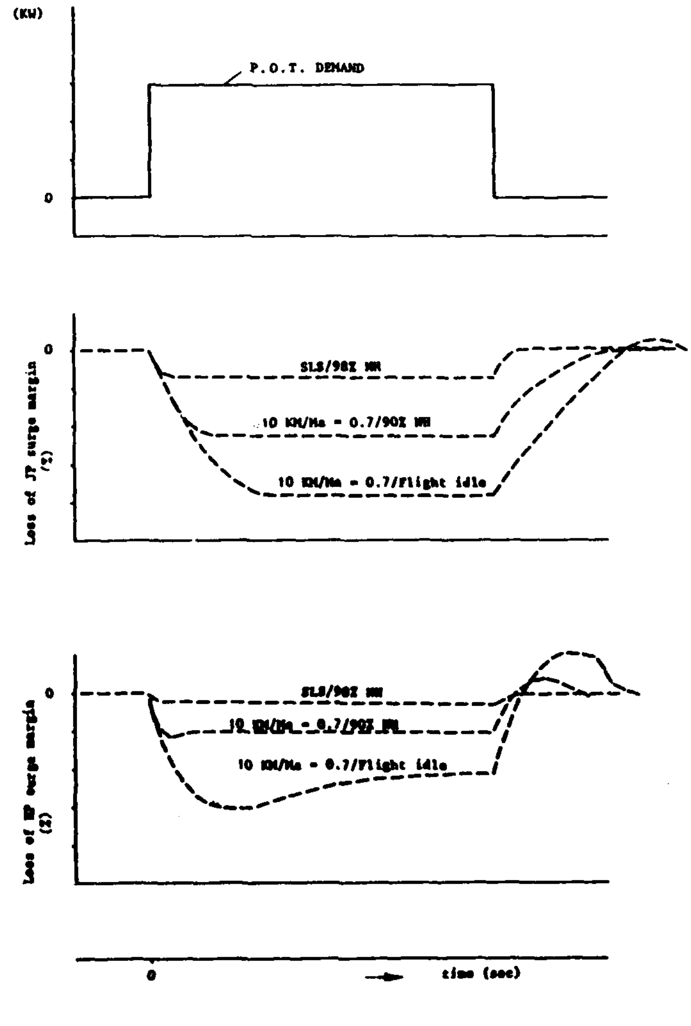


Fig. 11 THE EFFECT OF POWER OFF-TAKE ON ENGINE SURGE MARGIN

7. MEANS OF IMPROVING PERFORMANCE AND HANDLING QUALITIES

There are obviously two approaches to improve the compatibility situation:

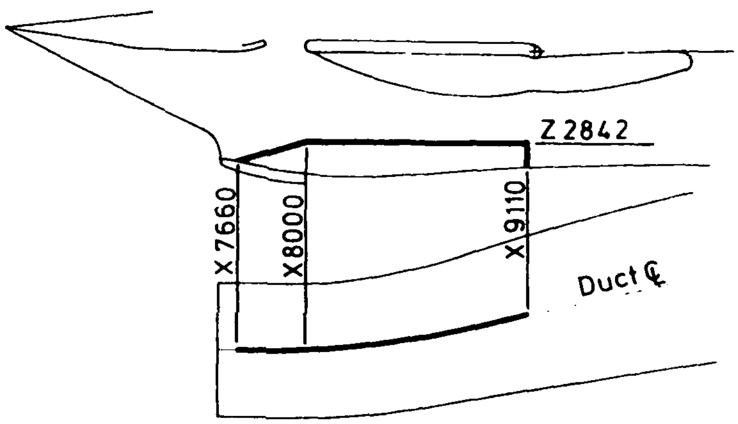
1. Propulsion system (engine and engine control):

   Any improvement in component efficiency of the engine improves thrust and handling quality. An improvement in efficiency may be gained either by detailed design changes to the engine (blading changes, improved sealing, reduced tip clearances, etc.) or by rematching the engine. A prerequisite to achieve performance improvements by raising the fan running line is adequate surge margin.

2. Airframe (intake):

   Costeffectiveness and time considerations pointed towards modification of the intake rather than the propulsion system:

   Various palliatives, designed to improve intake-engine compatibility and effective within the above described critical operational regions, as a minimum requirement, were fitted to several models and extensively surveyed in more than one wind tunnel. The range of fixes investigated included duct and cowl fences, duct guide vanes, a honeycomb flow straightener and a cowl slot. All of these, except for one, were rejected because they did not satisfy the prerequisites of practability and/or costeffectiveness. Although some were quite effective, the palliative answering the above criteria best is a simple cowl fence.



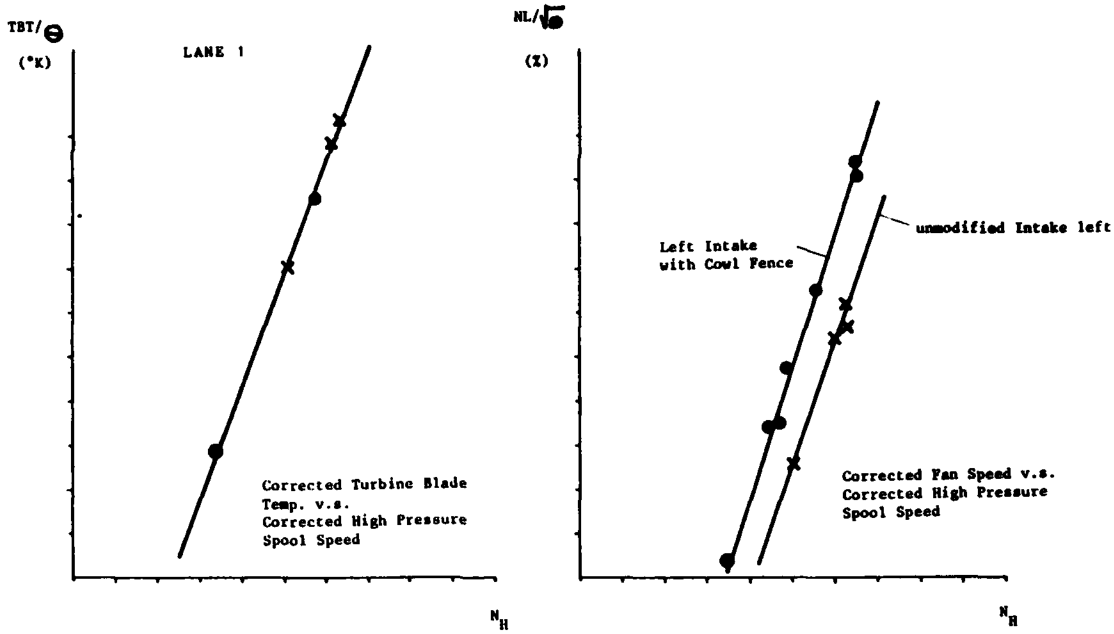
"Long" Cowl Fence [1]

A comparison of engine ground run results in the left hand intake with and without the cowl fence is illustrated in Fig. 12 .
The installation of the fence raises the fan speed by 2% for a constant N_H-speed, i. e. the intake swirl effect is compensated by +2.0 % N_L.

The SOT/N_H - relationship is not affected from the installation of the fence. This is in line with the observations from other engines which have run in both left and right nacelles. Thus SOT/N_H - relationship is the same for both installations. High incidence manoeuvres in Max Reheat were performed and a direct comparison of the engine handling qualities was obtained behind a normal and a fenced intake (see Fig. 13).

The engine was running surge free although the reheat restoration was increased, i. e. the fan running line was raised, which is obviously a result of the improved intake flow quality.

Fig.12 LEFT INTAKE FENCE EFFECT ON ENGINE FAN SPEED



Left Intake with Cowl Fence   Normal Left Intake

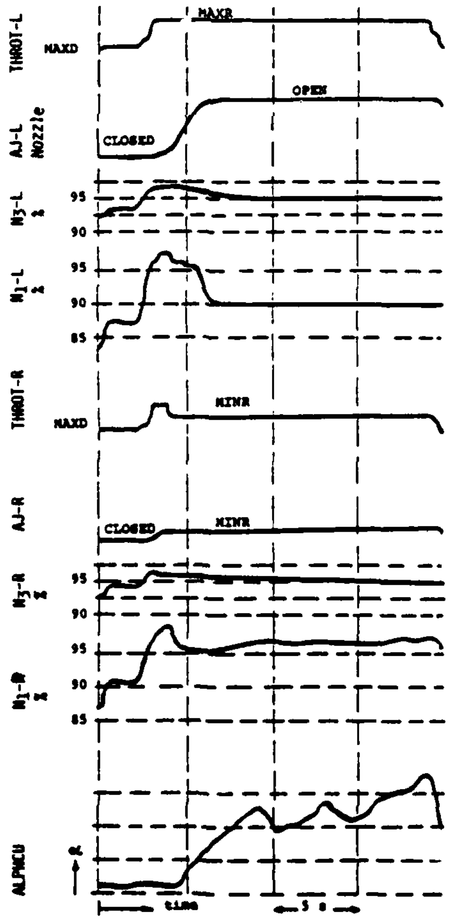


Fig.13

TIME HISTORY OF A HIGH INCIDENCE MANOEUVRE AT MAX. REHEAT POWER SETTING WITH FENCED INTAKE

Fitment of a cowl fence to the right hand intake, a mirror image installation to the left hand intake, resulted in an improvement in engine intake compatibility for the supersonic flight regime:
the separated flow from the 3rd (subsonic) ramp is being channeled in such a manner that the destabilizing swirl flow is sufficiently reduced to allow for surge free engine operation in the right hand intake in the supersonic area.

The results presented in Fig. 14 and 15 are gained from model tests and show clearly the effectiveness of the cowl fence in the reduction of swirl. The bulk swirl is reduced by a large amount whereas the twin swirl is stable.

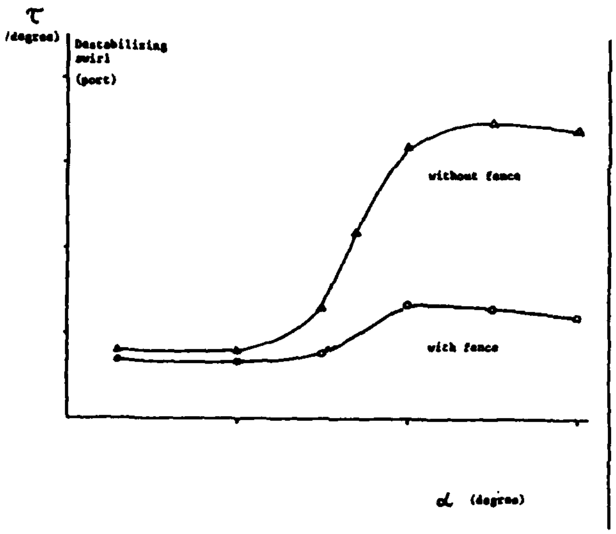


Fig.14 MAXIMUM SWIRL ANGLE ON 87 % DUCT RADIUS VS. INCIDENCE [1]

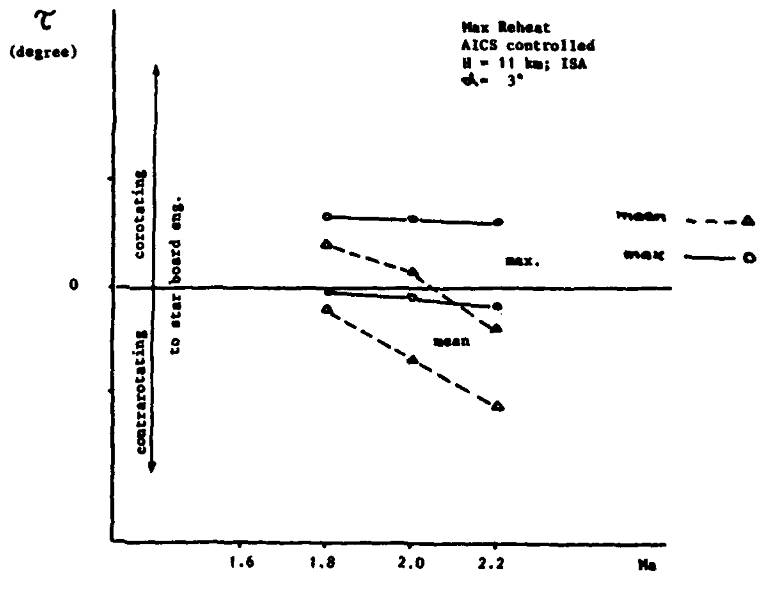


Fig.15 INTAKE SWIRL (SUPERSONIC) [1]

8. FUTURE REQUIREMENTS

The foregoing shows that a large amount of effort in engineering and flight test had to be invested in order to improve handling characteristics to todays well acceptable level.

In our case, a brand new aircraft/engine combination has definitely created a more difficult task.

This experience should now enable us to specify our future requirements more to the actual needs of a new system, taking into account expected advancements in avionic and weapon technology. In this context, the 1990 situation has been extensively assessed leading to two distinct combat arenas.
Contrary to earlier thinking that the expected weapon and avionics advancements would reduce requirements for the aircraft to the level of a mere weapon launch platform, it is now evident that superior aircraft manoeuvrability will be essential for combat effectiveness. In addition, a more dynamic way of manoeuvring is envisaged with only shorter sustained segments.

All of the above indicates that at least in some cases engine handling requirements will be affected.

<u>Short Range Combat</u>

As seen from Fig.16 the flight envelope has been extended to the left relative to conventional aircraft. In this regime the combination of three manoeuvres

- o post-stall-manoeuvering (PST)
- o drag modulation (DM) and
- o fuselage aiming (FAM)

can certainly affect the propulsion system.

Required angles of attack (α) may be 70° whereas yaw angles (β) could go to 20°.

This requires a thrust deflection device in both the pitch and yaw plane (cone). In order to attain such position within a minimum time, engine acceleration from steady-flight thrust-levels to max. thrust must be obtainable within 3 - 4 seconds. At the same time, the thrust deflection device should have a response speed of 60° per second.

A special inlet design is expected to deliver air to the engine face at a similar quality as in our present aircraft but within the extended envelope (α,β) of a new aircraft. Further wind tunnel tests are in progress. It is, however, very desirable to improve on the definition of distortion at the interface and to include all handling-relevant parameters including swirl [1].

It is also desirable to include a sub-routine to the customer performance decks allowing computation of surge margines using combined effects of power/bleed off-take, distortion/swirl and other installation influences. This would allow an early qualitative assessment of problem areas in the flight envelope.

Drag modulation can be enhanced by providing a very low installed idle thrust, since no moments will result due to thrust changes.

Installed idle thrust should become negative at around M = 0.4 at SL and Ma = 0.8 at H = 11 km.
This, coupled with a fast response under installed conditions is expected to affect engine handling capabilities.

The high thrust to weight ratio coupled with increased attitude control devices will allow the aircraft to be flown steady state at attitudes and time durations in excess of those specified in MIL-E-5007 D. Whether this will be required tactically is subject to further combat simulations. However, these extreme attitudes will more affect engine lubrication rather than engine handling.

<u>*Medium Range Combat*</u>

This combat arena is well separated from the short range area. Although high speed dynamic manoeuvres at moderate load factors (4g) are required for improved combat effectiveness, handling of the engine within the required thrust range should be less critical assuming a well designed intake with minimum swirl/distortion. The power and bleed off-take capabilities are usually adequate in this area unless altitude requirements are increased (zoom envelope).

Possible Solutions

The previous summary of problem areas can definitely not be complete. Experience has shown that many problems surface only when the power plant is operated under real, aircraft-installed, conditions.

One aspect that will help to solve future handling problems is expected with the advent of full-authority digital propulsion control systems. If these controls can be made to use some relevant aircraft parameters for working line adjustments (mode control), handling problems could be minimized.

As a first step it is suggested to use an angle of attack (α) and a power off-take (torque meter) signal for this purpose.

Both should be readily available and combined with a throttle speed signal they should enable surge free engine handling even under extreme conditions. Side slip (β) may also contribute to this effect.

It is hoped that digital controls will allow an easy exchange of performance for handling margin whenever the flight condition requires it.

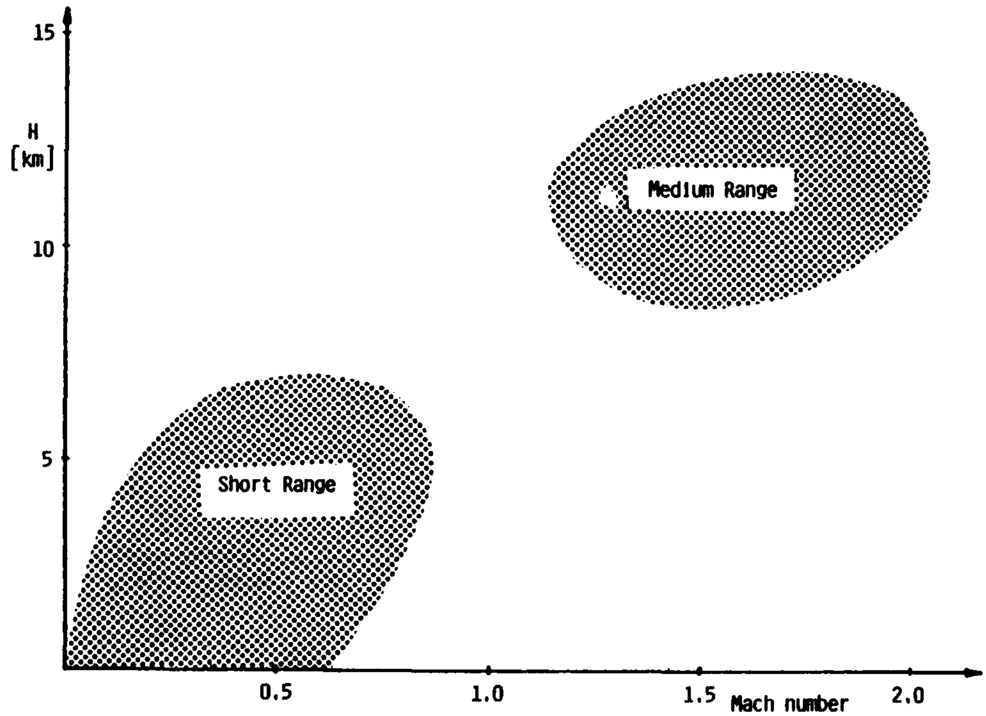


Fig. 16 ENVISAGED COMBAT ARENAS

9. CONCLUDING REMARKS

As described in the paper, development of the RB 199 in the Tornado aircraft with regard to engine handling involved from early, fairly common, problems through intensive analyses and trials to implemented solutions to the Forces today a well balanced, total system having excellent capabilities within the operational envelope.
It was demonstrated that by the development and introduction of the cowl fences an effective and low cost solution was found to a seemingly complex problem.
From this experience the following thoughts are offered for consideration:

Installation in an aircraft and inflight operation of an engine (handling) causes operating lines and surge lines to shift and surge margins to reduce. In addition, deterioriation and variations in built standard will further aggrevate the situation. The cumulative effect of all these shifts is indicated in a compressor chart (Fig. 17).

It should be noted that engine handling and power off-take capability as demonstrated in an high altitude test facility (ATF) and as demonstrated in the aircraft, differ. These differences occur because:

* Distortion patterns simulated in the ATF are probably not representative of the full scale aircraft.

* Intake swirl is not simulated in the ATF and known to have a large effect on engines without inlet guide vanes.

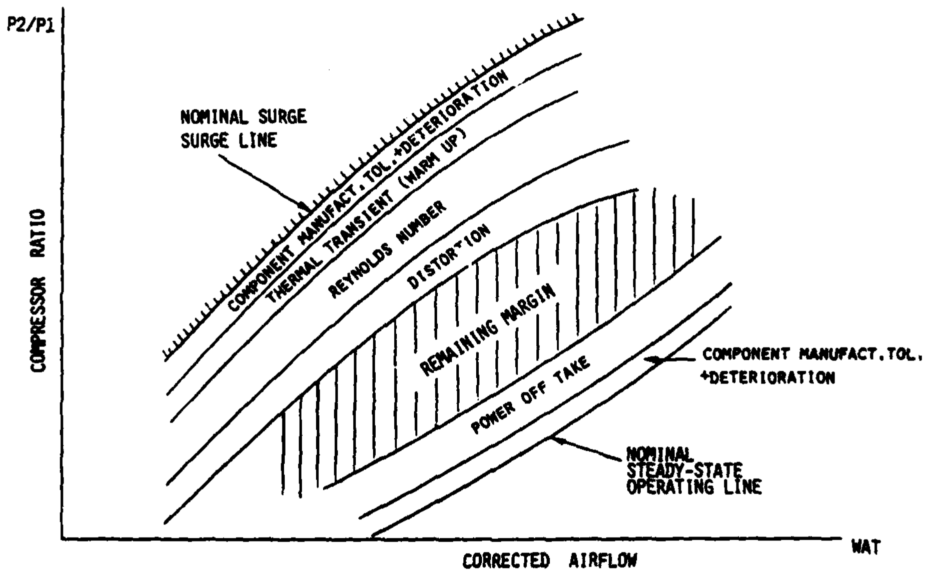
* It is difficult to simulate transient power off-take loads in the ATF.

Of great significance is the airframe/engine compatibility problem - a "handed" problem - as was experienced in the early unmodified Tornado. The question is, whether the dynamic total pressure distortion is the prime compatibility parameter especially for engines without inlet guide vanes. In the case of the Tornado engine, the existing swirl is of equal importance as the pressure distortion coefficient DC 60.

Projected engines or the continued development of existing engines for higher thrust will place even higher demands on the materials used (hot section). This in turn will aggrevate the above listed problems.
What can be done?: More sophisticated model testing must be developed, in order to uncover, at the earliest possible stage in development, any possible problem areas.

It is recommended that future digital engine control takes additional aircraft parameters (as angle of attack, angle of side slip (power off-take), a.s.o) into consideration in order to utilize fully the available engine handling capability.

FIG. 17 SURGE MARGIN DEGRADATION



10. REFERENCES

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OPERATIONAL ENGINE USAGE

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SUMMARY

The flight data recording system LEADS 200 was introduced into a F104G wing of the German Air Force for aircraft and engine maintenance purposes some years ago. The LEADS 200 airborne system components and ground-based central flight data processor are described in short below. The main evaluation routines for the engine data used by the maintenance personnel refer to the engine condition monitoring and fault diagnosis. Only for test purposes an engine cycle counting program, which registers the variations of selected engine parameters of each flight, has been added in order to obtain information on the operational engine usage. The records of these engine cycle counts have been evaluated for 1450 flights with 12 different aircrafts and set into relation to the LCF damage of the engine transients. Some results of this analysis are presented below and the impacts of the aircraft configuration of the mission type and of the pilot's handling upon the engine usage are discussed herein.

1. INTRODUCTION

The raise of thrust to weight ratios for modern military engines has resulted in a life limitation of the main rotating components due to low cycle fatigue (LCF). The engine manufacturer establishes the total life endurance of critical parts by a number of admissible engine cycles based on the expected flight mission profile.

The user of the engine will have task to record the engine cycles occurring during the missions in the operational service and to calculate the equivalent life consumption. Since these mission types may be quite different from the baseline flight mission an inflight engine usage monitoring system is essential in order to control the replacement of life limited parts.

An airborne flight data recording system (FDRS) used by some air fleets for engine diagnostic purposes has turned out as a suitable tool to evaluate the recorded engine parameters with regard to cycle excursions during the flight operation. This paper presents some results of a trial on engine cycle counting carried out by means of the FDRS LEADS 200 installed in a F104G fighter-bomber-wing of the German Air Force.

The aim of this study was to get an information on the spread of engine cycles obtained during aircraft operation in a wing service under real conditions and to analyze the influence parameters, which might be responsible for non-uniform engine usage.

2. SYSTEM DESCRIPTION OF THE FDRS LEADS 200

In the years 1974 to 1976 a total of 50 aircrafts of a GAF F104G fighter-bomber-wing was equipped with the FDRS LEADS 200. This system was a joint development by Dornier System GmbH., Friedrichshafen, Germany, and Leigh Instruments Ltd., Canada and was designed to support maintenance, aircraft, engine and subsystems fault diagnosis and crash evaluation. LEADS 200 was subjected to a trial in 1976/77 and has been in operational service since then.

2.1 SYSTEM CONFIGURATION

The FDRS LEADS 200 consists of an airborne recording system and of a ground-based central processing unit for the flight data evaluation as illustrated in Fig. 1.

Airborne subsystems are
- sensors,
- data acquisition unit,
- voice warning for the pilot,
- maintenance warning panel,
- crash tape recorder and
- maintenance cassette recorder.

The data picked up during the flight can be divided into four main parameter groups: airframe, engine, flight and electronic subsystem parameters. A total of 61 different parameters of digital, analog and synchro type and additionally an analog calibration signal and a synchronization are recorded. The data acquisition unit has a system sample rate capacity of 64 channels per second.

By the voice warning the pilot receives informations on critical aircraft system failures occurring during the flight, which may be fatal to flight safety. Critical engine warnings are for instance: exceedances of maximum RPM and turbine exhaust gas temperature (EGT) or an oil pressure drop.

Limit exceedances related to critical conditions will be additionally recorded by flags on the maintenance warning panel, which comprises 21 different displays. By the indication of these flags the flight line maintenance person receive a quick information on the condition of the aircraft and its essential subsys .

The inflight data storage is realised by magnetic tape recording on the maintenance cassette recorder and the crash recorder. The capacity of the maintenance cassette amounts to 3 hours.

The flight data are processed in the ground-based central computer station. After each flight the maintenance cassette is taken off from the aircraft and delivered to one of the flight line cassette reader terminals. Within a few minutes the maintenance personnel in the debriefing station receive a quick diagnosis of the aircraft in the form of a so-called Quick-Look output.

The main activities in the central data processing unit during the daily flight operation are:
- data check (parit check, bit shift, synchronisation loss),
- quick diagnosis routines (Quick-Look),
- data compression and
- data storage on magnetic tapes.

More detailed descriptions are given in [1] and [2].

2.2 USER SOFTWARE

Besides the above mentioned Quick-Look program the LEADS 200 user software comprises the following evaluation routines:
- detailed data evaluation with parameter plots and outputs in tabular form,
- engine parameter trend analysis,
- counting of g-loads and
- engine cycle counting.

The routine for the detailed data evaluation is only used on request for failure and accident analysis purposes and can be handled during the flight operation time, whereas the long-term programs are generally processed during the night. The following outputs can be obtained by the detailed evaluation routines:
- output of single flight data frames with scaled and unscaled values,
- output of selected parameters in time sequence,
- output of the on-off-history and
- analog plots of selected parameters.

Unlike the Quick-Look routine the engine parameter trend analysis is characterised by a larger observation time scale. The aim of an engine parameter trend analysis is the recording of the performance deterioration and the early detection of failures by means of observed anomalies in the trend curve of characteristic parameters.

The basis for the LEADS 200 engine parameter trend analysis is the comparison between the actual and the reference values of the engine parameters for steady state operating points. The reference values are obtained from the datas of the first 5 flights following an engine change. Further detailed information on the LEADS 200 trend analysis program is presented in [3] and [4].

The counting of g-loads is based on the records of acceleration measurements in some selected locations of the cell structure. The g-load counts are used for airframe fatigue life prediction and are processed outside the wing base.

The engine cycle counting routine was added to the LEADS 200 software in 1978, in order to analyze the engine usage in the operational service of a wing. This program is only used for test purposes and will be explained later.

2.3 ENGINE PARAMETERS

The F104G aircraft is equipped with the modified GE J79 engine known as J79-MTU-J1K. By LEADS 200 a total of 19 parameters are available for the engine condition monitoring. These parameters with specifications of the signal type, sample rate and channels used are listed in Fig. 2. In the engine cycle counting program only the RPM and the exhaust gas temperature (EGT) variations are considered in the evaluation procedure.

3. ENGINE CYCLE COUNTING PROCEDURE

The engine transients are represented by the variations of the thermodynamic parameters. The exhaust gas temperature (EGT) and the RPM were considered to be the most significant parameters for the evaluation of engine cycle counts. The plot in Fig. 3 illustrates the time-dependent changes of the EGT and the RPM occurring during a typical military mission. The curves 3 and 4 show the corresponding variations of the flight level (ALT F) and the air speed (IAS). The flight mission had a duration of about 1 hour and 20 minutes. This analog plot was produced by the detailed evaluation program of the LEADS 200 user software. It can be clearly seen from these plots that the engine transients in a real mission under operational conditions is characterised by a sequence of smaller and greater changes of the engine parameters.

The task of an engine cycle counting procedure will be to record and to classify the variations of the significant parameters and then in a further step to relate these cycles to those of the base-line flight mission and to calculate the equivalent damage with regard to low cycle fatigue.

By the LEADS 200 program system the variations of the EGT and the RPM are evaluated for each flight or engine run including the test runs on the ground and will be listed in the daily output of the total flight program. Fig. 4 shows an example of the daily output with the engine cycle counts (in the frame of dotted lines) and the g-load counts for flight No. 49 of the aircraft No. 2620 on February 25, 1980. The chosen classes for the counting procedure are printed in form of a matrix, as can be seen from the listing. Three thresholds 300 C, 550 C and 629 C were chosen for the counting of the EGT-changes. If one of these thresholds is exceeded, an event count is recorded. The counting procedure starts every time from the first value of a column and passes to the following line values until the last counted threshold is reached. For example the number 13 in the column 550 and in the line 629 indicates, that 13 EGT-changes have happened from a value below 550 C up to a value over 629 C. For the RPM counting the following 4 thresholds were selected: 60%, 85%, 95% and 99.5%.

The parameter changes are counted in form of semi-cycles, as can be seen from the schematic graph for the case of the RPM counting in Fig. 5: As the signal passes the defined thresholds or classes a marker will be set and if the signal falls below the reset level, an event will be counted. The width of the reset level for the RPM is 1,5% and for the EGT 12 C and is constant for all classes.

The relation between the selected thresholds and the different engine working ranges may be drawn from the crossplot of EGT versus RPM in Fig. 6. This curve represents the steady state line of engine operation for a typical flight mission and was established from the weighted mean values of RPM and EGT flight datas. Plots of this type are produced by the LEADS 200 trend analysis software. From this graph it can be seen that changes of the EGT between the thresholds 550 C and 629 C as discussed in the above mentioned example correspond to engine transients from cruise to military power settlings.

4. RESULTS OF THE ENGINE USAGE ANALYSIS

By adding the engine cycle counting program to the LEADS 200 software package there was no intention to use this routine in the operational service or even to trace the fatigue life consumption of the J79-MTU-J1K engine. The only purpose of this program, which had an experimental character, was to obtain some experience in developing methods for in-flight engine cycle counting and to get a better understanding of the following problems related to operational engine usage:
- spread of engine transients under actual conditions,
- impacts of aircraft configuration, mission type and pilot's handling on the engine usage and
- order of magnitude of the mean life consumption in relation to engine run time and engine flight hours.

The analysis presented here refers to an evaluation of engine cycle counting records covering a total of 1450 flights with 12 different aircrafts during a flight period of about 15 months in the years 1979 to 1981.

4.1 MIXED MISSIONS

In the normal flight operation of a wing the sequence of missions flown by one aircraft is more or less uncontrolled. Therefore the engine will be subjected to load cycles which may differ considerably according to the mission types.

Fig. 7 shows a plot of the number of events for EGT-changes from 550 C to 629 C (E 550→629) versus the engine run time (ERT) for the 12 aircrafts or engines respectively which are numbered from A to M. The event E 550→629 characterises the throttle settling from cruise to military power as was already explained above. This plot is an illustration of the wide spread of engine cycles occurring in the operational service and of a quite different usage of the engines. The histogram of the E 550→629/ERT ratio in Fig. 8 proves the large variability experienced. The mean value for the E 550→629/ERT ratio was found to be 9.16 events/hour. In some cases more than 40 events/hour were obtained.

The histogram of the E 300→629/ERT ratio in Fig. 9 shows a more pronounced frequency distribution with a mean of 3.06 events/hour.

The mean values for the main EGT- and RPM-changes in form of event/ERT ratios obtained by the mixed mission analysis are listed in Table I.

When comparing the corresponding column values it can be noticed that there exists a considerable spread both of EGT-event/ERT ratios and of RPM-event/ERT ratios between the observed aircrafts.

It has to be pointed out that the cycle counting by recording the EGT- or RPM-changes is only a rough measure with regard to the low cycle fatigue damage of the engine. Therefore

Table I

Average EGT- and RPM-cycles (mixed missions)

EGT

Aircraft	E300→550/ERT [1/h]	E300→629/ERT [1/h]	E550→629/ERT [1/h]
A	1.94	3.02	9.45
B	2.22	3.73	9.79
C	2.32	2.75	10.39
D	2.12	2.66	9.16
E	3.55	4.52	11.89
F	2.88	3.05	7.62
G	2.12	3.35	9.07
H	1.66	2.79	7.38
I	2.84	2.62	7.32
K	2.10	2.77	10.49
L	2.60	3.01	8.43
M	2.18	2.27	8.45

RPM

Aircraft	E60→99.5/ERT [1/h]	E85→95/ERT [1/h]	E85→99.5/ERT [1/h]	E95→99.5/ERT [1/h]
A	0.82	0.88	2.82	7.63
B	0.94	1.33	2.89	7.95
C	1.27	1.24	2.47	7.56
D	1.00	1.27	2.22	6.43
E	1.34	1.32	4.01	9.17
F	1.10	1.47	3.12	6.61
G	1.00	1.18	2.72	7.44
H	1.07	0.90	2.53	5.43
I	0.70	1.68	2.28	5.87
K	0.81	1.56	2.13	7.87
L	0.81	1.03	3.26	6.35
M	0.93	1.28	3.58	6.13

a stress analysis for critical engine parts was carried out in order to weight the recorded cycle counts with a fatigue damage factor. The bolt hole near the outer rim of the first turbine stage disc turned out to be most critical to LCF. Fig. 10 shows a graph of the relationship between the calculated damage factors representing the relative life consumption per cycle and the RPM cycle counts.

Based on these damage factors an estimation of the mean fatigue damage related to 1 hour of ERT was made for each aircraft. Table II contains a separate listing of the total number of flights, the total flight time, the total engine run time and the values of the mean damage rate per 1 hour of ERT for the 12 aircrafts.

Table II

Results of mixed missions analysis

Aircraft	Number of flights	Total flight time [h]	Total ERT [h]	Mean damage rate/ERT [1/h]
A	142	121.1	186.8	1.63
B	145	117.9	182.2	1.84
C	140	126.7	180.0	2.14
D	130	111.9	169.0	1.64
E	115	76.8	120.0	2.27
F	122	114.8	165.9	1.73
G	120	105.1	115.1	1.85
H	115	105.5	155.2	1.56
I	122	125.8	182.9	1.53
K	105	94.2	137.8	1.60
L	98	78.7	121.9	1.68
M	96	77.2	119.2	1.71

The mean damage rates obtained by the fatigue analysis show the same magnitude of data spread like the EGT- or RPM-changes listed in Table I. But it should be noted, that the damage factors used in this analysis are limited to the case of the turbine disc fatigue and may not be representative for the low cycle fatigue of other critical engine parts.

4.2 AIRCRAFT CONFIGURATION

In the majority of the evaluated missions only 3 different types of outer tank loading configurations were used:
- configuration A1 with 4 outer tanks,
- configuration C2, C3 only with tip tanks and
- configuration D3 without external loads (clean).

Since long range missions require an aircraft loading configuration A1 and a fighter training must be flown without external loads, the aircraft configuration characterises the main type of missions. Therefore the configuration analysis may give some indications on the causes which are responsible for the spread of engine cycles.

When the aircraft E with the highest mean damage rate and the aircraft I with the lowest mean damage rate are compared with regard to the tank configurations the following percentages related to the total flight number will be obtained:

Aircraft E: 12.2% A1; 54.8% C2,C3; 31.3% D3
Aircraft I: 36.1% A1; 59.8% C2,C3; 1.6% D3.

From these values it can be concluded that a flight program with a high portion of missions of the D3 type configuration leads to greater fatigue damage rates than in the case of a flight program with a high portion of long range missions with the configuration A1. This statement is confirmed by the results of the configuration analysis including the total of 1450 flights. Fig. 11 to Fig. 13 show the histograms of the relative low cycle fatigue damage related to 1 hour of ERT for the 3 main aircraft tank configurations in a separate evaluation. The following average values for the damage rate/ERT ratio were obtained:
- configuration A1: 1.55 1/h,
- configuration C2,C3: 2.23 1/h and
- configuration D3: 3.03 1/h.

Thus it turned out that the fatigue life consumption of the missions with the configuration D3 was nearly twice the rate of those with the configuration A1.

4.3 MISSION ANALYSIS

The mission analysis covered a total of 75 flights of only one aircraft which was considered to be representative. The evaluated missions were divided into 5 types:
- mission 1: long range and cross country flights,
- mission 2: flights with air to ground gun firing,
- mission 3: medium range flights,
- mission 4: air combat training and
- mission 5: test flights.

From the recorded EGT- and RPM-cycles a mean fatigue damage rate was calculated separately for the flights of each mission type in the same manner as mentioned before. But in this case the mean damage rate was related to the flight time in order to obtain a better measure of the spread in engine usage with regard to the mission type.

The graph in Fig. 14 shows for the flights of each mission type the mean damage rate related to 1 flight hour and the corresponding standard deviation. The results demonstrate that under operational conditions flights even of the same mission type have a great variation of engine load cycles which in consequence yields to a quite different engine usage.

4.4 PILOT'S HANDLING

On principle the LEADS 200 user software would allow to carry out an evaluation with regard to the individual pilot's handling because the code number of the pilot is recorded for every flight. But due to the fact that the missions in the operational service don't resemble each other it is impossible to quantify any differences in the engine handling between the pilots. There is one exception: the formation flight of two or more aircrafts. Some of these formation flights were analyzed. Fig. 15 to Fig. 17 show the analog plots of the characteristic engine and flight parameters versus the flight time for three typical maneuvers which had been recorded during a formation flight of 3 aircrafts. Fig. 15 illustrates the different engine handling by the leader and by the pilots at the No. 2 and at the No. 3 positions during the take off. Fig. 16 shows a comparison of the parameter variations for the 3 aircrafts during an attack maneuver and Fig. 17 during a climb maneuver. By these examples it can be concluded that the engines of the aircrafts at the No. 2 or No. 3 positions will be normally subjected to more cycle variations during the same flight period than the engine of the aircraft at the leader position. Therefore the different pilot's handling within a formation flight may contribute to the spread of cycles observed for the same mission type.

5. CONCLUSION

The results of this study show very clearly that there is a need for the individual engine cycle counting in the operational service of military aircrafts. In order to make the predictions of fatigue life consumption more precise the number of recorded engine parameters must be increased and the cycle counting procedures should be refined. From the experience gained by the FDRS LEADS 200 it can be stated that the implementation of an engine cycle counting procedure into an engine condition monitoring system is a promising solution for the engine usage evaluation.

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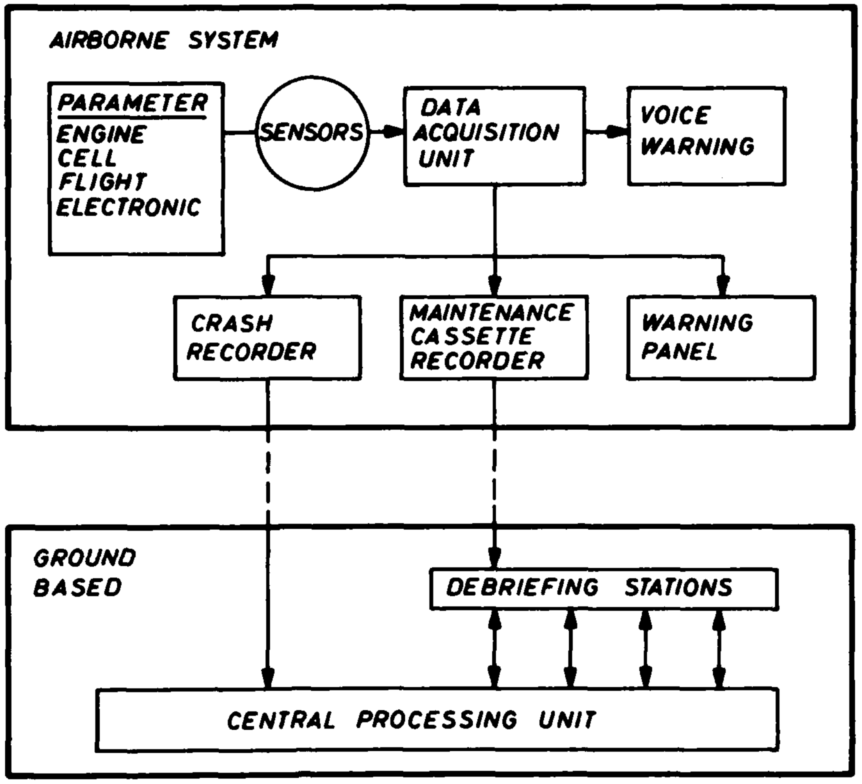

Fig. 1: LEADS 200 system configuration

	PARAMETER	SIGNAL D*	A*	S*	SAMPLE RATE 1/s	CHANNELS 1/s
1	RPM	X			1	12/8
2	NOZZLE POSITION (NOP)			X	1	1
3	OIL PRESSURE			X	1	1
4	MAIN FUEL FLOW			X	1	1
5	THROTTLE POSITION (THRO)		X		1	1
6	EXHAUST GAS TEMPERATURE (EGT)		X		2	2
7	COMPRESSOR DISCHARGE PRESSURE (CDP)		X		1	1
8	TOTAL AIR TEMPERATURE (TAT)		X		1	1
9	INLET GUIDE VANE ANGLE (IGV)		X		1	1
10	EGT-WARNING	X			1	1/8
11	COMPRESSOR INLET TEMPERATUR WARNING	X			1	1/8
12	OIL LEVEL LOW WARNING	X			1	1/8
13	FUEL LOW WARNING	X			1	1/8
14	FUEL BOOST PUMPS FAILED	X			1	1/8
15	FUEL SHUT-OFF VALVE	X			1	1/8
16	RPM-WARNING	X			1	1/8
17	IGNITION NO. 1	X			1	1/8
18	IGNITION NO. 2	X			1	1/8
19	ANTI ICE WARNING	X			1	1/8

*) D = DIGITAL, A = ANALOG, S = SYNC

Fig. 2: Engine parameter list

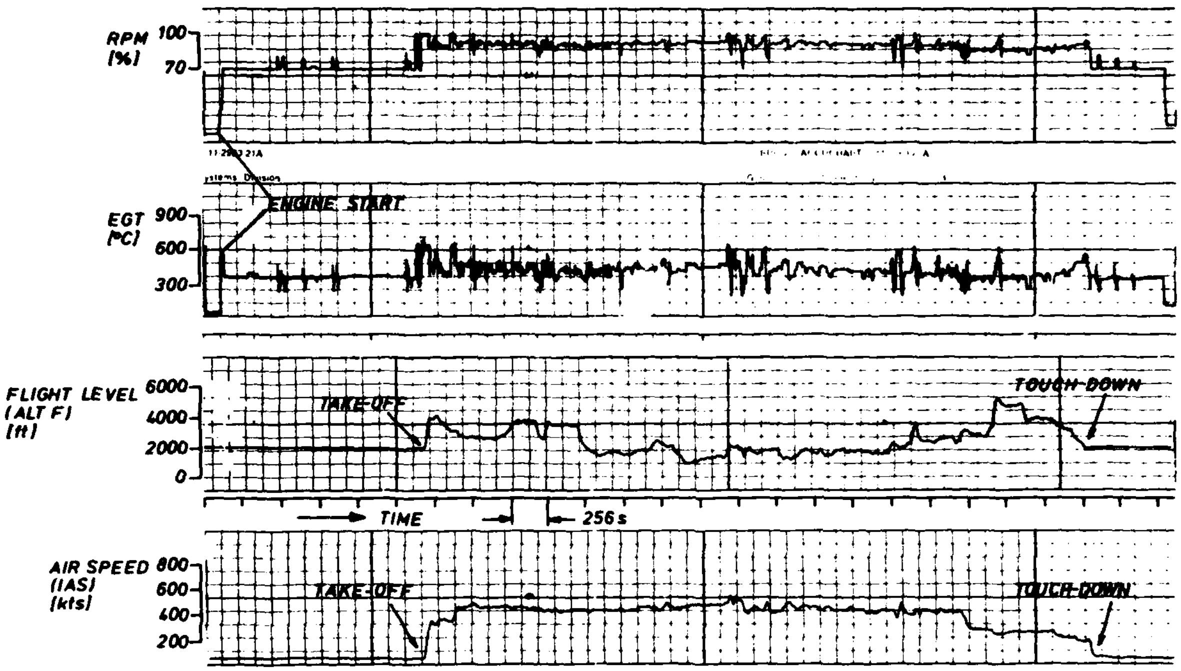

Fig. 3: Record of engine and flight parameters for a military mission

FLUG 49 DES AC 2620 VON 25. 2. 1980 MIT DER KONFIGURATION 630 (KONF: 630 DOSDA: 10056)

CONFIGURATION D3

FLIGHT TIME IN SEC : 2266
FT MIT FLAPS UP : 1064
FT MIT FLAP SONST : 1202

LASTKOLLEKTIVE:	+1,5G	+2,0G	+3,0G	+4,0G	+5,0G	+6,0G	+6,5G	0,0G	-1,5G	-2,5G
FLAPS UP :	22	16	9	2	0	0	0	0	0	0
FLAP SONST :	14	10	1	0	0	0	0	0	0	0
G - SUMMEN :	36	26	10	2	0	0	0	0	0	0

ENGINE RUN TIME IN SEC : 3138
MATRIZEN : SPALTE: VON WERT WIRD : ZEILE: NEUER WERT GEZAEHLT.

EGT - MATRIX : WERTE IN GRAD CELSIUS

	300	550	620	SUMME
300:	7	7	0	14
550:	2	16	18	34
620:	3	15	6	24

RPM - MATRIX : WERTE IN PROZENT

	60	85	95	99,5	SUMME
60 :	1	1	0	0	2
85 :	0	0	4	0	4
95 :	0	1	4	17	22
99,5 :	1	2	14	5	22

ENGINE CYCLE COUNTS

G - MATRIX : WERTE IN G

	+1,5	+2,0	+3,0	+4,0	+5,0	+6,0	+6,5	0,0	-1,5	-2,5
+1,5 :	17	19	0	0	0	0	0	0	0	0
+2,0 :	12	5	0	0	0	0	0	0	0	0
+3,0 :	5	2	1	2	0	0	0	0	0	0
+4,0 :	2	0	0	0	0	0	0	0	0	0
+5,0 :	0	0	0	0	0	0	0	0	0	0
+6,0 :	0	0	0	0	0	0	0	0	0	0
+6,5 :	0	0	0	0	0	0	0	0	0	0
0,0 :	0	0	0	0	0	0	0	0	0	0
-1,5 :	0	0	0	0	0	0	0	0	0	0
-2,5 :	0	0	0	0	0	0	0	0	0	0

Fig. 4: Printout of engine cycle counts

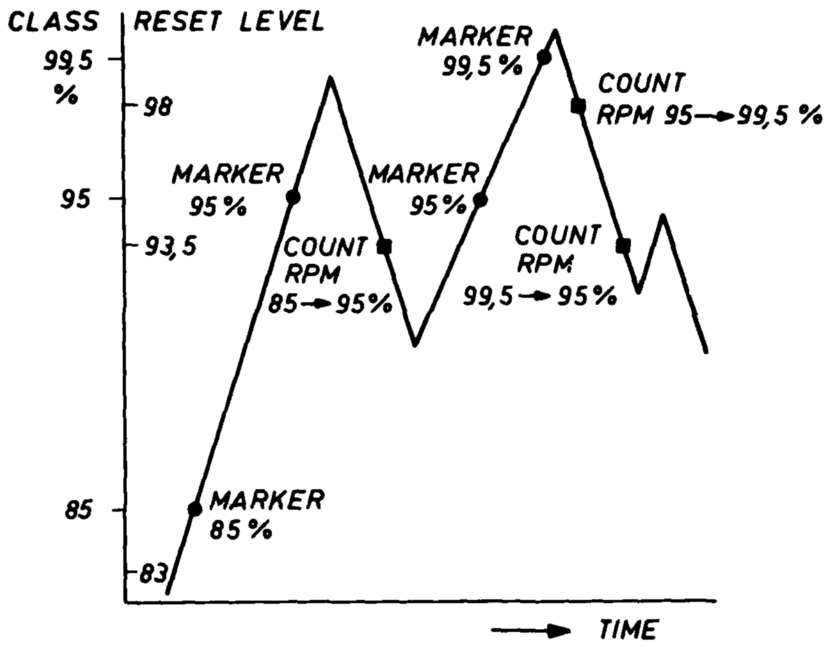

Fig. 5: RPM counting procedure

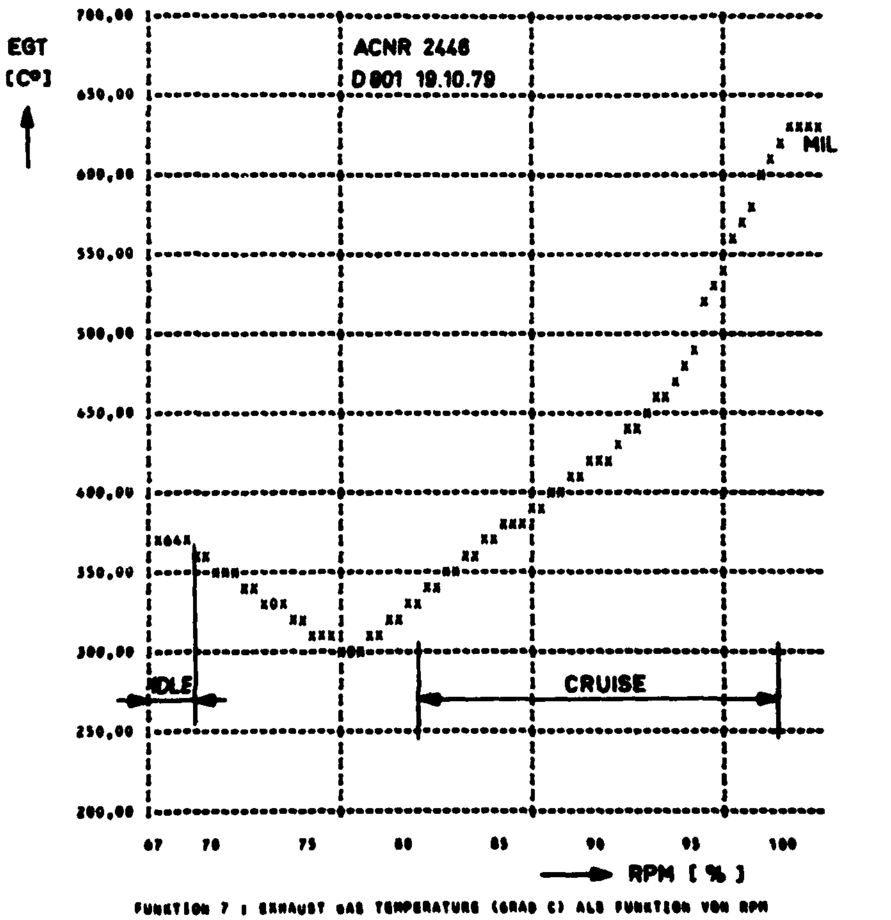

Fig. 6: Crossplot of EGT vs RPM for steady state engine operation

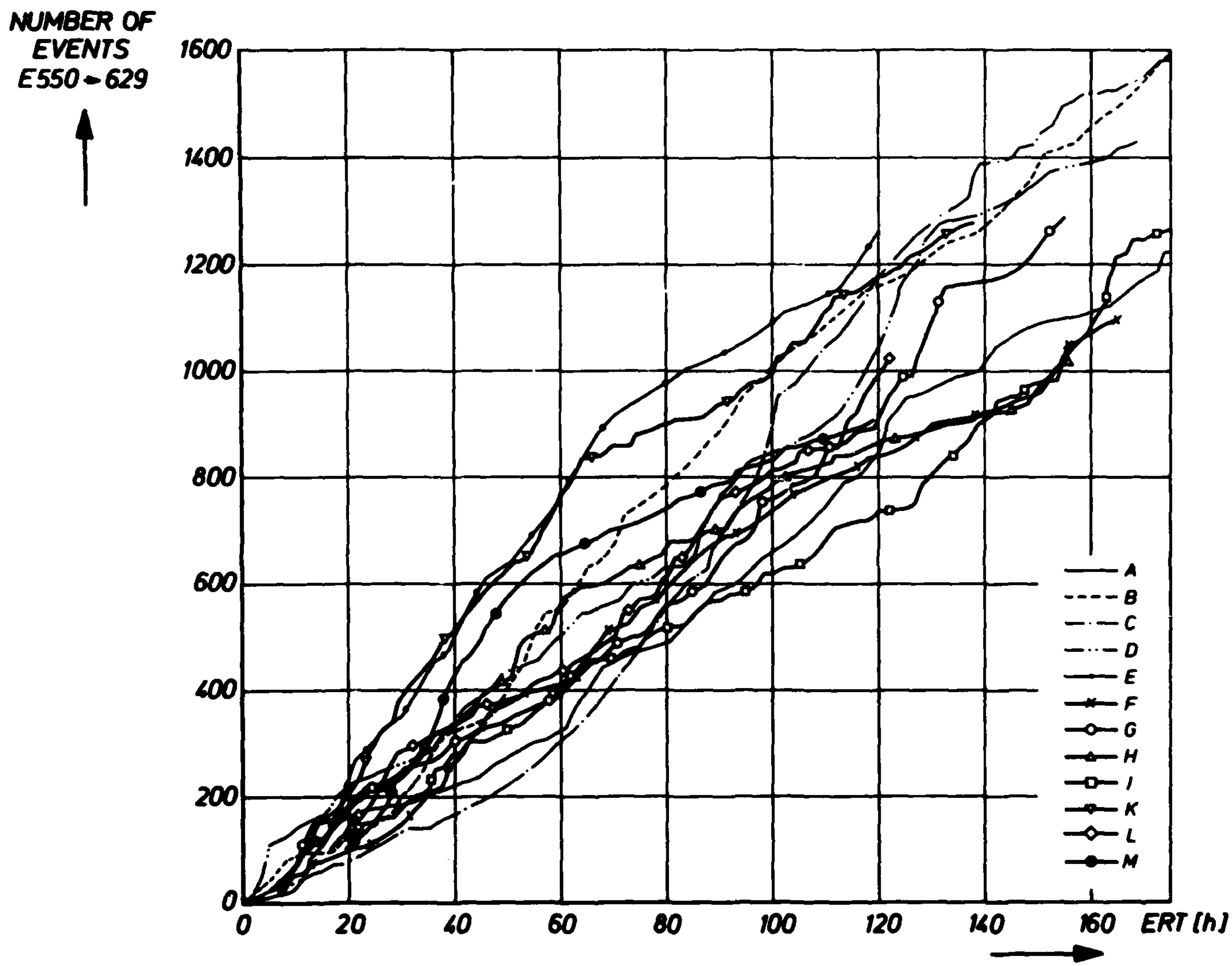

Fig. 7: EGT changes from 550 C to 629 C vs engine run time

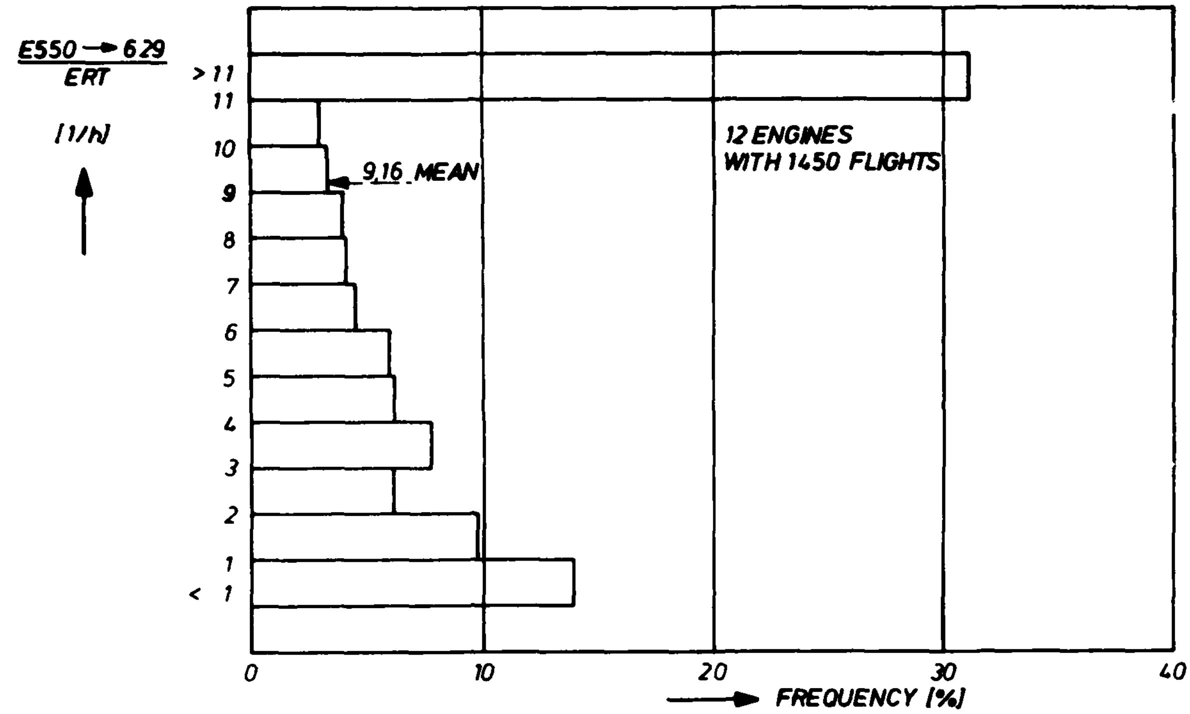

Fig. 8: Histogram of EGT changes from 550C→629C related to ERT

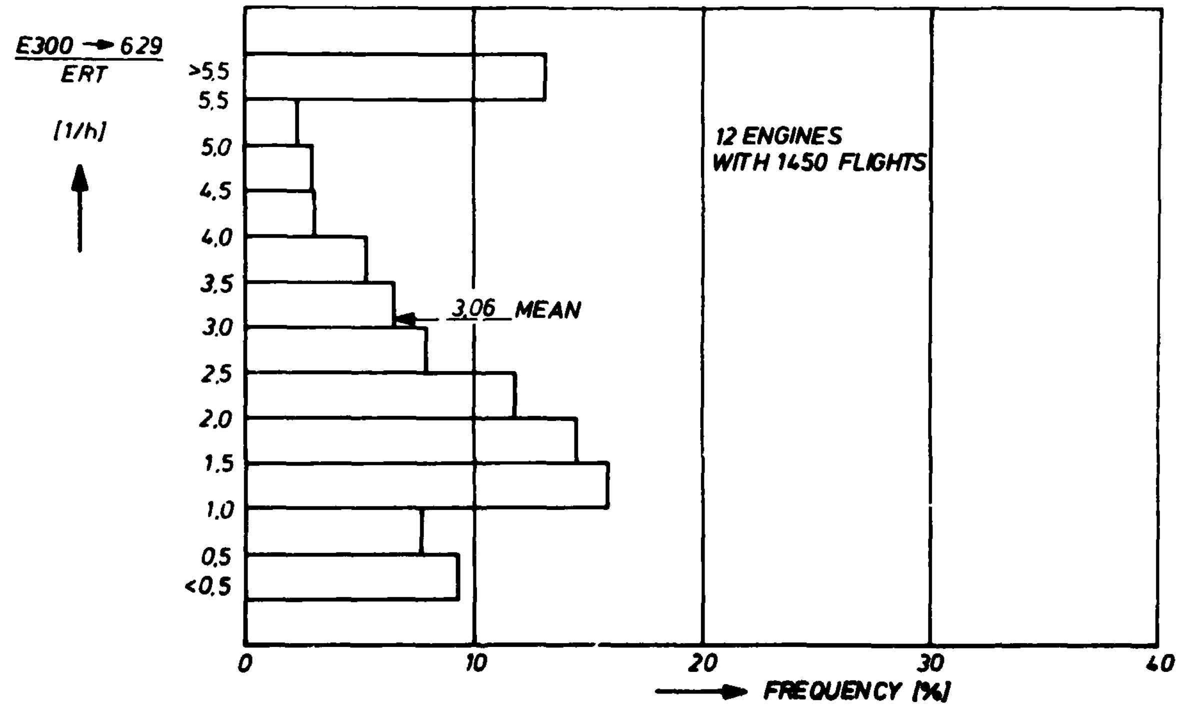

Fig. 9: Histogram of EGT changes from 300C→629C related to ERT

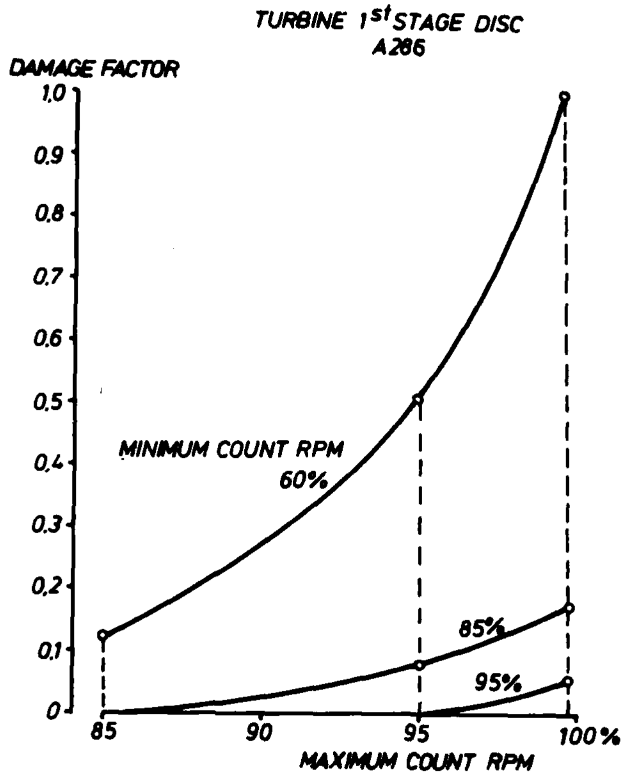

Fig. 10: Relative LCF damage of RPM cycles

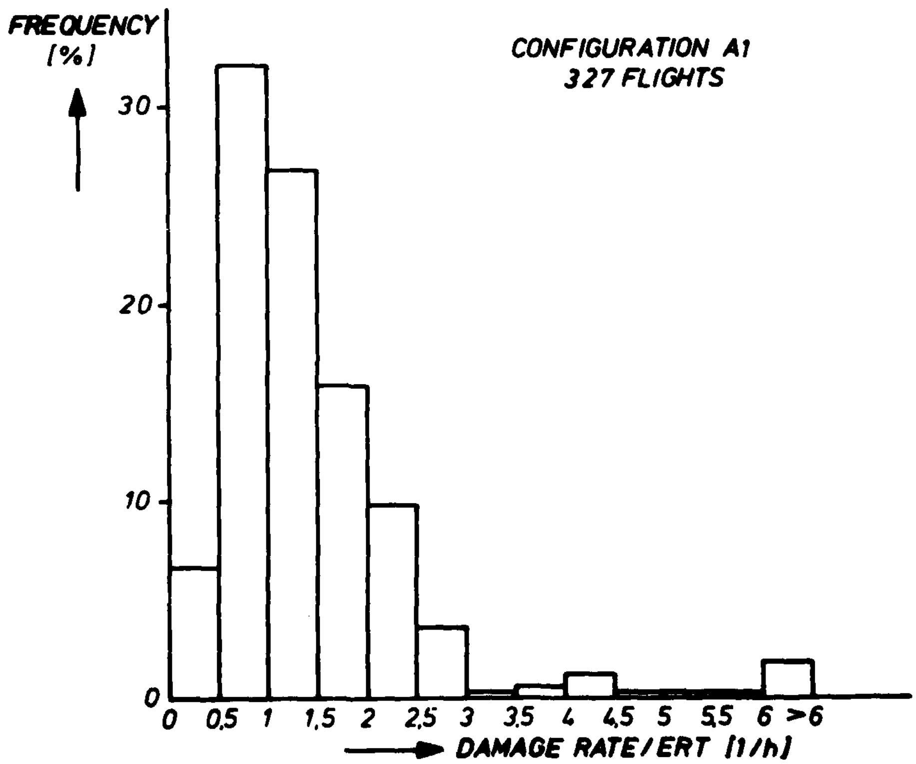

Fig. 11: Histogram of relative LCF damage/ERT for the aircraft configuration A1

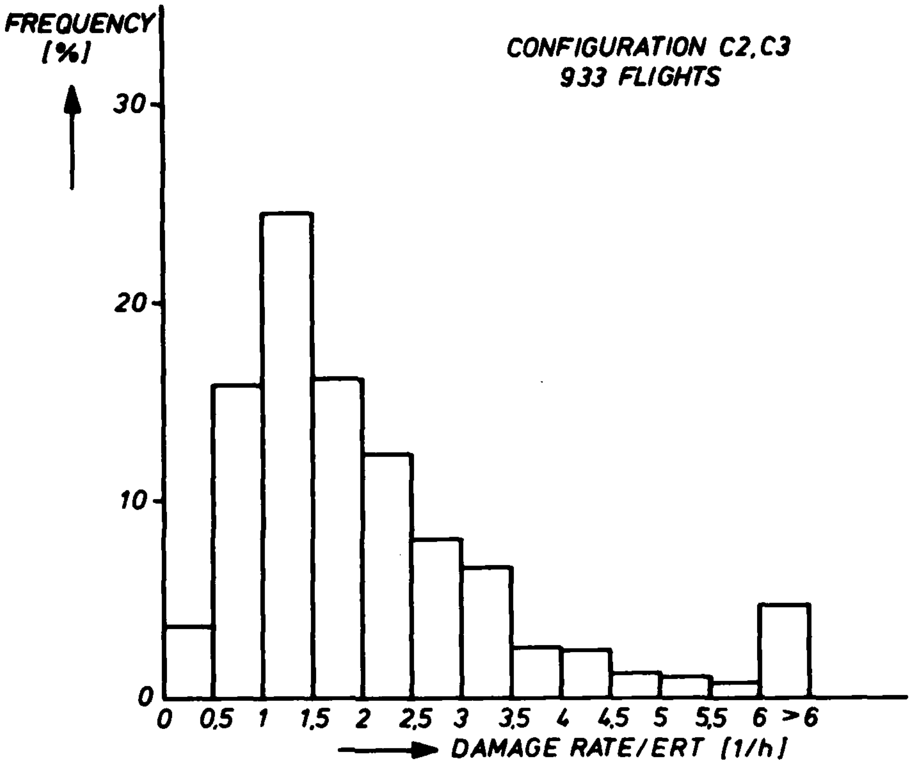

Fig. 12: Histogram of relative LCF damage/ERT for the aircraft configuration C2 and C3

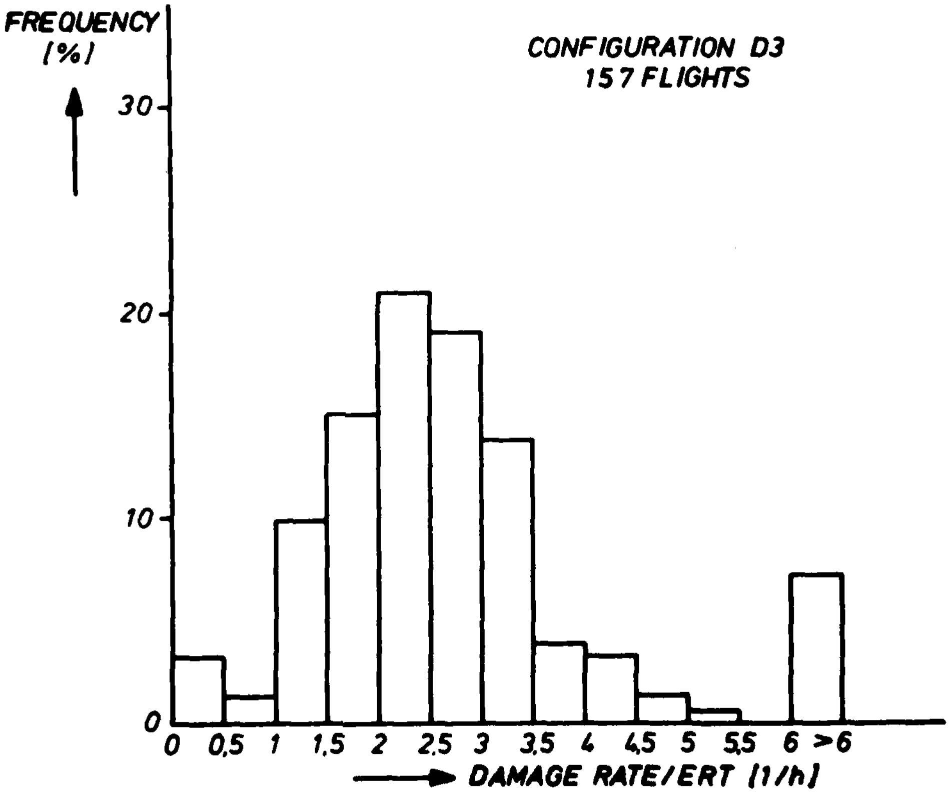

Fig. 13: Histogram of relative LCF damage/ERT for the aircraft configuration D3

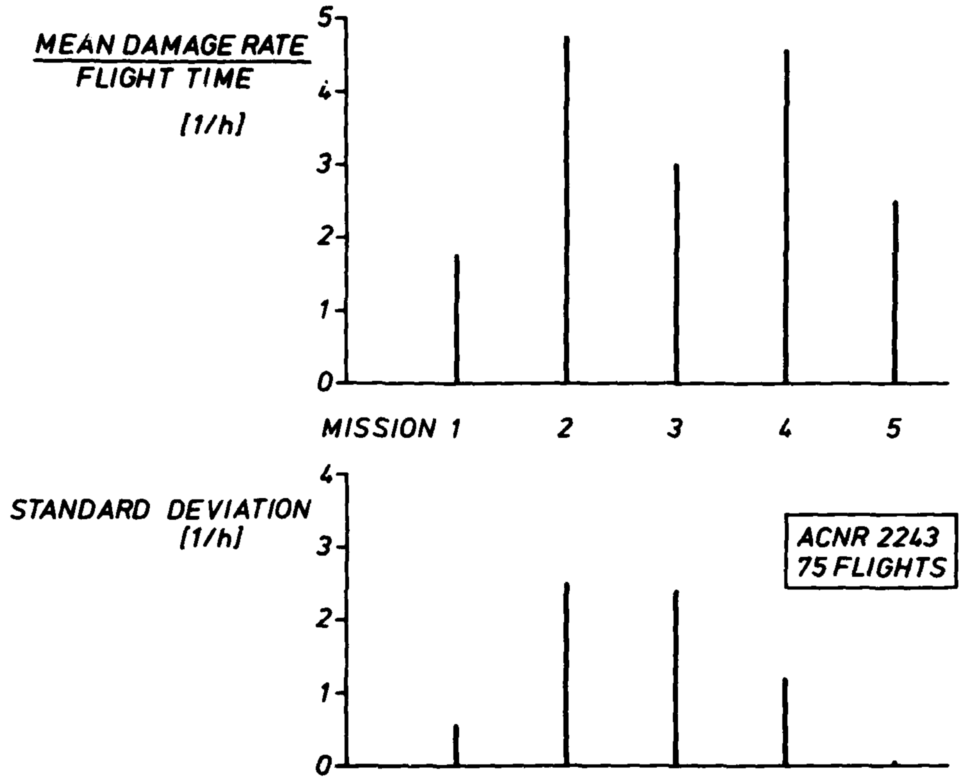

Fig. 14: Mission analysis

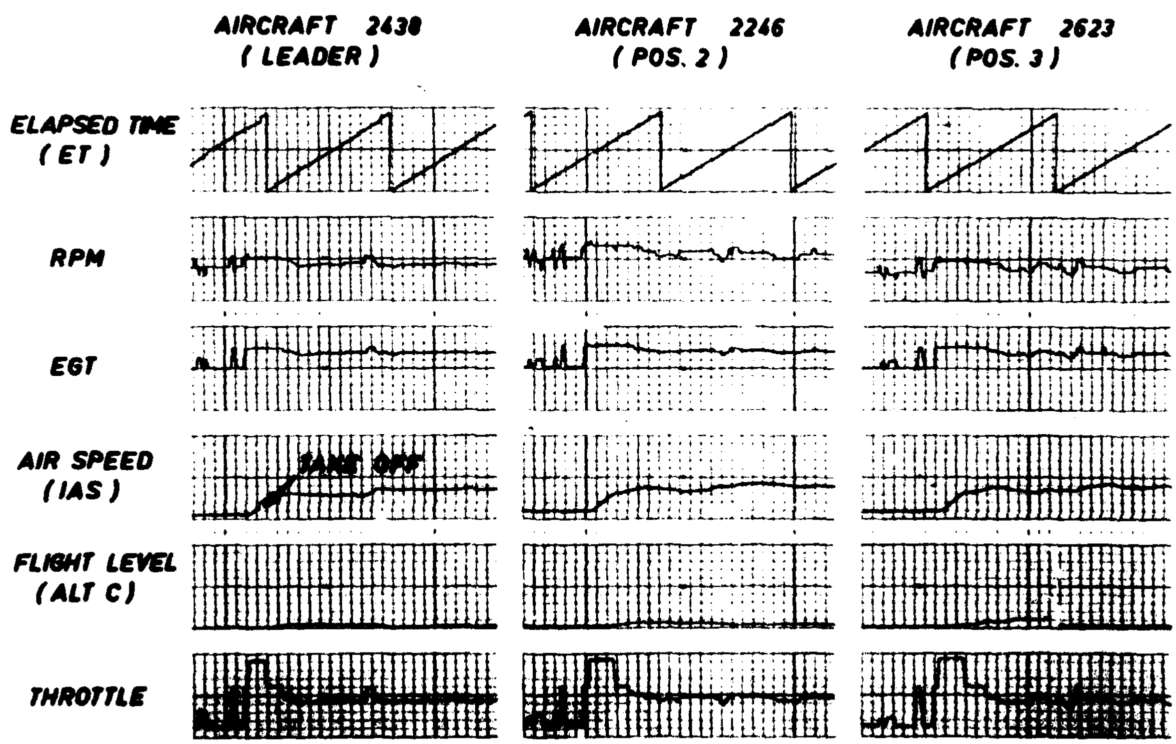

Fig. 15: Formation flight - take off

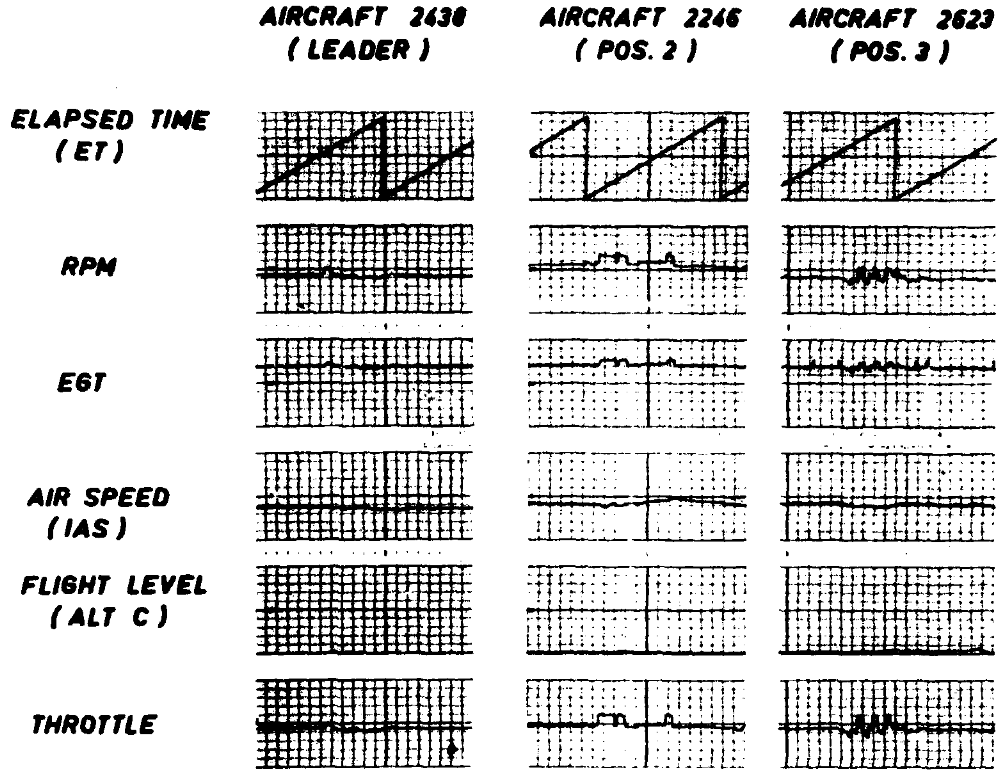

Fig. 16: Formation flight - attack maneuver

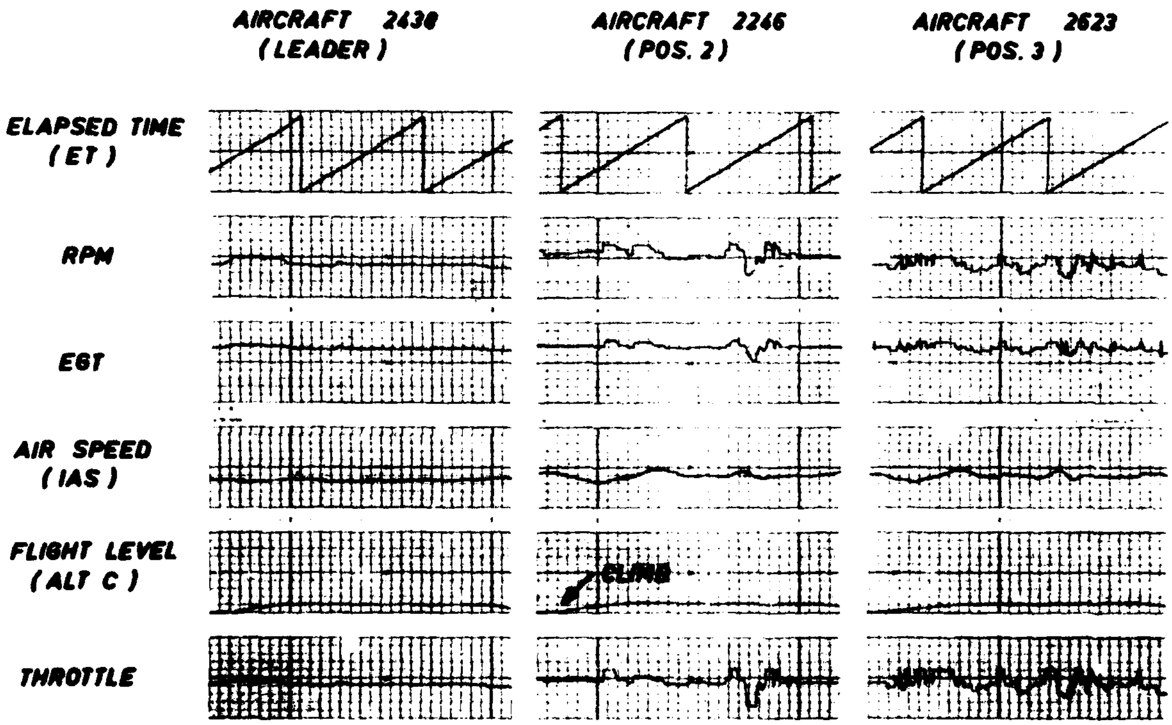

Fig. 17: Formation flight - climb maneuver

DISCUSSION

P.F. Neal, UK

Since the cyclic damage rate in rotating discs depends on the transient CF and thermal stresses, (which arise from temperature differences in rims and bores), how does your Leads 200 system estimate the cyclic damage rate, bearing in mind that excursions of EGT above a given level have a damage effect that depends on the transient temperatures in the disc which vary with position in the engine, noting that at the front of the compressor the principal stresses are dominated whereas at the rear of the compression system they are dominated by thermal stresses?

Author's Reply

It is clear that in the case of the turbine disc the temperature transients govern the cyclic damage. Therefore we used in the stress analysis a correlation between the RPM-changes and the disc temperature variations. But at that time, due to the limited capacity of the computer, we were unable to evaluate temperature-time-gradients in order to take account of different thermal transients in the turbine disc.

J.T. Bakker, Ne.

I noted that mean damage rate over flight time was higher for the air to ground gun firing than it was for the air combat training. I would like to point out that mean damage rate/flight time in the air to ground gun firing case is highly dependent on the weapon control computer on board. In the F-104 the pilot has to meet certain fixed parameters at weapon delivery (like air speed, dive angle etc.) because the aircraft possesses a fixed gun sight. This causes him to make many abrupt throttle transients just short for delivery. In more modern weapon control computers the impact point of weapons is continuously computed regardless of airspeed, dive angle etc. Hence, as the pilot has no need in obtaining fixed parameters during delivery, the number of cycles will be far less for the same amount of delivery passes.

Author's Reply

One reason, which may be responsible for the high damage rate per flight hour obtained for the missions of the No.2 type has already been mentioned in your question. It has to be considered that within the missions analysed no further distinction was made concerning the range distances, the aircraft positions in a formation flight etc. which in the case of the No.2 missions might have contributed to higher values and a wider spread in the damage rates against the case of the air combat training missions.

S. Olympios, US

Is your system used for diagnosis or prognosis (trending)?

Author's Reply

The LEADS 200 software routines 'Quick Look' and Detailed Data Evaluation are operationally used for diagnostic purposes whereas the airframe and engine cycle counting procedures and the engine trend analysis have a prognostic character and have been installed only for test purposes.

S. Olympios

Can your system fault isolate a problem to a particular component? EGT (high) for example could indicate a problem associated with the compressor, turbine or even fuel nozzle.

Author's Reply

In some cases a fault localisation can be made by means of a logic combination of different parameters, for example during the engine run-up check, but generally a component fault isolation is impossible due to the limited number of engine parameters (see Fig.2).

S. Olympios

Your results in the field are interpreted by engineers and at what level of maintenance (flight line or organizational, etc.)?

Author's Reply

The printout of the Quick-Look is interpreted by the flight line and maintenance personnel.

S. Olympios

Do you baseline the engine after every overhaul?

Author's Reply

In this program engine overhaul procedures have not been linked to engine flight data recording by LEADS 200. Therefore the engine baseline parameters needed for diagnostic and prognostic purposes will be established individually by recording the data of first 5 flights following an engine change.

PILOTABILITE DES MOTEURS DE COMBAT

Le point de vue du pilote

par

le Colonel Maurice ROUGEVIN-BAVILLE

Division MIRAGE 2000 de l'Etat-Major de l'Armée de l'Air

(FRANCE)

RESUME Pour permettre au pilote de combat de se consacrer à sa mission, il faut lui faciliter la conduite du moteur : suppression ou simplification des consignes, amélioration de la manette des gaz et des instruments de contrôle, mise au point de dispositifs annexes tels qu'automanette ou calculateur d'optimisation de la consommation.

Ces améliorations passent par une intégration plus poussée du moteur dans l'avion.

Le MIRAGE 2000 propose une solution simple et particulièrement efficace au problème du contrôle de la poussée.

*
* *

Dans un avion de combat, le pilote doit utiliser toutes ses facultés pour accomplir sa mission : découvrir son adversaire ou son objectif au sol, piloter sa trajectoire pour se mettre en position de tir, exécuter le tir à l'instant optimal, rechercher le renseignement, se protéger de la chasse ou des missiles adverses. Les systèmes d'armes deviennent chaque jour plus performants et plus complexes, . 's l'avantage est toujours à celui qui regarde dehors et qui peut consacrer la plus grande partie de son temps de réflexion aux problèmes tactiques. Il importe donc de libérer le pilote de tout souci technique d'utilisation de son système d'armes. C'est là le but principal recherché dans la conception des cabines des avions de combat modernes : pilotage tête haute, mains sur les commandes (concept HOTAS : Hands On The Throttle And Stick) :

"Le pilote regarde dehors et manoeuvre ses commandes en aveugle, toute action doit être possible sans regarder ses instruments".

*
* *

Le moteur n'est pas un équipement comme les autres : son importance est vitale, le pilote en est conscient et se montre toujours très soucieux de son bon fonctionnement. Est-ce une raison pour lui imposer une surveillance qui l'accapare inutilement ?

Or il est curieux de constater que, si les cabines, les commandes de vol et les visualisations ont profondément évolué depuis quelques années, la pilotabilité des moteurs modernes n'a guère changé : les performances sont en amélioration constante, mais les commandes et contrôles du F.100, du F 404, du RB 199 et du M.53 sont peu différents de ceux du J.57 ou de l'ATAR 9C.

Mais au fait, qu'est-ce que la pilotabilité du moteur ?

Pour le motoriste, c'est l'obtention d'un état désiré avec un effort manuel minimum, dans le temps le plus court possible, sans risques inutiles pour la sécurité.

Pour le pilote, la même définition convient, à condition d'ajouter : sans effort de réflexion et sans aucune surveillance.

Est-ce pour pouvoir employer les pilotes moins intelligents ? Pour leur permettre de dormir ? Bien évidemment non, il s'agit seulement de leur permettre de ne penser qu'à une chose : la mission.

Avant d'étudier la pilotabilité du moteur, essayons d'analyser les actions du pilote dans différentes phases de vol :

- au décollage, en montée, en accélération, il s'agit le plus souvent d'avoir la poussée maximale : c'est facile, il suffit d'avancer la manette,

- le problème se complique un peu pour le chef de patrouille qui doit laisser à son équipier une certaine marge de poussée pour lui permettre de suivre, sans toutefois trop dégrader les performances : de combien faut-il reculer la manette ?

- la navigation demande généralement le respect d'une vitesse, d'un horaire ... La méthode la plus classique consiste à afficher un régime de base, à voir ce qu'il donne, à corriger ... Bref, à piloter une vitesse en permanence,

- la tenue de patrouille vient compliquer les choses : il faut analyser très vite sa position et sa vitesse par rapport au chef de patrouille, en déduire une accélération souhaitable, modifier la position de la manette, attendre le résultat, refaire une correction, attendre de nouveau, recommencer, sans s'énerver, bien sûr,

- le ravitaillement en vol présente les mêmes difficultés, généralement aggravées par le fait que l'opération se situe à moyenne ou haute altitude (la réponse moteur est plus molle), que l'avion s'alourdit progressivement, que le panier dans lequel il faut introduire la perche de ravitaillement bouge toujours et qu'il faut faire vite pour laisser la place aux autres avions qui doivent ravitailler. Bien heureux quand ce n'est pas de nuit et dans les cirrus ....

- le combat contre avion est le moment crucial dans lequel le pilote a un objectif : tirer l'adversair[illegible] et deux soucis : surveiller ses arrières etsurveiller le pétrole ; la manette est pratiquement t[illegible] la plus loin possible en avant, avec parfois un retour brutal sur la butée du ralenti lorsque l'utili[illegible] plan vertical n'a pas permis une décélération suffisante, suivi d'un retour aussi brutal sur la b[illegible]ée avant pour reprendre la poursuite.

- le combat contre missile est la phase très brève dans laquelle il peut être nécessaire de diminuer quasi-instantanément, au moins pendant un court instant, la signature infrarouge du moteur,

- l'approche et l'atterrissage enfin, sont des phases dans lesquelles le respect de la pente et de la vitesse conditionnent la précision du point d'impact, donc la longueur de piste nécessaire et l'abaissement des minima météorologiques.

On en déduit que le pilotage du moteur se traduit par quelques mouvements simples de la manette :

- recherche d'une position connue : plein gaz sec, plein gaz PC, ralenti ... ou d'une position très voisine,
- recherche lente de la position qui donnera, au bout d'un certain temps, la vitesse voulue,
- variations rapides et si possible de faible amplitude autour d'une position quelconque,
- passage brutal du plein avant au plein arrière et retour tout aussi brutal au plein avant.

Bien entendu ces mouvements doivent être possibles dans tout le domaine de vol et dans toutes les conditions de vol, en toute sécurité ;

Le résultat doit être rapide, pour couvrir les cas critiques : combat, ravitaillement en vol, remise des gaz ; mais le dosage du mouvement doit être facile, pour libérer l'esprit du pilote.

*
* *

La recherche de la facilité doit porter sur quatre aspects : les consignes, les commandes, les contrôles et les dispositifs annexes.

LES CONSIGNES.

Le mieux est bien sûr de n'avoir aucune consigne : tout paramètre à respecter oblige à regarder un instrument, donc à quitter l'adversaire des yeux, avec toutes les chances de ne plus le retrouver en relevant la tête. Il est donc réellement ital de supprimer la surveillance de paramètres, donc de supprimer les consignes.

Cela n'est pas toujours possible dans tout le domaine ou dans toutes les conditions de vol. Si consignes il doit y avoir, elles doivent être peu nombreuses et simples à connaître : mieux vaut, en pratique, simplifier un domaine aux formes compliquées que surcharger l'esprit du pilote de nombreux chiffres d'altitudes, vitesses, nombre de Mach, régimes et températures à observer simultanément.

Une consigne très pénalisante est celle qui exige de stabiliser un paramètre avant de faire une autre action, par exemple attendre que la T4 soit à 700° pour passer la PC : le pilote doit quitter des yeux le monde extérieur pendant un certain temps ; heureusement, ces consignes ne se trouvent plus sur les avions modernes !

Les consignes doivent également être faciles à respecter. Par exemple, si le passage de la PC correspond à un simple "dur" à la manette, il est très difficile, en surveillant ses mouvements, de passer du plein gaz sec au mini PC : la manette dépassera presque toujours cette position ; il ne saurait être question de demander à un pilote de doser ses efforts en combat : une telle manette est incompatible d'une consigne telle que "pour allumer la PC, stabiliser en PC mini puis augmenter la charge PC".

Les commandes doivent permettre de respecter les consignes : une position caractéristique de manette doit se trouver facilement, sans avoir à regarder quelque part dans la cabine.

Il faut enfin savoir qu'un bon pilote cherche toujours à respecter les consignes. Cela a parfois des inconvénients : si un pilote s'aperçoit qu'il vient de faire une manoeuvre interdite (par exemple : couper la PC) il va, instinctivement, effacer son erreur (dans l'exemple : remettre la PC). Peut-on lui en vouloir ? Non, il vaut certainement mieux essayer de le couvrir contre des fausses manoeuvres instinctives ...

Dans la pratique, si une consigne doit subsister, la plus facile à respecter est "ne plus rien toucher". Peut-être, un jour, aurons-nous des moteurs assez intelligents pour suivre au plus près n'importe quelle demande du pilote sans jamais dépasser leurs possibilités.... permettant ainsi de supprimer les dernières consignes.

*

* *

LES COMMANDES

Les commandes actuelles sont toujours constituées d'une manette par moteur. Au fil des années, ces manettes ont été progressivement encombrées de nombreuses autres commandes : aérofreins, radio, volets, radar, contre-mesures ... ; cela prouve bien que la commande du moteur, au même titre que le manche, est une commande d'usage quasi-permanent et qu'il n'est pas possible de la lâcher.

Mais ces manettes ont, dans le fond, bien peu évolué : elles sont encombrantes ; leur débattement est très grand,pour assurer une bonne précision d'affichage du régime. mais du coup les commandes "annexes" ne tombent plus bien sous les doigts pour les positions extrêmes ; les efforts sont généralement importants, ce qui devient gênant en patrouille serrée ou en ravitaillement en vol.

En fait ces manettes sont conçues pour permettre l'affichage précis d'un régime moteur, alors que le pilote se moque éperdument du régime : c'est la poussée qui l'intéresse, pourquoi ne pas lui donner une poussée proportionnelle à l'angle de la manette ?

On peut aussi aller plus loin. Nous avons vu que le pilote cherchait, la plupart du temps, soit à aller sur une butée (ralenti, plein gaz sec, pleine charge PC), soit à trouver rapidement une position intermédiaire, soit à faire des variations rapides mais précises autour d'une position. Ne peut-on, dès lors, envisager une manette à débattement total réduit, avec, pour une même variation de l'angle manette, une variation faible, donc facile à doser, de la poussée autour de la poussée instantanée du moteur, et une variation beaucoup plus forte aux extrémités de la course ? Le pilote n'est pas gêné par une grande course de la manette et il a toujours une bonne précision d'affichage de la poussée (figure 1).

Bien entendu, dans le cas des bimoteurs, les deux manettes pourront être conjuguées afin de faciliter la synchronisation des moteurs. Aurons-nous un jour une seule manette pour deux moteurs ? ce serait bien pratique ...

Il reste à étudier la commande du moteur en cas de panne.

Envisage-t-on de changer de manche en cas de panne hydraulique, électrique ou électronique sur les commandes de vol ? Certainement pas. La manette doit permettre le pilotage de la poussée en secours ; il suffit, par exemple, de mettre plusieurs capteurs en parallèle à sa base ...

*

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LES CONTROLES.

Les instruments de contrôle doivent être bien adaptés à l'usage que l'on veut faire du moteur. Cet usage, pour le pilote, se résume en trois mots : poussée, consommation, sécurité.

- la poussée conditionne la performance, donc l'efficacité de l'avion;

- la consommation détermine l'autonomie, qui est toujours limitée et constitue la hantise du pilote de combat,

- la sécurité, enfin, est une nécessité évidente sur un mono ou un bimoteur.

Or, que donnent les instruments de contrôle actuels ? Un régime, une température, parfois un rapport de pression ou une section de tuyère, une indication d'allumage PC, un débit carburant, des voyants de pannes ... indications nombreuses mais inutilisées la plus grande partie du temps.

Une nouvelle indication est apparue, sur le MIRAGE 2000, il s'agit de l'accélération longitudinale de l'avion. Elle se présente, dans le viseur "tête haute", sous deux formes (figure 2) :

- au sol, des chiffres donnent en lecture directe le Jx,

- en vol, deux repères montrent par lecture directe à l'échelle 1 : 1 la pente potentielle, pente que l'avion peut prendre à vitesse constante avec la motorisation présente, ils montrent aussi, par rapport au vecteur-vitesse, si l'avion accélère ou décélère.

Cette indication s'est révelée remarquable par son utilité et sa facilité d'emploi :

- au sol, elle constitue une preuve formelle du bon déroulement de la phase en cours ; on mesure tout son intérêt quand on se souvient que bon nombre d'accidents aériens sont dus à une mauvaise appréciation par le pilote, de l'accélération ou du freinage pendant les phases critiques du décollage et de l'atterrissage,

- en vol, elle permet sans aucun effort et sans lecture d'un autre instrument de stabiliser instantanément une trajectoire déterminée, soit en amenant le vecteur-vitesse sur les repères de pente potentielle à l'aide du manche, soit, pour stabiliser une vitesse, en amenant les repères de pente potentielle sur le vecteur-vitesse à l'aide de la manette, soit en combinant les deux mouvements pour obtenir une pente donnée et une vitesse donnée ; l'action est quasi-instantanée et d'une très grande précision.

Mais cette indication, fort précieuse pour stabiliser une trajectoire ou pour connaître la marge dont on dispose avec la poussée actuelle ne suffit pas pour contrôler le moteur : elle ne donne aucune idée ni de la poussée disponible si on avance la manette, ni de la consommation.

Pourtant les autres instruments moteurs sont assez mal adaptés à la conduite d'un moteur de combat moderne :

- les valeurs de régime(s) et température(s) au plein gaz varient avec les conditions de fonctionnement et varient d'un moteur à l'autre ; faut-il les conformer ? leur utilisation serait plus facile mais ils perdraient toute validité pour la surveillance des valeurs limites.

- le débit carburant est évidemment nécessaire. Il est utilisé dans certaines phases de vol comme paramètre principal de conduite moteur, il a toutefois l'inconvénient d'obliger le pilote à une gymnastique intellectuelle pour tenir compte de l'altitude.

Une solution pourrait consister à présenter au pilote le rapport de la poussée actuelle à la poussée disponible dans les conditions de vol présentes ; il aurait les avantages sur le régime d'être plus "démultiplié" vers les fortes poussées, de bien indiquer la marge de poussée disponible et d'être directement lié à la manette.

Quel que soit le paramètre retenu, il est indispensable d'indiquer d'une façon immanquable le passage du secteur sec au secteur PC, en particulier en raison de l'accroissement considérable de consommation instantanée qu'il entraîne.

Bien entendu, un paramètre unique de contrôle peut convenir au fonctionnement normal mais ne suffira probablement pas pour permettre le pilotage en cas de panne ; d'autres indications sont donc nécessaires soit sur des instruments classiques soit sous forme de pages apparaissant automatiquement ou à la demande sur des visualisations cathodiques, selon la conception de la cabine. La disposition de ces indications est importante. Utilisées en cas de panne, elles doivent indiquer clairement les actions immédiates (paramètres hors limites) et les possibilités restantes (poussée disponible, restrictions de domaine ...).

On remarquera que l'une des contraintes actuelles rédide dans le fait que le pilote, lorsqu'il veut afficher un certain régime, doit surveiller les instruments jusqu'à ce que le moteur ait atteint le régime voulu, ce qui s'obtient par tatonnements et prend un certain temps. Il serait peut-être intéressant de présenter un paramètre commandé dont l'affichage serait instantané ; le pilote serait déchargé d'une tâche de surveillance désagréable et serait peut-être moins exigeant sur les temps de reprise de son moteur.On peut aller plus loin et ne plus présenter que le seul paramètre commandé. Certains penseront que le pilote préfère voir le résultat de son action plutôt que la recopie de l'ordre ; cela est faux : le pilote fait a priori confiance à son matériel. Seul un mauvais matériel lui inspire de la méfiance.

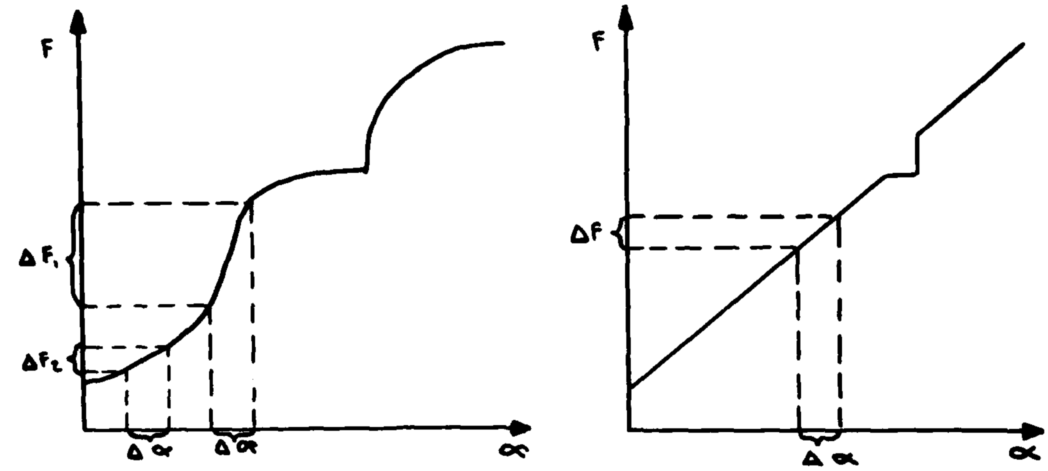
LES DISPOSITIFS ANNEXES

Les techniques numériques modernes permettent d'envisager des dispositifs destinés à faciliter la tâche du pilote ou à améliorer les performances. On peut citer :

- l'automanette : son rôle a souvent été limité à la tenue d'une vitesse en approche, alors qu'elle est beaucoup plus importante en interception ou en pénétration en basse altitude, quand le pilote ne devrait plus avoir à surveiller sa vitesse ou son Mach pour se consacrer entièrement à son objectif. Peut-être saura t-on un jour respecter automatiquement une heure sur l'objectif ? peut-être pourra t-on afficher directement une vitesse avec la manette ?

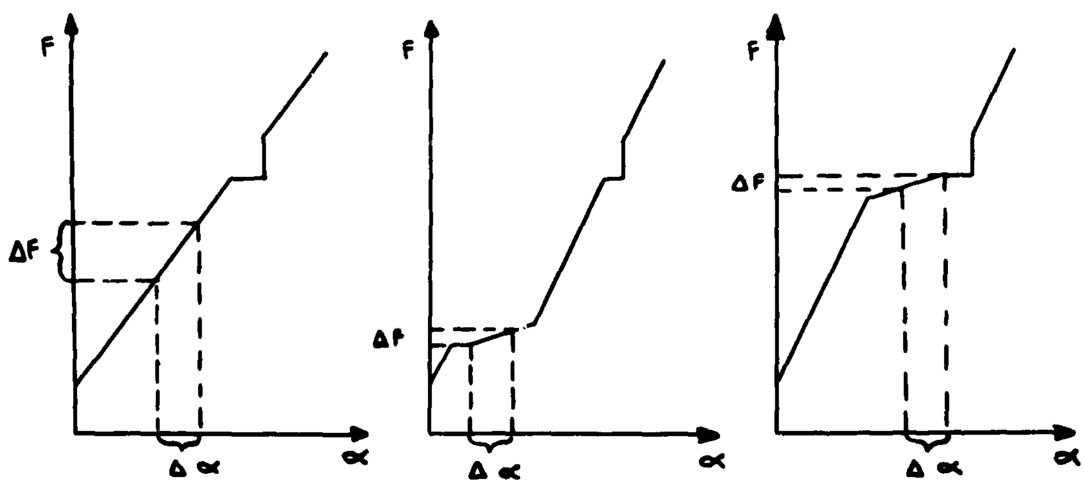
FIGURE 1

AMELIORATION DE LA LOI POUSSEE/ANGLE MANETTE



A : Loi actuelle (exagérée) :
à un même mouvement manette Δα
correspond un accroissement de
poussée ΔF variable.

B : 1ère amélioration :
la poussée est proportionnelle
à l'angle.



C : 2ème amélioration :
la course de la manette
est réduite.
Inconvénient : perte de
sensibilité.

D : 3ème amélioration :
on crée une plage de sensibilité
augmentée ; cette plage se déplace
en permanence, automatiquement, pour
rester centrée sur la poussée
instantanée.
A gauche, autour d'une poussée faible
A droite, au voisinage du plein gaz sec.

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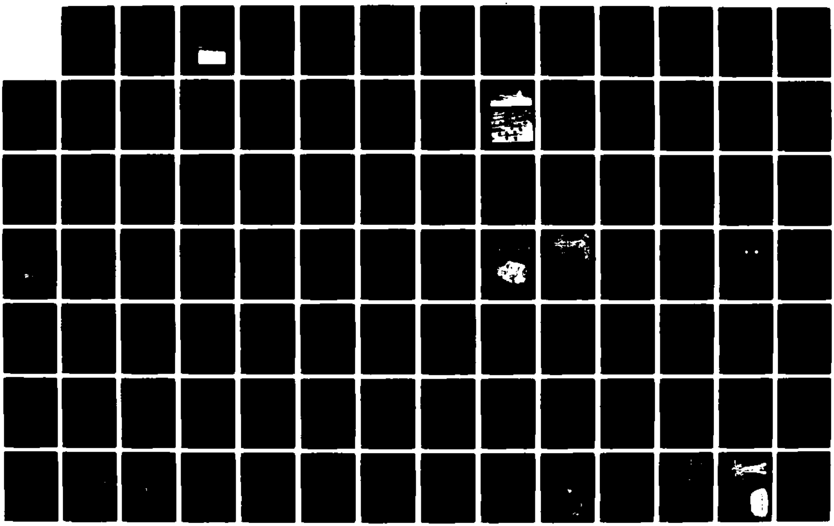
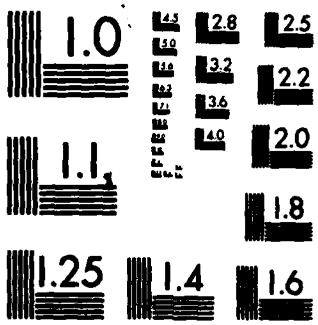




MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

- l'optimisation du fonctionnement du moteur. Selon la position et le mouvement de la manette (plein avant, fixe en position intermédiaire, mouvements rapides ...), le pilote cherche certaines performances (poussée maximale, consommation minimale, reprises rapides ...). Il doit être possible d'agir sur les paramètres de commande (section de tuyère, calages des aubes, vannes de décharge, spreaders...) pour obtenir la meilleure performance en fonction des conditions de vol. Il doit aussi être possible de faire comprendre au moteur la volonté du pilote, sans passer par un sélecteur de lois de pilotage dont la manipulation serait une charge supplémentaire.

- l'obtention d'une surcharge de combat. L'adjonction d'un décompteur de durée de vie pourrait permettre une utilisation à la fois sûre et efficace d'un tel dispositif,

- la coupure et le rallumage automatique, en cas de décrochage entretenu...

*

* *

Au cours de cette étude, il a bien peu été question de temps de reprise. Faut-il en déduire qu'ils n'ont pas d'importance ? Certainement pas. Un moteur doit exécuter rapidement un ordre qui lui est donné et fort heureusement les motoristes ont réussi, dans ce domaine, à donner des performances satisfaisantes à nos moteurs.

Mais toute mesure destinée à diminuer la charge de travail du pilote et à lui permettre de conduire son moteur sans se soucier de limitations, sans avoir à surveiller des instruments et sans devoir quitter sa cible des yeux accroîtra l'efficacité des avions de combat et compensera largement les quelques fractions de secondes qui n'auront pu être gagnées sur les temps d'accélération.

Le pilote a parfois tendance à oublier les progrès accomplis depuis de nombreuses années dans le domaine de la régulation des moteurs : optimisation des performances, limitations automatiques en fonction des conditions de vol, homogénéité du pilotage dans tout le domaine, passage automatique dans des modes dégradés ... n'ont été obtenus que grâce à des travaux difficiles et souvent obscurs, par lesquels les motoristes ont montré qu'ils étaient loin d'être en retard sur les autres techniques aéronautiques.

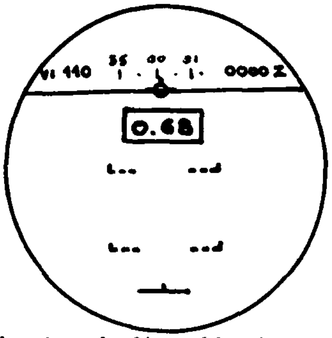
Les plus gros problèmes de pilotabilité des moteurs ont été résolus. Les réflexions qui précèdent proposent quelques axes de recherche pour améliorer encore l'efficacité globale des systèmes d'armes. Il semble que cette amélioration doive passer par une intégration plus poussée du moteur au système, tant en ce qui concerne son fonctionnement que les instruments associés.

Et le pilote pourra continuer à porter son jugement, concis et entier, à la descente de l'avion :

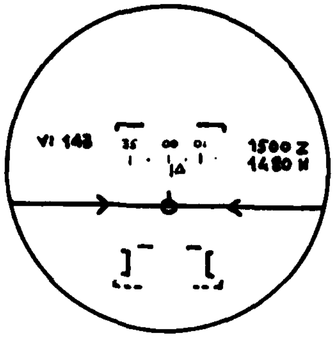
"Le moteur ? Pas de problème !".

F I G U R E 2

ACCELERATION LONGITUDINALE DANS LE VISEUR DU MIRAGE 2000.

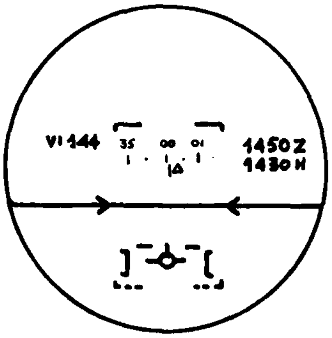


A - Au sol, l'accélération (positive ou négative) est inscrite dans un rectangle au centre du viseur.

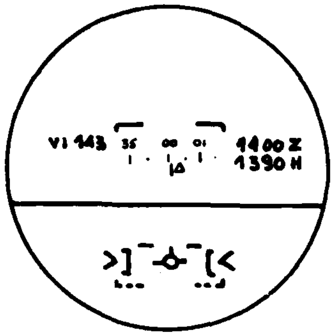


B -En vol, deux chevrons indiquent la pente que l'avion peut prendre, à vitesse constante, compte tenu du bilan poussée-trainée.

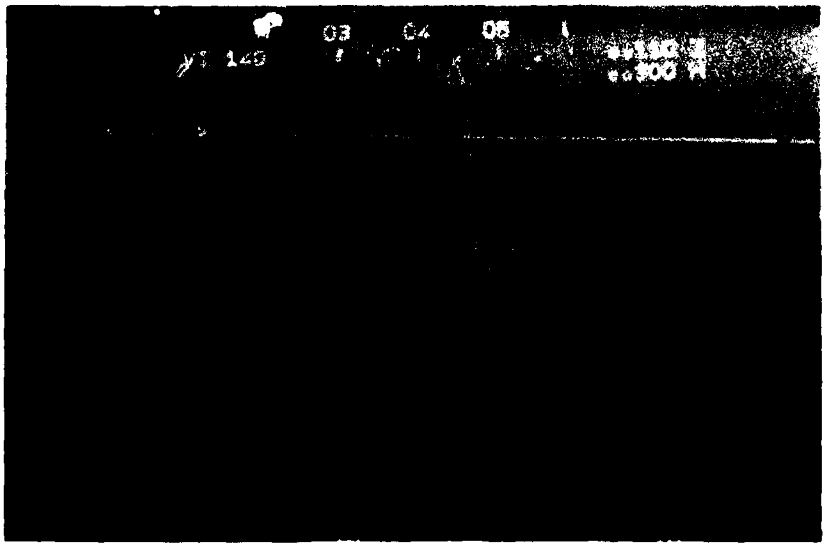
B1- Vol horizontal stabilisé.



B2 : le pilote, avec le manche, prend la pente d'approche finale ; l'avion accélère.



B3 : Le pilote, avec la manette, place les chevrons de pente potentielle en face du vecteur - vitesse : pente et vitesse sont stabilisées.



C : Vitesse, cap, altitude, hauteur radio-sonde, horizon, pente nominale, incidence nominale, pente potentielle, piste synthétique et ordres ILS dans un seul "instrument" intégré au viseur et superposé au monde extérieur ! facilité et sécurité !

DISCUSSION

Levison, Fr.

No-one could take exception with your desire to reduce the pilot workload to an absolute minimum at the point of combat by simplification of engine instrumentation.

Your concept of a purely secondary indication system for extra-normal operation only, causes me some concern. I feel that there is great danger in allowing the pilot to become unfamiliar with the normal behaviour of his vital propulsion system. By removing his "descriptive" instrumentation he is in danger of losing the "feel" of his machine and becomes less able to interpret the information presented to him in these difficult circumstances.

Just as in, for example, a motor race where the faster machine is often beaten by one which has a more sympathetic driver, is there not still the need to give the pilot the same close contact with his machine by using descriptive instruments for normal operation? Leave the fully automatic operation for combat models only!

Author's Reply

Un moteur moderne est extrêmement complexe, le pilote heureusement, en temps normal, n'a qu'une action limitée sur son fonctionnement par la seule manette des gaz: Il ne lui est pas possible d'optimiser ce fonctionnement, une instrumentation simple est donc suffisante.

En cas de mauvais fonctionnement, le pilote a besoin d'informations plus complètes pour utiliser au mieux les possibilités qui lui restent. Mais il est difficile de lui demander d'analyser la panne, il n'a ni les moyens ni les connaissances théoriques pour le faire. Il doit donc appliquer strictement les consignes qu'il a reçues, qui doivent couvrir la majorité des cas.

Personnellement je ne suis pas certain, que le fait de "sentir" son moteur en fonctionnement normal puisse permettre au pilote de mieux réagir dans les cas peu fréquents où il y a panne; le problème sera peut-être de mieux définer, dans certains cas (battements ou baisses de régime, fluctuations de température), à partir de quel moment il y a une panne.

Par ailleurs, en temps de guerre, toutes les phases du vol sont des phases où le pilote doit être très attentif à sa mission et donc ne pas être dérangé par des informations inutiles; en temps de paix, les vols d'entrainement coutent cher et on cherche à utiliser au maximum chaque phase de vol, par une utilisation intensive du système d'armes, il est donc difficile de faire une distinction nette entre les modes "tranquilles" et les modes "combat".

Bien entendu la philosophie que je propose devra être étudiée en essais et validée par l'expérience avant d'être définitivement adoptée.

J. Hourmouziadis, Ge.

It is understood that practically any engine parameter indication in the cockpit should be eliminated. Since it is not possible at the same time to eliminate all cases of engine malfunction or failure, on what information is the pilot expected to react in such a situation?

Author's Reply

Le pilote ne souhaite pas éliminer tous les instruments moteur de la cabine; il souhaite, dans la mesure du possible, qu'en fonctionnement normal un seul paramètre lui permette de piloter et de contrôler son moteur.

Bien entendu une information unique ne suffira pas, en cas de panne ou d'anomalie, pour l'avertir de la panne et l'informer des possibilités qui lui restent.

Il faut donc prévoir, en plus de cette information unique:

- *des systèmes avertisseurs* (voyants de pannes, lampes, signaux sonores ...) pour l'avertir de la panne (dépassement de régime, de température, de niveau vibratoire, baisse de pression ...) et lui donner des consignes immédiates.
- *des indications complémentaires* (par exemple, les paramètres actuels de pilotage du moteur), apparaissant automatiquement ou à la demande, pour lui permettre d'analyser les possibilités qui lui restent et donc de décider de la conduite à tenir à moyen terme.

Les systèmes avertisseurs existent. L'effort doit porter, semble-t-il, sur le paramètre unique en fonctionnement normal et sur les indications complémentaires en cas de panne.

FLIGHT TEST EXPERIENCE ON MILITARY AIRCRAFT ENGINE HANDLING

by
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SUMMARY

Most of our recent test experience on engine handling was obtained in the course of multinational official trials to assess the Tornado and Alpha Jet weapon systems, conducted by the German Official Test Centre (OTC) Erprobungsstelle 61 at Manching.

The object of these trials was to enable the specialists of the partner OTC's to recommend a clearance for the weapon systems necessary for a formal Release to Service.

To carry out these tests with an adequate method it was necessary to consider the operational role of the aircraft as well as to know the operation limits of the engines.

Testing carried out on Tornado as typical of the task of the German OTC is used to explain the methods applied and two examples of test results are given.

1. INTRODUCTION

Due to the high costs of modern weapon systems the previously common practice that two or even more contractors competed with their own designs has had to be replaced by a single and controlled phase-by-phase development. After each phase an assessment shows whether the achievements meet the objective of the phase and whether the assumptions, on which the development had been based, were correct. Provided they are positive, these reviews lead to a decision initiating the next phase of the development. This is the aspect under which the task of the German Official Test Centre Erprobungsstelle 61 should be seen. By nature, our test activities start rather late, i.e. at a time when the weapon system has already taken up its final shape to a large extent. If the previous phase decisions were made carefully, the aircraft now available should represent a useful solution. The evidence that this solution complies with the requirements of the Services must be demonstrated by official tests. The conclusions resulting from these tests are summarized in a recommendation stating the extent to which the aircraft can be operated by the Air Forces. This has successfully been demonstrated in the case of the Tornado weapon system. To save time, a step by step release was planned, enabling the Services a phase-by-phase changeover to the new weapon system. The first phase comprised a release required for training of crew instructors. Based upon this, a stepwise extension followed in accordance with the operational requirements, up to the final general release covering the specifications.

With the overall testing of a weapon system the assessment of the engine handling characteristics represents a specific and very important field, since engine handling affects the value of the weapon system considerably. Starting to prepare a test programme for the assessment of engine handling characteristics, we considered the following three essential aspects which determined the scope of testing:

- The fact that it is an official test.
- The mission of the weapon system.
- The specific technical characteristics of the engine.

(a) The official character of the tests influences the test objective rather than the test techniques. This objective has to be an assessment whether the technical solution meets the requirements of the Services. In more detail this official task can be defined as follows:

- checking the engine production standard with regard to the performance and characteristics required by the specifications
- checking the validity and applicability of the operating instructions proposed by the engine manufacturer
- assessment of the engines from the military-operational point of view
- gathering of information for checking and validating the national aircraft and engine documentation
- recommending a clearance for Service use.

(b) The missions of the weapon system for which it was designed and the resulting performance aspects have to be considered as far as they will affect the test programme. Based on mission frequency and priority, flight envelopes can be defined where the tests have to be conducted. Furthermore, the engine handling carried out during the tests shall comprehensively simulate the engine behaviour observed during typical flight manoeuvres. This requires the knowledge of the appropriate operational flight profiles, so that necessary power lever movements can be defined.

(c) To meet the operational requirements each engine design comprises many technical features which together contribute to a satisfactory engine operation. The knowledge of these features is useful to assess engine handling. This information was obtained from specifications and design documents of the engine manufacturer as well as from the test experience available from the engine and aircraft manufacturers.

One general problem is the fact that engines of a whole series have to be assessed at an early stage, at which time the final series production may not even have been started. In the course of series production manufacturing simplifications and modifications are carried out, which may have an effect on engine behaviour. In addition, there is also the variance that can be found in each batch. In order to obtain a statistically valid test result, tests would have to be carried out over a long period of time with a large number of series standard engines. This, however, would take too much time and would be highly expensive. The experience gained from prototypes and preseries standard engines is very helpful in this case. Because of modern production processes with small tolerances in manufacture - even for preproduction standard engines - the influence of variance on engine performance is reduced. In order to obtain reliable test results different methods can be applied.

In the case of the RB 199, official results have generally been obtained on a single engine basis. If unexpected test results had been attributed to an individual engine, results were verified by testing a second engine in another aircraft. This was especially required in case the official test results did not agree with those experienced by the manufacturer. If considerable influence on engine handling by modifications was expected, additional tests were considered necessary.

The manufacturer of the engine Larzac 04 has proposed a procedure to establish these influences. Assuming an "average engine", extreme behaviour is simulated by modifying the control characteristics. In defining the "average engine" typical geometric parameters, such as turbine stator cross sections and nozzle areas, are considered. Furthermore, control and performance parameters are used, taking into account design criteria, such as component characteristics, stall margin and observation of efficiency trends.

2. TEST AIRCRAFT

Our test experience of recent years has been gained with two technically quite different types of aircraft, the Alpha Jet and the Tornado.

Alpha Jet (Figure 1)

The Alpha Jet is constructed in a French version as a trainer and in a German version as an aircraft for close air support missions. It is a highwing aircraft with positively sweptback wings with dog teeth, and low positioned elevators. Two pylons can be fitted to each wing - the inboard pylons are twin-store carriers - to carry external stores and fuel tanks. A gun pod mounted to the bottom of the fuselage contains the 27 mm gun and 150 rounds of ammunition.

Larzac Engine (Figure 2)

The Alpha Jet is powered by two engines Larzac 04. The Larzac is a two-spool turbofan engine without augmentation consisting of:

- two-stage fan/separate bypass duct
- four-stage high-pressure compressor with 2nd stage bleed-air
- annular combustion chamber
- one-stage high-pressure turbine with cooled guide vanes and rotor blades
- one-stage low-pressure turbine - not cooled.

Tornado (Figure 3)

The Tornado has been developed trinationally and is a supersonic tactical combat aircraft with all-weather capability; its main missions in the interdiction-strike (IDS) version are operations against sea and ground targets. Special attention has been paid to carry a large variety of weapons over a long distance. In addition the Tornado is equipped with two 27 mm Mauser guns in the forward fuselage section. In order to reduce its vulnerability, Tornado is optimised for low level flight at high speeds.

A military requirement affecting the Tornado design was the capability to take-off and land on extremely short runways.

The Tornado will be used in Germany by the Air Force as well as by the Navy.

Characteristic design features are:

- two-seat high-wing aircraft
- variable positive wing sweep (25° - 67°)

- tailerons which can be operated together (in pitch) or differentially (in roll)
- two engines with augmentation
- thrust reverse.

Tornado's secondary power system (SPS) represents a new feature, i.e. the accessories (gearbox, hydraulic pump, generator) are no longer directly attached to the engine but one gearbox for each engine and one common auxiliary power unit (APU) attached to the right gearbox, constitute the separate SPS. The engines are connected to their gearboxes, by power transmission shafts. A cross-drive shaft permits the interconnection of the gearboxes. Connected by a cross-drive clutch assembly, which is controlled by the electronic SPS control unit, each engine can drive the gearbox of the other engine and take over its load in the case of speed reduction of the latter. On the ground each engine can be started by the APU but it is also possible to start an engine by the other (running) engine (via cross-drive) on the ground as well as in flight.

The operational impact of the SPS arises from the fact, that it enables the aircraft, together with Tornado's built in test equipment, to operate from dispersed operating bases, independently from aircraft ground equipment.

Engine handling in flight is affected by the SPS's capacity for high power off-take and its provision of additional air start procedures.

Engine RB 199 - Mk 101

The Tornado is powered by two engines RB 199 - Mk 101. They are compact three-shaft turbofan engines with augmentation. The core and bypass gas flows are fed into the augmentor without employing any extra mixing arrangements and expand through a common convergent nozzle with variable area.

Other features are:

- three-stage fan
- bypass ratio > 1
- three-stage intermediate-pressure compressor
- six-stage high-pressure compressor
- annular combustion chamber
- one-stage high-pressure turbine with cooled guide vanes and rotor blades
- one-stage intermediate-pressure turbine with cooled guide vanes and rotor blades
- two-stage low-pressure turbine
- blow-off valve behind the intermediate compressor to improve the engine handling under extreme conditions
- bleed air for air conditioning from stage 4 of the high-pressure compressor
- measurement of the blade-temperature of the intermediate pressure turbine (TBT) by means of pyrometers for control purposes.
- modular construction

Particular characteristics of this engine are its high bypass ratio for a particularly low dry specific fuel consumption in combination with a high boost augmentor. It is equipped with a thrust reverser for short landing distances. The specification requires very short response times on the ground as well as over the full flight envelope. These short response times near the ground are essential for the short distance landing in case of a necessary overshoot. Each engine is equipped with a full authority electronic control unit which controls the hydromechanical fuel systems for the engine and the augmentor. The augmentor control is unique in so far as it features a completely open loop arrangement in order to avoid the potential safety problems as experienced with other engines and to allow very fast response times inspite of the very large bypass ratio of approx. 1.2 for a military fighter engine.

3. OPERATIONAL ASPECTS AFFECTING ENGINE HANDLING

An analysis of the operational role of an aircraft with respect to engine handling does not only result in a catalogue of throttle manoeuvres necessary to provide the required thrust, but also shows those flight conditions where appropriate engine operation is essential. Considering a typical Tornado mission such as terrain following flight (TF) will provide special information which can be used to define adequate engine handling. Figure 5 gives an indication of a TF-flight.

During TF, the aircraft operates at low altitude at high subsonic speed, adapting the flight path as closely as possible to the terrain below the aircraft. This requires the engine to respond immediately to the throttle command to provide the demanded thrust. As TF normally does not require augmentation, mainly dry engine handling is affected, especially the acceleration and deceleration capability of the engine. Since the flight path consists of a sequence of pull-ups and push-overs the engine is permanently exposed to changing "g" loads which mainly affect the fuel and oil supply of the engine and necessitates also the consideration of oil and fuel system aspects.

Analysing the possible missions of the aircraft type involved, a catalogue of engine handling manoeuvres can be established, consisting of major elements such as slams, chops, reslams and modulated power settings, which also defines adequate flight manoeuvres like 1g-flight, wind-up-turn, push-over, inverted flight etc.

4. TECHNICAL BASIS FOR THE ASSESSMENT OF THE TEST RESULTS

The engine specification is the measure for the engine handling qualities. It reflects the military operational requirements, combined with the engineering possibilities of industry. These requirements have to be met.

The operation of any engine is limited to a specified operating range which must not be exceeded.

There are generally two types of engine limitations that have to be observed. The first type relates to flight conditions, aircraft manoeuvres etc which the pilot must not exceed while the second type relates to engine operational limits usually looked after by the automatic control system of the engine. It is a particular quality feature of any engine if the pilot has not to be bothered with specific instructions to watch gauges and to move the throttle only in a particular manner in certain critical areas within the flight envelope. In other words the power plant should be completely foolproof in this respect, which for instance had been a typical design goal for the RB 199. For a comprehensive test programme it is essential to

(a) cover all the critical flight conditions within the declared flight envelope

(b) carry out the type of handling tests which could occur in service but also in particular those which are expected to be the most critical ones to any engine.

Critical engine conditions to be searched for are for instance:

- stalls, either selfclearing or locking in
- engine flame outs
- augmentor flame outs
- augmentor buzz or screech
- exceedance of safe engine parameter limits.

As known from experience with other engines potential problems with respect to flame outs and stalls are for instance most likely to occur in the upper left hand corner of the flight envelope when slamming the throttle in combination with high power off-takes and high angles of attack. Therefore this region has been investigated for a large number of different aircraft/engine configurations in particular with the RB 199 in the Tornado. During these handling tests satisfactory engine operation was assessed. This assessment also included transient times taken by the engine to perform a commanded rating change either in the dry or in the augmentation mode which have to be compared against the specification.

4.1 PARTICULAR CHARACTERISTICS OF THE RB 199 ENGINE TO BE CHECKED OUT

While the above test specification is of a rather general nature allowance has also to be made for the inclusion of tests in specific areas which had been found critical during general engine flight development.

With respect to the RB 199 engine two problems are used as examples. The first one was an interface problem between intake and engine and the second one was an augmentor light up problem within a certain part of the flight envelope with F-40 type of fuel.

Since the German OTC has been involved in defining the magnitude of these initial problems and in supporting the required modification actions by the industry these two problem areas will be briefly dealt with.

4.2 THE ENGINE-INTAKE INTERFACE PROBLEM

Already during early development flying it became obvious that at subsonic flight all engines installed behind the left hand intake had a fairly consistent tendency to stall above modest angles of attack of the Tornado i.e. at values well below the maximum required angles specified. This was considered rather strange since the right hand engine did not give these problems at all.

The engine manufacturer started a detailed analysis of the problem and found the following explanation:

The RB 199 engine features a fan without inlet guide vanes in order to improve the bird strike capability and to save structural length and weight. The Tornado supersonic intakes on the other hand produce a considerable swirl of the air flow entering the engine. This in effect means a counter swirl for the left hand engine and a co-swirl for the right hand engine.

Model studies backed up by actual engine bench tests have then revealed a considerably increased HP-compressor sensitivity to flow pressure distortion from the intake in the presence of a counter swirl at the fan inlet face.

In order to overcome this problem a study had been carried out as to how and where to tackle this problem. It was found that the most cost effective solution was the introduction of a small flow straightener or fence in the left hand intake of the aircraft. Another paper from the aircraft industry will present more details of this topic. Aircraft testing with this relatively small modification incorporated in the left hand intake gave then fully satisfactory engine performance of the left hand engine up to the max. required levels of angles of attack both under steady state and transient engine conditions.

This comparatively minor modification to the left hand intake is now basic production standard and a retrofit action has been started with respect to early aircraft to incorporate this modification. To overcome handling problems with early Tornados without this fence, the fan running lines had been set down temporarily accepting a small performance loss.

4.3 THE AUGMENTOR HOT SHOT BLIP PROBLEM

At the end of the flight development phase augmentor operation had been very reliable and safe over the full flight envelope up to the handling boundary. The augmentor response times were faster than required by the specification for the early Mk 101 standard and will be even shorter for the Mk 103 standard validated at the moment. Some typical times can be taken from Fig. 6 and 7 as examples.

Practically all the development flying at the companies test centres had been carried out on Avtur (F35) type of fuel although the engine specification also requires satisfactory operation on Avtag (F40) fuel - still the primary Nato standard - which is used by the German Air Force and Navy as well as by the Italian Air Force. F40 fuel had been specified for the validation flight testing by the German OTC. In the course of this flight testing with F40 a small problem was highlighted which had not shown up when operating on F35 before. It occurred at higher altitudes prior to the actual firing of the so called hot shot through the turbines to light the augmentor. Due to a design weakness in the fuel control hardware an early injection of a limited amount of this hot shot fuel takes place lighting the augmentor fuel while the propulsion nozzle is still in its minimum (dry) position. The result can be a very light stall (pop surge) in the HP-compressor (not in the fan) which is selfclearing as can be seen on the test traces. This pop surge was sometimes so weak that it was not even registered by the pilot, but was only detected by the telemetry personnel monitoring the flight.

In the meantime the fuel system manufacturer has introduced a modification to the hot shot circuit to eliminate this problem. So far about a dozen engines have been flight tested by the contractor using 4 different modified control systems. These engines operating on F40 have been severely handled without producing "hot shot blip surges" any-where inside the flight envelope up to the augmentor handling boundary.

This modification in form of a changed restrictor and a spring rate in the fuel control is now basic for production controls and a retrofit action has been started.

It should be mentioned that this slight problem with early Mk 101 engines without the modified hot shot had not to be classified as a flight safety problem with any corresponding operational restrictions. An indication in the Flight Manual to the pilots on where this pop surge with F40 type of fuel may occur in the flight envelope will be removed as soon as the modification action has been completed.

5. SUMMARY OF ESSENTIAL RB 199 FLIGHT TEST RESULTS

The flight testing of several engines carried out under the control of the German Official Test Centre has confirmed satisfactory operation of the RB 199 engine. In particular with respect to the handling aspects the following statements can be made:

- the handling both in dry and augmentation is safe and has never led to any serious problems with respect to flight safety and/or engine integrity in flight
- the times required for a thrust change during dry handling of RB 199 Mk 101 engines comply with the Specification.
- the times required for a thrust change during augmentor handling of Mk 101 engines are shorter than specified and will be even better for Mk 103 standard. As an example only approximatly 1 s from min to max augmentation is required over the full flight envelope.

- development problems, such as the left intake/engine interface problem and the F40 augmentor lighting problem described in this paper, have been successfully investigated and solved.

6. CONCLUSION

Official tests of the German Test Centre are different from those of the Contractor. The test technique used is generally the same but the test objectives are different. They are not aimed at research or development but at the assessment of a final product offered to the government for Service use. Consequently the number of tests is normally less than the number of tests of the contractor. This in turn necessitates the selection of testpoints very carefully, considering engine characteristics and operational requirements as well as the test experience of the contractor.

The two specific examples given in this paper are typical of the tasks and duties of the OTC's with respect to engine trials. In each case a problem was encountered and a solution was developed by the contractor. A relatively short OTC trial was then conducted, to assess the proposed modification and to recommend a clearance for Service use.

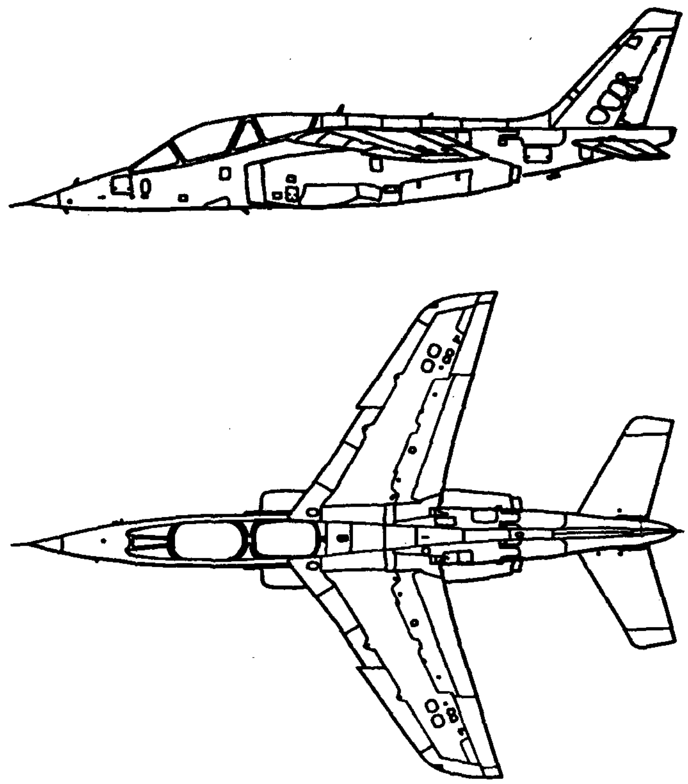
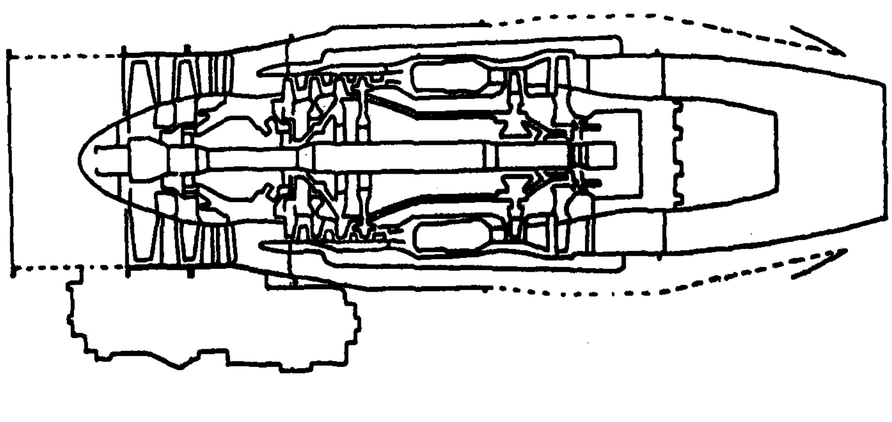


Figure 1   Close Air Support Aircraft Alpha Jet



igure 2   Turbofan Engine Larzac 04

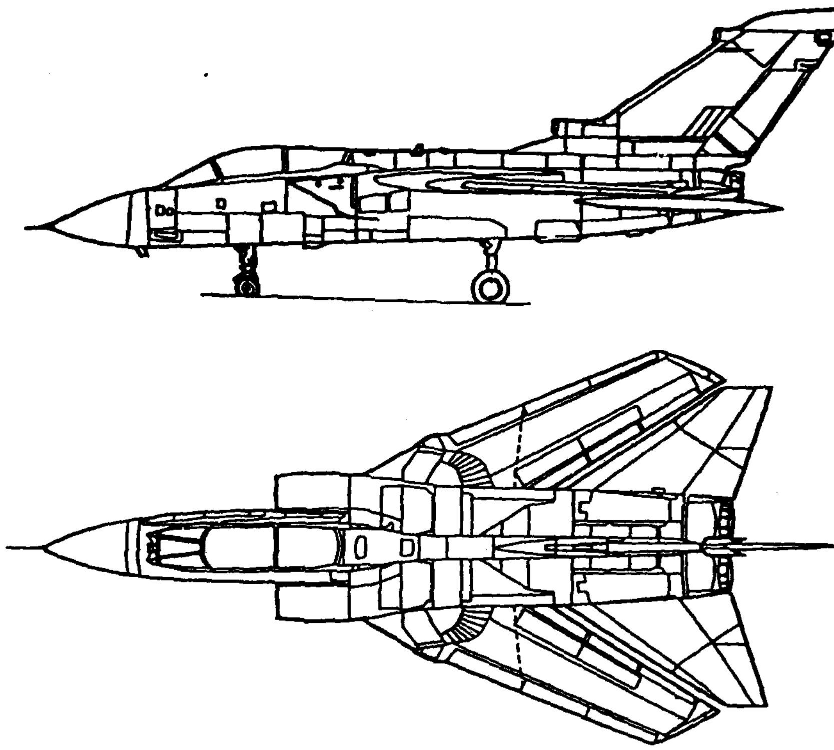


Figure 3 Multirole Combat Aircraft Tornado

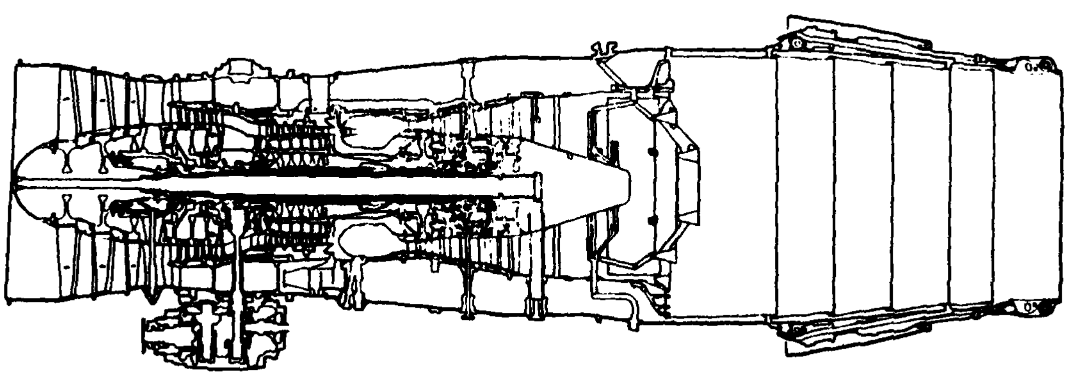


Figure 4 Augmented Turbofan Engine RB 199 - Mk 101

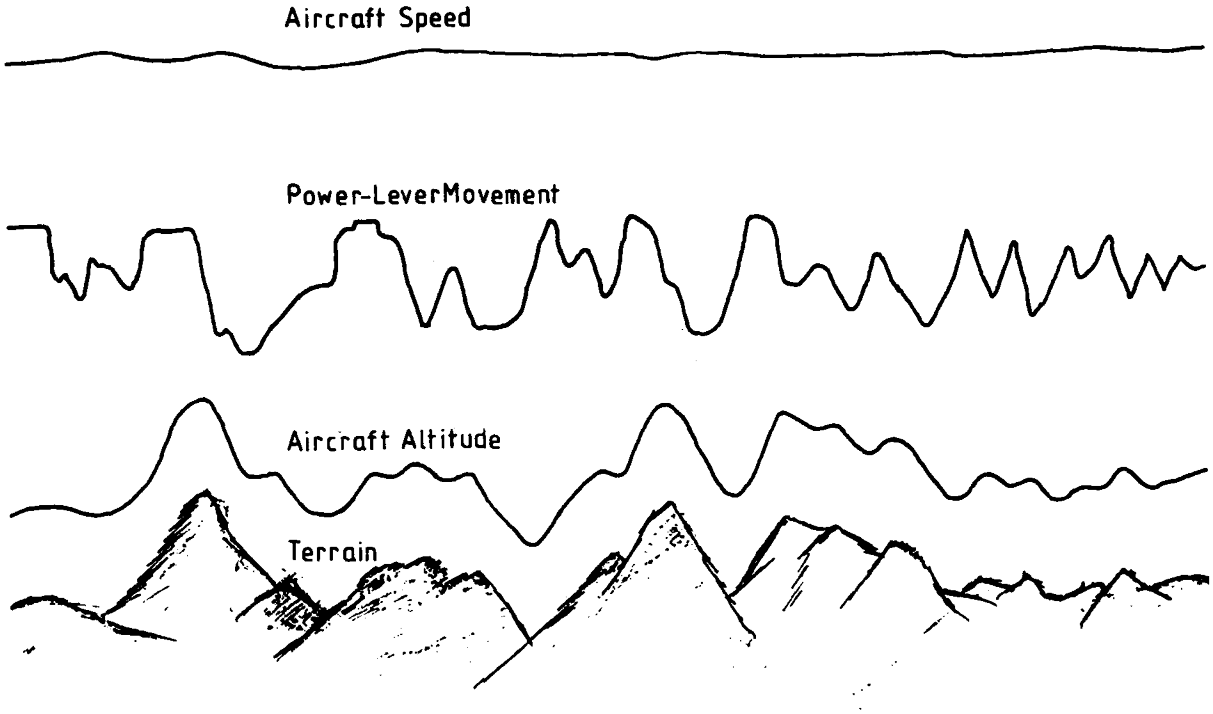


Figure 5 Parameters During Terrain Following Flight

Figures 6 and 7 Typical RB 199-Mk 103 Augmentor - Handling Times

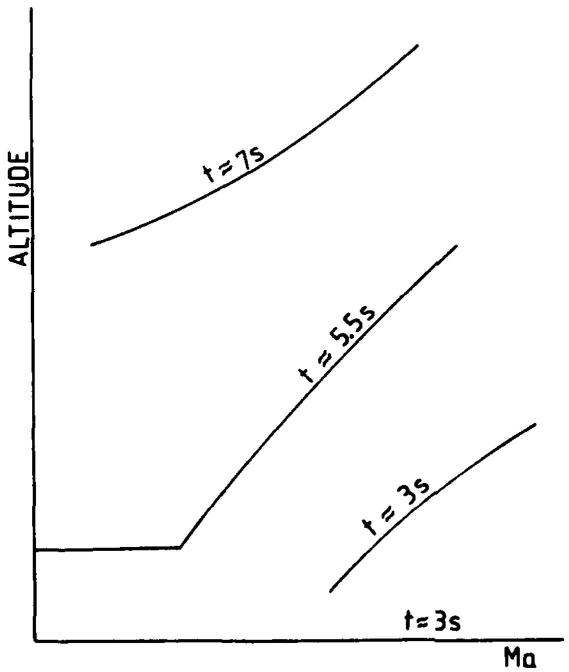


Figure 6 Idle-maximum Augmentation

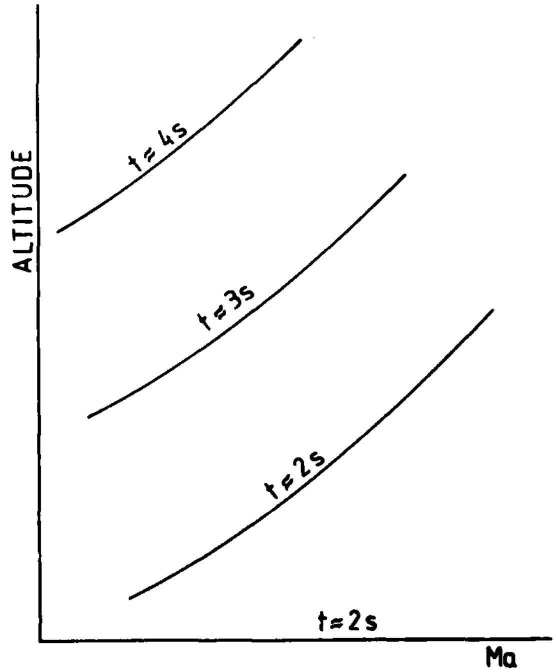


Figure 7 max. Dry - max. Augmentation

THROTTLE HANDLING RELATED TO J85 ENGINE

PERFORMANCE AND DURABILITY - CANADIAN FORCES EXPERIENCE

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SUMMARY

Because engine handling can impact two critical areas of engine operation - compressor stall margin and engine durability - the Canadian Forces undertook a flight test program with CF-5 aircraft, employed as an advanced jet trainer and close air support fighter, and Tutor aircraft, employed as primary jet tr iners and as a flight demonstration team aircraft. Both aircraft types use versions of the J85 engine.

The paper presents throttle handling data obtained over a wide variety of missions, from normal training to ground attack, ACM, and demonstration team flying. The methods used to relate mission profile data to mission severity factors are described. The resulting low cycle fatigue life predictions are compared with actual component overhaul rejection data.

In connection with CF-5 aircraft compressor stall investigations, the flight test program was successful in identifying throttle techniques involving the use of engine anti-ice, which improved compressor stall margin under certain conditions.

NOMENCLATURE

A/B	=	Afterburner
ACM	=	Air Combat Manoeuvres
AETE	=	(Canadian Forces) Aerospace Engineering Test Establishment
A/I	=	Anti-Ice
CDP	=	Compressor Discharge Pressure
CF	=	Canadian Forces
CIT	=	Compressor Inlet Total Temperature
EGT	=	Exhaust Gas Temperature
IGV	=	Inlet Guide Vanes
LCF	=	Low Cycle Fatigue
MFCU	=	Main Fuel Control Unit
MSF	=	Mission Severity Factor
N	=	Engine Rotational Speed
$N/\sqrt{\theta}$	=	Corrected Compressor Rotational Speed
NRCC	=	National Research Council of Canada
T2	=	Compressor Inlet Total Temperature (CIT)

1.0 INTRODUCTION

1.1 Canadian Forces (CF) interest in analysing J85 throttle handling was initiated during investigations of compressor stalls which occurred with the CF-5 aircraft, Figure 1, powered by two J85-CAN-15 afterburning engines. This investigation progressed from analytic and ground level static test cell investigations at the National Research Council of Canada (NRCC), to instrumented flight tests at the CF Aerospace Engineering Test Establishment (AETE). These tests found that altering throttle handling techniques could have a significant effect on reducing the frequency of certain types of compressor stalls.

1.2 During analysis of the flight test results, it became evident that dynamic engine parameters during manoeuvres were often very different from those assumed in early component life calculations. The same instrumented aircraft and engines were then used to develop more accurate profiles of engine usage for durability and lifing assessments. The same methodology was then extended to another aircraft type (Tutor training aircraft powered by a single non-afterburning J85-CAN-40 engine) for a similar investigation.

1.3 The Canadian Forces utilize approximately four hundred and fifty J85 engines of the two model designations in two different aircraft types. Both fleets of aircraft are utilized in a variety of roles, and the commonality of many of the rotating components allows additional insight into the effect of throttle handling on engine durability.

2.0 AIRCRAFT AND ENGINE DESCRIPTION

2.1 CF-5/J85-CAN-15

2.1.1 The CF-5 is a modified version of the Northrop F-5A/B aircraft built by Canadair Ltd., Montreal, Canada, powered by two J85-CAN-15 engines located side by side in the aft fuselage. This engine is a single spool afterburning turbojet with an eight stage axial compressor driven by a two stage axial turbine. The exhaust nozzle is fully variable through all throttle settings. Variable inlet guide vanes and inter-stage bleed valves are incorporated and scheduled on rotor speed and inlet total temperature. Nominal air mass flow is 20 kg/sec (44 lbm/sec) and maximum and military thrust ratings are 19.1 KN (4300 lbf) and 13.3 KN (3000 lbf) respectively. The CAN-15 engine shares its compressor components with earlier versions of the engine while the hot end components - combustor, turbine, and afterburner - are common with the later J85-21 version. In CF service, the CF-5 is used as an advanced jet trainer and ground/close support aircraft.

2.2 CT 114/J85-CAN-40

2.2.1 The CT 114 Tutor aircraft, designed and manufactured by Canadair, uses a single, non-afterburning J85-CAN-40 engine, with the civil engine model designation of CJ610. The CAN-40 has the same eight stage compressor and compressor variable geometry as the CAN-15, but with a fixed exhaust nozzle and a different combustor and turbine. The maximum rating for the CAN-40 is 12.7 KN (2850 lbf). The Tutor is used as a basic jet trainer and one squadron of aircraft is dedicated to the air demonstration role (the CF "Snowbirds", Figure 2).

3.0 CF-5 COMPRESSOR STALLS

3.1 General Compressor Stall Factors

3.1.1 The prime factors leading to compressor stalls in axial compressors have been identified for some time and are illustrated in Figure 3, which represents a compressor map for a simple, single shaft gas turbine with a fixed area exhaust nozzle. Included are steady state and transient compressor operating or working lines, which lie below the surge line. A hypothetical acceleration line from a low power setting (A) to a higher setting (B) has been sketched in.

3.1.2 In simple terms, a compressor will stall if the stall margin, represented as the relative distance between a point on an operating line and the surge line, is reduced such that the operating line intersects the surge line. This can be the result of a lowered surge line, a raised operating line, or a combination of the two. Some of the factors affecting the positions of these lines are included in Figure 3 and are summarized below:

Factors Lowering Compressor Surge Line	Factors Raising Single Spool Compressor Operating Line
Reduced Reynolds numbers	Engine accelerations
Inlet flow distortion	Reduced exhaust nozzle area
Compressor deterioration caused by wear, deposits, corrosion, or foreign object damage	Increased exhaust gas temperature
Increased tip clearances	Engine tolerances
Manufacturing tolerances	Fuel control tolerances

3.1.3 In addition to the above factors, at high or low corrected rotor speeds ($N/\sqrt{\theta}$), two areas exist where stalls can occur, which are usually avoided by proper functioning of the engine's control system. At low corrected speeds, a stall problem essentially results from different flow requirements at the front and rear of the compressor. Under these conditions, the rear stages are choked and the front compressor stages require a higher mass flow (to keep from stalling) than the rear stages are capable of passing. This type of stall has been overcome by engine designs incorporating multiple spools and/or variable inlet guide vanes (IGV), stators, or bleed valves. At the other extreme, a high corrected speed stall in effect represents a rotational Mach number limitation for the compressor. At this point, the compressor front stages have become choked and the rear stages are either stalled or near stall. Most engines incorporate a corrected speed limit (by reducing mechanical speed) under conditions of low inlet temperature (CIT or T2), beyond which the engine either cannot operate, or else does so with much reduced efficiency and stability margin.

3.2 J85-CAN-15 Compressor Stall Avoidance Design Features

3.2.1 The J85-CAN-15 control system provides countermeasures against both low and high corrected rotor speed stalls. To avoid low speed stalls, the engine contains both variable IGVs and bleed valves. To preserve stall margin under low CIT conditions and avoid high speed stalls, the main fuel control (MFCU) is designed to reduce maximum engine rotational speed when CIT is less than -26°C to keep the corrected compressor speed at or below 108%. In addition, the nozzle control system also varies the nozzle area to reduce engine exhaust gas temperature (EGT) as a function of CIT at temperatures below -26°C. These two control measures are referred to as "T2 Cutback" in J85 engine operation. Engine stall margin will be reduced, especially during throttle transients, if these cutbacks do not occur.

3.3 In-Service Stall Experience

3.3.1 As a result of pilot feedback providing information such as airspeed, altitude, attitude, throttle position, etc., just prior to stall, a base of three years of data was built up and a pattern of stall cause factors was determined. Figure 4 illustrates that compressor stalls generally fell into four regions of the aircraft level flight envelope. Each region has identifiable potential compressor stall cause factors:

Region A. Low speed flight at medium altitudes; stalls typically followed throttle transients. These flight conditions are experienced in certain training profiles and air combat manoeuvres (ACM). This region accounted for 52% of all reported compressor stalls and of these, 71% occurred with a CIT below -26°C. The stall inducing factors were a suspected lack of T2 cutback, coupled with inlet distortion associated with the higher angles of attack and aircraft yaw frequently encountered during ACM.

Region B. Stalls at higher altitudes, often occurring during throttle transients, accounted for 17% of all stalls, Of these, 41% occurred with CIT below -26°C. The suspected stall cause factors were lack of T2 cutback, decreased stall margin due to Reynolds number effects, and possible degraded compressor condition.

Region C. High speed, low altitude stalls accounted for 15% of all stalls. Under these conditions, CIT is high, hence corrected engine speed is low. If the corrected speed falls sufficiently, actuation of engine variable geometry will help to preserve stall margin. Stalls experienced in this region were attributed to improper variable geometry operation or rigging, and inlet temperature and pressure distortion associated with weapons delivery profiles and gun or rocket gas ingestion.

Region D. Seven percent of all stalls occurred under ground level static conditions during afterburner initiation at very low temperatures. The cause was suspected to be a lack of T2 cutback, since further investigation rarely found any engine or control malfunction.

3.3.2 From the above, it can be summarized that the major stall inducing factors were:

- Reynolds number effects;
- inlet distortion;
- low CIT operation; and
- engine transients.

4.0 STALL INVESTIGATIONS

4.1 Subsequent stall investigations concentrated on the effect of low CIT operation and suspected lack of T2 cutback, and on the effect of engine handling (transients) on stall margin.

4.2 Low CIT T2 Cutback

4.2.1 This work was reported in Reference 1. As a result of a comprehensive flight test program at AETE, it was determined that the MFCU was receiving erroneous T2 input signals due to engine installation effects; the air sensed by the MFCU had been warmed, primarily by the temperature sensing system heat soaking in the engine compartment.

4.4.2 Because of installation differences, the left hand engine was more affected than the right hand. Also, errors were increased under conditions of high angle of attack flight (Figure 5), where stall margin would be further degraded due to increased inlet distortion.

4.2.3 As a result of these errors, T2 cutback was either delayed or insufficient, producing a $N/\sqrt{\theta}$ overspeed (calculated to be as high as 111%) and EGTs 28°C above design, consequently reducing stall margin under these low CIT conditions. If an engine transient, particularly an afterburner (A/B) initiation, occurred simultaneously, operational experience was that a compressor stall frequently resulted.

4.2.4 On ground run-ups prior to take-off, the T2 sensing error is increased by prolonged running at a lower power setting before A/B initiation (as frequently happens during taxiing or awaiting take-off clearance). This was determined to be due to poor ventilation of the engine bay at these low power settings. It was discovered, as illustrated in Figure 6, that if instead of a throttle movement from the IDLE to A/B, the throttle was paused at 92% N for up to 20 seconds, the increased airflow through the engine bay and T2 sensor system would effectively reduce the errors.

4.2.5 Appropriate use of engine anti-ice (A/I) also can circumvent T2 sensing system errors under certain conditions. In the J85, the A/I system ducts hot 8th stage bleed air through the IGVs and bullet nose. This hot air, representing about 3% of compressor mass flow, can offset the T2 sensing error by raising the inlet temperature. Computer simulation using the General Electric status deck showed CIT was raised by an average of 7 to 8°C, lowering $N/\sqrt{\theta}$ by 1.2 to 1.4%.

4.2.6 Another effect of A/I is to reduce the compressor delivery pressure (CDP) and help preserve stall margin during A/B initiation. This is covered in the next section on the investigation of transient effects.

4.3 Transient Effects

4.3.1 The Engine Laboratory of the NRCC was tasked with overall program management and exploratory engine testing in a sea level static test cell (described in Reference 2). This testing progressed from development of steady state compressor operating lines by computer simulation and engine testing, through to dynamic testing to establish transient working lines for various throttle movements.

4.3.2 Steady State engine computer simulations and testing covering a wide range of inlet temperatures resulted in the determination of steady state compressor working lines, plotted on a J85 compressor map in Figure 7. The effect of varying inlet temperatures and the variable exhaust nozzle is evident by the lack of a unique steady state line which is the result of the engine and nozzle control system. The nozzle area is controlled by throttle angle until a target EGT is reached; beyond this setting, the nozzle is controlled to keep EGT constant. On Figure 7, points A to B represent the effect of reduction in nozzle area as a result of increasing rotational speed. At point B, the EGT has been increased to the target value at which time the electronic nozzle control takes over to hold EGT constant. Point M represents the MILITARY power position. Also evident on Figure 7 is the strong effect of CIT on the location of the working line.

4.3.3 Dynamic engine testing resulted in determination of transient working lines for several types of throttle movements as shown in Figures 8 a, b, c, which illustrate accelerations from IDLE to MILITARY, IDLE to MAX A/B, and MILITARY to MAX A/B.

4.3.4 What is evident from these working lines, is the reduced stall margin during engine accelerations. Movements into A/B produce the worst stall margin reduction, with substantial rotational speed rollback being frequently encountered upon A/B initiation with the transient line being displaced above the steady state line. This rollback is considered to be a result of jet pipe pressures that exist momentarily as the A/B fuel is ignited, until the nozzle control can react and increase nozzle area in controlling EGT.

4.4 Stall Margin Improvement

4.4.1 As a simple yet effective means of improving stall margin during A/B initiation, a technique involving manual selection of A/I, just prior to A/B selection, was developed. As mentioned previously, this reduces CDP and lessens the magnitude of any pressure pulse at the compressor delivery resulting from the A/B ignition. This technique was conceived and evaluated at the NRCC test cell, and as illustrated in Figure 9, the transient operating line is significantly lowered, improving stall margin.

4.4.1.1 As an aside, comparison of Figures 7, 8, and 9 will show slightly different steady state working lines. This was due to the fact that the test program progressed over a two year period and different engines were used in generating the data. The slight shift between the lines is considered to reflect normal engine to engine differences or variances in engine trim. In order to ensure valid comparisons, it was necessary to establish steady state working lines for each engine under test. Also, since CIT affects the position of the lines, and the test cell was an open atmospheric ground level facility, it was necessary to generate steady state lines as ambient temperatures changed (temperatures during the testing in Ottawa varied over a -25°C to +30°C range).

4.4.2 As part of the flight test program, the benefits of using A/I as a stall alleviation method during A/B initiation were evaluated. Figure 10a shows CDP during A/B initiation at 40,000 ft, 0.9M, with and without A/I activation. Clearly evident, is the pressure pulse associated with A/B ignition. The lower value of CDP resulting from use of A/I was estimated to provide up to a 30% improvement in stall margin.

4.4.3 Changes were made to aircraft operating instructions requiring pilot activation of A/I prior to A/B initiation during certain manoeuvres in area A of Figure 4. One example was a training exercise involving practice aborted landings carried out at 18,000 ft; the new throttle technique totally eliminated A/B initiation stalls during this manoeuvre. However, during ACM, which also frequently involves area A, it was considered that such a conscious pilot action would be difficult. Plans are now underway to incorporate an automatic A/I activation.

4.4.4 A disadvantage of A/I use is that it degrades engine net thrust by 6 to 7% at typical ACM flight conditions. Testing was continued to ensure that engine response would not be significantly degraded. This phase of the test program determined the optimum time interval for A/I activation, optimizing stall avoidance and engine response, was in the order of 8 to 9 seconds.

4.4.5 The effect of A/I was noted to vary according to flight condition, as illustrated in Figure 10 b. It should be noted that use of A/I is not a panacea for all stalls occurring during A/B initiation. In particular, use of A/I is considered to be potentially harmful in area C of Figure 4. This area is frequently entered during weapons delivery profiles, such as the strafe run shown in Figure 11. Flight in this region produces a low $N/\sqrt{\theta}$ engine condition due to high inlet temperatures. Since stalls in this area are frequently the result of improper variable geometry operation or rigging, and since the variable geometry control does not take into account the increased CIT due to A/I, activation of A/I could conceivably aggravate an already bad situation. For this reason, the modification proposed for automatic A/I activation only operates below a predetermined total inlet pressure.

5.0 DURABILITY

5.1 Early Status

5.1.1 It is interesting to note that approximately 600,000 hours of engine flying time were experienced by the Canadian Forces J85 fleet before a requirement for an ultimate life limit was recognized for cold section rotating components. This recognition was precipitated when a number of CAN-40 first stage compressor blade and disc-shaft failures occurred. The cause of these failures was subsequently determined to be low cycle fatigue (LCF) overlayed with periods of high cycle loading caused by a first stage aerodynamic instability. In the process of examining compressor cyclic design data, it was noted that a significant number of CAN-40 engines were approaching an hourly life which potentially equated to the cyclic limit of the fifth stage spacer. Figure 12 provides cyclic design life data for the rotating components and as illustrated, the pacing item in the compressor is the stage 5 disc. The majority of the subsequent life prediction analysis concentrated on this item.

5.1.2 The high time CAN-15 engines had many fewer flying hours than the CAN-40 lead-the-fleet engines; however, as for the CAN-40, there was no defined relationship which predicted CAN-15 life usage based on mission requirements.

5.1.3 The functions relating flying hours to life usage required definition for each of the roles in which the engines were exposed. No flight instrumentation had previously been dedicated, within the Canadian Forces, to the recording of throttle handling variations for life determination purposes. Some data did exist however, from other flight testing, which indicated massive differences in engine transients between mission profiles. For example, bleed valve movement, which in the J85 engine is a good indication of throttle transients, varied from 19 (equivalent) full travel movements per flight hour for CAN-40 training missions to in excess of 3,000 per flight hour in the Snowbird air demonstration roles.

5.1.4 To provide an initial assessment of the effect which various mission profiles had on life usage rates, it was decided to rely on data gained during previous pilot surveys to estimate the mission-particular throttle demands. The results of this analysis would be refined by undertaking a flight test program simulating the various missions. A final assessment of component lives would then be made based on an analysis of overhaul rejection data. It was hoped that this failure data would enable a comparison of the methods of mission profile data analysis and resulting variations in life prediction.

5.2 Life Prediction Methods

5.2.1 Prior to describing the methods used for measuring throttle movements and the resulting changes in LCF life predictions arising from refinements to the data acquisition process, it is necessary to briefly describe how General Electric (GE) determines mission related component lives for the J85 engine.

5.2.2 The GE procedures use the T-38 Talon, United States Air Force Air Training Command (ATC) missions as a baseline for determining component design life for afterburning versions of the J85 engine. In addition, mission profiles of the United States Navy (USN) T2C Buckeye aircraft, which use 2 J85 non-afterburning engines, were used as an additional consideration in evaluating the mission related life usage of the CAN-40 engine.

A differentiation is made between the afterburning and non-afterburning baselines, primarily to eliminate the overspeed variable which can by typical of afterburner termination. To illustrate this, Figure 13 shows the extent of engine overspeed on A/B termination. On a throttle movement from A/B to MILITARY, the nozzle may remain at an open position for an instant after the A/B fuel is shut off. This unloads the engine and a short duration overspeed occurs before both the nozzle closes down and the engine overspeed governor reacts. At the NRCC ground test facility, a means of reducing the overspeed by throttle action was investigated. Figure 13 demonstrates that if the throttle is moved from A/B to below MILITARY (typically 92 - 95% N), then back up to MILITARY, no overspeed results.

5.2.3 Through use of the component failure histories of the large ATC and USN fleets, combined with the original design data and spin pit test results, a baseline relating flying hours to cyclic life usage has been determined by General Electric. Fleets with different missions are then related to the baseline US mission mix by a Mission Severity Factor (MSF), which compares different missions' throttle movements and speed changes to the part cycle relationship of Figure 14. Dividing the cyclic life limits of Figure 12 by the MSF yields minus three sigma LCF life to dysfunction in terms of flight hours. Dysfunction is defined as the condition that exists when a crack has propogated to a point beyond which critical growth will occur resulting in excessive vibration or material separation.

5.3 Initial Analysis - Pilot Survey

5.3.1 Initial life prediction estimates were based on the results of extensive pilot surveys. Pilots from each squadron were requested to monitor and record power settings during flight for each type of mission. Each squadron then submitted averaged results and estimated the percentage of flying time spent performing each mission. These results were then combined and analyzed by GE. Figure 15 provides the breakdown of missions for both CF-5 and Tutor aircraft fleets. This analysis produced the type of throttle excursion plots shown in Figure 16. Note that the pilot surveys yielded throttle demand summaries which did not recognize short term excursions and in fact represented very docile flight profiles and corresponding simple power/speed, load/time summaries for the engine.

5.3.2 Although pilot surveys were undertaken for both CF-5 and Tutor aircraft, the data were only used in determining an MSF (given in Table 1) for Tutor training role CAN-40 engines. The MSF, so derived, was used for initial reprovisioning of CAN-40 replacement components.

5.3.3 However, it was recognized from the beginning that pilot surveys would give a very imprecise value of MSF. Consequently, while the pilot surveys were underway, plans were formulated for instrumented test flights to provide more accurate data. Determination of an MSF for Tutor Snowbird air demonstration aircraft used only the flight test data, which become available in time for the mission analysis. Because of a much lower fleet time for the CAN-15 and corresponding lessened urgency for a life prediction, flight test data were ultimately used for the CAN-15 MSF determination.

5.4 Life Prediction Refinement - Flight Test

5.4.1 CF-5. The flight test program for the CF-5 was carried out at AETE with missions flown by AETE test pilots and training squadron instructors. A dual cockpit CF-5D was chosen for the program, with instrumentation fitted in the rear cockpit. A single engine was instrumented and the following six parameters were recorded during flight:

a. Engine Speed;

b. Exhaust Gas Temperature;

c. Compressor Inlet Temperature;

d. Power Lever Angle;

e. Altitude; and

f. Indicated Air Speed.

In addition to the above listed parameters, a voice recorder was also used to describe manoeuvres and provide an additional time reference. Although only engine speed was used for life prediction purposes, the remaining parameters were recorded in anticipation of further enhancement of life prediction methods not related to LCF, such as turbine blade creep rupture life.

5.4.2 Figure 17 provides a graphical representation of throttle movement and engine speed transients during a four minute portion of a typical ground attack training mission, corresponding to that portion of the Figure 16a flight between the dotted lines. There are a significant number of engine speed excursions, two of which exceed the 30% threshold for influencing the cyclic life usage of the GE analysis indicated in Figure 14. The pilot survey output however, contains no information of throttle excursions which would effect component life using the GE analysis. Figure 18 illustrates the extent of throttle movements measured during an ACM 180° engagement. Once again, considerable engine speed cycles are evident.

5.4.3 Tutor. Two separate flight test programs were carried out to gather data on missions flown by Tutor aircraft; one for the Snowbird air demonstration role, and one for the training missions.

5.4.4 The Snowbird engine usage data were gathered during air show practices in conjunction with an aircraft structural loads survey. Engine parameters recorded were:

a. Power Lever Angle;

b. Exhaust Gas Temperature; and

c. Engine Speed.

5.4.5 The air demonstration squadron utilizes nine aircraft for air shows and data were collected for what were considered to be the most and least severe positions. The mission mix for the squadron consists of ferry, practice and show flight time with ferrying representing approximately 1/3 total flight time.

5.4.6 A four minute portion of the data gathered at the most severe position (Right Outside Wing) during an air show practice is provided in Figure 19, corresponding to the area between the dotted lines of Figure 16 b. Once again, the throttle excursions were extremely numerous; for this, and all other formation flights, the number of throttle movements far exceeded any projections.

5.4.7 Although in-flight data were obtained for CAN-40 engines in the training role, it was ultimately decided that a MSF could be based on a statistical analysis of the failed components, since a sufficient number (180, stage 5, 6, and 7 compressor discs) had by now been identified and analysed as to mode of failure.

5.5 MSF Conclusions

5.5.1 The MSFs determined at the various phases of the program are listed below in Table 1:

ENGINE TYPE	MISSION SEVERITY FACTOR (MSF)		
	PILOT SURVEY	FLIGHT TEST	COMPONENT FAILURES
J85-CAN-15	(NOTE 1)	1.26 (1.50 NOTE 2)	NOT YET DETERMINED
J85-CAN-40 SNOWBIRDS	(NOTE 1)	1.5	2.5
J85-CAN-40 TRAINING	.94	NOT DETERMINED	1.6

TABLE 1

NOTE: 1. Pilot survey data were not used in determining MSF due to availability of flight test data.

2. 1.5 represents MSF with ground run time included in analysis.

5.5.2 It is obvious from the flight test and component failure data that pilot surveys, while better than nothing as a first estimate, are not sufficient to accurately define engine usage for component lifing purposes. This is not unexpected, since in an environment with a high pilot workload such as ACM, weapons delivery profiles, or formation flying, it would be impossible for the pilot to note all throttle or engine speed movements. Additionally, even if consciously watching gauges, pilots could not see short duration engine speed exceedances during certain throttle manoeuvres (Figure 15, A/B termination).

5.5.3 A better mission definition is obtained from instrumented test flights; however, this does not provide a complete answer, as can be seen from Table 1, where MSFs calculated from test flights under-predict the lives based on actual component failure data. There are several probable reasons for this. The first is that the mission profile data from test flights were obtained from a few flights by a small sample of pilots and subsequently extended to apply to all engines on an average fleet wide basis. This approach misses aircraft to aircraft and pilot to pilot variations. Additionally, test flights were flown by test pilots or otherwise experienced aircrew. It is considered that throttle movements would be much more severe in actual use by student pilots.

5.5.4 The knowledge of what throttle manipulation occurs must be expanded to allow for better fleet life predictions, or, ideally, individual engine life usage calculations. For the CF-5 and the Tutor, cycle counters will be installed in ground test facilities and approximately 15% of the total fleet engines. The counters will effectively count engine speed excursions in bands, and equate them to part cycles according to the relationship earlier provided. Once again, the count to operating hour function will be compared to USAF or USN missions, and fleet life predictions made. This analysis will make use of a larger data base and will recognize mission type or mix changes with time.

5.5.5 The CF18A and B Hornets being purchased by the Canadian Forces bring the data source for life usage analysis closer to the ideal, in that each individual engine's total operating time will be monitored by an on-aircraft computer. Additionally, an In-Flight Engine Condition Monitoring System continuously monitors high pressure compressor speed, and records full and part cycles. Although the definition of part cycle counts is restricted to one band width for speed excursions, the system offers a significant improvement over the fleet sample data base.

5.5.6 As a final comment, the MSF was calculated by relating Canadian mission throttle demands to those of a US baseline. This does not take into account any effects due to climatic differences. The aircraft under discussion were flown from bases in the Canadian mid-west where temperature extremes from winter to summer of -40°C to +35°C are common. Average minimum ground level January temperatures are in the order of -25°C. This compares to substantially warmer temperatures in the American south and south-west where the baseline US aircraft predominantly operate. The climatic effect on centrifugal inertia loads would not be expected to be large, although compressor loading is higher in the colder, low level, regime where most throttle handling occurs (for example, CAN-15 compressor work on a -30°C day at 100% N, 3200 ft, .62M, is 15% higher than that of a standard day and 24% higher than a +30°C day). To our knowledge, this aspect of component lifing - a cold and low as opposed to a hot and high operation - has not been investigated to any great extent and is an area where we plan further work.

6.0 CONCLUDING REMARKS

6.1 An investigation by the Canadian Forces and National Research Council of Canada to determine the cause of compressor stalls with J85-CAN-15 engines installed in the CF-5 aircraft progressed from analytic computer simulations, to ground static test cell engine running, to instrumented test flying. Several stall cause factors were identified, among them, some factors which may be considered inherent in axial compressor operation, some which were a function of the engine installation, and some which resulted from engine transients and were responsive to changes in engine handling technique.

6.2 Compressor stalls were rarely considered to be the result of a single cause factor, but rather were the additive result of more than one factor; consequently, there is no single compressor stall cure. However, for those stalls occurring during A/B initiation, it was found that concurrent selection of engine anti-ice had a marked effect on improving stall margin.

6.3 In assessing the effects of throttle handling on engine durability, the Canadian Forces has repeated the experiences of other operators to the point of performing instrumented test flights to define mission profiles and associated engine usage. Information on throttle movements obtained from pilot surveys was shown to be inadequate, especially for those missions with a high pilot workload, such as ACM, weapons delivery, and formation flying.

6.4 Discrepancies between mission severity factors based on test flights and those derived from component overhaul rejection or failure data, indicate that actual in-service usage is yet again more severe. This is seen as another justification for individual engine usage indicators, either cycle counters or more sophisticated systems to integrate all significant mission profile data.

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2. Rudnitski, D.M., "Gas Turbine Engine Transient Testing", AGARD Conference Proceedings No. 293, Paper 22, October 1980.

3. Macmillan, W.L., Nelson, K.D., and McGimpsey, R.M., "J85-CAN-15 Compressor Stall and Flameout Investigation", 1978 Symposium of Flight Test Engineers, Arlington, Texas, October 1978.

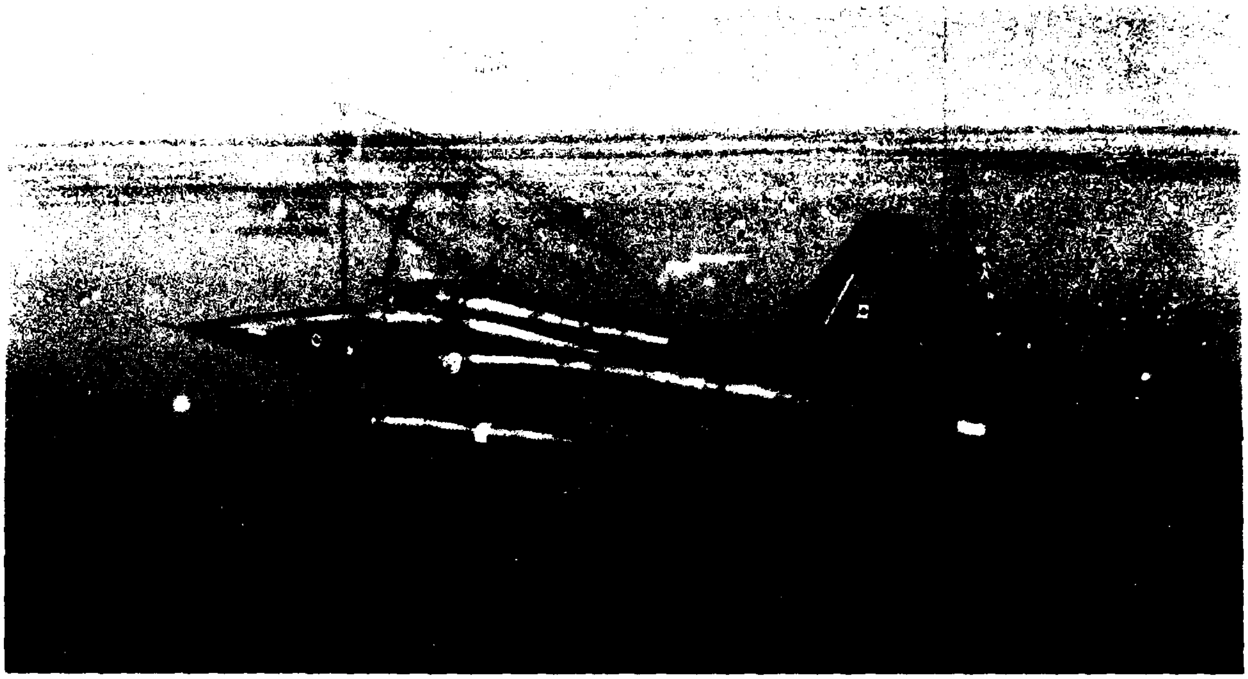


FIGURE 1: CF-5 AIRCRAFT

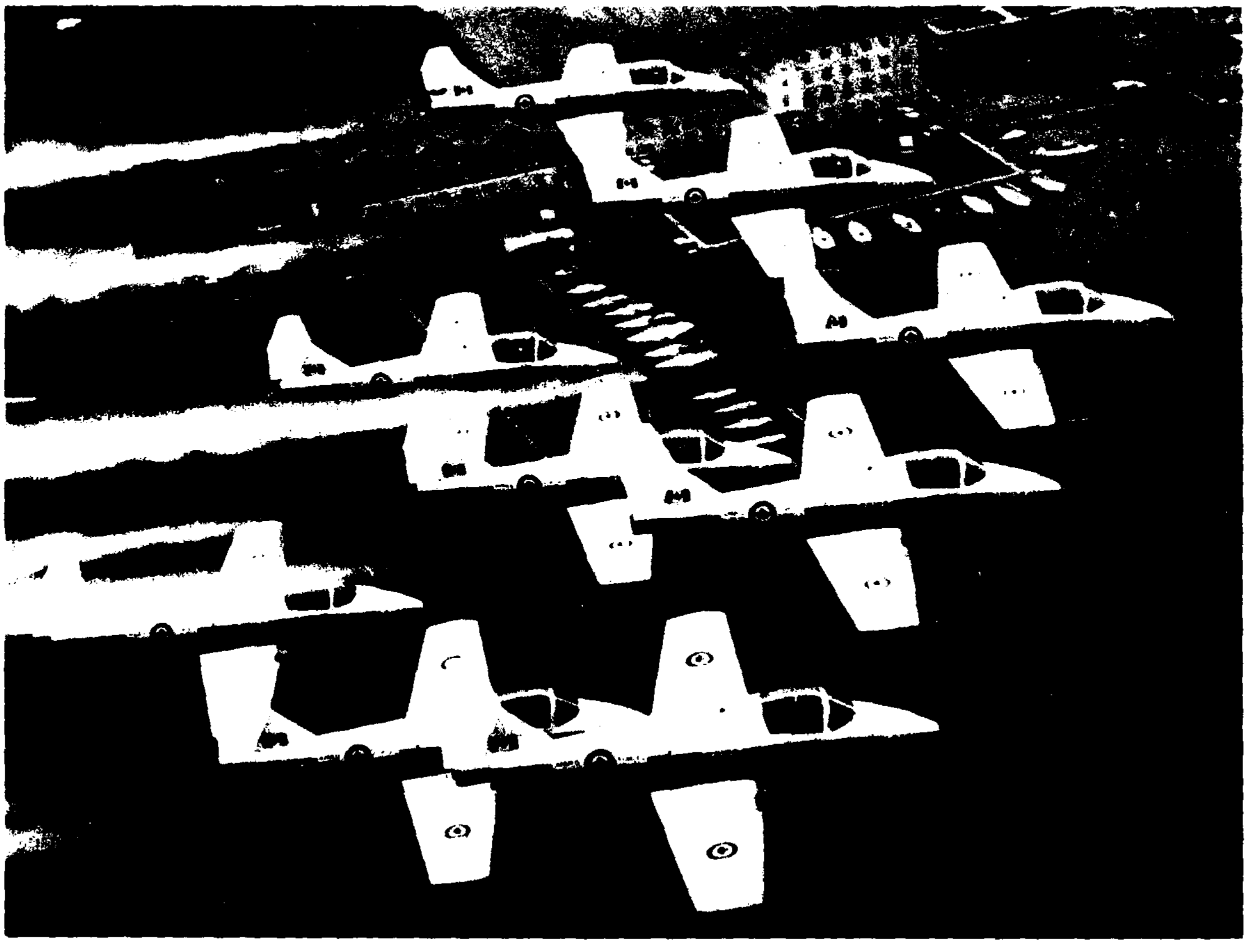


FIGURE 2: CT114/TUTOR; "SNOWBIRDS" AIR DEMONSTRATION TEAM

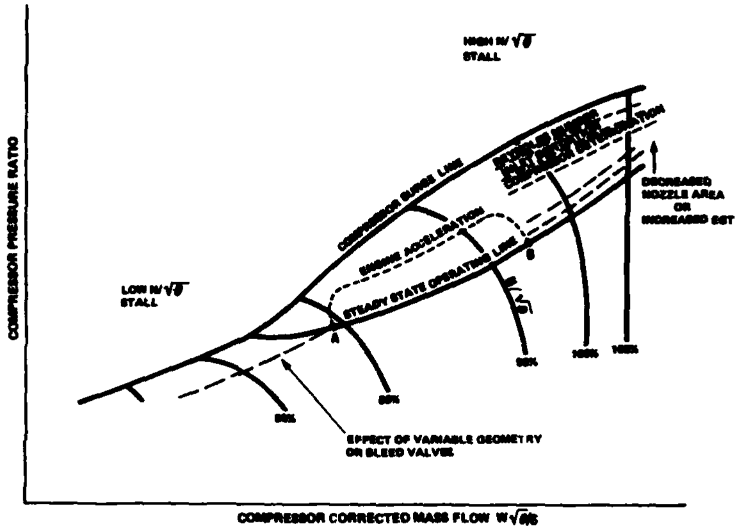


FIGURE 3: GENERALIZED COMPRESSOR MAP FOR A SINGLE SPOOL, FIXED EXHAUST NOZZLE TURBOJET (REF 1)

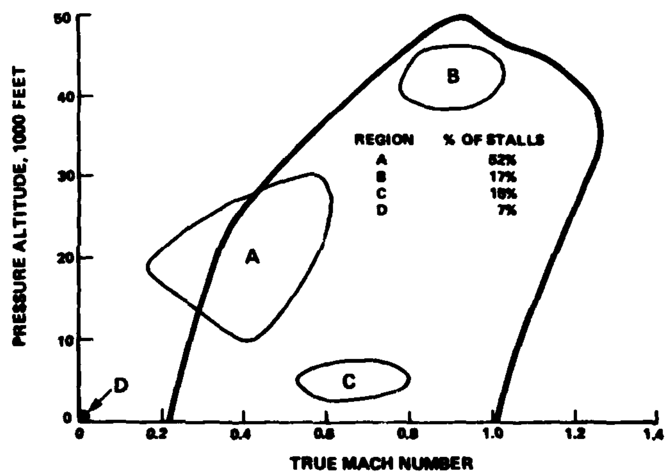


FIGURE 4: CF-5 LEVEL FLIGHT ENVELOPE SHOWING FLIGHT REGIONS WHERE COMPRESSOR STALLS OCCURRED (REF 1)

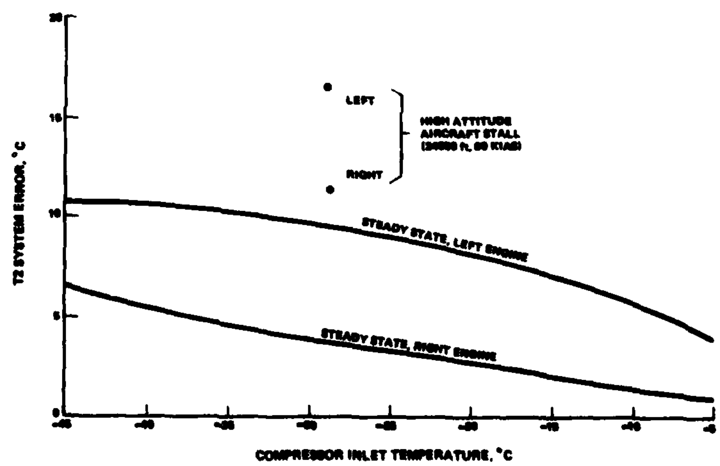


FIGURE 5: IN FLIGHT T2 SYSTEM ERRORS

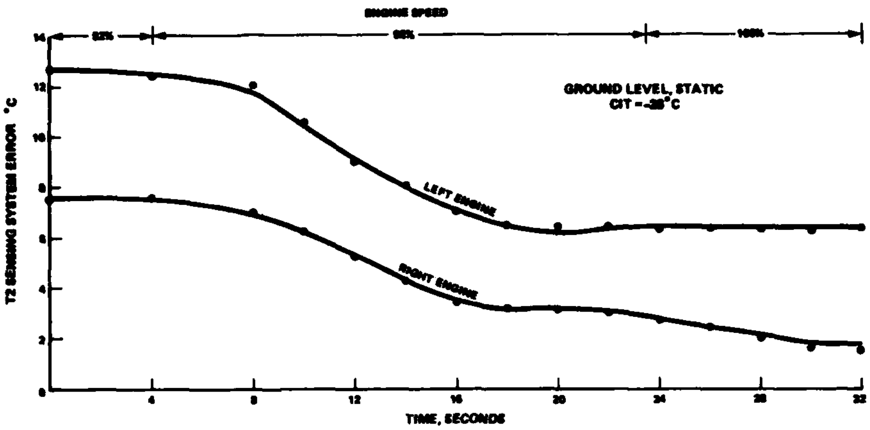


FIGURE 6: GROUND LEVEL STATIC T2 SYSTEM ERROR, EFFECT OF ENGINE SPEED

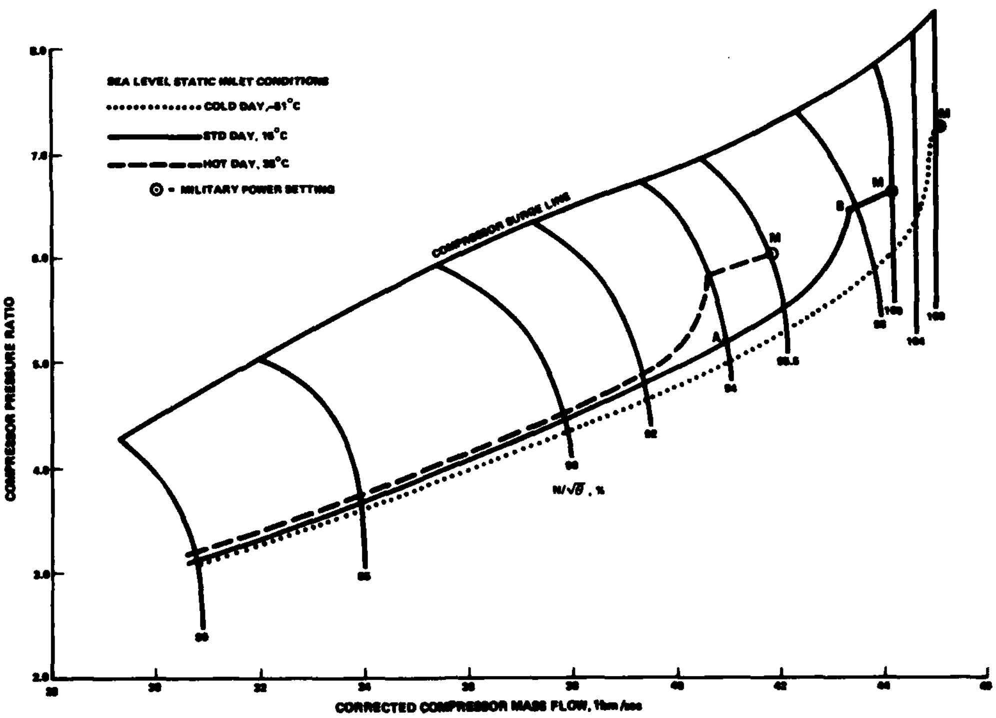


FIGURE 7: J85-CAN-15 COMPUTER GENERATED STEADY STATE COMPRESSOR WORKING LINES

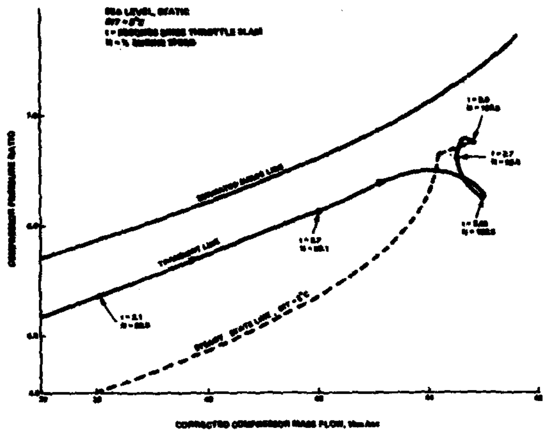


FIGURE 8a: TRANSIENT WORKING LINE; BURST, IDLE TO MILITARY

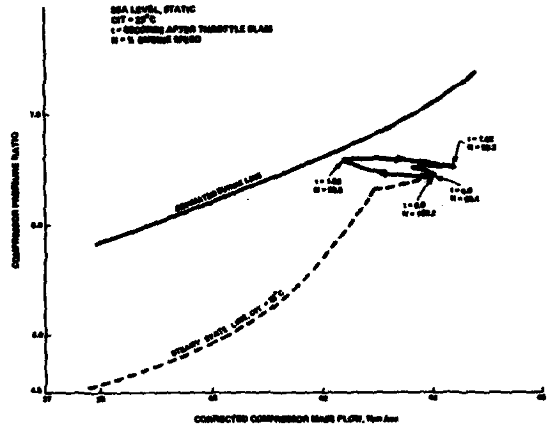


FIGURE 8b: TRANSIENT WORKING LINE; BURST, MILITARY TO MAX A/B

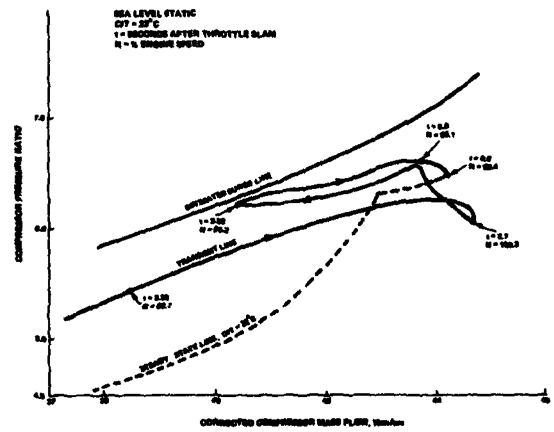
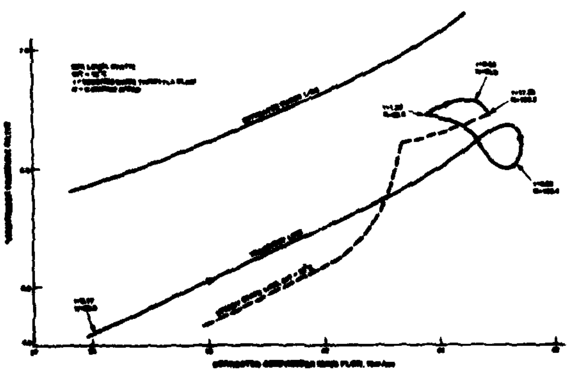
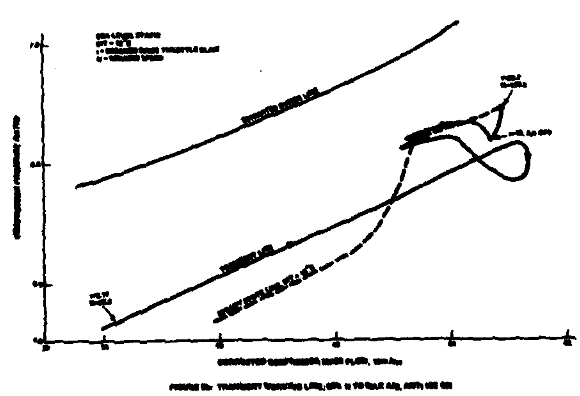


FIGURE 8c: TRANSIENT WORKING LINE; BURST, IDLE TO MAX A/B





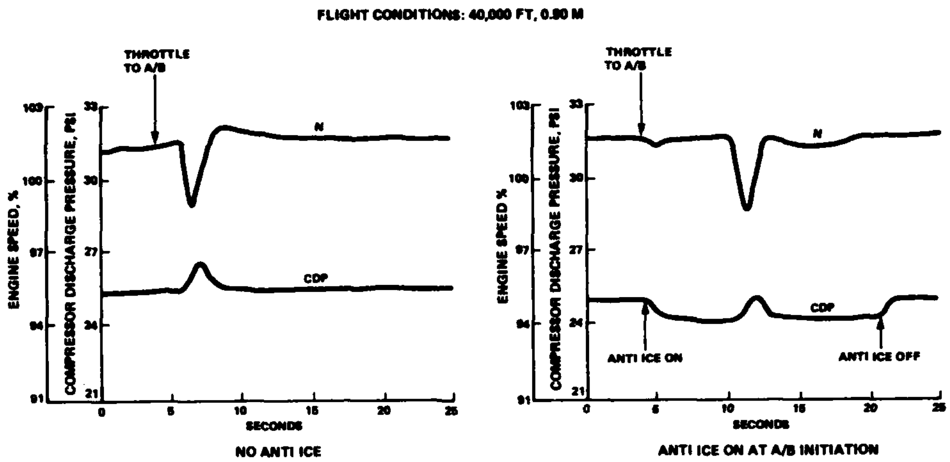


FIGURE 10a: IN FLIGHT A/B INITIATION, WITH AND WITHOUT ANTI ICE (REF 3)

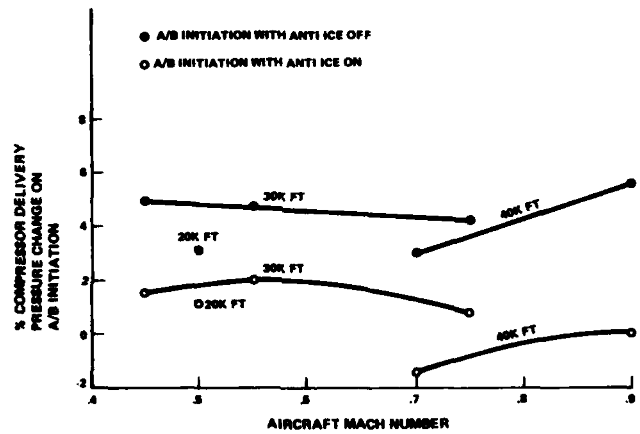


FIGURE 10b: IN FLIGHT A/B INITIATION WITH AND WITHOUT ANTI ICE USE

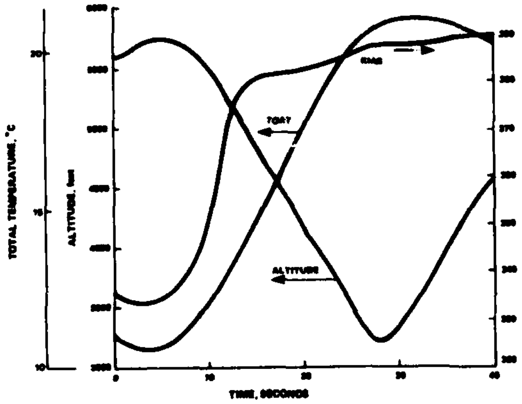


FIGURE 11a: FLIGHT PARAMETERS, STRAFE RUN

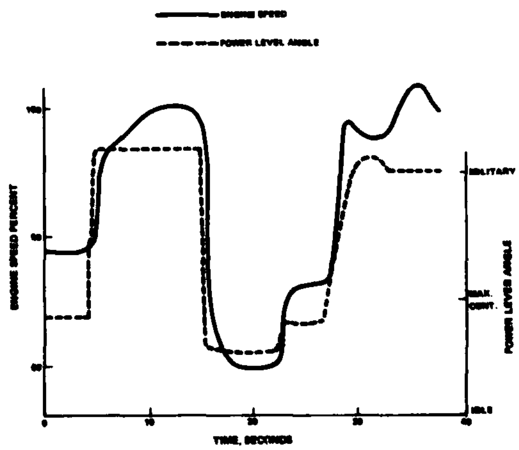


FIGURE 11b – ENGINE PARAMETERS, STRAFE RUN

PART		PREDICTED LIFE CYCLES (-3σ TO DYSFUNCTION)
1.	Compressor	
	a. Disc - Stage 1	17,000
	2	100,000
	3	70,000
	4	5,700
	5	4,500
	6	5,400
	7	5,600
	8	7,400
	b. Spacer - Stage 1	6,000
	2	6,400
	3	4,700
	4	5,500
	5	6,700
	6	6,800
	7	6,700
2.	Driveshaft	6,700
3.	Turbine Discs Stage 1	5,700
	2	6,800

FIGURE 12: J85 ROTATING COMPONENT *LCF* PREDICTIONS

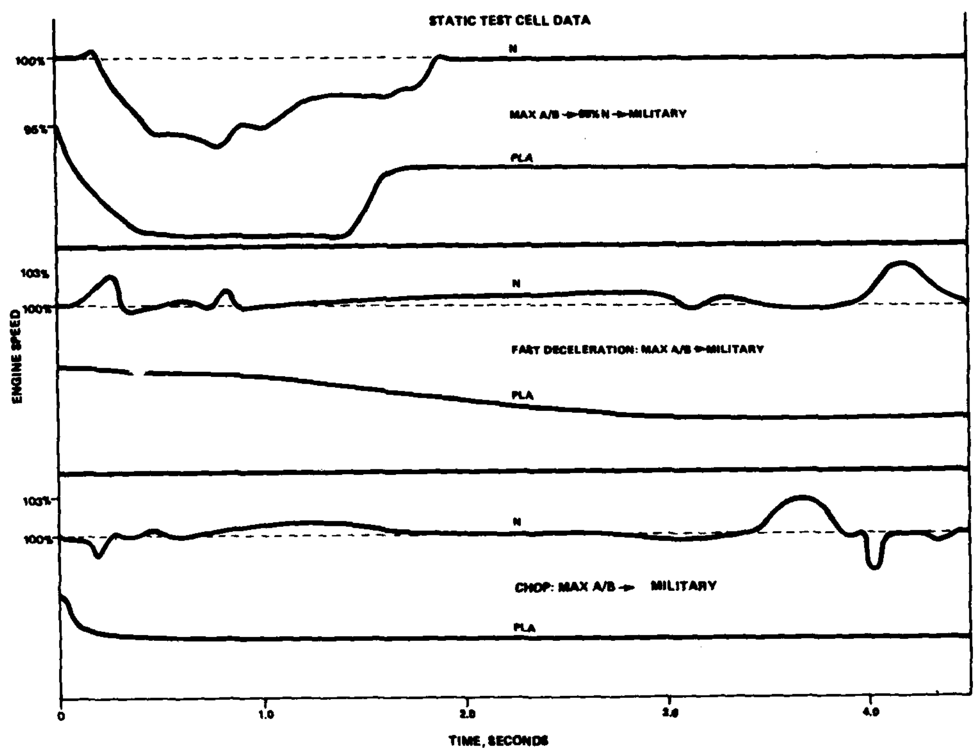


FIGURE 13: EFFECT OF A/B TERMINATION THROTTLE MOVEMENT ON ENGINE SPEED EXCURSIONS

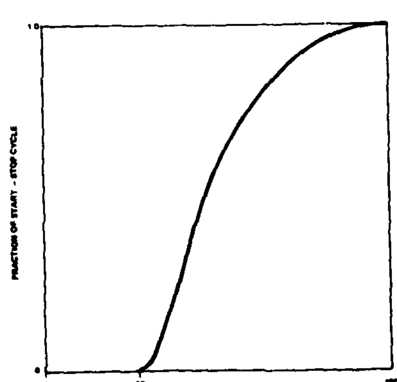


FIGURE 14 FRACTION OF FULL CYCLE VS SPEED CHANGE

CF-5 MISSION PROFILE DESIGNATORS

1. Pilot Familiarization/Basic Handling/Transition/Clearhood (Day or Night);
2. Formation (Day or Night);
3. Instrument Training/Ferry Flights (Day or Night);
4. Navigation Training;
5. Reconnaissance;
6. Air-to-Air Refuelling;
7. Air-to-Air Gunnery;
8. Dart Tow;
9. Advanced Handling/BFM/ACM;
10. Ground Attack on the Manned Range With or Without Practice Ordnance Dispenser;
11. Ground Attack on the Tactical Range With or Without Practice Ordnance Dispenser;
12. Ground Attack on the Tactical or Manned Range With Heavy Ordnance (Live or Practice);
13. Air Displays;
14. Test Flights;
15. Ground Run or Taxiing Only; and
16. Unclassified.

CT114 MISSION PROFILE DESIGNATORS

Training

1. Clear Hood Training
2. Formation (2 Plane)
3. Cross Country
4. Navigation
5. Instrument Flights
6. Test Flights

Demonstration

1. Ferry
2. Air Show

FIGURE 15

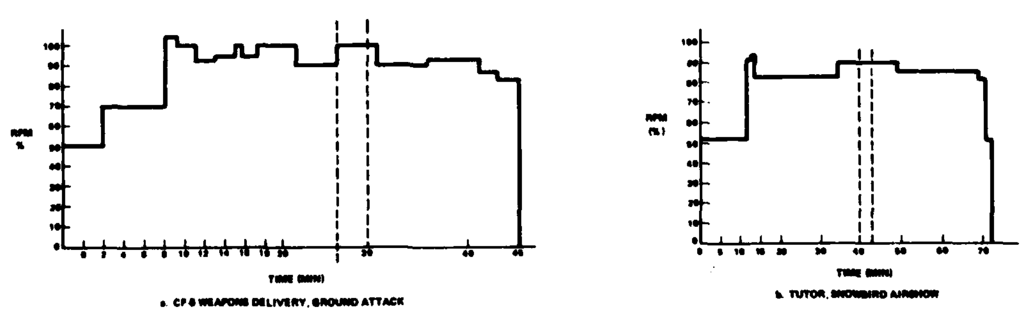


FIGURE 16: ENGINE SPEED EXCURSIONS, PILOT SURVEY DATA

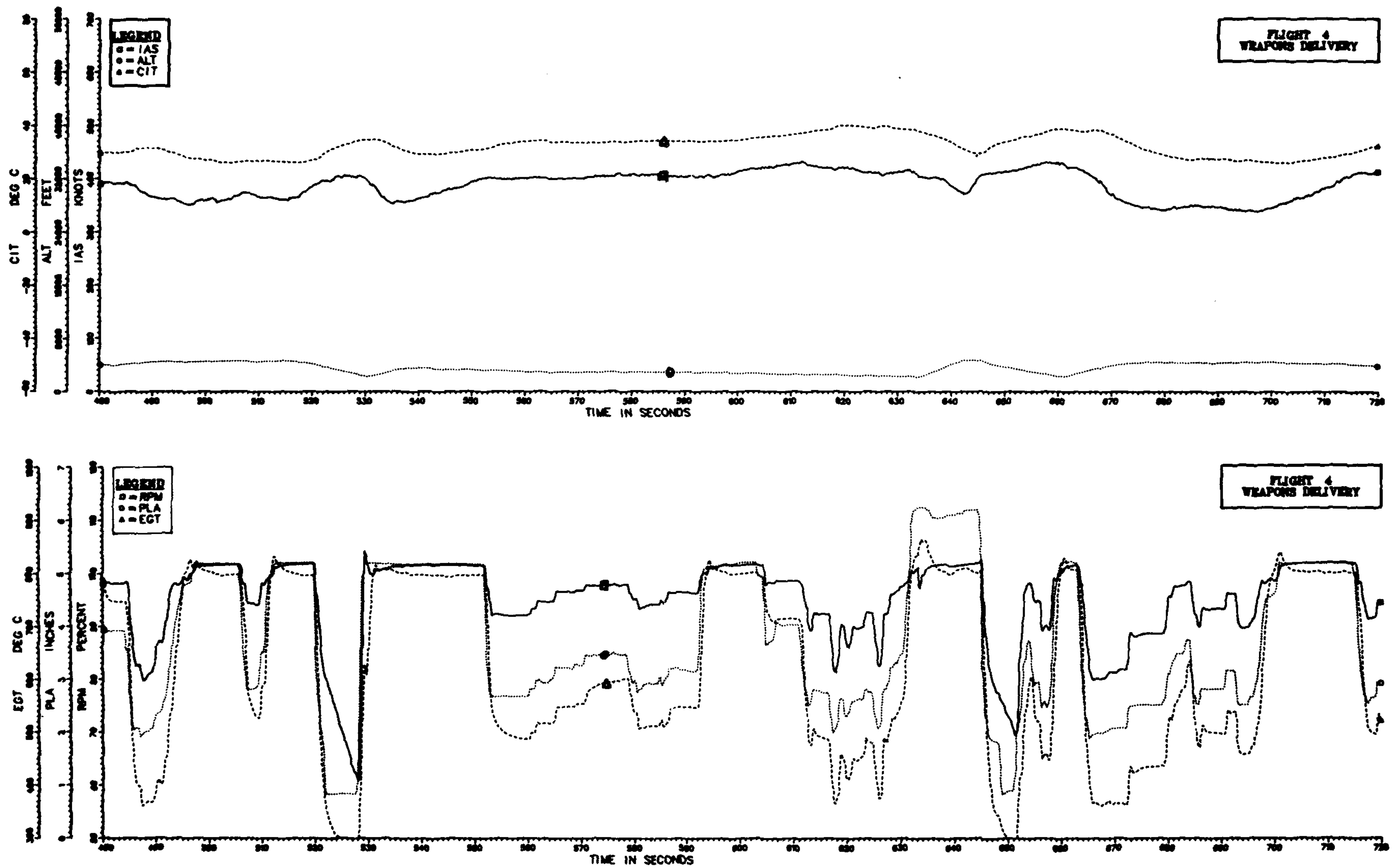


FIGURE 17: FLIGHT AND ENGINE PARAMETERS; CF-5 WEAPONS DELIVERY

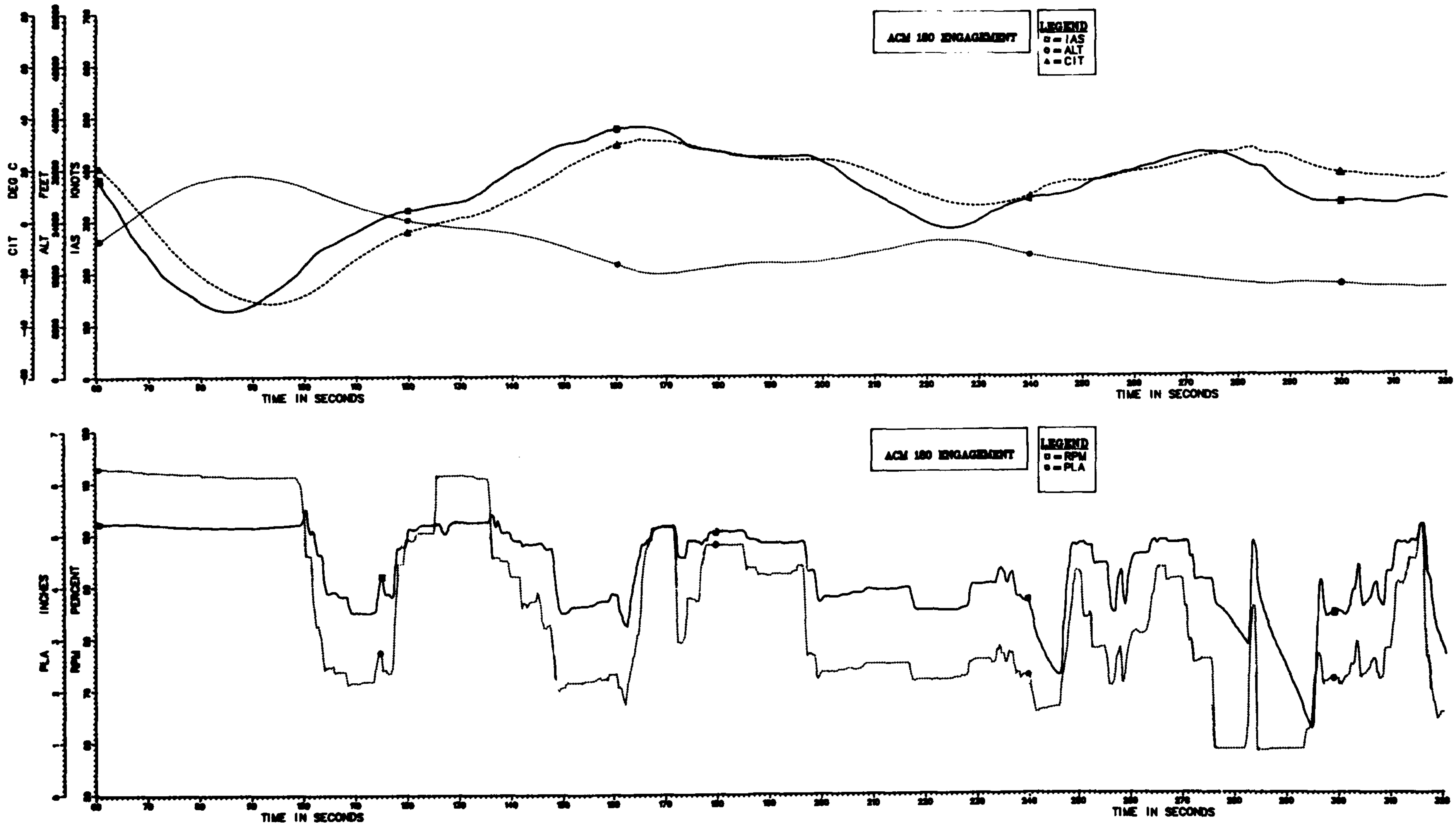
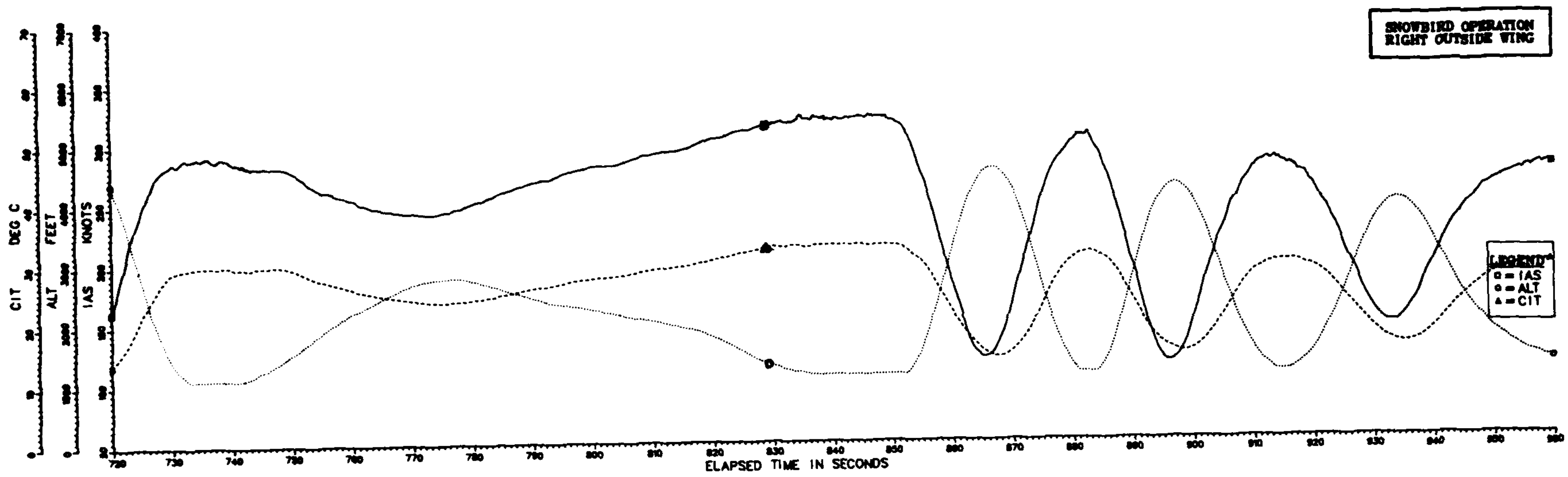


FIGURE 18: FLIGHT AND ENGINE PARAMETERS; CF-5 ACM - 180° ENGAGEMENT



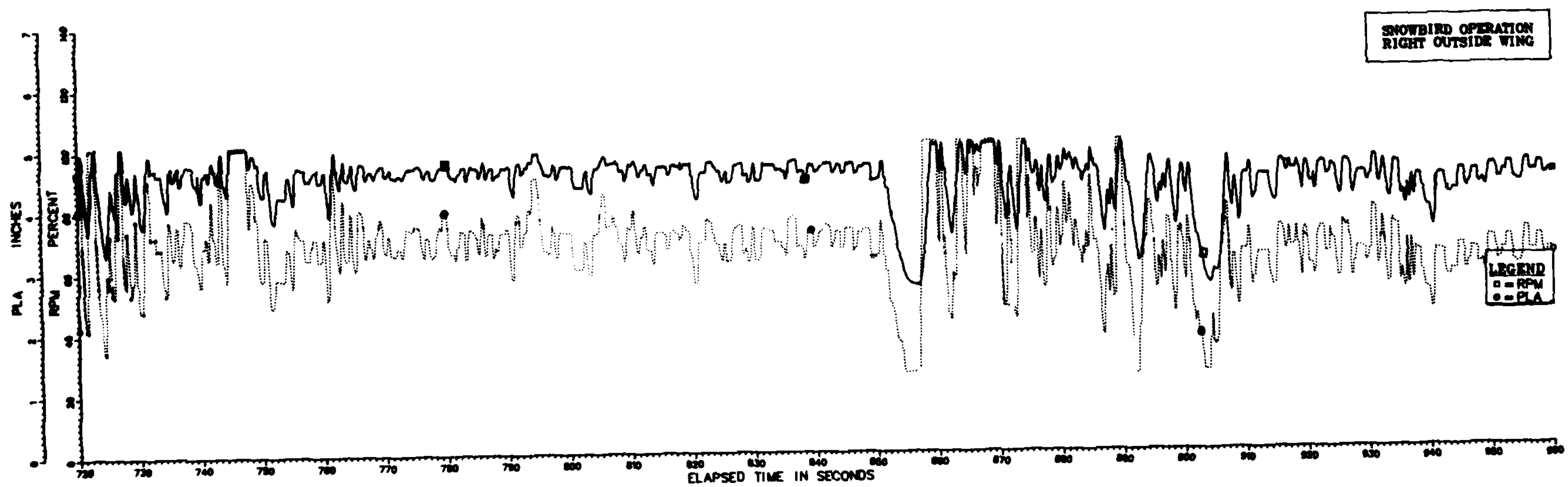


FIGURE 19: FLIGHT AND ENGINE PARAMETERS; PORTION OF "SNOWBIRD" SHOW - RIGHT OUTSIDE WING

DISCUSSION

J. Hourmouziadis, Ge:

Fig.8 and 9 show engine transients in the compressor map. How were compressor parameters measured during transients?

D. Rudnitski's Response:

1. The compressor pressure ratio P_{3s}/P_{2t} was obtained by measuring the static pressure at the compressor exit with a close-coupled strain gage type pressure transducer. The line length from the pressure tap to the transducer was less than 12 cm. The frequency response of the transducer was 6.000 hz.P_{2t} measured at the inlet to the bellmouth entry section.

2. Airflow was deduced by measuring the wall static pressure in the bellmouth entry section to the engine. Inlet total temperature and pressure were also measured and, knowing the flow area, the mass flow could be calculated. Again the strain gage type pressure transducer was close coupled to the bellmouth to ensure satisfactory pressure response.

3. The signals from the transducers were digitized by a mini-computer system which subsequently reduced the data and plotted the results.

OPERATIONAL AND MAINTENANCE ASPECTS OF THE INTRODUCTION OF AN ADVANCED FIGHTER TYPE

by

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INTRODUCTION

In 1979 the first F-16 aircraft, powered by a Pratt & Whitney F-100 - PW200 engine, entered service into the Royal Netherlands Air Force (RNLAF) at AFB. Leeuwarden. To-day AFB-Leeuwarden is fully equipped with two attack squadrons and a training squadron. Further introduction of the F-16 into the RNLAF is now taking place at AFB. Volkel.

The F-16 is replacing the F-104G, powered by the General Electric J-79 engine. The F-16 F100 is the first turbofan engine in use in the RNLAF. This paper will deal with those operational- and maintenance aspects, which were found significantly different from previous RNLAF experience.

ENGINE DESCRIPTION

The F-100 engine is a low bypass ratio (.67), high pressure ratio (24:1), dual spool, afterburning turbofan engine. It has a 3-stage fan, a 10-stage compressor, an annular combustor, a 2-stage high- and a 2-stage low pressure turbine, a mixed flow afterburner and a variable area balanced beam nozzle.

The F-100 is designed as a modular engine so that parts associated either physically or functionally can be removed as units.

Engine Control Systems

The primary engine control system consists of a hydromechanical Unified Fuel Control (UFC) and an Engine Electronic Control (EEC). The UFC receives information from the throttle, the core RPM sensor, the fan discharge temperature sensor and a number of other sensed parameters. With these inputs the UFC provides the proper basic engine operation from idle to maximum afterburner by providing output commands to schedule the rear compressor variable vanes, the nozzle position, the core engine- and afterburner fuel flow in the afterburner segments 1 through 5. The EEC finetunes the UFC's basic operating schedules in many areas to optimize engine operation. It is the primary source of commands for special functions which are mainly safety features. These features are:

- fan inlet guide vane scheduling;
- nozzle area reset, opens the nozzle with landing gear down at low throttle settings to minimize thrust in landing and during ground operation;
- minimum idle RPM control;
- stall recovery logic;
- RPM lock up at high Machnumbers;
- segment 5 afterburner lock out, prevents ignition of the last afterburner segment under certain airspeed/altitude conditions (see fig. [illegible]).

Apart from the primary engine control system the F-100 engine is equipped with a Backup Control (BUC), which has to be selected manually by the operator. The BUC schedules gas generator fuel flow (biased by altitude) and rear compressor variable vanes as a function of throttle setting. During BUC operation the fan inlet variable vanes are fixed in a cambered position and the exhaust nozzle is kept closed.

The engine starting system is self sufficient, employing a radial, single-stage compressor and a single-stage turbine, which operates on aircraft fuel. The Jet Fuel Starter (JFS) is capable of starting the engine on the ground as well as in the air. The power required to crank the JFS is supplied through a hydraulic motor, which is driven by two hydraulic accumulators. The accumulators are adequate for two starts on the ground during normal conditions and one start in flight.

The ignition system provides constant ignition of the gas generator and initial ignition for the afterburner. Ignition power is provided by an engine-driven alternator.

OPERATIONAL ASPECTS

Engine ground operations

The engine is started with assistance of the JFS. A second start attempt for the JFS is sometimes necessary and usually successful. The JFS turns the engine up to approximately 27% RPM, which is more than sufficient for proper main fuel pump operation and adequate airflow for the engine start. From selecting JFS it takes approximately 1 minute before the engine is operating at EEC closed-loop idle. Starting a warm engine (Shut down more than 30 minutes, but less than 3.5 hours ago) requires a motoring time of 40 seconds at 27% RPM before ignition is commanded by placing the throttle from off to idle. This is due to the rotorbow present in the engine. Hung or hot starts are extremely rare. During a hung start caused by an exceptional low fuelflow the pilot can increase the starting fuelflow with about 100 pph. to assist engine acceleration. A BUC engine start is sometimes performed to check out the system and to familiarize the pilot with the techniques needed for an inflight BUC start. The BUC start requires more pilot attention, because the fuelflow and hence the FTIT, during the start is controlled by the pilot via the throttle.

After start up the pilot has to verify proper transferring to - and operation in the UFC (EEC off) and BUC mode. Due to transfers from UFC to BUC at a too low RPM numerous flame outs occurred, however this problem has been cured by raising the RPM at which BUC is selected. Apart from this check the engine readings are checked on the runway at 80% RPM (since the brakes are not capable to hold the aircraft at military power) to verify proper operation before flight.

Although the EEC resets the nozzle to open with the landing gear down, which reduces the thrust by approximately 30%, the remaining idle thrust is still approximately 700 lbs. The brakes have to be used regularly to avoid too high taxi speeds. A reduced idle thrust circuitry was tested, but it showed an undesirable deterioration of the stall margin at near idle transients. It was therefore not installed.

Military thrust is sufficient for most aircraft weights and ambient temperatures to ensure safe abort conditions from rotation speed on a standard length runway. This has noticeable advantages with regard to noise abatement. Maximum afterburner take offs are only performed with heavy weight aircraft configurations and during engine check flights (to ensure proper engine operation on the ground).

Flight operations

At powersettings below afterburner the presence of the EEC is noticeable to the pilot, especially during closed formation flying. Shortly after a power adjustment is made and the throttle is left at the desired setting, the finetrimming of the EEC alters the thrust slightly and a new throttle input in order to maintain formation position is required. Especially during the first closed formation flights this phenomena introduces a tendency to overcontrol out of phase with EEC trimming. The problem is overcome by deliberately reducing the magnitude and number of power adjustments.

Engine response to throttle inputs up to military power is less direct than a turbojet engine, however the response is adequate for all flight phases. The slower response is most noticeable during the landing phase, in which large, rapid power corrections are sometimes necessary.

At higher altitudes the EEC will raise the idle RPM to keep the combustion pressure at or above a minimum set value. At supersonic speeds the idle RPM is raised further to avoid airframe inlet instability. At 1.4 Mach or greater the idle RPM is equal to military power RPM. At high altitude/low airspeed conditions the military power RPM and FTIT are reduced by the EEC in order to maintain stall margin. At high supersonic speeds the RPM and FTIT will increase slightly above ground level to maximize performance.

Selecting afterburner from military power normally results in light off within 2 seconds. When afterburner is selected from any other than military power throttle position the afterburner will not light before a certain decrease in RPM-rate is reached. An idle to afterburner transient takes about 4 seconds from throttle advance to light off at sea level. This time will increase somewhat with altitude.

The afterburner operating envelope is divided into three regions determined by flight tests and operational experience.

Region 1 : Unrestricted afterburner operation. Mislights, blowouts or stalls should not occur. Some afterburner rumble at lower speeds is possible.

Region 2 : Most afterburner transients can be expected to be successful. Afterburner mislights, light-off stalls, rumble-associated blowouts or stalls may occur during throttle transients into afterburner, but should not occur during steady state afterburner operation.

Region 3 : Steady state afterburner operation or afterburner cancellation only. Afterburner mislights, light-off stalls, blowouts and rumble-associated stalls are probable from all powersettings. Afterburner rumble and blowout are possible during steady state operation.

Segment 5 of the afterburner is locked out by the EEC as a function of combustor pressure and air temperature, therefore the line can vary by ±0.2 Mach.

During aircombat the pilot often manoeuvers from region 1 into regions 2 and 3 without realizing the consequences for afterburner operation. A subsequent stall is usually just one "Bang", because as soon as the EEC senses the stall it reduces the afterburner fuelflow to segment 1 regardless of throttle position. This usually clears the stall. The pilot has to retard the throttle to military power to reset the afterburner fuelflow logic. If the stall does not clear automatically, the throttle must be retarded rapidly to military power. This action will open the nozzle 10% wider than normal to restore the airflow. A stall in regions 2 and 3 is not regarded as abnormal operation and is not reported as long as the afterburner functions normally in region 1.

Inflight problems

Engine stalls are considered a problem when they occur in afterburner in region 1, or at other power settings at any speed and altitude. Non-afterburner stalls are very rare and are generally caused by inlet distortions (highly unlikely when the aircraft is flown within the authorized manoeuvre envelope) or control system malfunctions. One of these malfunctions could be the loss of the EEC input to maintain a sufficient high idle thrust. Afterburner stalls in region 1 can be caused by many reasons such as: high engine pressure ratio, ignition system problems, misscheduled engine compressor geometry, etc. The afterburner related stall rate in the F-16 is currently 3.5 per 1000 flight hours and is still decreasing.

Engine stall recovery technique is to rapidly retard the throttle from afterburner to military power where the stall recovery logic of the EEC is most effective. When the stall does not clear or occurs at a lower than military power setting, the throttle has to be retarded to idle. If the stall still does not clear the continuous stall cycles will result in a decay of RPM to a point where there is insufficient energy left for selfrecovery. This phenomena is called stagnation with the F-100 engine. Stagnation characteristics are: RPM decay below 60%, no engine response to throttle movements and an increasing FTIT. For recovery the engine has to be shut down and restarted. The current F-16/F-100 stagnation rate is approximately 1 per 10,000 flight hours. The stagnation rate has been reduced by the installation of a proximate splitter which brings the bypass stream separating device closer to the fan.

During back-pressure pulses from an afterburner associated stall only the outside of the fanblades are now affected. The inner (core) stream remains intact.

In order to obtain a successful UFC airstart several conditions have to be met. RPM has to be above 25% (Below 15% RPM main fuelpump pressure is insufficient) and FTIT should be below 700° C (to avoid a hot start). To ensure that the FTIT is below 700° C before RPM drops below 25%, it is essential that a stagnated engine is shut down as soon as possible to avoid excessive FTIT and subsequent cooling time. The RPM spooldown rate varies with altitude and airspeed. At low altitude the RPM spooldown is rapid and very little benefit is obtained by increasing the airspeed. At higher altitudes the spooldown rate is slower and varies considerably with airspeed. An airspeed of 350 KCAS should maintain RPM above 25%. Below 20.000 feet the JFS should be used to maintain RPM above 25% regardless of airspeed. If a stagnation occurs at very low altitude, time may be insufficient to complete the airstart before the aircraft has to be abandoned because of the time between stagnation and thrust restoration (approx. 1 min.).

Apart from stall/stagnation most other engine problems are countered by setting the throttle at midrange and switching off the EEC. The safety features provided by the EEC are then lost. Therefore RPM range is restricted between 80 - 88% RPM above 15000 feet and 70 - 88% RPM below 15000 feet until landing is assured.

When the initial problem is not solved by switching off the EEC the remaining option for the pilot is to switch to BUC. The BUC is installed to provide a "get home" - capability. Because of the simplicity of the BUC it has a restricted envelope.

The high altitude restriction is because of fuelflow scheduling inaccuracies due to the low levels of fan discharging pressure encountered. The high airspeed restriction exists because the rear compressor variable vanes are scheduled by the throttle only and not as a function of temperature. This may cause axial flutter of the vanes, which may result in compressor damage.

The BUC airstart envelope is more restrictive than the UFC airstart envelope.

The high altitude restrictions are based upon a combination of the above mentioned fuelflow scheduling inaccuracies and the main combustor fuel nozzle characteristics. The pilot sees very little engine response early in the airstart sequence. When the fuelflow gets high enough the fuel nozzles become more efficient and FTIT will rise very rapidly, leading to a probable hot start. The low airspeed restriction is based upon the difficulty the pilot has in advancing the throttle at a rate between too much fuel flow (hot start) and too little fuelflow (hung start). As airspeed decreases the margin between these two conditions gets smaller. Switching to BUC to solve engine problems or performing an airstart in BUC was until today never necessary in the RNLAF.

MAINTENANCE ASPECTS

New maintenance concept

Comparing the maintenance concept of the J79 and the F-100 engine shows a change from preventative maintenance to inspection maintenance. The goal of this inspection maintenance with the F-100 is to inspect the engine without the need for disassembly. The results of this are reduced manhourconsumption for preventive maintenance, increased engine/aircraft serviceability, lower possibility of introducing assembly failures into the engine.

A comparison of the maintenance schedule of the J-79 and the F-100 is presented in the table below:

		J-79	F-100
airbase level	installed engine	pre-, thru- and postflight	pre-, thru- and postflight
		oil sampling every 10 hours	oil sampling every flight
		50 hours inspect.	Hourly postflight (50 h)
	engine in shop	100 hours	Phase I (100 hours)
		200 hours	Phase II (200 hours)
		400 hours	-
depot level		800 hours	1350 cycles (±675 hours)

Each inspection of the F-100 consists of the preliminary inspections plus some extra checkpoints typical for that inspection.

For example: Phase II includes an Hourly Postflight, a Phase I and the extra checkpoints typical for Phase II.

To inspect the F-100 without disassembling several techniques are used, some new and some older techniques, the latter are used in another setup to provide a full scheduled inspection.

Borescope inspection

To visually check blades, vanes and combustion chamber.

Rigid borescope: with direct damage measuring capability to determine if an engine can remain in service or should be disassembled for repair. To inspect fan 2nd and 3rd stage blades, compressor 4th, 6th, 7th, 12th and 13th stage blades and turbine 1st, 2nd and 3rd stage blades.

Flexible borescope (150 cm length): with fiberoptic, to inspect combustion chamber, 1st stage turbine stator vanes and fuel nozzles.

Supersnake (flexible borescope, 260 cm length): to inspect the 2nd stage turbine vanes through the combustion chamber of an assembled engine.

JOAP - analysis (Joint Oil Analysis Program)

Immediately after every flight an oil sample is obtained by the crewchief and locally checked for metal wear (size of wearparticles less than 0,005 inch). To inspect bearings (nr. 1 to 5), gearbox and oilpumps.

Magnetic chip detectors

One detector on the gearbox and three on the oil pump. Visually checked for larger wearparts, to inspect bearings and gearbox for malfunctioning.

Test sets

Several types of testers are used for preventive inspections and trouble shooting, such as:

Engine Trim Box (ETB), to trim installed engines.

Mach Nr. Simulator, used in correlation with the ETB to provide electrical Mach. nr. signals.

Supervisory Control System test set (S.C.S.), for static and dynamic tests to check inputs to the EEC and effectors controlled by the EEC. The SCS-tester is one of the most important testers for trouble-shooting.

Ignition test set, provides power to simulate voltage for ignition (variable) to check stored energy in exciters and functioning of sparks.

Engine Starting System test set (E.S.S.), to test engine starting system and ESS electronic controller.

The results of these inspection methods in two years of F-100 use at AFB. Leeuwarden are:

Borescope-inspection

Found many cases of F.O.D.-damage during inspections or after 1st stage fan damage discovery (F.O.D.-rate is 20 times as high as J-79, due to runway and taxi upperlayer, small dent limits of blades and the low airintake position of the F-16). Several special inspections after a certain occasion (one time inspections) performed without disassembling the engine (turbine air seal).

JOAP-analysis

One occasion of a loose hub nut and a damaged compressor disk.

Chip detectors

One occasion recently, probably of a damaged nr. 1 bearing-housing.

Modular concept

The advantages of the modular concept are:

In case of corrective maintenance beyond airbase level allowance, the malfunctioning module of the engine is sent to depot, instead of the complete engine. Since introduction of the F-100 only two complete engines have been sent to depot and only for workload purposes.

Shorter repair time of the engine, because of the interchangeable modules.

An advantage in total workload is not noticeable.

Computer supported maintenance

Events History Recorder (E.H.R.)

On the engine is installed the EHR, which provides a direct readout of:

- Engine Operating Time (E.O.T. in hours, when F.T.I.T. > 260° C.
- time F.T.I.T. (Fan Turbine Inlet Temperature, measured at 3rd stage turbine inlet) > 922° C.
- time F.T.I.T. > 957° C (in 1/100 of an hour)
- Low Cycle Fatigue (L.C.F.), is number of cycles from 73 - 89% and return, i.e. going through a thermal cycle.

In addition the recorder provides warning indications for:

- N1 sensor fail (F.T.I.T. > 842° C).
- Hot Start (F.T.I.T. > 842° C and N2 > 46%).
- Overtemp. B (F.T.I.T. 3 sec > 1020° C).
- Overtemp. C (F.T.I.T. 1,5 sec > 1062° C).

Many inspections are based on the EHR-readings which has the advantage that inspections are based on actual engine usage, instead of predicted engine usage based on aircraft hours.

Maintenance data (349-form)

Every maintenance action is recorded on a form by the mechanic:

- Nomenclature of aircraft/engine/part.
- Type of maintenance (unscheduled, type of inspection, modification and others).
- Action taken (repaired, removed, replaced, calibrated and others).
- When malfunction is discovered (before flight, in flight, in inspection, functional checkflight, maintenance, calibration and others).
- Reason for malfunctioning (broken, cracked, foreign object damage, engine failed to start, hot start and many other reasons).
- Figures for manhour consumption.

Multinational Maintenance Management Data Collection System (M.M.M.D.C.S.)

A time sharing world wide computer system. In the system all the engines and important parts (81 in each engine) are loaded by part- and serialnumber. Updating the computer with:

- Aircraft/engine data, such as EHR-readings, manual cycles (cycle from 0% to intermediate) and the number of JFS-starts.
- Maintenance data (349-form).

This together with the loaded engines/parts provide a life time tracking for separated parts. Also tracked are modification status and engine configuration status.

The benefits of the computer-system are fully developed by the world wide input of figures, resulting in trends, which are easily recognized in an early stage, resulting in: investigations for malfunctions, control of supplies, modification design, control of configuration, change of inspection intervals, periodic replacementschedule of parts, updating books.

For local management the MMMDCS-system is useful to: control modification status, control of time change items, provide figures for monthly reports.

Trim procedure

The J-79 engine was a power controlled trim, the F-100 is an EPR-controlled trim.

$$\text{Engine Pressure Ratio} = \frac{\text{Afterburner inlet pressure}}{\text{Fan inlet pressure}}$$

(EPR = Pt6/Pt2)

The max. $N2/\sqrt{Tt2}$ (when Tt2 < 0°C) and max. F.T.I.T./Tt2 (when Tt2 > 0° C) are trimmed at intermediate on the EEC by setting the EPR at a certain value. The EPR is representative of thrust and stall margin.

The average of AFB. Leeuwarden engines after two years of use is raised from 5.5 to 7 clicks (setting of F.T.I.T./Tt2), with a maximum of 10 clicks.

EEC clicks (F.T.I.T./Tt2)	F.T.I.T. (°C) min.	F.T.I.T. (°C) max.
5	904	919
6	909	924
7	915	930
8	921	936
9	926	941
10	932	947
11	937	952

Engine checkflights

The number of engine checkflights is approximately 20 per 1000 flight hours. This high figure is caused by the regulation, that an engine checkflight is required not only for maintenance to major engine components and sensors, but also when a serviceable engine is removed from one aircraft and is installed into another aircraft (Engine Swapping). This swapping is common practice in the RNLAF when an aircraft is down for maintenance for a longer period.

The engine checkflight profile is similar to the profile used for most fighter aircraft. Slamaccelerations from idle to maximum power are performed at various airspeeds and altitudes. Also proper engine operation is checked during supersonic flight. Besides the primary operating mode, the other modes (EEC off and BUC) are also checked in the air for proper operation. As the JFS is an important feature during airstarts and flame out landings its maximum start altitude of 20.000 feet is also tested.

The main reason for engine rejection are afterburner sequencing stalls in region I. These stalls usually occur at higher altitudes when the last permitted segment is lighted; i.e. below the segment 5 lockout line segment 5, above the lockout line segment 4. Retrimming the engine's afterburner fuelflow is usually sufficient to cure the problem.

The effects on organization and personnel

Effects on the organization and personnel are:

- More engine specializations due to the use of testers, borescopehandling and interpretation, JOAP-analysis, MMMDCS-computeroperating.
- More forms have to be filled in by the mechanic: EHR-readings by the crewchief, maintenance history by all the mechanics.
- Every maintenance action is fully described in a Technical Order (T.O.). Maintenance on the J-79 was depending on the mechanics own skill, supported by the T.O.'s. The F-100 T.O.'s describe the engine from troubleshooting to repair and operational check outs.

CONCLUDING REMARKS

- The F-100 engine has brought many new technologies into the RNLAF on both operational and maintenance side.
- Operationally the safety features installed in the engine (EEC and BUC) are appreciated, on the other side pilots had to learn to live with a somewhat restricted afterburner envelope.
- Because of the different modes of operation possible, the pilot has to remember many limitations and restrictions in case of arising emergencies; it should be emphasized however, that the engine has a good safety record so far in the RNLAF.
- The new maintenance concept has proven to be of advantage compared with the old system.
- The high number of FOD engines is not due to the combination F-16/F-100 alone. The AFB Volkel FOD-rate is expected to be less.

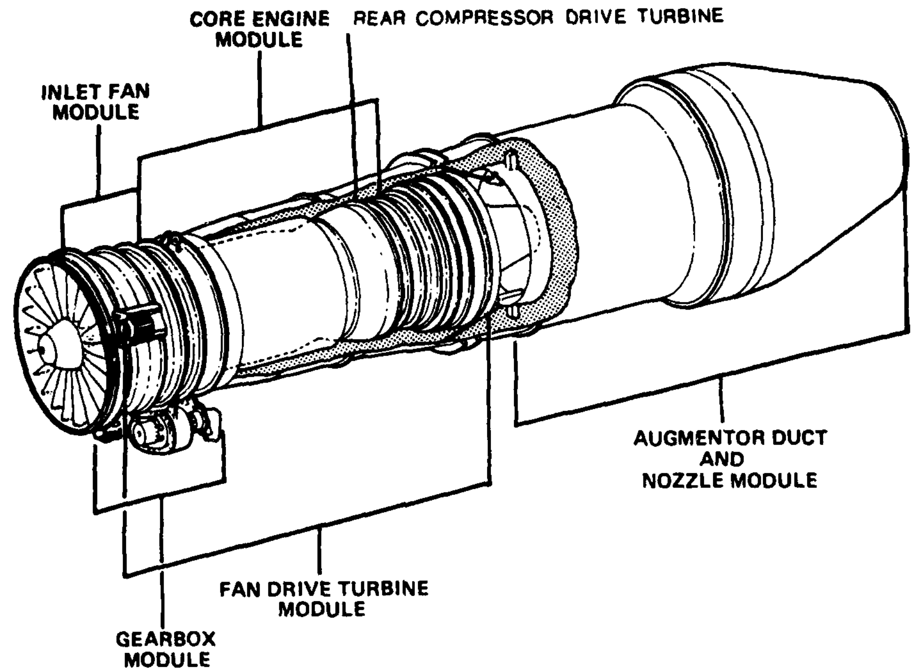

Fig. 1 Engine modules

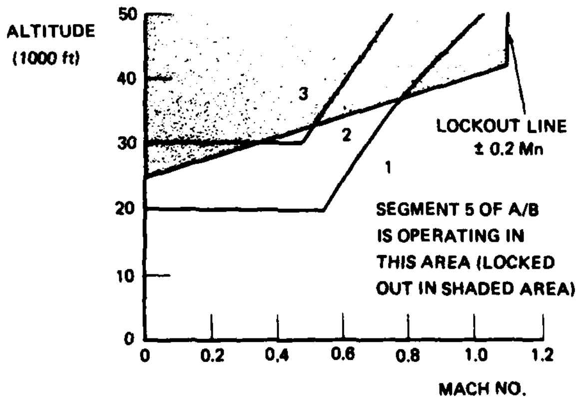

Fig. 2 Afterburner operating envelope

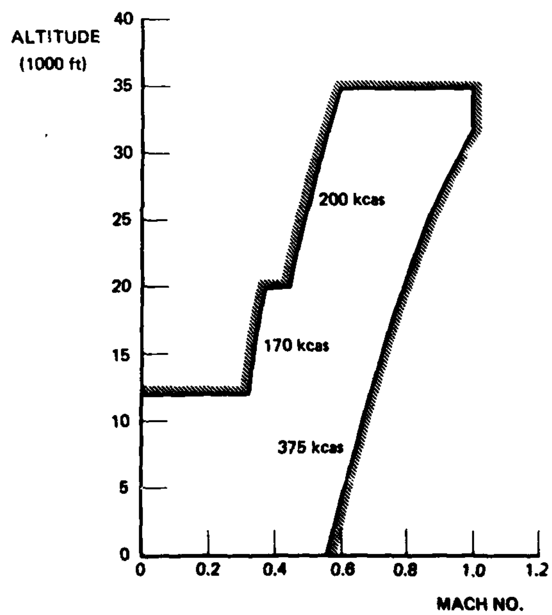

Fig. 3 BUC operating envelope

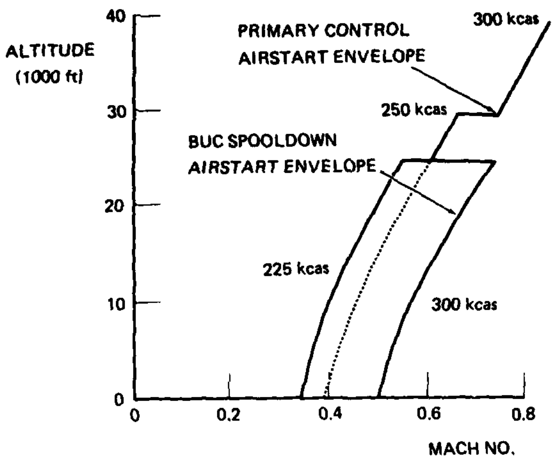

Fig. 4 Airstart envelopes

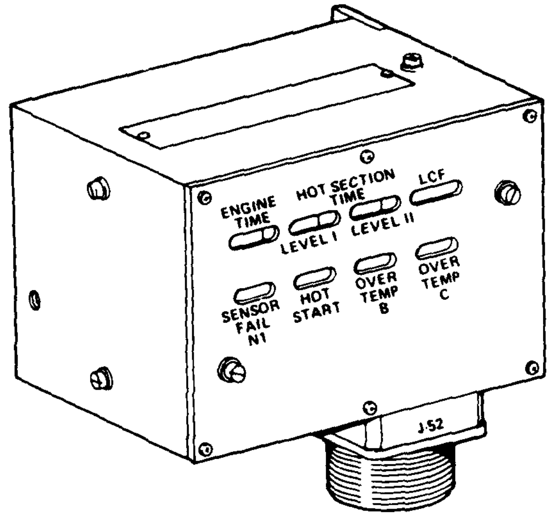

Fig. 5 Events history recorder

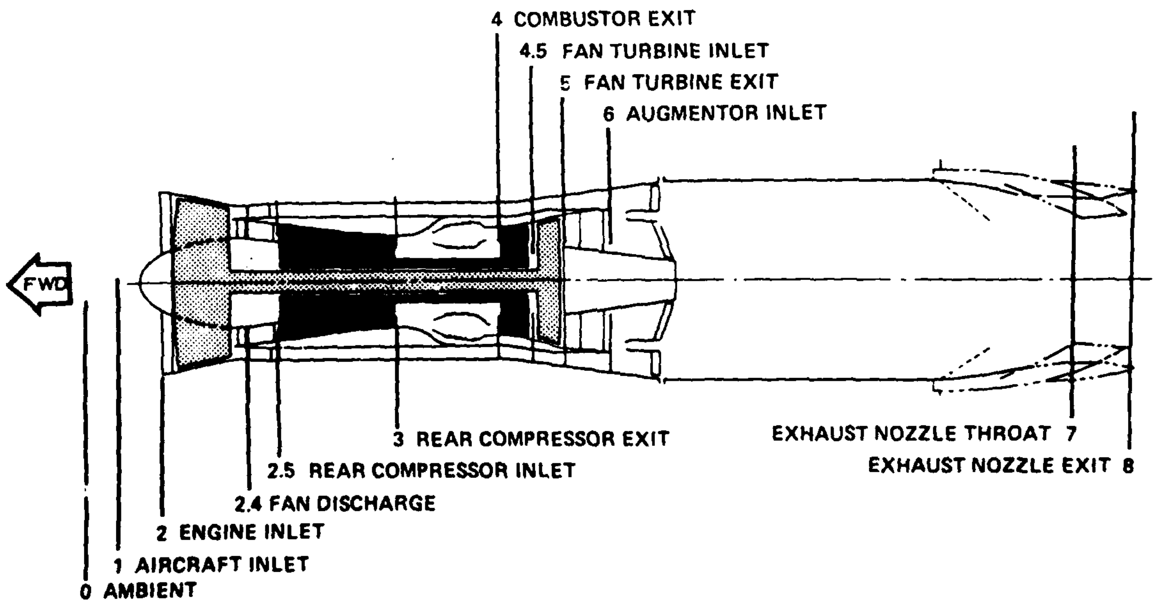

Fig. 6 Engine stations

DISCUSSION

A. Wulf, Ge.

What kind of ground running is required after module change; on aircraft – off aircraft?

Author's Reply

There is no additional maintenance required for change of afterburner duct and gearbox modules. For change of fan turbine, core engine and fan compressor module the engine is checked out in the test cell and will undergo a subsequent functional checkflight afterwards.

W. Heilmann, Ge.

Referring to Fig.2 – Afterburner operating range.

(1) What are the reasons for the limitations in the indicated areas (1); (2) and (3)?

(2) Have you got a buzz warning in the cockpit?

Author's Reply

(1) The afterburner operating envelope is divided into three areas based on the probability of A/B transient success. These areas were determined through instrumented flight testing and operational F15 and F16 experience. As the aircraft transitions to a higher region (higher altitude/lower speed), the A/B transient success rate decreases. This is due, in part, to changes in Reynolds number which reduces engine stall margin thus making it more sensitive to afterburner pressure pulses. In addition, fuel flow scheduling is biased by augmentor duct pressure and is therefore more sensitive to noise at higher altitudes.

(2) No.

G. Dahl, Ge.

You showed us on your history recorder, that you can find an NR-failure. Which procedure do you use to make the decision that the sensor fails? Do you have three sensors or do you have analytical redundancy?

Author's Reply

The Engine Electronic Control senses one failed fan speed sensor and will indicate this on the Events History Recorder for maintenance purposes only. The good sensor remains operational. The EEC is also capable of sensing a dual sensor failure and will indicate this to the pilot by means of a caution light. The pilot then has to observe EEC off limitations as described in the paper.

P.F. Neal, UK

What is the length of time necessary to crank a hot engine to achieve a successful start and does this affect subsequent capability to start the engine with the jet fuel starter at sea level and altitude?

Author's Reply

Motoring time after previous flight is only required when shut down took place between 30 min. and 3½ hours prior to start.

Whenever an immediate airstart is needed motoring time is not required. The Jet Fuel Starter will normally be at speed within 10 seconds up to its maximum starting altitude of 20,000 ft.

Overall you can say that the reaction time of the Jet Fuel Starter is not a factor in an airstart procedure.

E. Hienz, Ge.

When you do experience compressor stalls, is it the LP or the HP compressor that surges?

Author's Reply

Afterburner-related stalls normally are restricted to the LP compressor. The back pressure from the stall will reach the LP compressor via the duct and usually will cause a stall only on the outside of the blades. The core engine (HP compressor) airstream remains unaffected.

M. Rougevin-Baville, Fr.

Avez-vous sur F16/F100 des positifs ou des procedures de rallumage en vol rapide?

Author's Reply

Yes, we have. Actually this is the normal – so-called spool down – airstart procedure. All the pilot has to do is to place the throttle at midrange and the engine will start automatically providing remaining RPM was greater than 15% for fuel pump and ignition system reasons.

S. Olympios, US

a. Is your maintenance based on "condition" or on "hard times"?

b. Is your history recorder sufficient to provide diagnosis/prognosis?

c. What is available to trouble shoot the black boxes? Do you have a fault detection location subsystem to effect the above?

Author's Reply

a. F100 maintenance is based on the concept of "On Conditions"; that is, whenever a part is found to be unserviceable to the tech order limits it is removed. A replacement schedule based on "hard times" (cycles) is also used for "do not exceed" times or cycles.

b. The Events History Recorder presently in use on the F100 engine does not have diagnostic capability. A recorder with this capability was developed which would, upon command from the cockpit, record and store approximately 30 seconds of transient data from several pre-determined critical parameters. Maintenance personnel would be able to de-program this recorder after the aircraft landed. The USAF decided not to purchase this system.

c. The engine's black box is tested indirectly through the use of a ground tester. This tester permits the maintenance technician to perform all the functions of the engine black box manually. In so doing, the inputs to the black box are checked and then the technician manually activates solenoids or stepper motors the same as the black box would. If no problems are found the black box is assumed to be at fault and therefore replaced. Direct testing of the black box is only done at depot level.

EXPERIENCE WITH THE KHD APU T312 FOR A MODERN FIGHTER TYPE

by

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SUMMARY

A short description will be given of the APU T 312 as a lightweight single spool gasturbine. The installation into the aircraft compartment requires a very short overall lenght. The solution as to an optimized air inlet will be shown up. A survey will be given on in-service experience of the deterioration of compressor efficiency due to air contamination and eventual corrective actions. Cold start capability associated with modern aircraft lubricants will be discussed as well as a proposed inflight restart procedure.

1. INTRODUCTION

Requirements established in the late 1960's for new combat aircraft led to concepts which significantly differ from previous aircrafts. New concepts had to be conceived not only for airframe and engine but also for most of the equipment. Expecially for fighter type aircraft the concept of independance of ground support on the airbase led to the idea of a Secondary Power System (SPS) which contains its own power source by means of a small gas turbine, the Auxiliary Power Unit (APU).

This paper will report how the requirements for a powerful though light gasturbine were met by the APU T 312 and what experiences were gained in respect to installation and operation. In detail, the following topics will be discussed:

- Basic design of the APU
- Installation for optimized air inlet
- Contamination of the Compressor
- Cold start capability
- Inflight restart of the APU

2. BASIC DESIGN OF THE APU

Compared with large aircraft like bombers or transport aircraft the main and essential peculiarity of the equipment for a fighter type is the requirement for both extremely light weight and small size. Hence the APU has to follow this concept.

The SPS for the "Tornado" aircraft is shown in Fig. 1. Being powered by two main low bypass turbofan engines the aircraft has a SPS consisting of two gearboxes interlinked by a cross drive shaft. The starboard gearbox carries the APU which provides the energy for the whole SPS. For ground operation such as pre- and aft-flight check of the aircraft systems, for stand-by and for starting the main engines via a hydraulic torque converter, the APU delivers the necessary energy as mechanical shaft power. This is transmitted into the gearbox via a dry friction disk clutch which will be automatically engaged as soon as the APU has nearly reached its 100 % speed after start up. To compliment these SPS design features the APU T 312 has been defined as a single shaft gas turbine engine. Besides other characteristics, for example the constant output speed, such a single shaft APU has the great advantage of low weight and small size. (Ref. 01 and 02).

A view of the APU T 312 is given in Fig. 2 which shows the compact design. The APU has an axial air inlet. The exhaust duct is arranged to the upper side to cope with the installation arrangement in the aircraft. All necessary accessories are positioned onto one side and to the bottom of the APU.

Fig. 3 shows a cross-section with the main components: A two stage axial-radial compressor, reverse flow annular combustion chamber, two stage turbine, exhaust duct and a planetary type gearing which reduces the output speed to 8.000 rpm. The APU has its own oil pumps. The oil reservoir is in the Starboard gearbox to which the APU is connected by means of a V-clamp. The fuel supply is controlled by a hydraulic governor. The starting sequence as well as overload and overspeed protection are achieved by an integrated Electronic Control Unit that also manages the operations of the gearbox.

As was previously mentioned the T 312 is of small size and low weight. The overall lenght is 510 mm, including all accessories the diameter is 380 mm. The rated shaftpower is 105 kW with a 10 % contingency power. The ratio of rated power to weight is 2,7. Fig. 4 compiles the power to weight ratio of gas turbines produced to date. The APU T 312 is in comparison found to be most successful as regards this parameter.

3. INSTALLATION FOR OPTIMIZED AIR INLET

Because of the compact design of the figher aircraft the room available for the overall SPS installation is only small. Hence optimized air feed design for the APU was very difficult to achieve. Inspite of a total lenght of but 510 mm there was only a gap of 70 mm left between the bell mouth entry area of the air inlet and the adjacent wall of the bay for the SPS. That distance from the wall compared with the inlet diameter of 150 mm resulted in a ratio of less than 0,5, Fig. 5. In addition to this the airflow enters the compartment by an opening in the fuselage, only 270 mm away from the center line of the APU. This definitely leads to an inlet distortion followed by compressor surging and an intolerable loss of APU shaft power. Additional loss would result from elevated temperature of the air in the SPS bay due to heat rejection from the gearboxes and especially the outer turbine housing of the APU itself. To avoid these negative effects extreme design effort was required in this area. (Ref. 03)

From geometrical arrangement it is apparent that the only means for improvement was the application of a short inlet duct. This eliminated undue heating of the inlet air. Fig. 6 shows the duct and its location in the SPS bay as well as the flap with its hydraulic actuator. This flap will be opened during APU operation only, in flight it is closed to avoid drag losses. To overcome inlet distortion, guide vanes had to be located within the duct. To find the optimum, a series of tests was conducted on a compressor rig using a full scale model of this inlet duct, Fig. 7. In the inlet area of the axial compressor a rotating probe was located which had 3 sensors for the measurement of total pressure and 3 sensors for the measurement of flow angle. Besides this static pressures at the inner and outer diameter were measured. A glass window allowed to check flow pattern.

To compare test results for different designs the following data were calculated:

- Total Pressure
- Distortion Coefficient over an angular segment of 60^{o}; DC 60.
- Max and min deviation of the inlet flow angle.

Alltogether eight configurations with different guide vanes in circumferential, secantial and radial positions were tested, including the "clean" duct without any vanes. Results obtained with the "clean" duct, Fig. 8, showed an average loss of 4 % total pressure and a swirl in both the upper right and lower right corners. This could also be made visible with wool tufts. It was clearly demonstrated that means had to be provided to straighten the flow. The eight configurations tested are shown in Fig. 9, the results are compiled in Fig. 10, with the best values being underlined. In selecting the optimum, two aspects had to be considered:

- Shaft power is determined by:
  Total pressure, and
  Distortion Coefficient DC 60

- Surge margin is determined by:
  Inlet flow pattern, and
  Distortion Coefficient DC 60

The decision was finally made for configuration "h". It produced a tolerable Distortion Coefficient and as regards the flow pattern it provided for the lowest deviation from the straight inlet flow. For verification a test with configuration "h" was conducted on an APU. This confirmed the APU starting without surge and with a power loss in the range of 6 %. Finally it can be reported that experience with this inlet duct on production aircraft demonstrated surge free operation.

4. CONTAMINATION OF THE COMPRESSOR DUE TO POLLUTED AIR

As is well known from the theory of gas turbines the compressor absorbs a considerable part of the energy produced by the turbine. With a shaftpower turbine this is in the range of 2/3. To achieve low specific fuel consumption it is therefore mandatory to provide compressors with high efficiency. It is, however, necessary to maintain the high efficiency of the compressor throughout the whole time between overhauls (TBO). Two major influences are known which may foul the compressor. These are Foreign Object Damage (FOD) and polluted air.

An intolerable decrease of compressor efficiency by FOD can only be eliminated by a major repair or even by the replacement of the whole compressor. The magnitude of loss of compressor efficiency by polluted air depends upon several different factors.

- Size of the compressor.

The smaller the gas turbine the more will the compressor be susceptible to contamination. This is obvious because the physical size of the compressor and hence its surface which may get contaminated does not increase or decrease proportionally with power or air mass flow, i.e. a relative compressor surface for a small gas turbine is much larger than that for a big one.

- Location of the air inlet.
The location of the air inlet has an important influence. If the inlet is located under the body of the aircraft and the opening is showing to the ground,then the APU acts literally like a runway vacuum cleaner. The higher the inlet is arranged the less will the compressor be affected.

- Operating conditions.
Compressors of large main propulsion engines are less effected by contamination simply because they operate most of their running time in clean environment which always prevails in higher altitude.

For the APU concerned all three influences tend to the negative.

- Size
Being an APU of the light weight class the size of its compressor is very small by design.

- Location
As was shown before,the location of this APU in the very narrow SPS bay left no other choice than having its air inlet underneath the aircraft body. Only by these means the necessary distance between the inlet (from underneath the aircraft) and the exhaust (at the most upper side of the bay) could be realized.

- Operation
Acting only as a ground operated gas turbine the APU has to cope with the most severe conditions of polluted air. Another aspect particular to APU's is its short running time during one duty cycle. As soon as the APU has been stopped and cooles down the accumulated dirt can condense. Thus one layer after the other clogs onto the surface of the compressor.

The experiences gathered with the APU T 312 in respect to compressor contamination are not yet sufficient as the aircraft has entered actual service not too long ago. Todays results are complied from prototype aircraft. However, they allow to already indicate some tendencies. Fig. 11 shows the loss of shaft power versus APU running time. As to be expected,the influence of air pollution can clearly be seen. Compared with actual aircraft service, operation on a testrig usually provides clean air which is even partially filtered by the air inlet silencers which are obligatory because of the noise abatement laws of the local communities.

With some of these APU's compressor washing was conducted. The washing procedure was not optimized,neither to the washing fluid nor to the method, the same procedure was applied as is still in use with the services. Fig. 12 demonstrates the loss of shaft power that remains after the washing procedure had been applied. Though with only a few results it can already be seen today that a washing interval of approximately 20 APU running hours results in a remaining loss of nearly zero, whereas with 50 hours intervals for example the remaining losses are in the range of 10 %. This, however, leads then to a continous accumulation of loss versus the time, resulting the following disadvantages:

- Higher fuel consumption

- Higher Turbine entry temperatures than rated. This is possible because of the contingency rating which is designed for short time peak load capability only. This in the end leads to

- Shorter Life of all hot parts, expecially the turbine rotors.

- Earlier maintenance as soon as the contingency rating has been used up and the APU shuts down because of the overload protection device.

Actions that need be taken to avoid these disadvantages are:

- Provide compressor washing intervals short enough to prevent remaining losses to accumulate.

- Use better chemical washing fluids to definitely yield improved and even longer lasting results.

- Investigate in more depth the application of "fired" washes with full speed instead of "cranked" washes with 15 to 20 % speed.
(Note: "fired" means the APU is running with the combustion chamber light up. "cranked" means cranking the APU with the electric starter motor only but without igniting the combustion chamber)

- Further potential areas of improvement are, soaking periods between two washing procedures or the method of inducing the washing fluid with an internal spray ring or only a single spray tube. Last not least an external spray may be sufficient.

5. COLD START CAPABILITY

Requirements of modern aircraft demand a cold start capability down to -40°C. Two points have to be considered for the starting phase of the APU:

- The mechanical resistance and the ventilation losses, the so called rotor drag which the starter motor has to overcome.

- The combustion chamber which has to light up completely.

As to rotor drag the Specification called for the same oil to be used for both the main engines as well as for the SPS. Therefore the APU has to cope with an oil MIL-L 23699. The viscosity of this oil, Fig. 13, increases by a factor of 150 when the temperature decreases from +15° C to -40° C. As a result of this drag increases. Fig. 14 displays the rotor drag lines for both temperatures. The performance of the electric starter motor is nearly constant over this temperature range resulting in steady-state cranking speeds of 3 % at -40° C and 20 % at +15° C. Due to the APU control system the fuel does not begin to flow before 8 % speed is reached. Therefore additional steps had to be taken to safeguard APU light up with temperatures in the range of -40° C. Because of weight it was not possible to increase the power of the electric starter motor. The problem could also not have been solved by an earlier light up of the combustion chamber at for instance 5 % speed as the point of self sustained speed is not lower than 15 %. The turbine would have to fill up the power deficit with even more overtemperature than anyhow prevails during the start up phase. The only way left was to reduce the rotor drag.

In Fig. 1 the power transmission from the APU to the gearbox is shown. As soon as the APU has reached nearly full speed the clutch will be engaged. For this engagement the oil supply pump of the APU has to provide oil flow of a pressure of 30 bar. Tests had shown that this pressure already builds up at 2 % speed. Therefore a device was developed to deload the pump during the starting phase, still ensuring full pressure in time for clutch engagement. During the whole procedure, however, a min level of pressure is maintained in order to guarantee the lubrication of the APU.

The second feature for successful cold starting is early and all round light up of the combustion chamber. The earlier the combustion chamber lights up the more the turbine will assist the starting procedure. Another important factor is high efficiency of the combustion process especially during start up. This will be achieved only if the combustion chamber lights up all round immediately after activation of the torch igniter. Fig.3 also shows the cross section of the combustion chamber. It is of the annular reverse flow type. Fuel will enter by twelve vaporizer nozzles. The torch igniter consists of a high energy spark plug and an adjacent atomizing swirl spray nozzle. This torch igniter will light up safely for a variety of fuels including Jet A 1 and even at very low temperatures. However, it is known that vaporizer nozzles are slow and rather unwilling to light up at very low temperatures. The reason is obvious. Whereas once the combustion chamber is burning these nozzles show excellent behaviour, with cold fuel and cold air no vaporization takes place. At these starting conditions only those nozzles start to fire which are directly within the flame cone of the torch igniter. This then explaines why only a few, say three or four nozzles start burning. Those nozzles, however, do not spread a penetrating flame to their neighbours. In consequence the combustion chamber does not light up around the whole circumference. Test showed this effect with fuel Jet A 1 and temperatures below -26° C. It therefore became necessary to introduce two further starter nozzles of the atomizing swirl type, altogether three evenly spaced, Fig. 15. With this arrangement a safe light up at temperatures even at -40° C is reached.

The experience with the aircraft in actual cold environment verfied a safe start of the APU after both design features - oil pump management and additional starter nozzles - had been introduced.

6. INFLIGHT RESTART OF THE APU

For the fighter type aircraft as it was specified, no requirement was set for an inflight restart of the APU. Tests, however, had demonstrated that an inflight restart was possible over a wide range of the flight envelope. The tests reported in this paper were conducted with the T 312's predecessor, an APU for VTOL aircraft. The core engine of both powerplants is identical. Though in the end not applied to the fighter type APU, it is deemed interesting to discuss this performance parameter.

The whole test series were conducted in a high altitude test chamber. This provided the altitude pressures together with the temperature conditions and could deliver air flow up to Mach 1. Fig. 16 shows the envelope of successfully tested starts. When these tests were performed the question was still open whether the starter was required throughout or if starting by windmilling only was possible. Therefore tests were performed in two modes.

- Starts with the electric starter motor
  with and without ram air.
  Result: Inflight restart is possible up
  to an altitude of 11.000 Meters.

- Starts with ram air only.
  Result: Inflight restart is possible to the right of the line marked ▼.

If windmilling restart of the APU were desired it has to be noted that naturally a special ram air inlet has to be arranged for an effective air pressure recovery. Also the sequence of opening the inlet flap has to be subjected to a function of the APU run up speed.

Fig. 17 finally shows available power of the APU versus altitude. It can be seen that for emergency cases a useful operation of the APU is possible up to a flight altitude of 4500 to 7500 meters only, as the SPS then requires about 80 kW. This would be perhaps sufficient for civil aviation in an extreme emergency case. For a fighter type, however, means other than the APU have to be adopted.

7. CONCLUSIONS

Modern fighter type aircraft of today require a light but powerful APU as an energy source for the SPS. A small single shaft gas turbine can best fulfill this task.

The installation of an optimized air inlet leads to configurations of an inlet duct with an arrangement of guide vanes to yield the best compromise between pressure loss and inlet flow distortion.

Contamination of the compressor due to polluted air demand better compressor washing procedures to avoid decay of efficiency followed up by higher fuel consumption, higher turbine entry temperature and hence shorter life of all hot parts.

Cold start capability even at very low temperatures can sufficiently be reached with modified oil management schemes during start up. This will be the more important as new type of lubricants often require oil of higher viscosity. Small gas turbine having fuel vaporizer nozzles, demand reliable and complete light up of the combustion chamber during the starting phase. This is reached by introducing additional swirl type starter nozzles which will be closed near the end of the starting phase.

A proposed procedure shows that in a wide range of the flight envelope a restart of this APU is possible. In the end this was not applied to the fighter type as a special ram air inlet could no be realised.

Finally as a result of thorough development the handling experiences with this APU in actual aircraft service show that operational requirements are reliably met.

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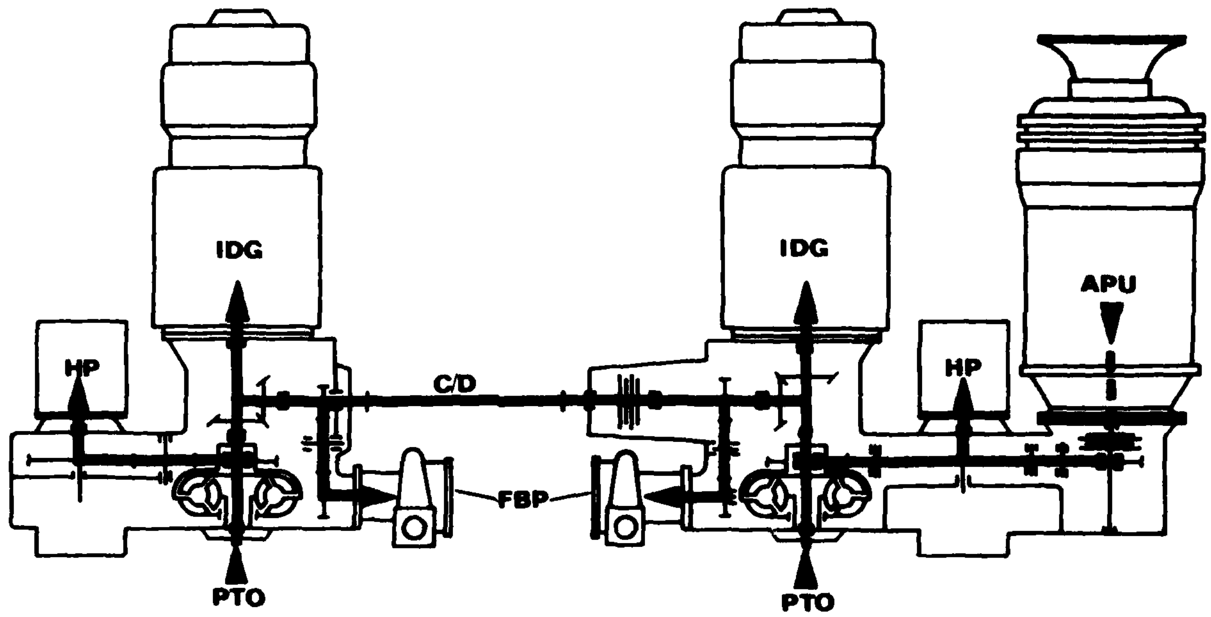

FIG. 1 SECONDARY POWER SYSTEM

FIG. 2 APU T 312

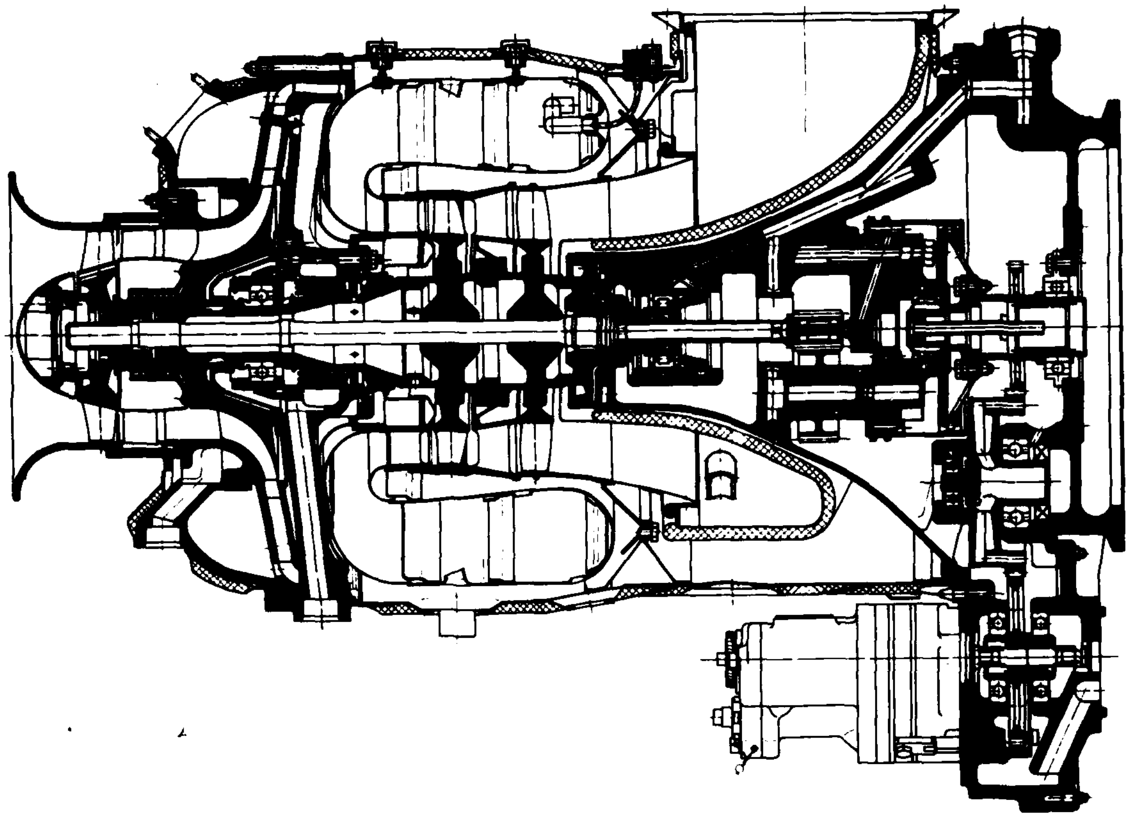

FIG. 3 APU T 312 CROSS SECTION

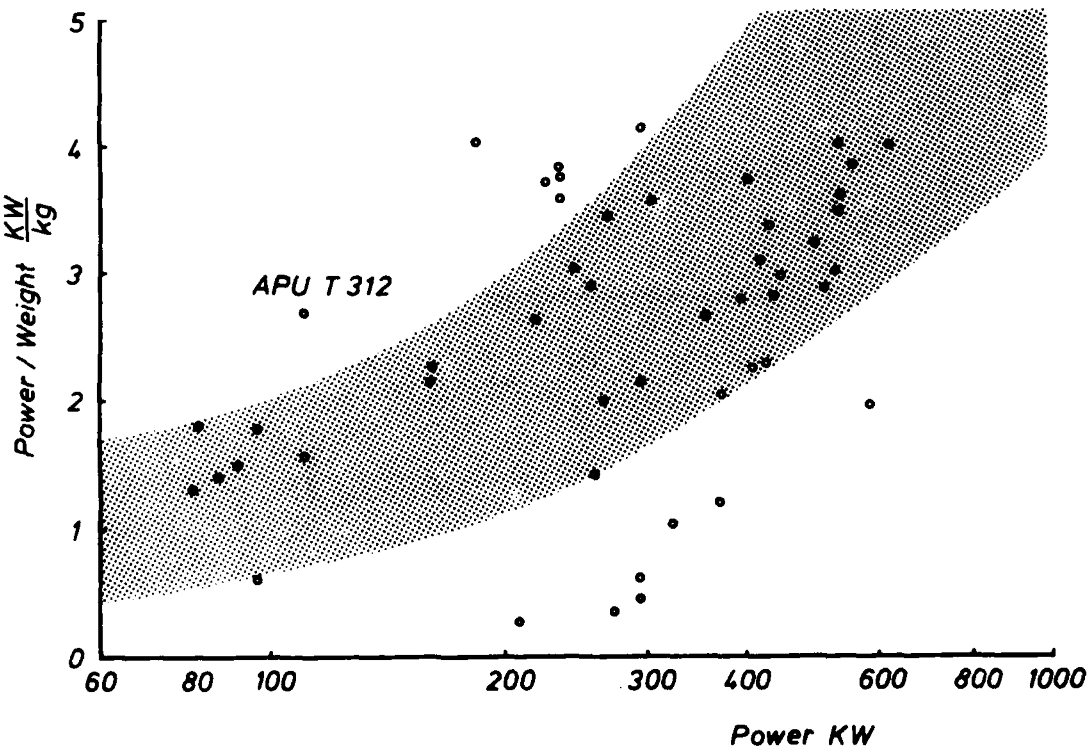

FIG. 4 POWER TO WEIGHT RATIO OF PRODUCED GASTURBINES

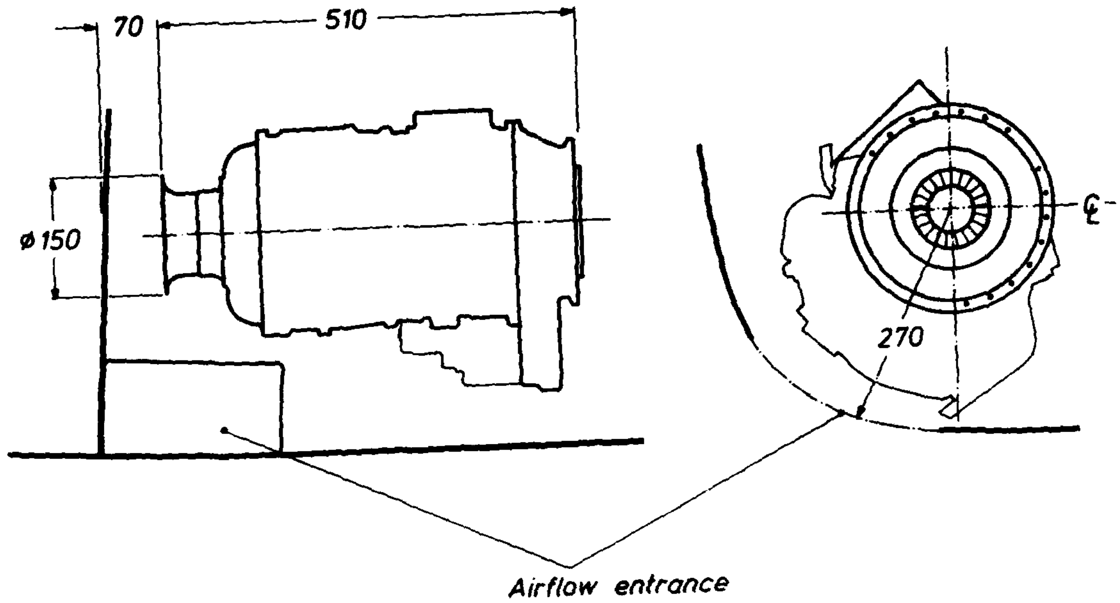

FIG. 5 APU INSTALLATION IN AIRCRAFT SPS - BAY
- without inlet duct -

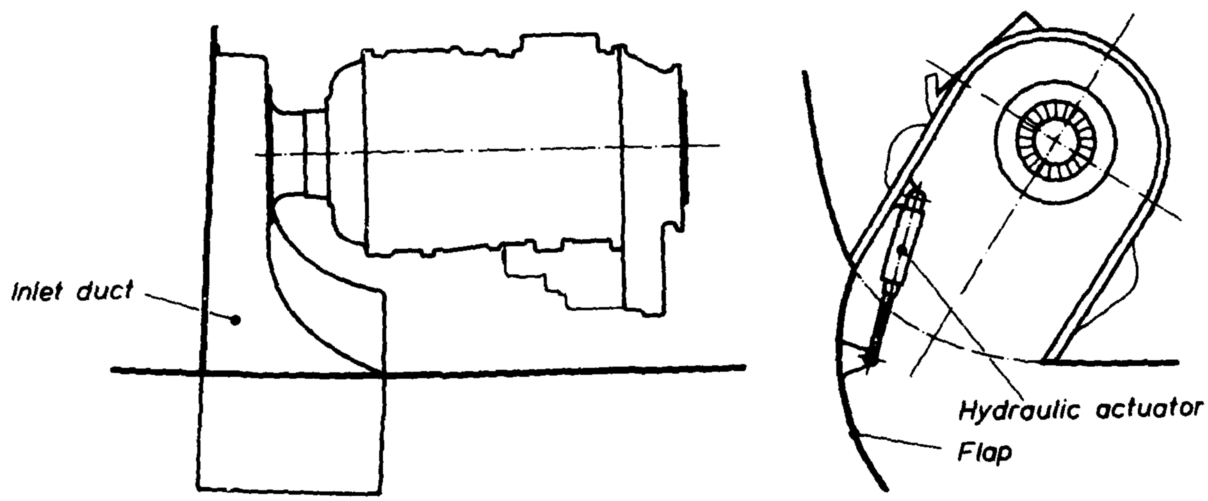

FIG. 6 APU INSTALLATION IN AIRCRAFT SPS - BAY
- with inlet duct -

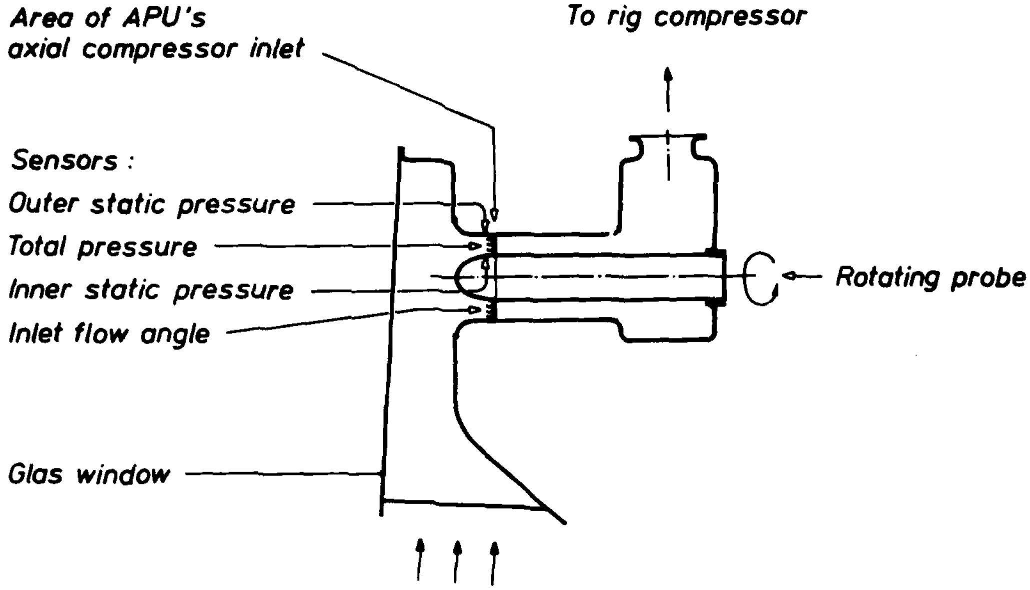

FIG. 7 APU INLET DUCT ON TEST RIG

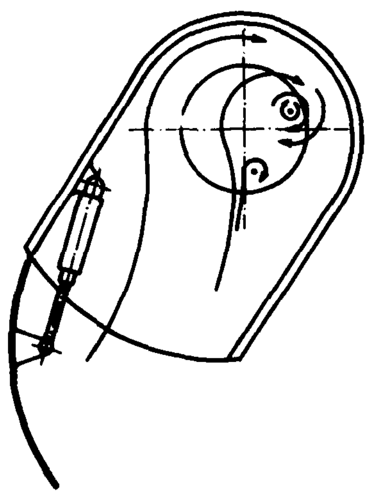

FIG. 8 TEST RIG RESULTS WITH „CLEAN" DUCT

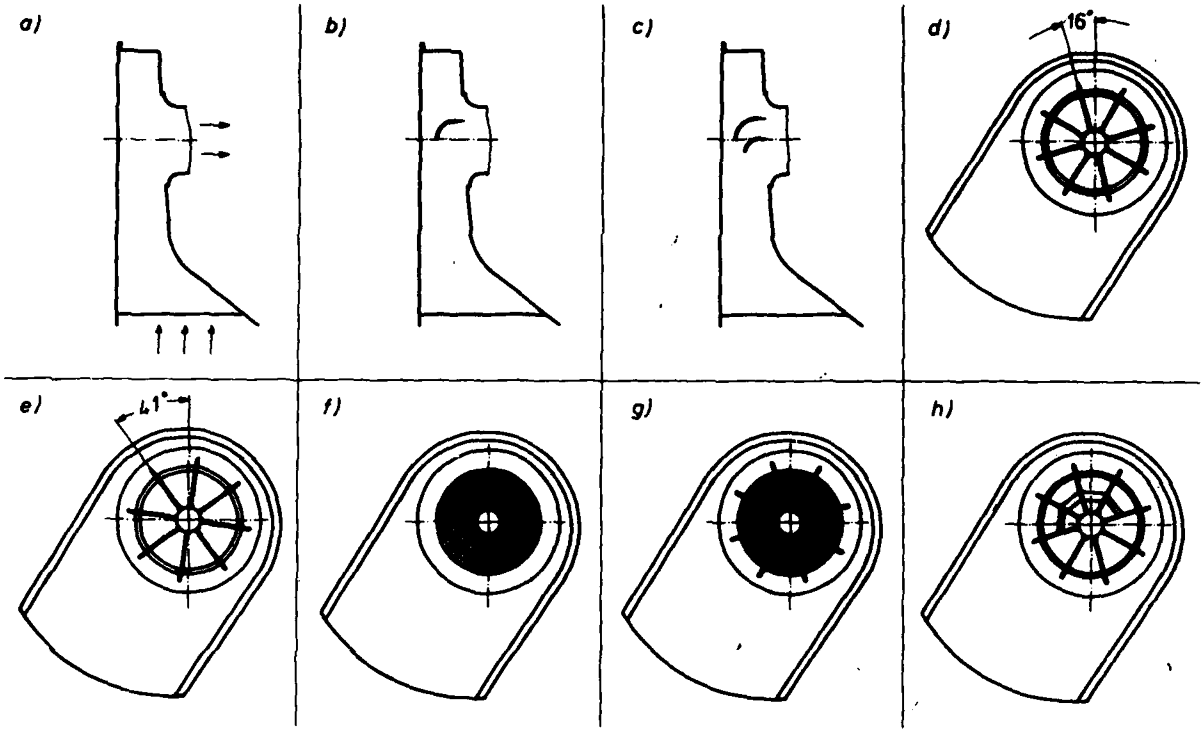

FIG. 9 APU INLET DUCT TEST CONFIGURATIONS

	CONFIGURATION	TOTAL PRESS (relat.)	DC 60COEFFICIENT max	min	INLET FLOW ANGLE DEVIATION	
a	"CLEAN DUCT"	0,9614	0,1995	-0,7249	+ 6,2	- 16,5
b	1 GUIDE VANE, secant.	0,9750	0,1256	-0,4825	+ 9,4	- 10.8
c	2 GUIDE VANES, secant.	0,9884	0,0538	-0,1207	+ 7,2	- 6,0
d	8 GUIDE VANES, radial,Pos.I	0,9922	0,0227	-0,0714	+ 5,0	- 6,0
e	8 GUIDE VANES, radial,Pos.II	0,9919	0,0406	-0,0500	+ 5,0	- 4,8
f	SCREEN, MESH 2mm	0,9743	0,0642	-0,2295	+ 6,0	- 14,6
g	COMBINATION OF d) and f)	0,9751	0,0343	-0,0557	+ 5,5	- 6,2
h	8 GUIDE VANES,radial,Pos. I + 4 circumferential VANES	0,9884	0,0464	-0,0126	+ 3,3	- 4,1

FIG. 10 APU INLET DUCT TEST RESULTS

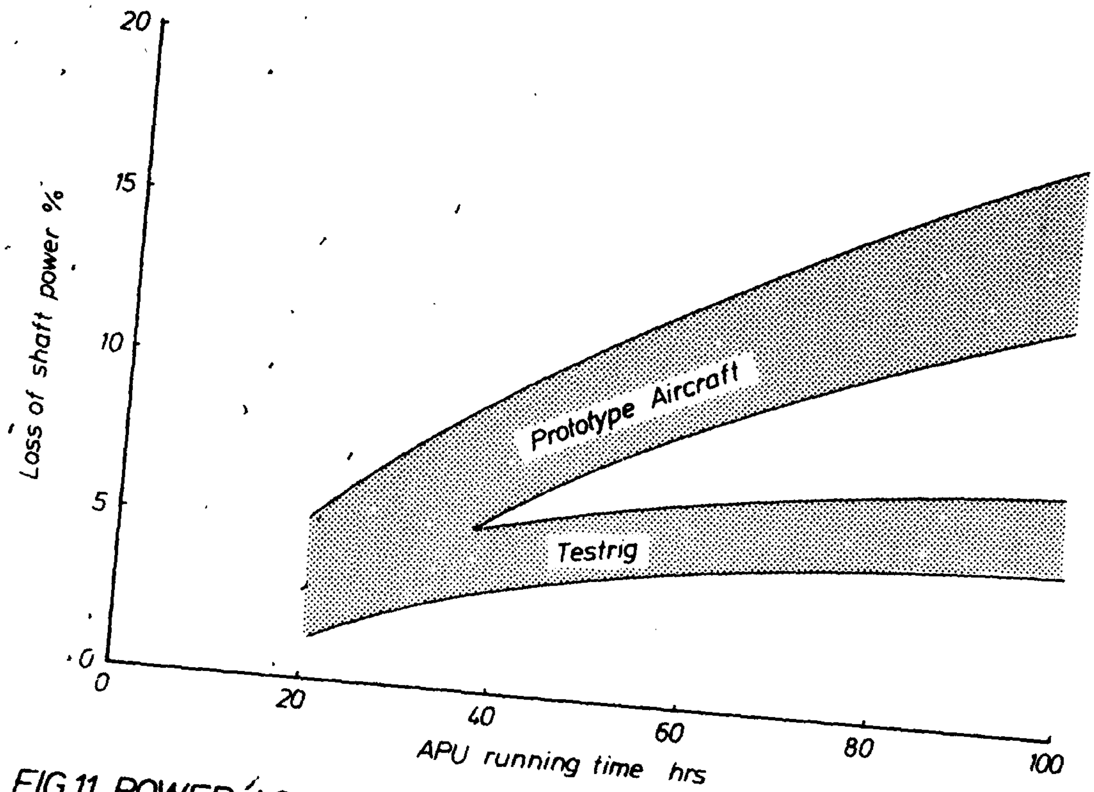

FIG.11 POWER LOSS DUE TO COMPRESSOR CONTAMINATION

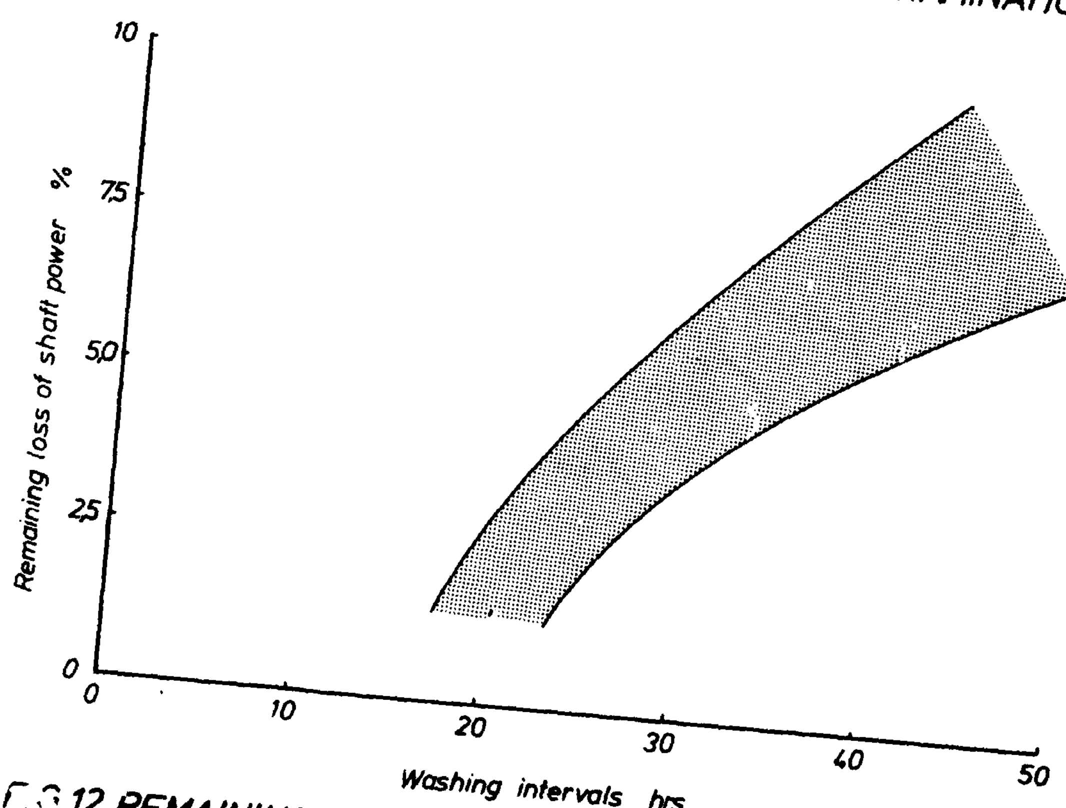

FIG.12 REMAINING POWER LOSS VERSUS WASHING INTERVALS

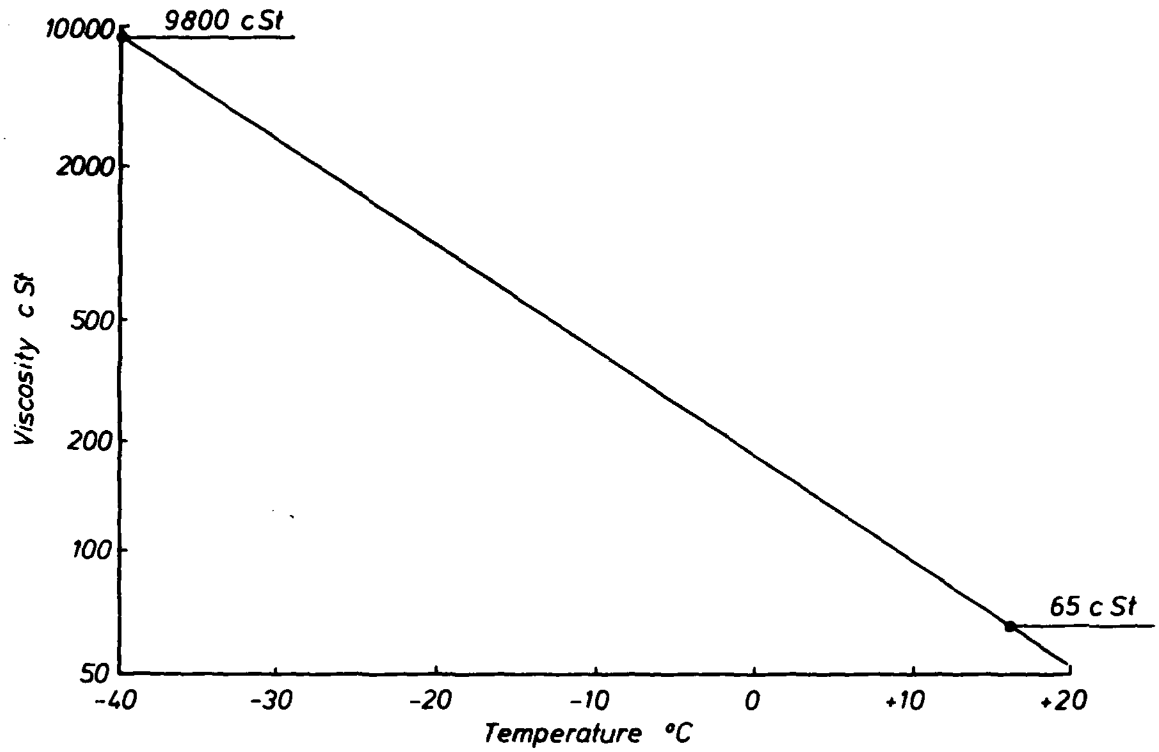

FIG.13 VISCOSITY MIL-L 23 699

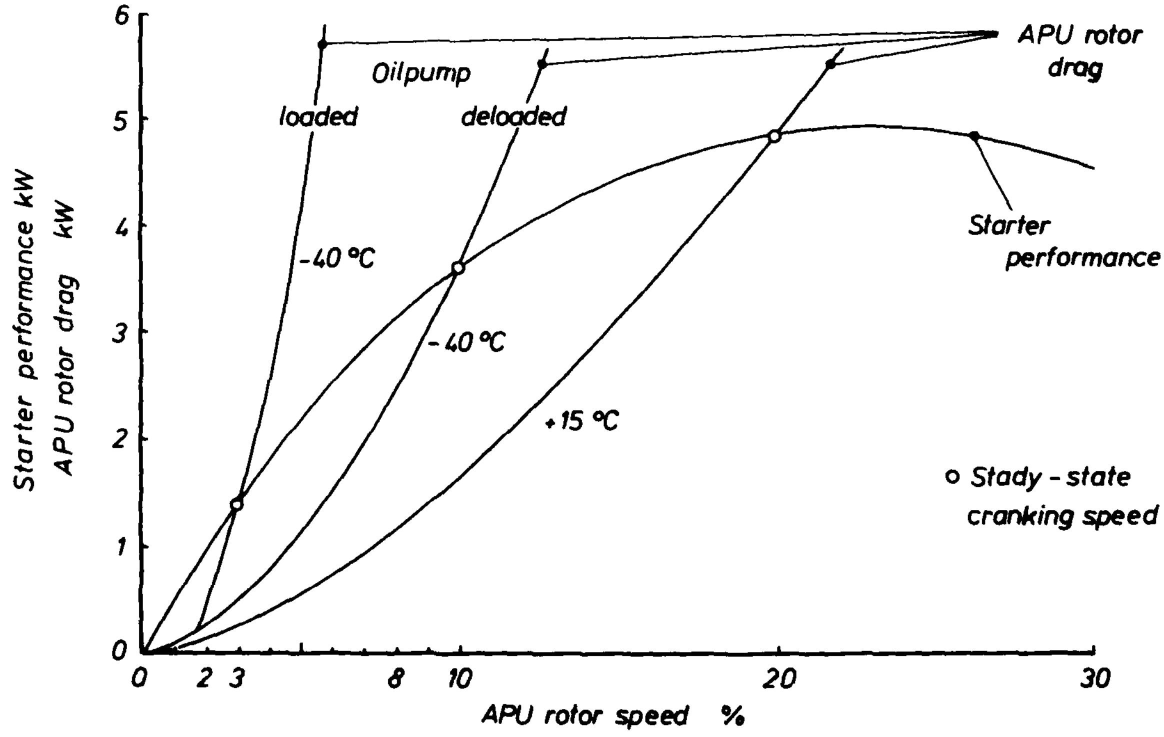

FIG.14 STARTER PERFORMANCE / APU ROTOR DRAG

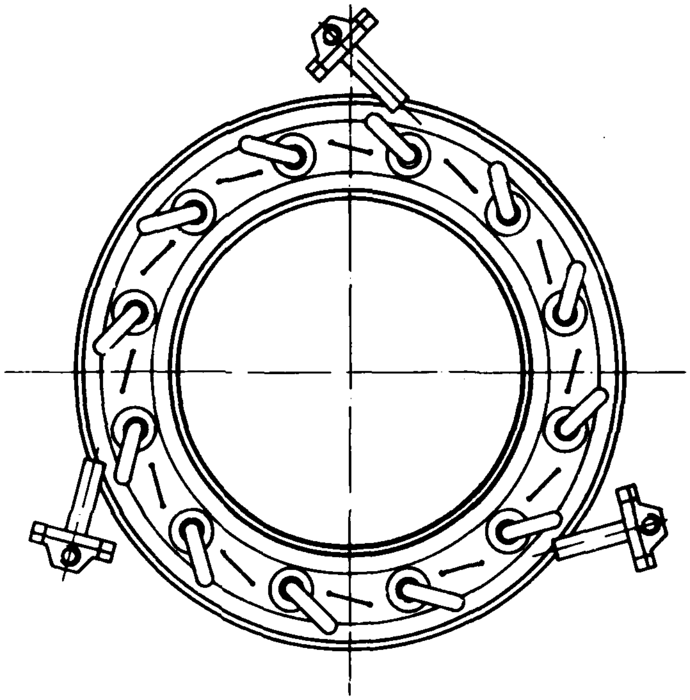

FIG. 15 APU T 312 COMBUSTION CHAMBER
– with 3 starter nozzles –

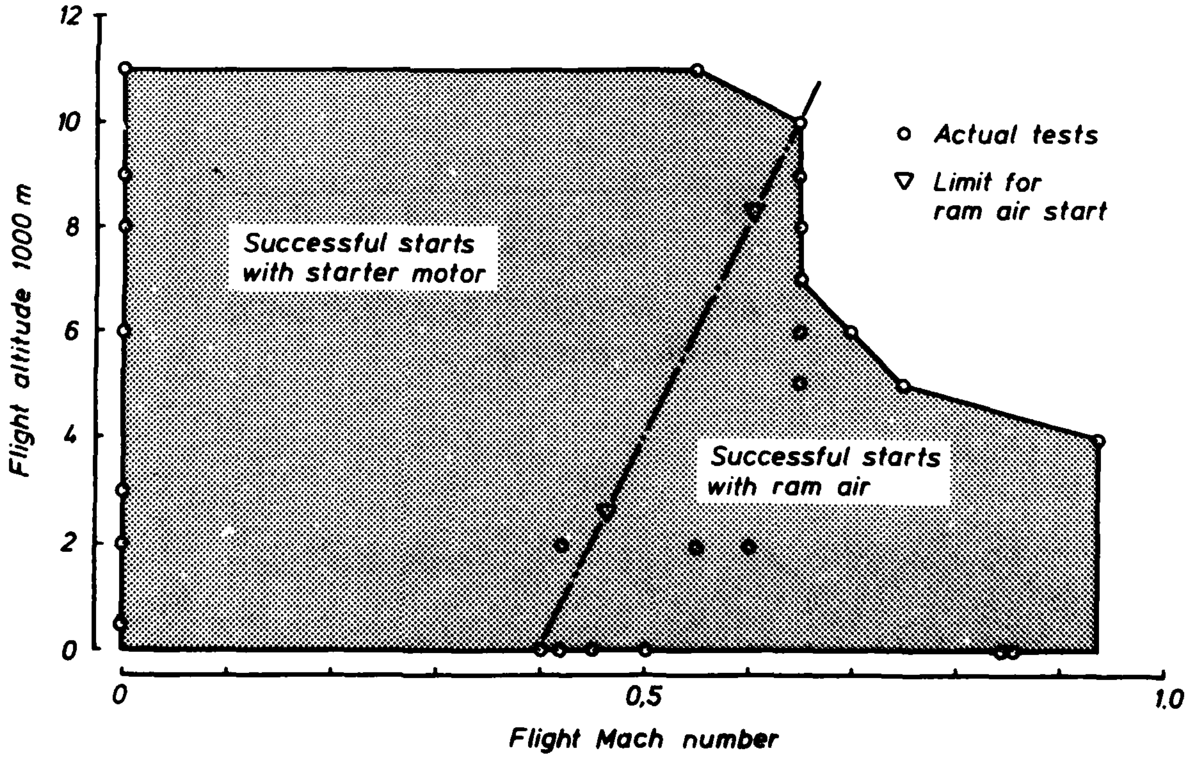

FIG. 16 APU START ENVELOPE

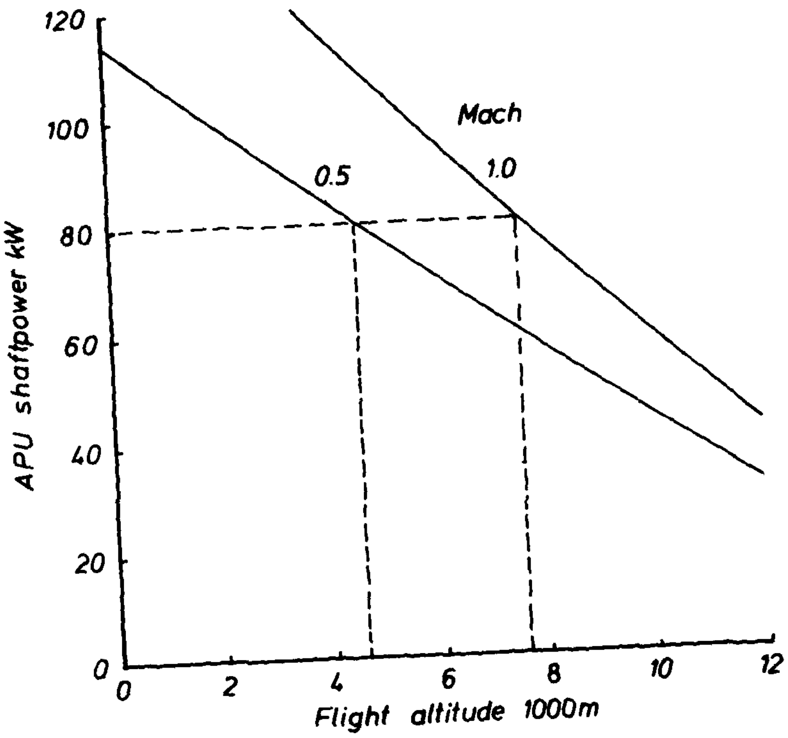

FIG. 17 APU INFLIGHT SHAFTPOWER
-with ram air-

DISCUSSION

G.J. Bramer, Ne.

1. What is the failure rate of the APU (the number of failure starts/running hours)?
2. Is the APU capable of delivering bleed air?

Author's Reply

1. The failure rate of the APU is only known to us with the prototype aircraft, there the failure rate was modest. The rate for series production aircraft, however, is not known.
2. The APU is capable of delivering bleed air up to a maximum of 0,2 kg/sec with a pressure of 3,8 bar.

G. Dahl, Ge.

20 h APU running time; what does that mean in flight hours?

Author's Reply

During the phase of prototype aircraft the APU running time was approximately equal to flight hours.

With series production aircraft it is planned for APU running time to be much less than flight hours. Actual figures are not known.

S. Olympios, Hughes Helicopters

a) Was this APU developed for a specific a/c i.e. the "Tornado"?

It would appear to me that if the installed configuration was known one should have selected a compressor more forgiving to inlet flow distortion caused by the small axial space (70 mm).

b) Does your installed configuration allow any maintenance task to be performed while the APU is on the A/C?

Author's Reply

a) The core engine, especially the axial/radial compressor, was taken from a previously developed APU. To some extent it is true that with another type of compressor one could have managed better with the inlet distortion, but by no means without an inlet duct housing.

b) Yes, all necessary first stage maintenance can be performed with the APU on aircraft.

MECHANICAL AND THERMAL EFFECTS ON THE TRANSIENT AND STEADY STATE COMPONENT PERFORMANCE OF GAS TURBINES

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SUMMARY

Gas turbines in a flight installation are subjected to varying mechanical and thermal loading which produces relative deflections between rotating and stationary components. The relative movement usually occurs during transient operation of the engine and results in varying tip clearances, annulus spoiling and increased/decreased seal mass flows. If rubbing and wear occurs during the transient phase, steady state performance of the compressor and turbines may be impaired usually leading to higher fuel consumption, loss of surge margin and higher flame temperatures.

The mechanical design of components to minimise transient performance loss can be severely inhibited by cyclic life, weight, mechanical integrity, operational requirements and production testing considerations. It follows that the optimised mechanical design of components to achieve the best overall performance of gas turbines is usually a compromise based on a trade-off between these conflicting requirements.

The origin and magnitude of the transient variation in radial and axial deflections during engine operation in flight together with an assessment of the losses produced during transient and steady state engine operation are discussed.

1. INTRODUCTION

The design process for gas turbines in airborne applications has undergone a progressive change in the last decade as the influence of the transient mechanical and thermal behaviour of the components, (particularly the rotating assemblies), has assumed increased importance. Designing for transient conditions is currently a necessary requirement to ensure steady state performance objectives can be both achieved and maintained with the minimum of subsequent Service deterioration.

The performance of aero gas turbines at steady state conditions is still, however, the basic means through which the relative merits of competitive designs are evaluated. In addition to achieving the steady performance level required, the engine must also operate in a diverse spectrum of transient conditions, all of which offer a potential source of deterioration in overall performance. Furthermore, a given engine in several installations may be maintained and overhauled in different ways dependent largely upon the operators and their requirements, (this is particularly true of commercial operation). To ensure the best performance return for a given design configuration with minimum in-service deterioration therefore imposes the condition that all the operational requirements must be integrated into the transient analysis of the initial design such that the certification requirements, cyclic life, transient and steady state performance and minimum subsequent deterioration are as compatible as possible.

2. CLEARANCE CONSIDERATIONS

There are two basic areas in a gas turbine where clearances are associated with performance loss. Of prime importance are the clearances in the main annulus where leakage of the working fluid from, or round, the gas path almost always results in a cycle loss. In addition the secondary air system, which cools blades, vanes, discs, etc and seals against internal leakage and ingestion of annulus gas in the turbines, also introduces losses in performance as seal clearances increase.

There are usually two types of seal in the annulus flow path. For unshrouded blades and vanes the clearance problem is to minimise the circumferential flow of fluid from the pressure side to the suction side of each individual blade on each stage. For shrouded blades and vanes a labyrinth seal is used to restrict the axial flow of leakage fluid through the stage. In the internal air system labyrinth seals are extensively used.

In addition to clearance effects the annulus geometry must also be designed to avoid severe mismatch between rotating and stationary components causing spoiling of the annulus line. It is pertinent to note that since a gas turbine involves the stage by stage change of density of an approximately constant mass flow the changes in area must necessitate a hade of either the inner or outer annulus line, or both. This implies that axial movements must also be considered since they can have serious effects on the radial clearance and annulus geometry control. Typical examples of the effect of axial movements are shown in fig.1. Large axial movements can also change the number of fins in engagement with the seal member in the internal air system and introduce further handling and transient problems.

3. OPERATIONAL REQUIREMENTS

For a given operating cycle there is usually one point at which the Cold Build Clearances, (CBC), can be set to give a minimum condition. The efficiency of the mechanical design can then be judged by the degree to which clearances can be controlled at other phases of flight and test bed running and by the amount of deterioration incurred with time in Service due to rubbing conditions.

For commercial operation CBC's are usually adjusted to ensure that the annulus tip clearances are small in the flight regime where maximum block fuel usage occurs. This is usually the cruise phase for long and medium stage lengths but may be in both climb and cruise for a short range operation.For military operation the requirement could be for close clearances during take off or for combat but is essentially tied more closely to the airframe mission requirements.

Before flight operation all engines must undergo a Test Bed run to a Production Test Schedule, (PTS) to ensure that the engine meets the required flight performance. While running at sea level static conditions a number of engine conditions occur which are not typical of subsequent service operation.

Taking a Production Test Schedule, PTS, (typical of a large commercial transport engine), the initial performance curve consists of 5 points with a 4 minute soak at each condition. At max conditions, which are reached after approximately 20 minutes, all the rotating discs have a rim radial deflection due to thermal expansion which is in excess of that usually produced during a Service take off. In this condition the decel, (or the decremental performance points), must be carefully controlled to avoid severe rubbing conditions in both the later compression stages and the initial turbine stages where radial thermal deflection of the disc rim is a large fraction of the total deflection.

Similarly to set the correct times for the Acceleration and Deceleration Control Units,(ACU/DCU), the engine must undergo a slam accel. followed by a decel. By careful design of the PTS, taking account of the deflection v time response of the relevant components, (based on a computer model calibrated by engine measurements), this phase of the PTS can be achieved without prejudicing the steady performance by avoiding heavy rubbing of the seals and blade tips.

Other relevant conditions in the PTS where the transient movements can lead to seal rubs are:

(i) during running-in handling

(ii) during the check for overshoot of gas generator temperatures following an accel.

and

(iii) other running involving prolonged soaking at high ratings

In each case the deflection v time for blade, vane tips and internal seals for the relevant components must be controlled to avoid severe transient rubbing during a decel.

Clearly it is possible to design a Production Test Schedule which could pass an engine to flight with an abnormally high performance level which would be largely lost when flight loads are imposed. A careful balance must therefore be achieved between the amount of performance loss that can be tolerated in the PTS relative to the loss predicted to occur in service operation.

4. DATUM REQUIREMENTS FOR CLEARANCE CONTROL

In order to estimate the design efficiency of passive and active control of clearances, a perfectly matched design can be defined as:

(i) radial deflections of the rotating and static components, (controlling clearances), for each stage are equal for all transient and steady state operation.

(ii) axial deflections of the rotating and static components are equal at each axial plane measured from the rotor thrust bearing centre for all transient and steady state operation.

(iii) the rotating and static components are maintained round and concentric with each other at each axial plane.

(iv) there is no local tilting of the blades, discs, vanes and other static components from a symmetric plane.

If the above 4 conditions could be met in an ideal design then radial and axial clearances at each running condition of the engine would be fixed by the cold build clearance, (CBC), necessary for assembly.

Putting the above 4 conditions more simply, (i) above represents the conditions for a passive or active matching of the rotating and static components for radial clearances only, based on the assumption of complete symmetry. Conditions (ii), (iii) and (iv) therefore represent the non-uniform deflections which potentially can largely destroy any gains in radial clearance control from a passive or active system.

5. REQUIREMENTS FOR ANNULUS GEOMETRY CONTROL

The previous 4 conditions above are directed towards an ideal control of main annulus tip clearances and also seal clearances in the internal secondary air system of the engine. However,if for any turbine or compressor stage a blade tip radial clearance at the outer annulus line could be designed to satisfy these conditions the inner annulus line is not automatically constrained to maintain the correct annulus geometrical relationship at all engine conditions.

Annulus areas at the engine design point have to be corrected to sea level, (SL) cold static conditions to determine the drawing dimensions for manufacture. Since the annulus is only smooth at this design point, off-design conditions usually produce either spoiling of the inner annulus line, or loss of stator tip clearance in an unshrouded design, and hence results in steady performance losses in addition to transient losses from any mismatch in response characteristics together with probable rubbing of seals.

Since the inner annulus line conditions are related to the outer annulus line conditions principally through the stator blade or NGV aerofoil thermal expansion then it would seem logical that if the rotating aerofoil tip deflection relative to the inner annulus line was equal to that of the stationary vane aerofoil deflection relative to the outer annulus line then spoiling,unshrouded stator tip clearance increase, changes in capacity etc. at transient and steady conditions could also be eliminated in an ideal design meeting clearance control datum requirements.

The practical difficulties in achieving this requirement will be referred to subsequently.

6. EFFECT OF COMPONENT MECHANICAL DESIGN AND STEADY PERFORMANCE REQUIREMENTS ON TRANSIENT DEFLECTIONS

6.1 AXIAL COMPRESSORS

In multi-stage axial compressors the principal components of symmetric radial deflections controlling clearances are the thermal growth of the casing and the combined CF and thermal growths of the rotor. The relative magnitudes and response times effectively determine the radial clearance variation. In general, large clearances in the early stages of a multi-stage compressor tend to reduce low speed Surge Margin and efficiency whilst large clearance in the rear stages affect high speed efficiency and surge margin. As a consequence of the low hub/tip ratios at the front of the compressor the large blades result in a high proportion of the total radial deflection depending directly on (rotational speed^2). In the compressor rear stages the high hub/tip ratios reduce the percentage of radial disc rim deflection due to rotational speed but the temperature levels at current compression ratios for commercial engines, (in excess of 30:1), impose a certification and cyclic life duty which can only be met by the high thermal expansion alloys and some titaniums. Since the transient radial temperature gradients in the discs increase, in general, with the number of stages of compression, disc design in the rear stages result in geometric and material properties giving effectively a 2-rate rim radial deflection transient response. At the rear compressor stages, therefore, the thermal component of the rim radial deflection can be in the region of 50% of the total rim deflection. Compressor casing design in the distant past had very little element of transient clearance control in their design philosophy. Hence there was a monotonic radial growth response of the casing to the forcing influence of the annulus temperature rise which resulted in a wide variation of rotor tip clearance, during and following, transient operation. A typical tip clearance curve, Fig.2, shows the initial closure on the accel. to high power followed by the increase in clearance, (and subsequent closure), during steady high power running. Following a decel. from high power, (typical of a descent from cruise), the thermal contraction of the casing tended to overtake the reduction in blade tip radial deflection, (due to centrifugal effects), after a short time interval at low power. If rubbing conditions occur either at the initial accel. pinch point or following the pullback from high power the tip clearances increase for all subsequent engine operation.

Currently, there are several solutions to this basic tip clearance problem. Ideally, a full active system where casing position reacts to a feedback error signal from a measurement of instantaneous tip clearance, would satisfy the radial control conditions imposed by the Operating Requirements. However,a semi-active system,where the casing is controlled from a scheduled signal or a passive system using thermal slugging or low coefficient of expansion casing materials are typical of the current solutions to radial control of tip clearances. By the above means of controlling blade tip clearances radially a degree of control of the conditions at the inner annulus line is achieved since at steady conditions rotor and stator aerofoils do not have a significant difference in temperature. In addition, shrouded stators have their inner platforms forming a smoother inner annulus line and unshrouded stators have a better degree of control of their tip clearance. However, in addition to radial movements there are axial movements to consider and particularly the relative movement of rotor and casing. Because the density change through the compressor requires a contracting area the hade introduced at either, (or both), walls of the annulus automatically gives a component of radial movement from any relative axial effect, (Fig.1).

The performance effects of relative axial movement between the rotor and casing depends on the compressor aerodynamic and mechanical configuration chosen.

Taking the effect of hade on the inner annulus line, (and still assuming symmetry of rotor and casing), relative axial movement increases or decreases the stator tip clearance. This is most marked in the stages with low hub/tip ratios. With shrouded stators, relative axial movement can introduce spoiling at the inner annulus line, forming "cliffs and waterfalls" at the platforms and the magnitude is similarly influenced by the stage hub/tip ratio. With hade on the outer annulus line relative axial movement introduces a change in the blade radial tip clearance for both shrouded and unshrouded designs.

Considering the combined radial and axial relative movements during a transient accel. or decel. it is a good approximation to assume that the forcing influence for the thermal expansion radially and axially of the casing is related directly to the annulus temperature rise, the rate of expansion being controlled by the material thermal diffusivity and the local heat transfer coefficients. Since the rotor has radial and axial components of movements related directly to rotational speed and pressure level, (for bearing loads), wide variations in the locus of relative movement can occur. Although the compressor can achieve a relatively stable performance condition comparatively quickly thermal equilibrium conditions occur considerably later in time. Since the relative radial and axial movements are not usually in phase, opening or closing of tip clearances, annulus spoiling, capacity changes, rotor/stator axial pitching etc. usually occur at different times. The interaction of the mechanical and thermal effects can therefore cause, for example, loss of engine Surge Margin at ostensibly steady engine conditions.

In addition to the clearance effects in the annulus, the performance of multi-fin labyrinth seals in the secondary air system relies directly on the number of fins in engagement. Relative axial movement of the fin and seal during transient operation of the engine can produce significant changes in bearing loads, cooling air mass flows etc, which can also introduce additional transient movements. It is advantageous, however, if unavoidable seal rubs can be arranged to occur at a location on the seal lining away from the position where normal running and maximum performance is required and this may be possible by careful analysis and design. Conversely if the rotor is perfectly matched axially to the casing so that there is no relative axial movement, (which is desirable to avoid annulus losses), then any radial mismatch in the rotating and stationary deflections during transient operation, causing rubbing, opens up the seal clearance for all other running.

In order to minimise any mechanical interaction with the aerodynamics of axial compressors, to ensure steady performance objectives can be met and maintained, (and still making the basic assumption of symmetrical conditions), the magnitudes and response rate of the relative axial and radial movements must be established for the critical parts of the operational spectrum for the engine. These basic data are necessary in order to ensure that asymmetries in the deflection characteristics do not destroy any performance benefits obtained by clearance control schemes.

6.2 ASYMMETRIC DEFLECTIONS

The departure from symmetric conditions can be simply defined, relative to the rotor blade tip, as out-of-roundness, axial distortion, eccentricity and tilt and skew of the casing linings.

6.2.1 Out of Roundness

Out-of-roundness, (OOR), is assumed to be the circumferential variation from the intended geometry in an otherwise symmetrical plane. This effect can be induced in a number of ways both thermally and mechanically. Unbalanced local thermal gradients, single air offtakes from a circumferential casing bleed, and circumferentially varying environmental conditions can all contribute to out-of-roundness of the linings from thermal effects. Split casings, shrouded stator split lines, engine primary structure distortions transmitted to the lining support structure are examples of mechanically induced OOR.

The effect of OOR on clearances also depends on the abradability of the linings. If local rubbing removes lining material transiently then a local increase of tip clearance results at other engine conditions if OOR does not change significantly. If the lining does not wear and effectively 'picks up' metal from the blade tip then the blade tip radius is reduced, the circumferential clearance increasing and the compressor stage becoming effectively smaller in diameter.

6.2.2 Axial Distortion

Axial distortion is assumed to be the dimensional variation axially from an otherwise symmetric plane.

This effect occurs if the lining is supported, for example, by the engine primary casing which is subjected to bending moments from thrust line offsets, flight and inertia loads etc. It also occurs if the disc rim has 'swash' (as a result of the manufacturing process), when the blade tip cuts a wider groove in the linings than that which would be produced by symmetric blades.

6.2.3 Eccentricity

Eccentricity is taken to be the translation of the casing centre line relative to the rotor centre line causing high and low clearances at diametrically opposite positions. Although this gives rise to 'compressors in parallel' losses the effect could be different in a compressor with a large number of stages where the flow streamline twist from inlet to exit may be of the order of 90° to 180°. It is possible that there may be further losses of Surge Margin and efficiency under these conditions but no currently published data has been found to quantify the effect. In addition to asymmetric clearances at the blade and, unshrouded stator tips there are spoiling effects at the inner annulus line in shrouded stator stages and asymmetric stator interstage seal clearances.

6.2.4 Tilt and Skew

Tilting of the casing centre line relative to the rotor centre line can be defined as a rotation of the casing resolved in the vertical plane and skew as the horizontal component of the rotation, the axes being aligned with the plane of suspension of the engine in the airframe.

Bending moments from flight loads, thrust line offsets, inertia loads etc. when combined with torque reactions can cause a complex movement of the casing relative to the rotor in terms of tilt and skew depending on the mechanical arrangement of the casing in the engine structure. Non-uniform thermal conditions surrounding the lining support structure can also induce tilt and skew. This type of deflection could, it is suspected, have widely different effects dependant on the particular combination of aerodynamic and mechanical design chosen but again no published data has been found to quantify the performance effect. Clearly, clearances vary axially along the compressor as do the spoiling effects from shrouded stators and if streamline twist does have any effect on performance with eccentricities of the casing present then tilt and skew would probably compound the loss.

6.2.5 Summary of Asymmetric Deflection Effects

Asymmetric deflections, when considered in the context of the wide range of mechanical and aerodynamic configurations possible, can result in a complex time dependant sequence of radial and axial movements. In nearly all cases there will be a loss of performance at steady conditions and there may also be further losses during transient operation.

Compressor characteristics, (or chic's), are normally obtained as a carpet of steady state equilibrium points on a rig test with symmetric linings and clearances. Making the simplifying assumption that the chics do not vary during symmetric transient operation still leaves the question of the effect of asymmetric lining deflections unanswered. Virtually no published data have been found on this subject; unpublished past data available to the author has shown that an eccentricity of 0.010 in. and a tilt of 0.030 in. at the rear of a 6 stage compressor, (relative to the front stage), when coupled with take-off clearances severely deteriorated the compressor Surge Margin by up to 0.45 of a Pressure Ratio. In this particular case control of tip clearances and re-location of the casing eliminated the loss.

6.3 TURBINES

Considering initially symmetric deflections in the turbine stages, the gas path conditions provide an environment more severe than in axial compressors. The mechanical and thermal effects producing deflections are still similar, however, to those in axial compressors,although deflection magnitudes tend to be larger and the basic response times for the rotating and static components tends to be longer. Again, as in the rear stages of the compressor the only materials currently available to meet the rim loading and metal temperature requirements are the high expansion alloys which develop radial and axial gradients during transient conditions leading to a 2-rate rim deflection response. If the boundary conditions are dissimilar at front and rear of the disc, transient,(and possibly steady state), bending can give a wide locus of movement at the blade tip where the radial and axial deflection components have different response times. In a similar manner to axial compressors the change in blade length and gas path conditions leads to the CF component of deflection increasing rearwards through the turbines.

Axially, rotor thrust bearings are usually positioned remote from the immediate environment of the turbine stages so that a relatively long axial path exists from the blade tip to the lining. In addition further axial deflections due to shaft tensions and bearing load reactions in the stationary structure are usually of low magnitude but relatively fast time response compared to the shaft and casing thermal expansion response.

When the transient axial and radial deflections are computed for the operational requirements of the engine a locus of movement of the blade tip is obtained, Fig. 3. Two types of tip clearance have to be considered, the shrouded and the unshrouded stage;in both cases the problem is to match the shroud seal segment response to minimise clearances. For a shrouded turbine it is possible by judicious use of thermal matching techniques and by controlling the maximum cut in the seal to occur at an axial position away from the running position to obtain a good clearance control. For an unshrouded stage axial bending of the blade, either through its own chordwise temperature distribution or by turbine disc rim rotation, results in a basic loss of clearance and may affect the integrity of the blade itself. If LE and TE cuts are unavoidable during some engine operation then very careful design of the Running-in Handling during initial Production testing is necessary.

Symmetric control of turbine tip clearance does not however, automatically control the annulus geometry and capacity changes coupled with spoiling losses at the inner annulus line add to the basic aerodynamic losses if mismatches occur. In addition the secondary air system seals are subjected to similar radial and axial movements to those in the compressor and further movements both transiently and steady state can occur.

By combining the predicted radial and axial movements the performance losses from blade tip clearance, turbine capacity, spoiling and bleed flows can be estimated from rig testing.

6.3.1 Effect of Asymmetric Deflections

Asymmetrics in the stationary structure relative to the rotating components can be defined as previously, Section 6.2.

6.3.2 Out-of-Roundness, OOR

Out-of-roundness of the blade tip seal linings usually results from an asymmetry in the segment supporting structure and can originate in a number of ways,

(a) hot annulus gas leaking behind the shroud segments can induce local thermal gradients in the support structure

(b) if the shroud segments are supported directly from the main engine structure, engine support link reactions, bending of the carcase from thrust line offsets are typical ways of inducing OOR in the linings

(c) a variable stiffness structure, (unless designed to produce uniform deflections), can induce OOR effects

(d) cross-key location of two components can lock-up if not correctly designed for the local deflection conditions and again induce OOR effects.

and

(f) asymmetric cooling of the shroud supporting structure or the engine carcase can also produce asymmetric linings.

In addition to loss of basic clearance in tip seals and secondary air system seals, OOR must also introduce spoiling losses at inner and outer annulus lines.

6.3.3 Axial Distortion

These effects are basically as indicated for Axial Compressors.

6.3.4 Eccentricity

Offsets of the shroud segments relative to the turbine blade tip can also originate from a number of sources. Typically -

(a) 'squeeze films' in the rotor bearings do not centralise the rotor during the initial start and run up to Idling and asymmetric rubs can occur,

and

(b) bending of the main engine carcase can offset the rotor shaft centre relative to the shroud segments, (particularly under flight and inertia loading).

The effect of eccentricity of the turbine linings is that the flow operates at high and low clearances, diametrically opposite. There is also inevitably effective flow area changes and spoiling losses at the annulus walls which are not uniform circumferentially. No published data have been found to quantify these effects.

6.3.5 Tilt and Skew

Tilt and skew of the linings is mainly pertinent for multi-stage turbines where the centre line of the seal segments varies radially from the rotor centre line. Depending on the engine suspension arrangement for reacting thrust and torque transient tilting and skew may result. In general, this effect is mainly relevant in the later stages of turbine and since the engine is usually supported near this position the magnitude of this effect is unlikely to be as significant as in axial compressors.

6.3.6 Summary of Asymmetric Deflections in Turbines

Asymmetries in the turbine stages again can lead to transient and steady state performance losses. Although little published data is available to quantify the effect, (as in the case of compressors), unpublished data does confirm that significant efficiency losses are possible from for example, tilting of a NGV in a turbine stage.

7. CONCLUSIONS

The integration of the operational requirements for gas turbines into the analysis of movements and deflections in compressors and turbines contributes towards efficient design for achieving and maintaining maximum transient and steady state performance. As would be expected, compromise is often necessary because of basically conflicting requirements. High expansion alloys, prone to large temperature gradients, are necessary to meet cyclic life and certification requirements in the rear compressor stages and initial turbine stages but pose severe transient clearance and control of annulus geometry problems. Engine casings, designed to satisfy containment requirements may not be sufficiently stiff to avoid excessive transient rubs occurring. Clearly this leads to a trade-off between engine weight increases against the possible reduction in fuel burn and knowledge of the symmetric and asymmetric behaviour of the engine is then necessary to arrive at a compromise situation.

In the deflection analysis of an engine considerable emphasis is placed on the necessity for good rig and engine test bed measurements, particularly when a whole engine deflection analysis is to be carried out. Without this data, which effectively calibrates deflection computer programs, delineates actual load paths due to interferences etc.,the effect of symmetric and asymmetric deflections on the transient and steady state performance of an engine would be difficult to evaluate.

As a typical example of the application of the above analysis the current RB 211 series of engines have been developed to achieve low deterioration rates in addition to small losses in sfc during the Production Pass-off testing. Extensive measurement in test bed and rigs coupled with a transient deflection analysis, (starting with the engine primary structure), has shown that good control of deflections during transient and steady state operation of the engine is possible. Analysis of the deflection behaviour of the HP Compressor casing and rotor has shown that radial tip clearances of the order of 0.015 in. to 0.020 in. are possible by reduction and control of asymmetric deflections. Simple passive control of tip clearance and low coefficient of expansion material in the casings would then be sufficient to achieve the above clearances at cruise conditions. At this level of tip clearance the weight and probable unreliability of complex active systems was not justified for this installation. Primarily the isolation of the compressor casings from the main structural casings with short, stiff structures, Fig. 4, allows good transient and steady state performance to be obtained,with low deterioration rates in Service,Fig.5.

Similarly simple control of HP turbine tip clearance using a thermal control ring tailored to the blade tip radi and axial locus of movement minimises HP turbine efficiency losses during transients, (Scheme 38), Ref.2.

In conclusion,therefore,a full deflection analysis of the engine for the spectrum of operational conditions to be met is a necessary part of the engine design process for the increasingly stringent requirements imposed on future designs. A full engine measurement capability coupled with a large computational capacity is also necessary. However,asymmetric deflections have received very little attention in current publications and if progress towards better performance and fuel economy is to be made some measure of this effect will be required in the future.

8. ACKNOWLEDGEMENTS

I would like to thank the Directors of Rolls-Royce Limited for permission to prepare and present this paper. My thanks also go to the Rolls-Royce Engineering Illustrations Department for their invaluable assistance in executing the diagrams.

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9. FIGURES

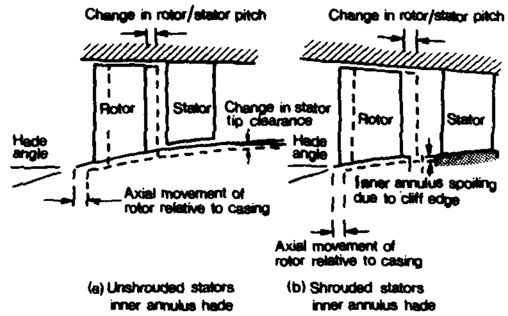

Fig.1 Effect of Axial Movement of Rotor Relative to Casing

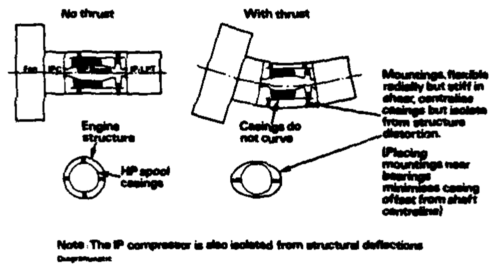

Fig.4 RB 211 HP Spool-Isolation of Casings from Engine Structure

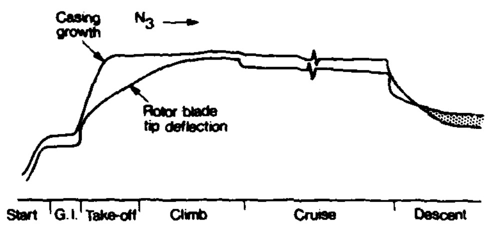

Fig.2 Typical Tip Clearance Curve

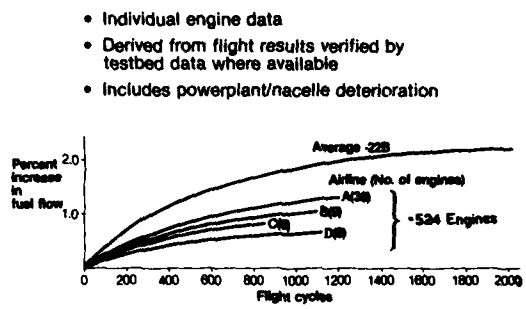

Fig.5 Relative Performance Deterioration

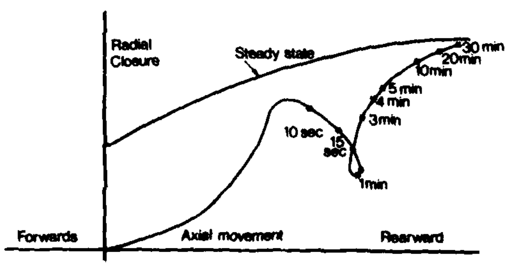

Fig.3 Locus of Movement for Shrouded Turbine Blade Tip

DISCUSSION

K. Robinson, UK

The paper shows the performance advantages to be gained by an effective, active tip clearance control. It also showed that such a mechanism would be very complex with the likelihood of significant degradation in reliability and increase in overall cost. Does the author expect that there is, or will be in the near future, any practical advantage to be gained by active tip clearance control?

Author's Reply

Yes I do expect active clearance control will eventually come into more widespread use in the future but a better understanding of the mechanical and thermal behaviour of the rotor/easing transient movements and symmetric and asymmetric deflections is a necessary pre-requisite to ensure that these systems will be effective.

The degradation in reliability due to the complexity of fully active clearance control will be largely dependent on the success with which designers can minimize the control problems set for active control. By a fuller understanding of the engine transient movements and by initially achieving the best passive matching possible the residual transient problem left may well be coped with by a simple, rather than a complex, fully active system of control.

Currently, however, I believe the best engineering solution to close tip clearance control in turbines and compressors should be based on an understanding of the asymmetric transient and steady state mechanical and thermal movements in the components which offers a good solution in that the combined use of axial and radial movements in a passive system offer simplicity and reliability with a good degree of clearance control.

W. Heilmann, Ge.

You showed x-ray photos and explained that you use the data derived from the photo in your sophisticated engine clearance model. But the x-ray results can only be obtained in one plane of the engine - how do you manage the application of these results if you have also eccentricity and distortion of the parts in the considered area?

Author's Reply

X-ray results are used to give a better understanding of the deflections occurring transiently and at steady state in the area under test. By locating a clearance probe in the plane of the x-rays the two measurement systems can be cross calibrated.

Other probes mounted circumferentially in the plane of the x-rays then give the eccentricity distortion and out-of-roundness of the component considered.

X-ray data is also taken in a number of planes in such a way that all the results can be read across from each other to enable an effective axial calibration of the clearance model.

J.A. Rowlands

We have an engine with a problem of axial clearance in engines which have had overhaul and re-use of components (NGV's). These may have reduced stiffness or whatever following re-use. How much importance do you put on checking for axial or radial movement on components after many 1000's of hours of earlier use in transport a/c or many 1000's of hours of use in fighter bomber types?

Ancillary comment on Question

Spey 202 in Phantom has problems of LP NGV bow causing seal to impinge on LP1 disc at blade roots and trepan material away. We lose material and eventually blades with firtree roots. The cure is seen to be 100% replacement of NGV's on overhaul – a very expensive solution.

Author's Reply

I do not recall any evidence of reduced stiffness (as defined by component geometry) occurring after many hours of Service use. If this did occur it would not be easy to take into account during the design phase of the engine from the point of view of clearance control.

I would suspect that your problem is a specific local problem, possibly caused by creep in the component concerned. Creep can be regarded as a variable deflection with time and is usually checked for during overhaul.

Clearly it is important to check for creep effects when considering clearances and the control of clearances in turbines for example. If creep does occur, the difficulty as I see it is that increased deflections (either radial or axial) occur when the component reaches the appropriate stress/temperature condition, but this condition may not be linear or even have a monotonic rate with Service use. This would make good tip clearance control a complex problem.

Ph. Ramette, Fr.

What are the new techniques which are used for clearance measurements at Rolls Royce?

Are x-rays used both for axial and radial displacements?

Author's Reply

(i) Capacitance probes and stepper motor probes are currently used for clearance measurements in engines and rigs. Other systems are under experimental evaluation and, subject to back-to-back testing with the measurement systems above, could well replace the current systems in the near future.

(ii) X-ray measurements are used for both axial and radial displacements since they indicate deflections of all the components in the x-ray picture (at that instant) in the plane of the shot. This information is complementary to tip clearance probes and axial probes which measure point movements at various circumferential positions.

MAITRISE DES JEUX AXIAUX ET RADIAUX
SITUATION ACTUELLE ET PERSPECTIVES

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RESUME

La prédiction et le calcul des jeux axiaux et radiaux des turboréacteurs ont une importance croissante en raison de leur répercussion directe sur les performances et la manoeuvrabilité.

Ce document présente les différents critères que chaque type de jeux doit remplir, que ce soit pour des moteurs civils ou des moteurs militaires, en précisant les méthodes de calcul habituellement utilisées.

Il décrit plusieurs exemples typiques de problèmes rencontrés, ainsi que les actions correctives à apporter, celles-ci devant résulter de compromis performances/masse/coût.

Les perspectives d'amélioration à court et moyen terme des modèles de prévision et des techniques de contrôle sont ensuite abordées.

1.- INTRODUCTION.

Le besoin de réduction des jeux dans les turbomachines s'est fait sentir, il y a de nombreuses années, puisqu'il est vite apparu que les jeux radiaux en sommet d'aubes (les premiers concernés par les études sur le sujet) avaient un impact important sur les performances des différents composants du moteur, et donc sur le moteur complet (Fig. 1).

Le souci continu d'amélioration des performances encore accru par l'augmentation brutale des coûts du carburant a amené à perfectionner le dimensionnement des jeux radiaux en sommet d'aubes, mais également à contrôler soigneusement les étanchéités des différents labyrinthes, qu'ils assurent l'étanchéité au pied de la veine ou servent de réglage pour le refroidissement interne du moteur (Fig. 2).

L'objectif n'étant plus seulement de minimiser les débits de fuite (parasites) principaux pour un cas de fonctionnement donné, mais de minimiser tous les débits de fuite dans un grand nombre de cas de fonctionnement, ainsi que leur taux d'augmentation en fonction du temps d'utilisation (Fig. 3). (Réf. 1).

Le jeu de la concurrence, et les exigences croissantes des avionneurs amènent à rechercher des rapports poussée/masse toujours plus élevés, tout en contrôlant sévèrement les coûts de réalisation. Si l'optimisation des jeux radiaux contribue - à masse donnée - à l'augmentation des rendements, donc de la poussée, l'optimisation des jeux axiaux entre rotor et stator permet de diminuer la masse et le coût à poussée donnée en même temps qu'elle permet de faciliter les recouvrements de plate-forme en paroi intérieure de veine et, bien sûr de limiter l'encombrement du moteur.

Aucun interface rotor - stator ne doit être négligé, mais sa caractérisation - en terme de déplacement - et son optimisation passe par l'intégration et le calcul de nombreux paramètres.

2.- OBJECTIFS ET CRITERES DE DIMENSIONNEMENT.

La variété des situations que l'on peut rencontrer dans un moteur ne permet pas l'utilisation d'une démarche commune à tous les types de jeux : chacun est un cas particulier qu'il convient de traiter en considérant sa fonction particulière et son environnement.

On peut cependant dégager pour chaque catégorie un certain nombre de règles générales.

2.1.- Les jeux axiaux.

- Paroi intérieure de veine -

L'objectif est de minimiser l'entrefer rotor/stator afin de limiter les recirculations d'air entre les écoulements principal et secondaire, et de guider l'air réinjecté dans l'écoulement principal, afin de diminuer les pertes aérodynamiques (Fig. 4).
Un critère impératif est d'interdire tout contact rotor/stator, même lors de fonctionnements accidentels.

- Labyrinthes -

Les critères sont les mêmes qu'il s'agisse des étanchéités en sommet d'aubes, ou des labyrinthes internes. La partie du rotor susceptible d'entrer radialement en contact avec l'abradable doit toujours être en face de celui-ci (Fig. 5 a).
Dans le cas d'un labyrinthe étagé, on doit donc éviter un désengagement des léchettes (Fig. 5 b) et un contact axial léchette/abradable (Fig. 5 c).

- Aubages -

Il faut bien sûr éviter, même lors de fonctionnements exceptionnels, tout contact axial entre aube fixe et aube mobile.

Dans certains cas, on peut dimensionner un jeu axial entre une grille fixe et la grille mobile voisine, de telle manière que, lors d'une rupture d'arbre, la rétention axiale du rotor soit assurée par le contact entre ces deux grilles.

2.2.- Les jeux radiaux -

- Critères généraux -

Les jeux à froid doivent être suffisants pour permettre le montage du moteur. Il est également nécessaire de vérifier qu'ils ne risquent pas d'engendrer des frottements à basse vitesse qui sont générateurs de déformations importantes.

- Sommet d'aubes -

Si l'objectif est toujours d'obtenir des jeux faibles dans un grand nombre de cas de fonctionnement, sa mise en application dépend de la technologie des deux parties assurant l'étanchéité (Fig. 6) :

- La géométrie du rotor:une aube à talon muni de léchettes tolère des pénétrations plus importantes qu'une aube sans talon, surtout si celle-ci a une faible épaisseur (sensibilité à l'usure et aux déformations).

- La nature de l'abradable : Les abradables résistant aux hautes températures sont beaucoup plus durs que ceux utilisés dans les parties froides (Réf. 2 - Réf. 3).

- Les conséquences d'un cas de fonctionnement exceptionnel : Il importe de tenir compte de l'ampleur des consommations de jeu dues à des facteurs de charge exceptionnels (vrille à plat, excentrages résultant de la perte d'aubes) et de s'assurer que les pénétrations importantes qui en résultent ne peuvent entraîner des dommages secondaires.

- La zone considérée : Une variation de jeu de 0,1mm n'a pas la même importance pour un compresseur BP que pour une turbine HP.

La variété des situations rencontrées amène donc à se donner des critères de dimensionnement évoluant entre deux extrêmes qui sont :

. Eviter dans presque tous les cas des interférences entre le rotor et le stator.
. Rechercher, dans tous les cas de fonctionnement à régime élevé, une pénétration des léchettes dans l'abradable. Ce choix, lorsqu'il est possible permet de s'affranchir de l'effet des tolérances.

- Labyrinthes -

Leur fonction principale étant de limiter les débits parasites existant inévitablement dans le circuit de ventilation du moteur, il peut paraître séduisant de choisir les jeux à froid les plus faibles possibles, l'abradable se chargeant d'encaisser les variations de déplacement relatif.

Cependant, compte tenu de l'importance de l'intéraction qui existe entre les labyrinthes et les autres moyens de calibration du circuit de ventilation, des variations significatives des jeux, suivant les cas de fonctionnement ou lors du vieillissement du moteur, se traduiraient par des déséquilibres de l'ensemble du système. Les conséquences peuvent être :

. Une dégradation des performances.
. Un échauffement des pièces (aubages, disques).
. Une variation importante de la poussée axiale.
. Une inversion du circuit de pressurisation des enceintes entraînant des fuites d'huile.

On s'efforcera donc de garantir une bonne stabilité des jeux en fonctionnement, en maîtrisant les dilatations différentielles entre le rotor et le stator. Cette démarche conduit à des jeux en fonctionnement qui, pour les cas de dimensionnement (décollage par exemple) vont dépendre du rayon du labyrinthe,de la température de l'environnement et du régime de rotation du rotor.

Dans certains cas, on peut rechercher un jeu en fonctionnement très faible qui ne permet pas a priori une stabilité du jeu au cours du temps. Il est alors nécessaire de contrôler les dilatations, notamment en régime transitoire, du rotor (modification de l'inertie thermique) et du stator (refroidissement contrôlé par exemple).

Ces critères sont parfois difficiles à respecter, car ils ne sont pas toujours compatibles avec d'autres critères de dimensionnement, tels que la stabilité thermique ou la stabilité aéroélastique du labyrinthe .

3.- LES METHODES DE DETERMINATION DES JEUX.

3.1.- Les paramètres à prendre en compte.

Les déplacements relatifs entre les pièces fixes et mobiles sont dûs à de nombreux facteurs dont l'importance varie suivant le type de jeu étudié et sa position dans le moteur.

Cette importance relative est illustrée (Fig. 7) pour un labyrinthe en aval d'un compresseur HP, et pour le jeu en sommet d'une aube de turbine BP.

Les facteurs prépondérants peuvent être classés en deux catégories :

- Ceux sur lesquels les moyens d'action sont très limités :déplacements d'origine mécanique, tolérance sur les cotes axiales.
- Ceux que l'on peut, dans une certaine mesure, minimiser ou contrôler :
  . Les tolérances radiales (par rectification à grande vitesse).
  . Les déplacements d'origine thermique.

. La sensibilité des structures aux facteurs de charge.
. Les ovalisations liées à des sollicitations non axisymétriques.

C'est sur ces trois derniers points que doivent porter les efforts d'optimisation des jeux lors des phases de dimensionnement et de développement des moteurs.

3.2.- Approche théorique - Calcul des déplacements.

Les différentes étapes du calcul des jeux nécessitent la prise en compte et l'intégration de la plus grande partie des paramètres nécessaires au dimensionnement du moteur (Fig. 8).

A partir des données de base que constituent les paramètres thermodynamiques et aérodynamiques (pressions statiques, efforts sur les aubages), on peut calculer la ventilation d'ensemble du moteur, en se donnant des jeux de labyrinthes en fonctionnement (l'expérience permettant de choisir des valeurs réalistes), ainsi que l'ensemble des efforts d'origine aérodynamique s'exerçant sur les diverses pièces du moteur. Il est alors possible de procéder aux calculs thermiques puis mécaniques tant en fonctionnement stabilisé qu'en régime transitoire. Les déplacements radiaux et axiaux dûs aux diverses sollicitations sont alors déterminés. L'intégration à ces déplacements, des tolérances, des jeux à froid réalisables et des critères de pénétration dans l'abradable, permet de calculer les jeux en fonctionnement. Lorsque les évolutions de jeu ainsi obtenues ne sont pas satisfaisantes, il est nécessaire de modifier soit la géométrie de la zone considérée, soit son environnement jusqu'à ce que les objectifs fixés soient atteints.

3.3.- Choix des conditions de fonctionnement dimensionnantes.

On a vu au § 3.2. que les éléments les plus importants à prendre en compte sont les déplacements mécaniques et thermiques, ainsi que ceux dûs aux facteurs de charge. Ces derniers sont relativement indépendants des conditions de fonctionnement et sont définis par les spécifications. Ils seront donc superposés de manière adéquate aux autres déplacements.

En ce qui concerne les effets thermiques et mécaniques, les cas de fonctionnement à étudier découlent de l'objectif général d'optimisation des jeux pour lesquels on souhaite idéalement obtenir des valeurs faibles dans tous les cas et stables, c'est-à-dire non susceptibles de dégradations, ces dernières pouvant résulter soit de déformations, (non identifiables a priori) soit de contact rotor/stator lors de certaines évolutions des paramètres de fonctionnement du moteur.

On voit que - schématiquement - l'optimisation des jeux (ou des déplacements) consistera en une minimisation de l'écart entre la distance rotor/stator ΔR obtenue pour chaque point de fonctionnement et la valeur minimale de ΔR (ΔRm) susceptible d'être rencontrée à l'intérieur de l'enveloppe des cas de fonctionnement possibles.

Cette enveloppe recouvrant un nombre de cas de figure extrêmement important, il serait irréaliste de vouloir les étudier avec la méthode décrite au § 3.2. - C'est pourquoi, de manière générale, l'étude se limite aux évolutions transitoires au sol (accélération et décélération Ralenti/Plein gaz, et réaccélérations), et à certains cas de fonctionnement stabilisés caractéristiques :
- Plein gaz sol, croisière, montée...pour un moteur civil.
- Cas les plus fréquemment utilisés pour les moteurs militaires en fonction des missions retenues.

3.4.- Vérification expérimentale.

Plus encore que tout autre paramètre, dans la mesure où ils sont obtenus à la fin de la chaîne complexe de dimensionnement du moteur, les jeux doivent faire l'objet d'une vérification expérimentale - celle-ci, bien évidemment, ne peut intervenir qu'au stade des essais du moteur complet.
Elle peut être effectuée par mesure directe ou par mesure indirecte.

- Mesure directe -

On peut citer trois méthodes, par ordre chronologique d'utilisation.

Chacune d'elles requiert une caractérisation précise lors du montage des jeux qui seront étudiés et analysés.

- Les témoins de déplacement -

Ce sont des plots à base de graphite fixés sur le stator (Fig. 9) en surépaisseur de telle manière qu'ils entrent en contact en fonctionnement avec le rotor. La profondeur de l'empreinte mesurée donne la variation maximale du déplacement relatif obtenu pendant l'essai.

Cette méthode, utilisée principalement pour les déplacements axiaux, puisque les abradables peuvent fournir la même indication pour les déplacements radiaux, à l'avantage d'être facile à mettre en oeuvre, mais présente un inconvénient important.

. Elle ne donne que la valeur maximale du déplacement relatif obtenu pendant un ensemble de conditions de fonctionnement, et uniquement dans le sens de la réduction du jeu.

- Les rayons "Super X".

Cette méthode s'est généralisée depuis quelques années malgré une mise en oeuvre relativement lourde (Réf. 4), car elle présente l'avantage de donner le jeu réel en fonctionnement, y compris pendant les évolutions transitoires (Fig. 10).

Elle permet également de mettre en évidence et de quantifier les déformations des pièces.

Elle est cependant difficile à utiliser dans les zones proches de l'axe moteur où les problèmes d'accessibilité et de précision de mesure en limitent l'intérêt.

La complexité de l'installation fait qu'elle est inutilisable lors des essais en vol.

- Les capteurs de proximité.

Les capteurs utilisés sont de deux types : optiques (Réf. 5) ou capacitifs (Réf. 6 et Réf. 7).

Les servitudes liées à leur taille, leur implantation et à leur protection vis-à-vis de l'environnement, limitent leur utilisation à la mesure des jeux radiaux au sommet des aubes mobiles.

Ils ont cependant l'immense avantage de fournir des mesures en continu et de donner, avec une chaîne de mesure adaptée, les jeux moyens ou aube par aube.

Leur utilisation lors des essais en vol ne soulève pas de difficulté majeure.

- Mesure indirecte -

L'implantation de thermocouples sur les pièces fixes et mobiles permet de réajuster les calculs des déplacements d'origine thermique, mais aussi, lorsque les mesures sont suffisamment nombreuses, de vérifier la présence ou l'absence d'ovalisation des structures.

3.5.- Synthèse des informations disponibles.

La confrontation entre les valeurs calculées et les résultats d'essais est essentielle pour étalonner et valider les hypothèses utilisées.

Lorsque des écarts apparaissent, entre jeux calculés et jeux mesurés, il importe d'abord de savoir si ces écarts sont imputables aux dilatations thermiques ou aux déplacements d'origine mécanique. Il est donc fondamental de disposer d'une instrumentation suffisante en thermocouples lors de la mesure directe des jeux.

Une fois les écarts calcul/mesure expliqués, il faut en vérifier les conséquences au niveau de l'étude complète précédemment réalisée et définie au § 3.2.

4.- EXEMPLES DE PROBLEMES RENCONTRES ET D'AMELIORATIONS APPORTEES.

Nous avons vu que les préoccupations relatives au problème des jeux sont très variées, aucune situation ne ressemblant complètement à une autre. Cette multiplicité de cas de figure se retrouve lorsqu'il s'agit d'interpréter les essais réalisés et d'en déduire les modifications à apporter à la technologie ou au réglage des conditions d'environnement des pièces concernées.

Parmi les différents cas pouvant se présenter, les quelques exemples ci-après illustrent les enseignements qu'apporte la comparaison entre les valeurs calculées et celles résultant d'essais.

4.1.- Refroidissement d'un carter de turbine.

La technologie étudiée (Fig. 11) est un carter de turbine basse pression de moteur civil. Le principe de contrôle des jeux en sommet d'aubes est un refroidissement du carter par des rampes alimentées par de l'air de sortie de la soufflante. Ce refroidissement est dit "semi-actif" : une vanne permet deux réglages du débit d'air : l'un est utilisé dans la plupart des cas de fonctionnement, notamment au décollage et lors des phases transitoires pour éviter des interférences importantes entre le rotor et le stator, l'autre, fournissant un débit plus important, permet de diminuer les jeux lors du régime de croisière.

Ces débits ont été optimisés pour remplir deux objectifs principaux :

- Obtenir les meilleures performances en tenant compte de deux influences contraires : l'effet bénéfique de la diminution des jeux, et l'effet néfaste des prélèvements d'air (Fig. 12).
- Maintenir à des températures identiques les deux crochets maintenant chaque distributeur, de façon à éviter leur déversement qui serait incompatible avec le maintien de jeux axiaux faibles à la paroi intérieure de la veine.

Les premiers essais ont montré des écarts entre les températures mesurées et calculées en certains points du carter (Fig. 13), dus à une estimation imparfaite des débits de recirculation entre la paroi interne du carter et l'écoulement principal. Ce phénomène a été simulé et vérifié par le calcul.

Une amélioration des étanchéités et un ajustement du perçage des rampes de refroidissement ont permis de revenir à la répartition des températures voulue, et donc à des déplacements du stator corrects.

4.2.- Ecoulements parasites dans un compresseur.

Les écoulements parasites existants dans les compresseurs, et spécialement la recirculation autour des pieds des redresseurs (Fig. 14) ont pour conséquence d'abaisser le rendement des différents étages, mais aussi d'entraîner une diminution de la marge au pompage. Simultanément, les différents mélanges entre les débits d'air prélevés ou réintroduits dans l'écoulement principal ont une influence importante sur les températures de la partie extérieure des disques.

Une étude faite sur la première version d'un compresseur (calcul itératif entre les débits parasites, leurs températures de mélange, les températures des disques et les jeux) a montré :

- Que des fuites importantes subsistaient au travers des pieds d'aubes.

- Que cet air était réintroduit dans l'écoulement principal de manière localisée (Fig. 15 a), avec des valeurs élevées, rapportées en pour cents du débit du compresseur, même lors de fonctionnement à altitude élevée, lorsque la marge au pompage se réduit naturellement.

Une campagne d'essais utilisant les rayons "Super X" a permis de confirmer les hypothèses faites sur les jeux des labyrinthes sous redresseurs, et les températures de mélange calculées ont pû être comparées avec les indications fournies par les thermocouples placés dans les enceintes entre les grilles d'aubes.

L'amélioration de l'étanchéité des pieds d'aubes mobiles et l'ajustement des jeux des labyrinthes sous les redresseurs ont permis de réduire notablement ces écoulements parasites, et de mieux répartir les inévitables réintroductions dans l'écoulement principal (Fig. 15 b).

4.3.- Arrêt d'un moteur à grand Mach.

Un moteur installé sur un avion militaire volant à fort Mach (intercepteur par exemple) pourrait subir, pour une raison quelconque, indépendante du générateur de gaz lui-même, une extinction, suivie d'une période d'autorotation jusqu'à une altitude où le réallumage soit possible.

Cette éventualité, que l'on doit prendre en considération, entraîne une brusque variation de l'état thermique du moteur, génératrice d'importantes modifications des jeux radiaux et axiaux. Il faut donc vérifier, d'une part, qu'il n'y a pas de contacts radiaux pouvant entraîner un blocage du rotor et interdire le réallumage du moteur, d'autre part, qu'aucune interférence axiale ne se produit.

La simulation expérimentale de cette panne fournit les paramètres de vol et les conditions de fonctionnement du moteur en fonction du temps (Fig. 16).

Le calcul des températures et des déplacements de tous les points importants du moteur, effectué lui aussi en fonction du temps, permet de repérer les points critiques du moteur.
Sur l'exemple présenté (Fig. 17), on constate, sur un labyrinthe étagé, un contact axial entre une léchette et l'abradable de la léchette voisine.
Cette étude ayant été menée pendant le phase de conception, on a pu éviter une intervention ultérieure coûteuse en augmentant, sur le dessin initial, l'écartement entre les léchettes de ce labyrinthe.

4.4.- Déversement d'un distributeur de turbine HP.

Des défauts d'étanchéité sont apparus entre les plates formes et le support d'un distributeur haute pression. Ces défauts pouvaient s'expliquer par un déversement du distributeur (axe d'empilage de la pale non perpendiculaire à l'axe du moteur), entraînant un mauvais contact avec les supports inférieur et supérieur (Fig. 18).

Les mesures "Super X" réalisées ont montré un déversement du pied des aubes du distributeur vers l'arrière, lors des phases d'accélération entre le ralenti et le plein gaz. Celui-ci était dû à un temps de réponse différent des structures maintenant la plate forme supérieure et la plate forme inférieure de ce distributeur (Fig. 19).

De plus, en raison de la présence du flux secondaire, un déversement subsistait en fonctionnement stabilisé, les deux structures étant à des températures différentes.

Plusieurs modifications pouvaient permettre de remédier à cet inconvénient :
- Modification du temps de réponse de l'une des structures par augmentation d'épaisseur.
- Isolation de la structure extérieure vis-à-vis du flux secondaire.
- Changement de matière de certaines pièces.

A toutes ces solutions lourdes et coûteuses a été préférée une légère augmentation de la longueur du carter extérieur, qui introduit un déversement vers l'avant à l'arrêt, restitue la position correcte au plein gaz, tout en évitant les contacts partie fixe/partie mobile en transitoire.

5.- LES PERSPECTIVES.

La détermination, le contrôle et le réglage des différents jeux font maintenant partie systématiquement de la conception des moteurs civils, dont les conditions de fonctionnement en utilisation sont bien définies (Réf. 8, 9, 10, 11). L'optimisation des jeux est donc possible pour ces fonctionnements bien particuliers.

Une telle optimisation est beaucoup plus difficile pour les moteurs militaires dont l'analyse des conditions réelles d'utilisation (Fig. 20), montre que toutes les évolutions des paramètres sont envisageables, et que les régimes transitoires ont une grande importance.

Un des principaux axes d'études qui reste à approfondir est donc l'extension des plages de fonctionnement à prendre en compte lors de l'optimisation des jeux et des déplacements.

Cette démarche ne peut aboutir qu'en remplissant plusieurs conditions, d'ailleurs liées entre elles :

a)- Trouver des moyens d'action sur les jeux de plus en plus sophistiqués.

b)- Disposer de modèles de prévisions très performants dans une large gamme de cas de fonctionnement.

c)- Disposer de moyens de vérification et de validation des modèles de calcul de plus en plus perfectionnés.

La condition (b) est à la fois la source et la conséquence de la condition (a).

5.1.- Les systèmes d'optimisation.

Ceux-ci se sont appliqués successivement en ce qui concerne les jeux radiaux aux turbines HP, aux turbines BP, puis aux compresseurs HP. On est passé des systèmes passifs aux systèmes semi-actifs, et les prochaines générations de moteur possèderont un contrôle actif complet des jeux.

Un tel système sera en fait une chaîne de régulation complète permettant de minimiser à chaque instant les jeux. Les modèles de calcul (voir § 5.2) devront permettre d'identifier les paramètres influents dans tout le domaine de fonctionnement, et d'en déduire les lois de régulation remplissant l'objectif fixé.

Certains des paramètres fondamentaux à étudier prioritairement sont l'altitude, le Mach, les régimes de rotation, la température de sortie de la turbine (EGT), les conditions ambiantes (ΔISA), les temps de réponse des pièces, et les caractéristiques de l'air de refroidissement.

Une étape importante de cette analyse est le choix de la manière dont chacun des paramètres doit être traité : faut-il le négliger ? utiliser des lois continues ? définir des seuils ? La réponse à apporter dépend - cas par cas - du compromis nécessaire entre masse, coût et performances.

L'aboutissement de ces études très complexes passe par des études plus localisées visant à dégager des tendances permettant de restreirdre l'enveloppe des investigations ultérieures.

Un exemple de ces études préalables est la quantification des possibilités de modulation des jeux d'un compresseur HP de moteur civil en fonction de la technologie retenue.

La figure 21 présente deux technologies types : un carter "lisse", dont la partie structurale est très proche de l'écoulement principal, et un carter "suspendu" dont la partie structurale est éloignée de l'écoulement principal.

Une étude paramétrique de l'efficacité d'un refroidissement par impact d'air sur la paroi externe permet de sélectionner les solutions les plus prometteuses.

La figure 22 montre les évolutions des jeux radiaux en sommet d'aubes obtenues avec plusieurs hypothèses lors d'une accélération entre le ralenti-sol et le régime maximum.

5.2.- L'amélioration des modèles de calcul.

* Des méthodes de calcul précises et reposant sur des modèlisations fines sont utilisées depuis longtemps. Elles subsisteront pour fournir les résultats les plus exacts possibles, mais dans un nombre limité de cas de fonctionnement.
En effet, leur lourdeur de mise en oeuvre, même réduite par une automatisation poussée, n'en fait pas l'outil idéal pour des études paramétriques. Ces méthodes élaborées ne permettent pas non plus, en raison de la finesse de leur schématisation, de modèliser en une seule fois une partie étendue d'un moteur, malgré l'augmentation de la puissance de calcul disponible.

* De nouveaux modèles de calcul se développent, dont les caractéristiques principales sont une utilisation de méthodes de résolution plus simples, et une facilité d'utilisation, tant en ce qui concerne les conditions aux limites que la géométrie.

Les deux principales applications à développer ont pour objectifs :

- Etablir les coefficients d'influence et les lois de régulation optimales pour réaliser les objectifs évoqués au § 5.1.

- Prédire aisément, dans tout le domaine de fonctionnement du moteur, et de manière analytique, les évolutions des jeux. De tels modèles peuvent être localisés pour la détermination des jeux radiaux les plus critiques, ou globaux, c'est-à-dire capables de schématiser l'ensemble du moteur pour calculer tous les déplacements relatifs entre les parties fixes et mobiles.

Ces différents modèles doivent bien sûr être validés. Cette validation peut s'obtenir par la comparaison des résultats avec ceux obtenus avec les modèles existants (mais dans un nombre relativement limité de configurations), ou par des mesures, telles que les "Super X", les capteurs (optiques ou capacitifs). Cette validation sera d'autant meilleure que l'on peut espérer disposer à terme de capteurs dont la précision, la fiabilité et la tenue à la température seront améliorés, et dont la taille sera suffisamment réduite pour leur permettre d'accéder aux labyrinthes internes.

6.- CONCLUSION.

Les jeux radiaux et axiaux sont dépendants de la plupart des paramètres intervenant dans le dimensionnement d'un moteur, et constituent d'une certaine manière une synthèse des différentes études réalisées.

Vouloir les maîtriser et les ajuster plutôt que les constater nécessite une analyse qui doit intervenir dès l'avant-projet : c'est en effet à ce stade que se définit la technologie d'ensemble.

Compte-tenu de la grande variété des cas à prendre en considération, les agencements technologiques retenus doivent être perfectionnés en faisant largement appel aux différentes techniques de calcul et d'optimisation, puis être validés ou ajustés grâce aux essais.

Cette démarche est maintenant utilisée systématiquement pour les moteurs civils, l'étape ultérieure est l'application des méthodes d'optimisation aux moteurs militaires , pour lesquels de nouveaux modèles de calcul permettront de simuler, sans une lourdeur excessive, toutes les évolutions représentatives d'un avion de combat moderne.

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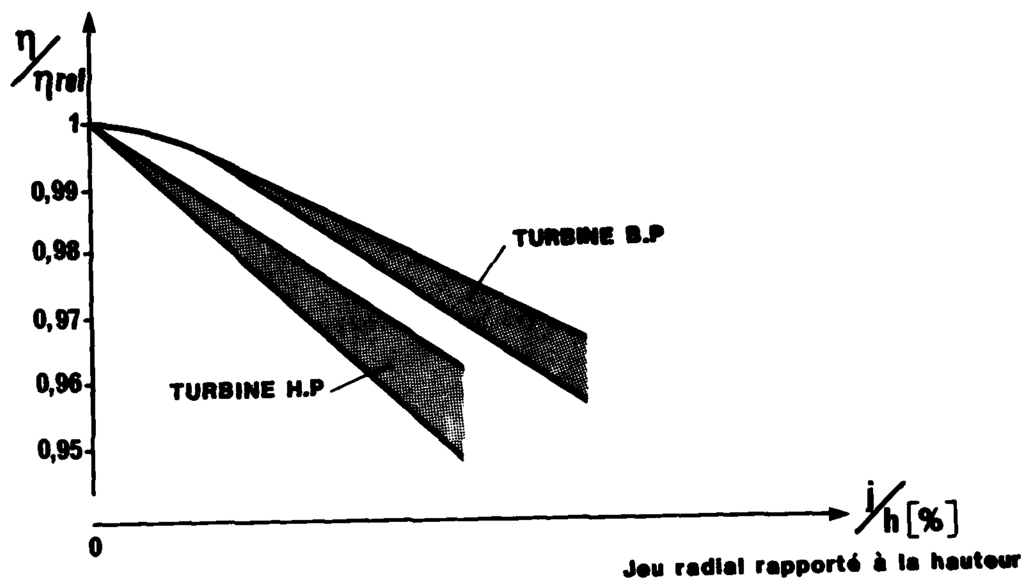


Fig.1 Influence du jeu radial sur le rendement de turbine

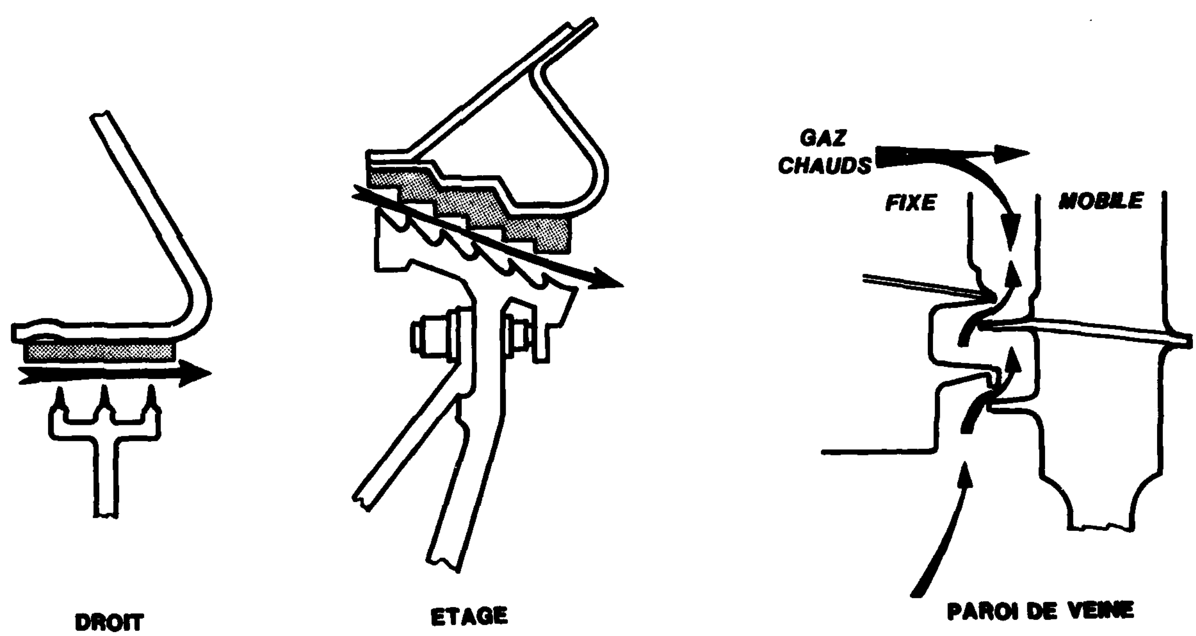


Fig.2 Principaux types de labyrinthes

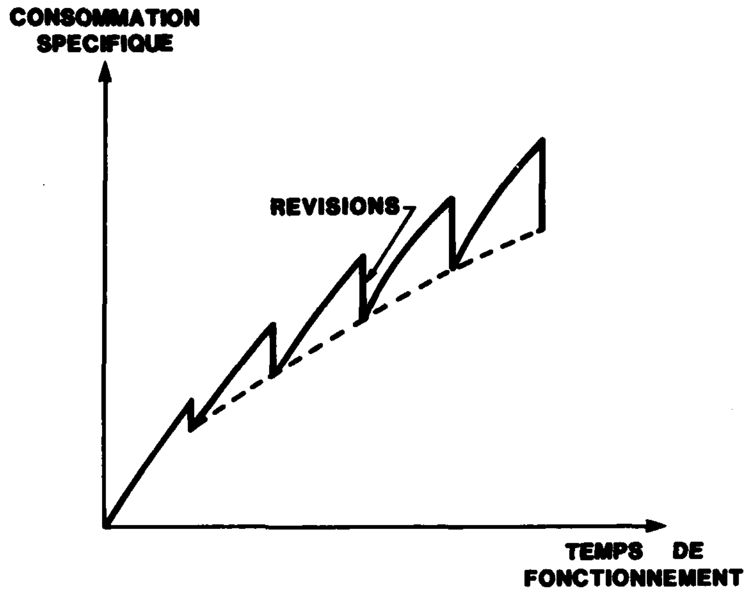


Fig.3 Degradation des performances (d'après réf.1)

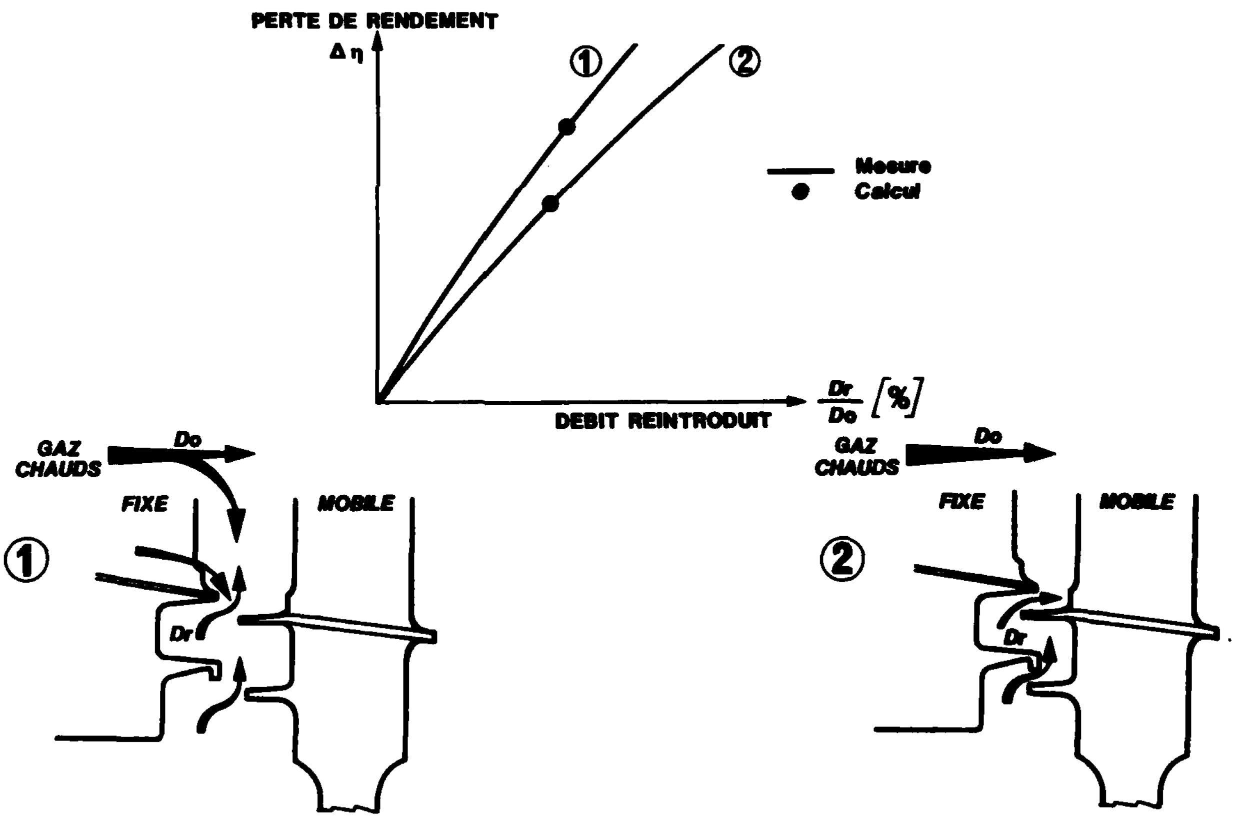


Fig.4 Paroi interieure de veine: influence des jeux axiaux

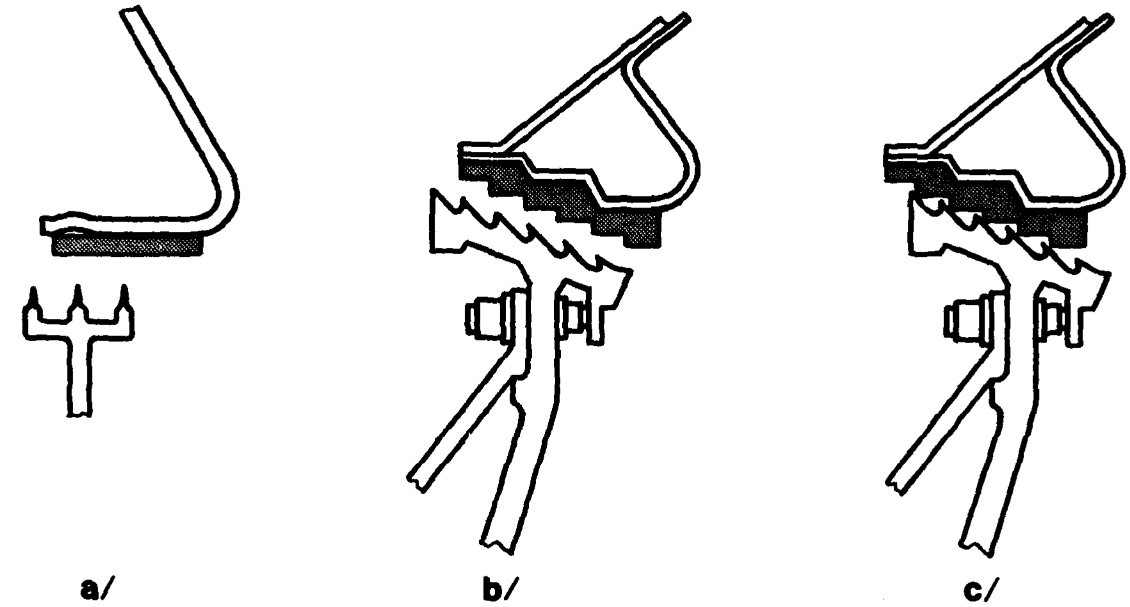


Fig.5 Deplacements axiaux des labyrinthes

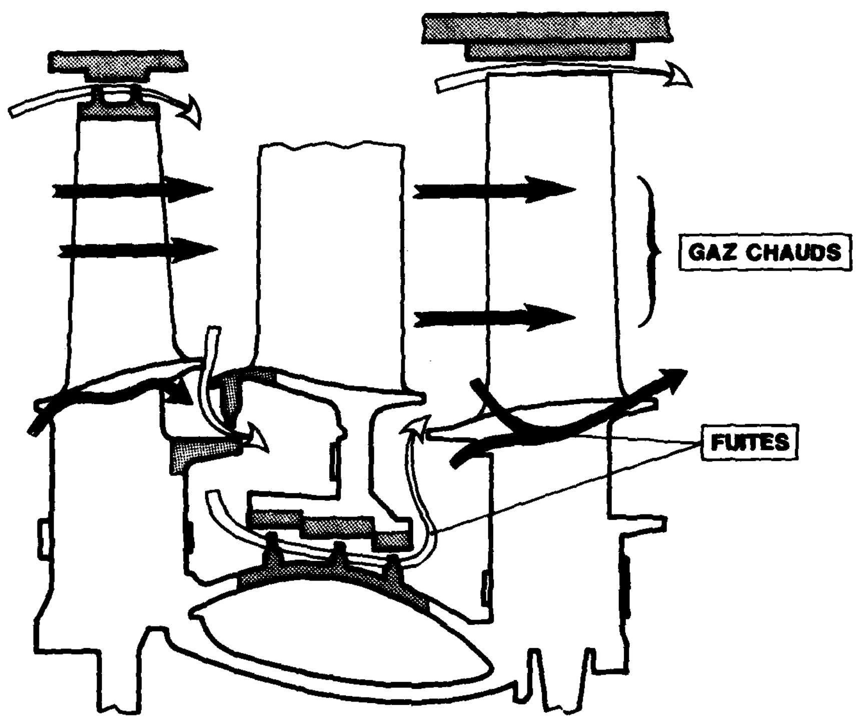


Fig.6 Les systemes d'etancheite dans une turbine

PARAMETRES	COMPRESSEUR HP		TURBINE BP	
	AXIAL	RADIAL	AXIAL	RADIAL
ΔL centrifuge	+	+++	+	+++
ΔL thermique	+++	+++	+++	+++
Tolérances	+++	++ou +	+++	++
Déflections Aéro.	+	/	+	/
Facteurs de charge	/	++	/	++
ΔL vibrations	+	/	/	/
Jeux roulement	++	+	+	+
Pompage	++	/	++	/
Excentrage	/	+	/	+
Balourde	/	+	/	+
Déformations/ Ovalisations	+	+	+	++

+++ Influence très importante
++ '' importante
+ '' secondaire
/ '' négligeable

Fig.7 Importance relative des parametres de dimensionnement des jeux

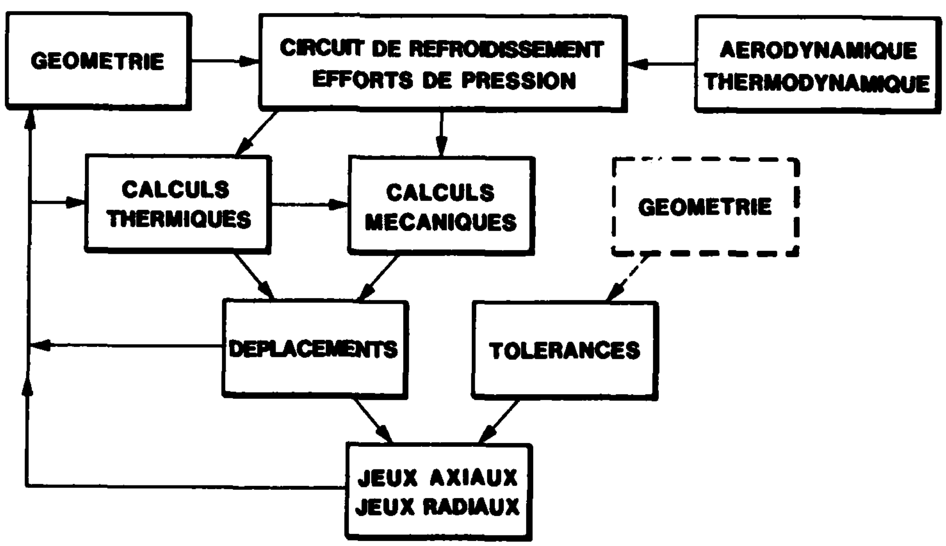


Fig.8 Organigramme de calcul des jeux

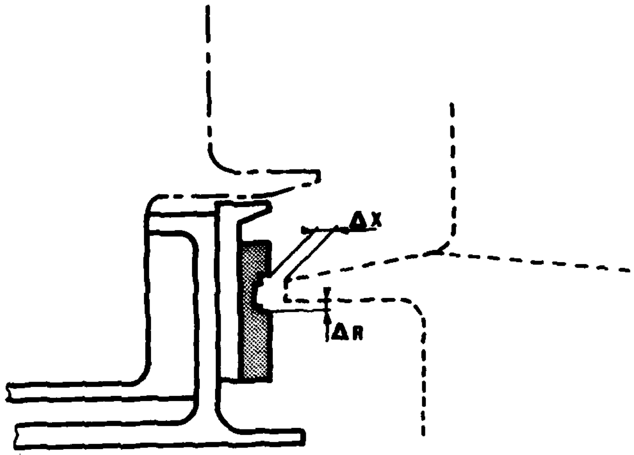


Fig.9 Temoin de deplacement

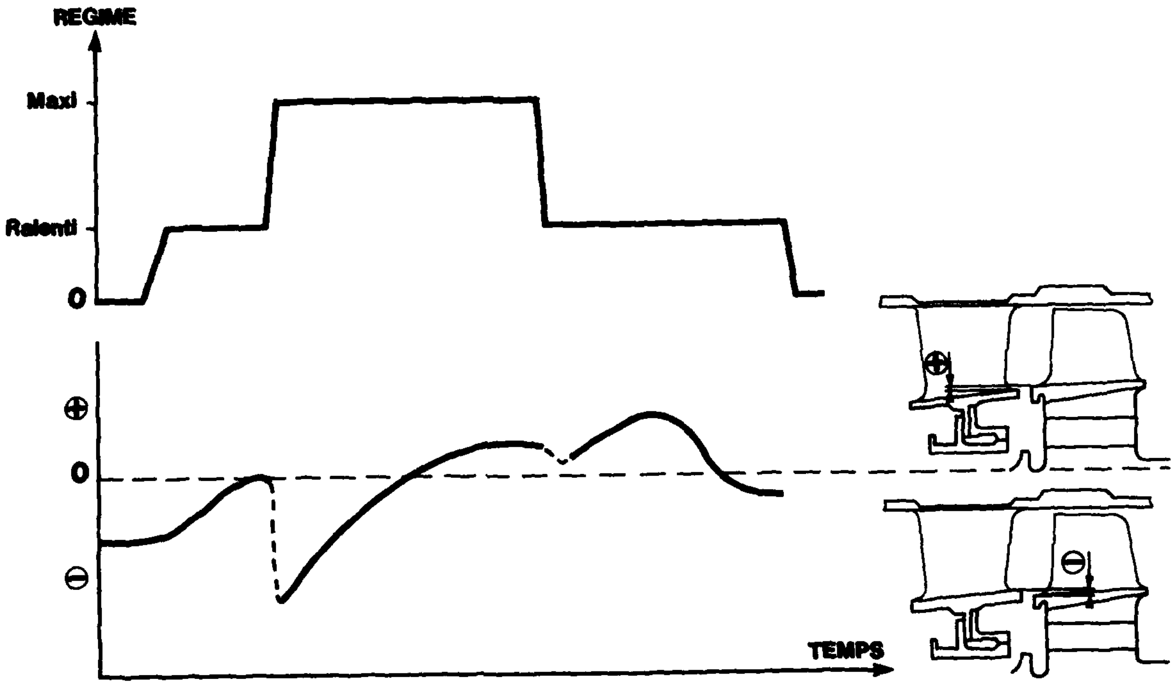


Fig.10 Mesure par rayons "Super X"
de l'alignement de la veine principale

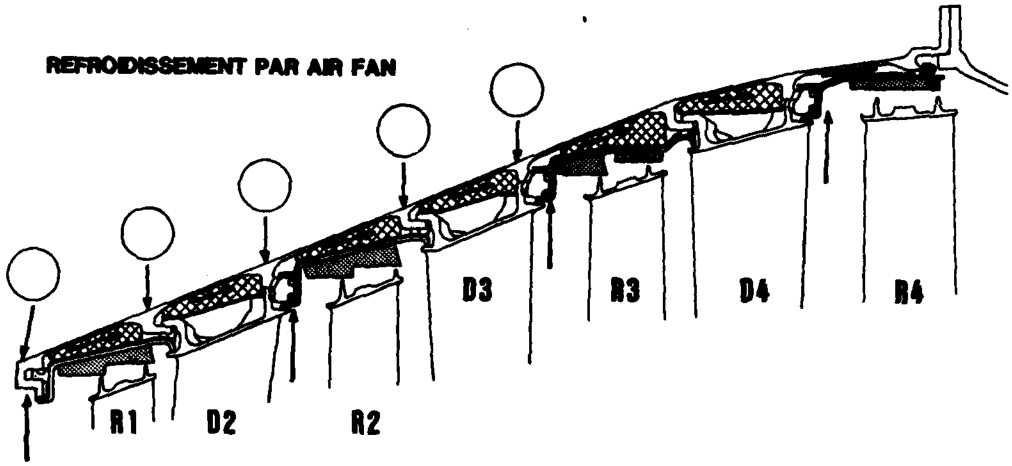


Fig.11 Carter de turbine BP – environnement

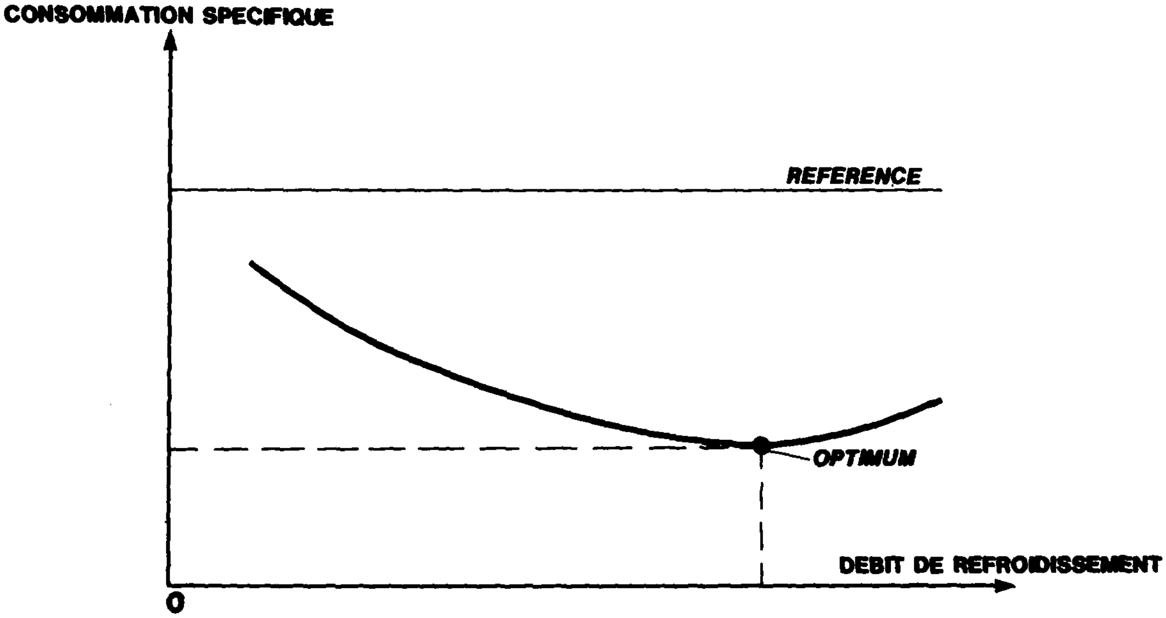
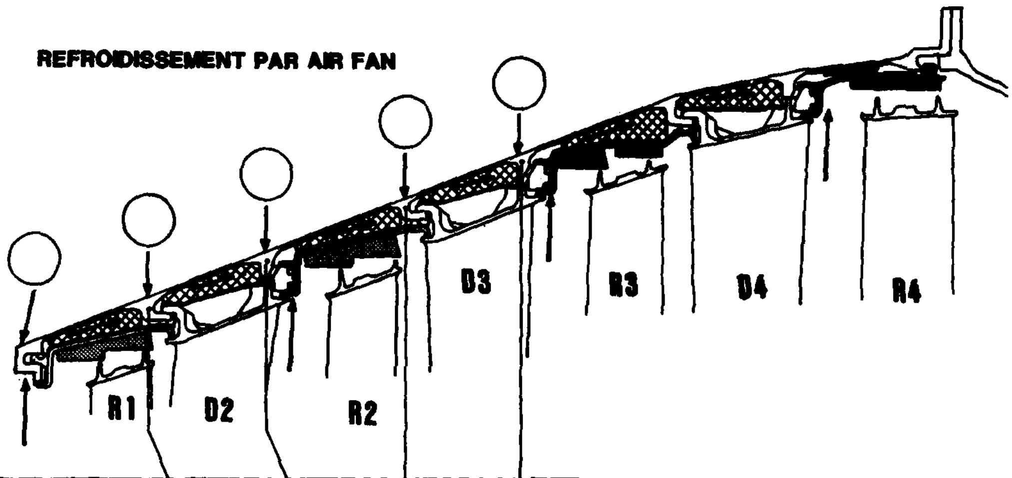


Fig.12 Carter de turbine BP
optimisation du debit de refroidissement



PREDIMENSIONNEMENT	455	446	439	443
PREMIER ESSAI	344	513	421	471
VERSION MODIFIEE	449	385	433	385

CONDITION DECOLLAGE

TEMPERATURES EN °C

Fig.13 Carter de turbine BP – ajustement du refroidissement

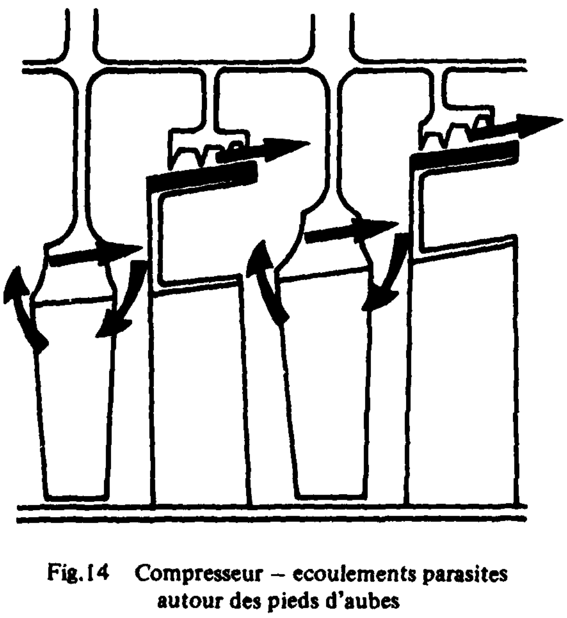


Fig.14 Compresseur – ecoulements parasites autour des pieds d'aubes

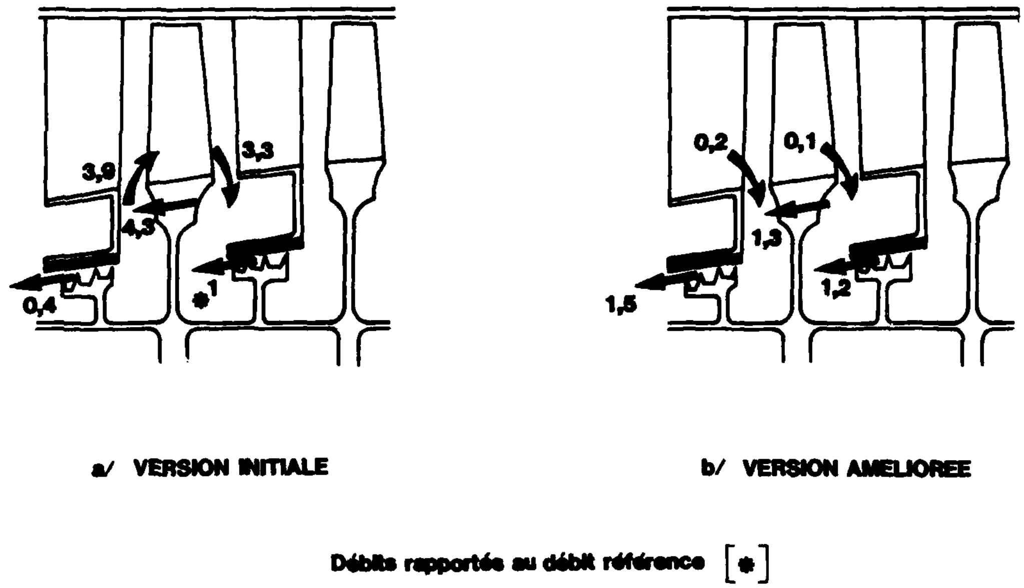


Fig.15 Compresseur - reduction des ecoulements parasites

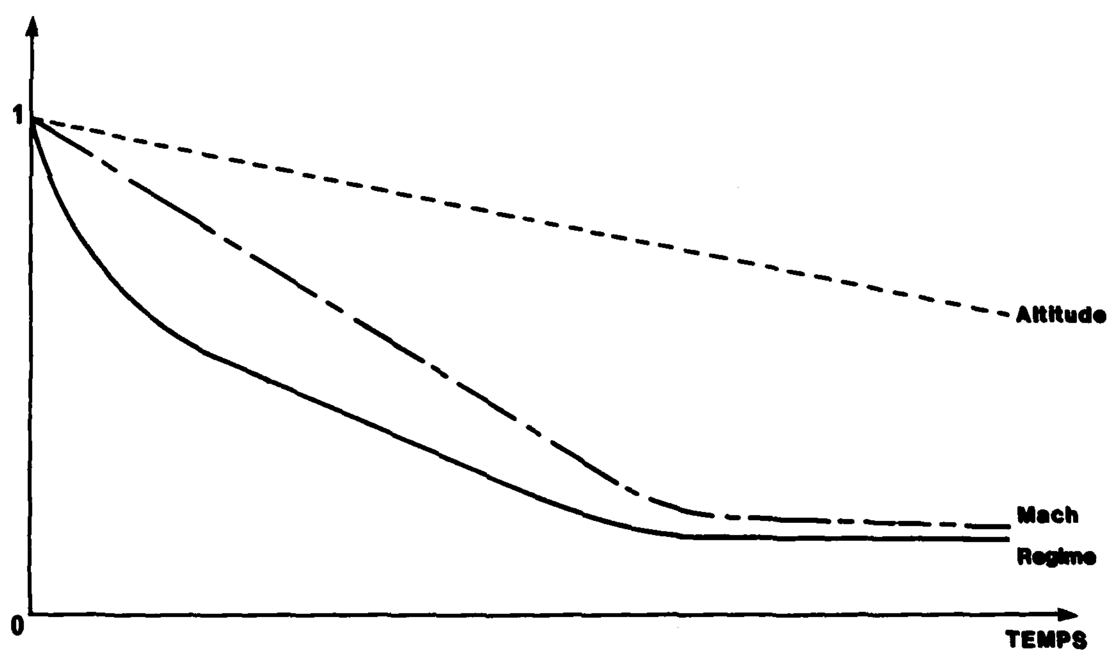


Fig.16 Arret moteur a fort Mach
evolution des parametres de vol

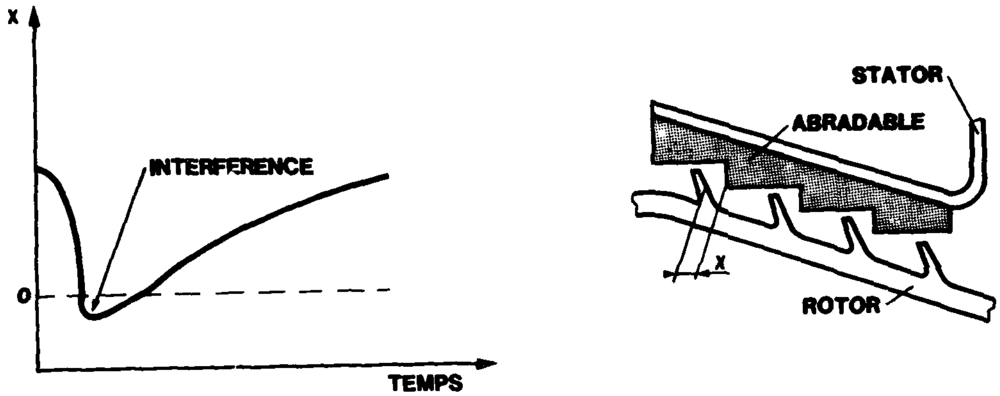


Fig.17 Arret moteur a fort Mach – evolution d'un jeu axial

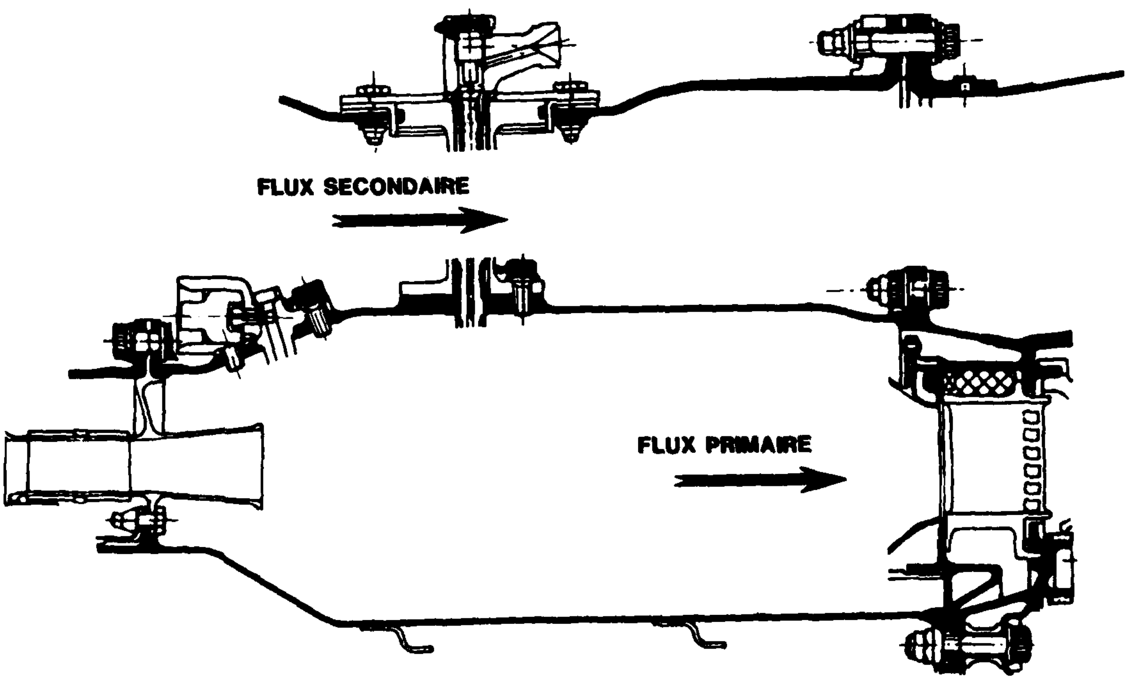


Fig.18 Distributeur de turbine HP – technologie des supports

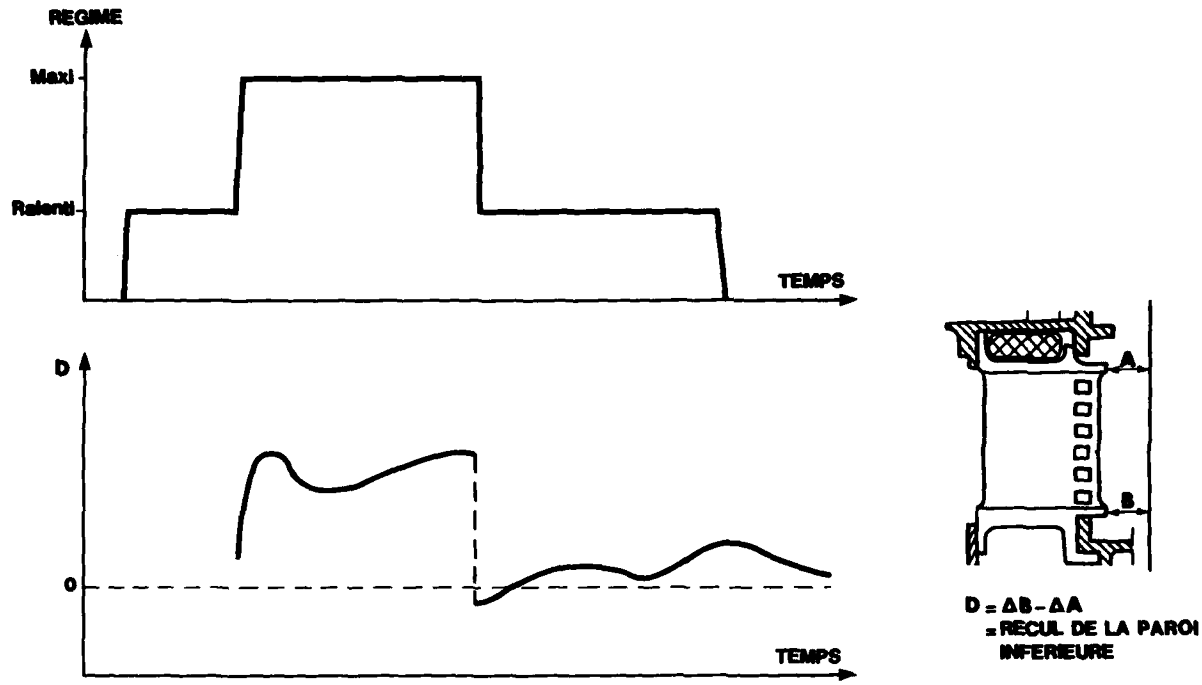


Fig.19 Distributeur de turbine HP
mesure du deversement par "Super X"

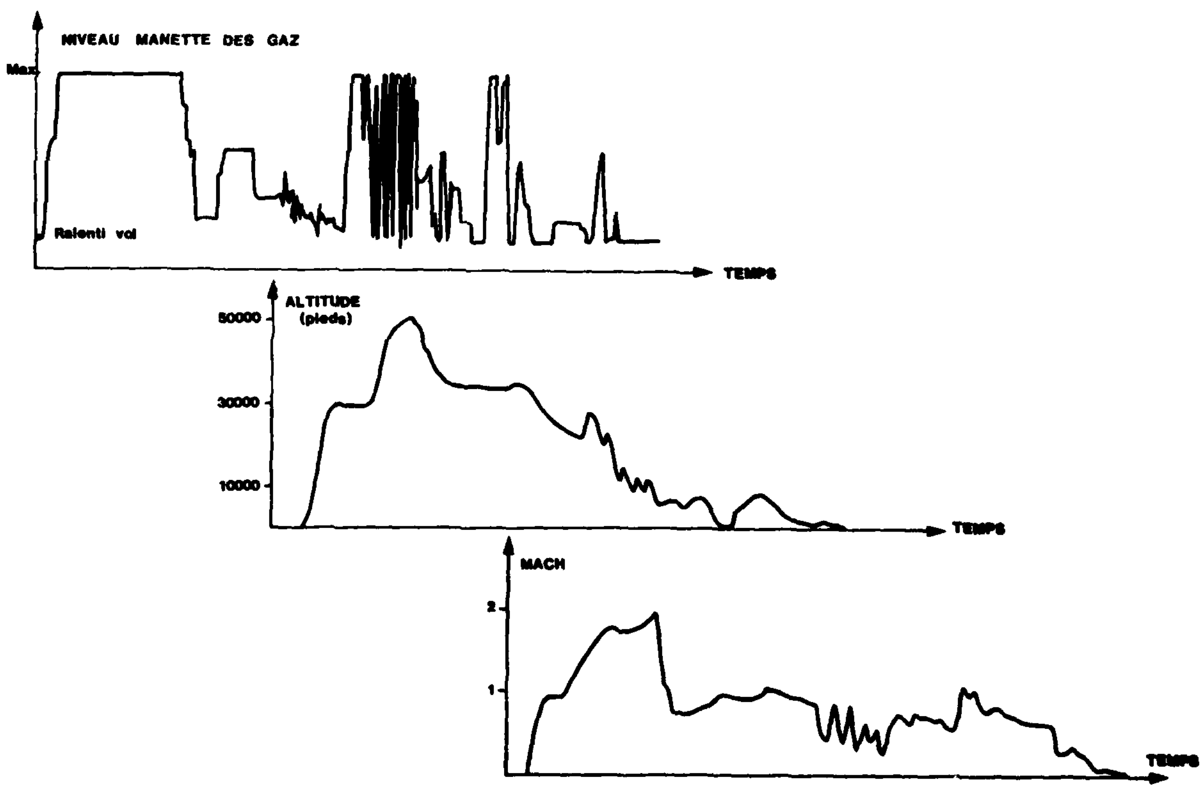


Fig.20 Profil de mission moteur enregistre
sur un avion de combat moderne

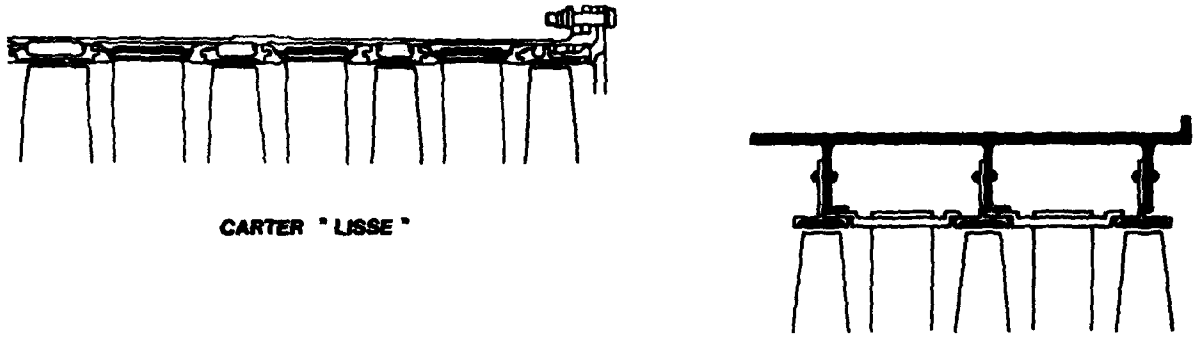


Fig.21 Carter de compresseur – exemples de technologies

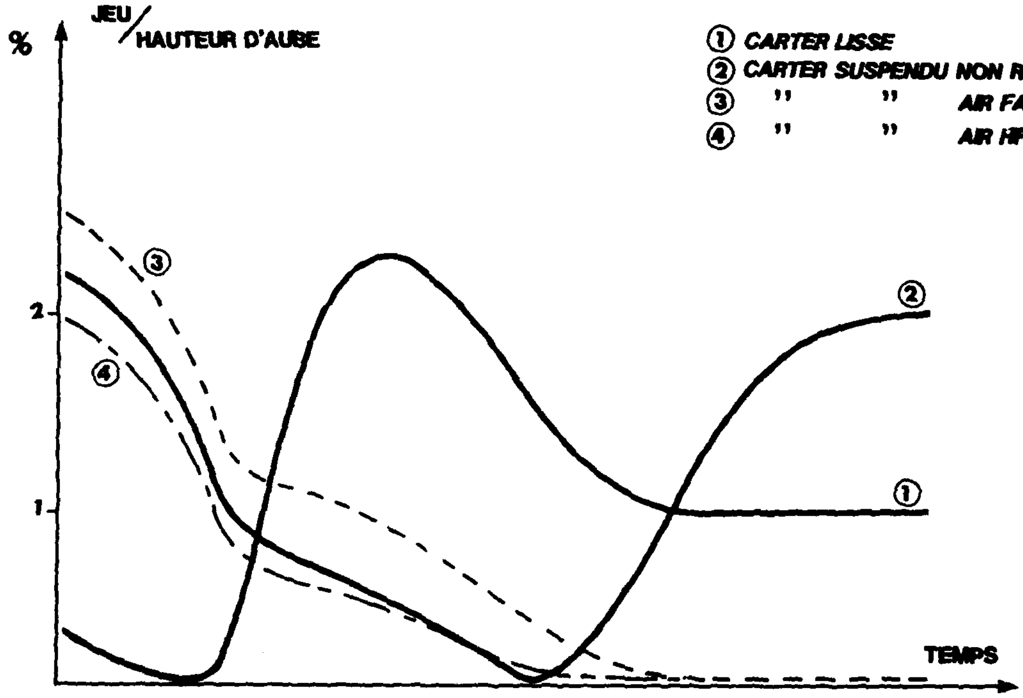


Fig.22 Compresseur – jeux radiaux en sommet d'aubes.
Influence de la technologie et du refroidissement
pendant la phase d'acceleration

DISCUSSION

W. Heilmann, Ge.

Ref. Fig. 7. Importance *relative* des *paramètres de dimensionnement* des *jeux*:

J'accepte que l'approche théorique doit être fondée sur un système axisymétrique. Mais je pense que l'excentrage et les déformations/ovalisations n'ont pas une influence secondaire.

Author's Reply

L'approche théorique *comporte* effectivement deux phases qui peuvent être traitées simultanément: d'une part le calcul axisymétrique pour déterminer la valeur des jeux moyens, d'autre part la quantification des déformations locales qui permet de déterminer les jeux *minima*. La recherche du jeu optimal doit prendre en compte les deux resultats obtenus.

L'importance des déformations non axisymétriques est très dépendante de la technologie des parties structurales du moteur. Elle peut donc, suivant les cas, être plus grande (ou plus faible) que celle que j'ai indiqué.

TURBINE TIP CLEARANCE CONTROL IN GAS TURBINE ENGINES

by

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SUMMARY

High aerodynamic efficiencies of the turbomachinery of modern aero-engines are only possible if tight tip clearances are achieved by proper thermal matching of rotor and casing. This paper describes various possibilities of passive and active tip clearance control systems that are designed to avoid large rub-ins as well as excessively large gaps during transient operation whilst maintaining very tight clearances during steady-state operation. The impact of these systems on the mechanical design is discussed.

Advanced clearance control configurations are described for two actual examples, i.e. the gas generator turbine of a shaft power engine for helicopters and the low-pressure turbine of a large turbofan engine for transport aircraft. The results of the analytical and mechanical designs are presented and the benefits are demonstrated.

1. INTRODUCTION

Advances in analytical and experimental methods in aerodynamics have raised the compressor and turbine efficiencies of modern aero-engines to 90% and more. Thus, further improvements by aerodynamic means alone will be difficult. Future designs will have to focus very carefully on features such as reduced leakage flows and extremely tight clearances throughout the flight cycle.

This paper addresses the problem of achieving and maintaining tight tip clearances, i.e. the gaps between the rotor blade tips and casing. It will be shown how tip clearances affect the fuel economy of today's aero-engines. Then, based on a description of the growth behavior of rotor and casing, principal methods to reduce and control these clearances are discussed and the necessary theoretical tools outlined. Two different examples of turbine tip clearance control, one for a helicopter engine, the other for an engine for a transport aircraft will be presented along with the benefits to be achieved.

2. PROBLEM DEFINITION

Fig. 1 shows that tip clearances have a strong influence on the aerodynamic efficiencies of compressors and turbines (Ref. 1, 2). An increase in rel. tip clearance (based on blade length) of 1%-pt. will reduce the efficiency by about 1.5%. The consequence of such an efficiency drop on the fuel consumption of an engine is indicated at the bottom of Fig. 1. An efficiency drop, for instance, of 1% in each of the four components LP and HP compressor, HP and LP turbine of a two-shaft turbofan engine increases the fuel consumption by about 2%. The same efficiency drop in the three components of a turboshaft engine with free power turbine increases fuel consumption even by 3.5%.

Fuel consumption is of paramount importance for modern aero-engines. Its influence on the life cycle cost of an aircraft is illustrated in Fig. 2 (Ref. 3 to 5) for the example of a fighter aircraft program. Assuming that its cost comes to, say, $ 20 billion within 15 years, then a fuel consumption increase of 2% is equivalent to a life cycle cost increase of $ 280 million or about $ 20 million per year. The same tendency applies to helicopters and transport aircraft (Ref. 6). Thus, there is a strong incentive to achieve high efficiencies by tight tip clearances.

Handling of aero-engines characterized by numerous rapid accelerations and decelerations may have a strong effect on tip clearances if special measures are not taken. This effect on clearances is caused by the difference in the radial displacement behavior of the rotor and casing during transients, as indicated in Fig. 3 for an acceleration. The rotor and casing radial displacement pattern is similar for compressors and turbines. Owing to the higher temperatures in the turbine area, however, the effects are usually more pronounced there. During a fast acceleration from idle to take-off condition the rotor expands very rapidly at first owing to the centrifugal forces and fast thermal expansion of the blades. This takes about the time until the take-off rotational speed is reached, i.e. in the order of 5 seconds. Then the rotor continues to grow, but at a much slower rate. This is due to the slow thermal expansion of the disks, which have a relatively large mass, whose bulk is remote from the main stream. The time constant of the disks is typically of the order of 5 to 10 min.

In contrast to the rotor, the casing, with no centrifugal forces acting, starts growing slowly at first, until heat soaks in. But then the growth rate of the casing exceeds that of the rotor because the thin-walled structure has a small mass and is more directly exposed to the main stream.

As a result of the different expansion rates of rotor and stator, the blade tip clearance (Fig. 3 bottom) is reduced rapidly as the engine is accelerated, reaches a minimum some seconds after take-off power has been reached and then increases again. At take-off of the aircraft the blade tip clearance is further reduced by rotor and casing deflections caused by thrust and maneuver loads. The maximum clearance reduction by load deflections need not coincide with the clearance minimum resulting from the different expansion rates of rotor and casing - but it can, as shown in Fig. 3. Thus, at the so-called "pinch-point", blade tip clearance is reduced to zero and this condition then defines the blade tip clearance at idle, the cold clearance and the resulting clearances at other power settings such as cruise if nothing is done to control them. Although not shown in Fig. 3, it should be noted that a similar "pinch-point" can be found after a deceleration of the engine from a steady-state high-power setting followed by a reacceleration when the casing has cooled down while the disk is still hot - a so-called "hot reslam". However, since the problems arising from this pinch-point regarding steady-state blade tip clearances are the same, we will confine ourselves to the discussion of the acceleration phase.

Fig. 3 also shows that, even if tip clearances at steady-state cruise can be controlled, large transient clearances may occur, causing a corresponding efficiency drop in the engine components, leading to a considerable thrust droop, if a temperature or speed limiter is used. The minimum thrust, which in some engines has been measured to be about 15% below the full-power value, may occur after about 20 - 40 seconds, when the aircraft may have reached the end of the runway and be about to take off. This is precisely when maximum thrust is required, not minimum. In addition large intermittent gaps may even cause modern highly loaded compressors to surge.

Obviously, a radial displacement and tip clearance behavior, as shown by the solid lines in Fig. 3, is unacceptable and measures to reduce and control the transient and steady-state clearances have to be taken. By proper matching of the thermal behavior of rotor and casing, a radial displacement and tip clearance pattern, as shown by the dashed curves in Fig. 3, may be achieved, resulting in a clear improvement in the situation. Therefore, efficient and practical methods to control tip clearances are called for.

3. PRINCIPAL METHODS OF TIP CLEARANCE CONTROL

Methods to influence the thermal behavior of a compressor or turbine usually require to make the rotor thermally faster and the casing thermally slower. For the rotor this can be achieved, for instance, by venting the cavities between the disks by a small amount of air taken from the main stream. However, measures to speed up the rotor thermal response are limited and not very effective. Therefore, clearance control methods concentrate on the casing. To slow down the casing one may simply add mass which, of course, is not exactly desirable in aero-engines. Therefore, more effective methods of casing construction have to be developed.

One may distinguish between two principal ways of tip clearance control:

1) Passive clearance control. The secondary air system and the casing are carefully designed toward the desired thermal behavior. Then, during engine operation, the casing reacts passively to changes in engine power setting. There is no possibility of controlling the thermal expansion of the casing independent of the engine running condition.
2) Active clearance control (ACC). This involves means by which, using the engine control system, the temperature of the casing can be actively influenced to achieve optimum values more or less independent of the engine running condition. This can be achieved, for instance, by a cooling system whose coolant flow can be turned on and off to attain the desired casing temperature.

4. COMPUTATIONAL METHOD

The design of an effective compressor or turbine tip clearance control system calls for a very intricate computational method to predict the transient temperature fields and radial displacements. The structure of a corresponding computer program system is depicted in Fig. 4 (Ref. 7). The left-hand column shows the computation sequence, and the center column the data files and subroutine library. The first step is to take the relevant geometry data and generate a finite element grid. Then the thermal property values are evaluated. After that the boundary conditions (heat transfer coefficients, air temperatures, etc.) for each surface element are computed using cycle, air system and mission data and calling-up the relevant subroutines. These subroutines contain relationships describing the heat transfer process for typical situations, such as flat plates, channels of various geometries, rotating disks, etc. In some cases detailed boundary layer calculations are performed to evaluate the heat transfer coefficient. The next block, the heart of the program, solves the heat conduction equation for one time step. Then the procedure is repeated for all time steps throughout the flight mission. The temperatures are stored and, as indicated in the right-hand column in Fig. 4, are presented together with the geometry in isotherm plots, in graphs as function of time, and as mean values. In some cases, heat fluxes and temperature gradients are of interest. These results are used by the heat transfer engineer for making an initial evaluation as to whether the desired thermal behavior has been achieved. Then, the results are processed further in programs such as NASTRAN or MARC that determine the stress distribution and the displacements.

For describing the generally highly complex geometry with curved surfaces it is convenient to employ isoparametric quadratic or cubic elements as shown in Fig. 5 (Refs. 8 and 9). They can be used directly for stress calculations without the necessity for interpolation onto another finite element grid. In this way, computation is simplified and there is no unnecessary loss of accuracy.

5. PASSIVE CLEARANCE CONTROL SYSTEM FOR A HELICOPTER ENGINE

In this chapter, an example for a passive clearance control system is presented. Fig. 6 shows a schematic view of a turboshaft engine in the 900 kW-class, being developed at MTU (Ref. 10). It has a two-stage gas-generator turbine surrounded by an annular combustion chamber, which doesn't leave much space for an active clearance control system. Therefore, passive clearance control is used, i.e. a conventional segmented liner design in stage 1 and a simpler, but nevertheless highly efficient design, in stage 2.

The arrangement for stage 2, as shown in Fig. 7, consists of two concentric rings, elastically suspended separately at the inner wall of the combustor casing. The inner ring or liner, which forms part of the turbine annulus, has only a small mass. It is insulated on the inside by a layer of zirconium oxide sprayed into the honeycombs, and is impingement cooled at the outside through the outer ring. This outer ring has a considerably higher mass, which enables it to act as a containment ring for the turbine wheel. It is also insulated at its inner surface by a ZrO_2 layer, which controls the heat transfer from the inner to the outer ring.

It is evident that this liner arrangement is a very simple solution, which in addition has the advantage of complete tightness against hot gas leakages compared with the conventional segmented liner arrangement. Its blade tip clearance control capability is shown in Fig. 8. The calculated radial growth behavior of the rotor forms the basis for the sizing of the inner and outer ring. To avoid severe rubbing under high load deflections, sufficient blade tip clearance should be available at the pinch-point. The magnitude of possible load deflections depends on rotor and casing stiffnesses as well as on the bearing arrangement, and is calculated together with the rotor-dynamic behavior of the engine. The desired blade tip clearance at the pinch-point and the thermal expansion rate of the liner then define the radial position of the liner and the corresponding blade tip clearance at idle. Looking first at the radial position of the liner at idle, it can be seen that the blade tip clearance at full power would be too large if the liner ring were not cooled. But cooling alone is not sufficient, because it cannot avoid a large transient clearance, which leads to a considerable temporary power loss, as explained earlier. To close this transient clearance, the casing (or outer ring in the described design) comes into action. Its slow response to changes in the engine operating condition keeps the inner liner ring from expanding too quickly. Thus, the blade tip clearance can be controlled within close tolerances over the whole operating range.

This turbine tip clearance control system has been built and is currently being tested in a demonstrator engine to evaluate its potential.

6. ACTIVE CLEARANCE CONTROL SYSTEM FOR A TURBOFAN ENGINE

The HP compressor and the turbines of new turbofan engines, such as the PW 2037 which is shown in Fig. 9, are designed with an active clearance control system as an integral part of the design. The method chosen to control the clearances for the PW 2037 will be explained for the LP turbine. The principle is to cool the casing when the tip clearances are to be reduced. This is effected by extracting air behind the fan and ducting it through a pipe to a manifold that feeds a number of pipes surrounding the LP turbine casing circumferentially, as can be seen in Fig. 9 and in greater detail in Fig. 10. The axial location of each pipe corresponds to the place where a casing hook supporting the liner segments is situated. Each pipe has a row of holes facing the casing. The bleed air is blown through the holes against the outside of the casing. In this way, the bleed air, which is colder than the turbine casing, can be used for an effective impingement cooling of the casing to achieve a desired temperature and shrinkage of the casing, resulting in tight tip clearances. Since the amount of cooling flow can be regulated by valves, it is possible to actively control the casing temperatures, and thus the clearances, to attain the values optimal for each point in the flight cycle.

For the design of this system careful calculation of the transient temperature fields of both the rotor and the casing for the whole flight cycle with and without various levels and schedules of case cooling was carried out. The transient clearances were computed from these temperatures. An example of a part of the finite element grid of the rotor is depicted in Fig. 11 (for clarity the grid in the blade and blade root area is not shown). The main stream as well as the secondary flow and hot gas ingestion, which have to be modelled very carefully, are indicated. Proper convective heat transfer coefficients for each surface element and temperature values for the surrounding air are evaluated at each time step. Because of the very complicated flow pattern between the co-rotating disks with radial throughflow, the models developed by Owen and his coworkers (Ref. 11) are used for the disk heat transfer coefficients.

As an illustration of a result obtained using these models, the temperature distribution 40 seconds after an acceleration from idle to take-off condition is shown in Fig. 12. Strong axial and radial temperature variations can be observed.

For the design of the cooling system the pipe diameters were chosen as small as possible to minimize the extra weight, but large enough to ensure a relatively uniform circumferential distribution in the cooling flow rate. The diameter and pitch of the impingement holes as well as the distance between the pipes and the casing were optimized toward minimum cooling flow rate using correlations that gave very similar heat transfer coefficients as those in Ref. 12.

To check the validity of the temperature calculation, both the rotor and the casing of an LP turbine installed in an engine were extensively instrumented and a comprehensive test program performed. Good agreement between measured and calculated temperatures could be observed, as shown for 2 measured points in Fig. 13. This supports the validity of the tip clearance predictions.

The tip clearance behavior resulting from the calculated and measured temperatures can be seen in Fig. 14 for the first stage of the LP turbine. The transient radial displacement of the rotor and casing is plotted. The dashed curve indicates the radial displacement of the casing when the ACC system is turned on 40 seconds after the acceleration to take-off power. This results in a tip clearance reduction of about 1.2% at cruise condition, which, according to Fig. 1, is equivalent to an efficiency increase of approximately 1.8% for that particular stage. Similar, but decreasing improvements (owing to the increasing blade lengths) are attained in the subsequent stages of the LP turbine. The overall benefit of this clearance reduction in the LP turbine, taking account of the extra weight and the loss due to the extraction of cooling air (less than 0.5% of the core mass flow), is a lowering of the fuel consumption by about 0.4 - 0.5%, which is certainly a benefit worth striving for.

Development engines of the PW 2037 program are currently being tested. So far the predicted saving in specific fuel consumption has been confirmed.

7. CONCLUSIONS

It has been shown that the aerodynamic efficiencies of compressors and turbines of modern aero-engines can be further improved by careful design for tight tip clearances. The principal methods of passive and active clearance control systems are discussed. The design of these systems necessitates very intricate computational methods. With two actual examples, one for a passive control system for a helicopter engine, the other for an active control system for a turbofan engine of a transport aircraft, the achieved reductions in tip clearances are demonstrated. For the turbofan engine, the active clearance control system of the LP turbine alone lowered the fuel consumption by about 0.4 - 0.5%. This improvement more than offsets the additional complexity and cost.

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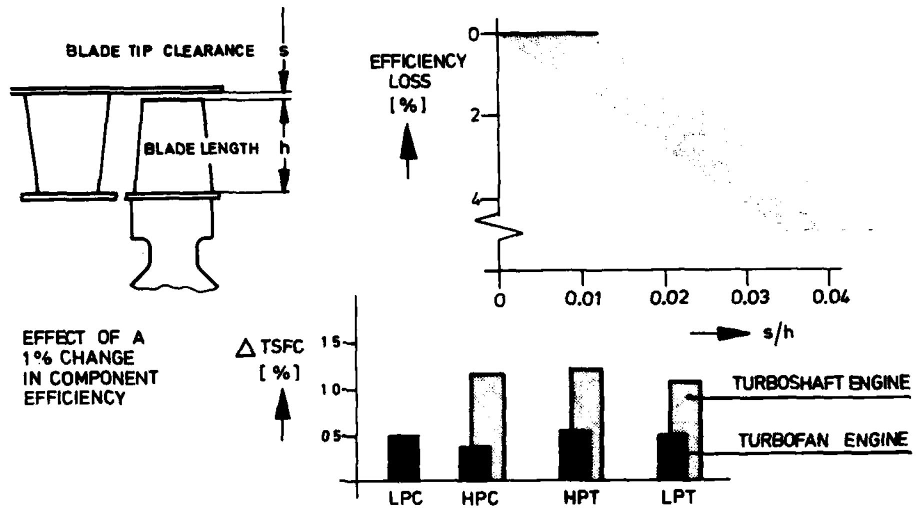

Fig. 1   Influence of blade tip clearance on engine fuel consumption

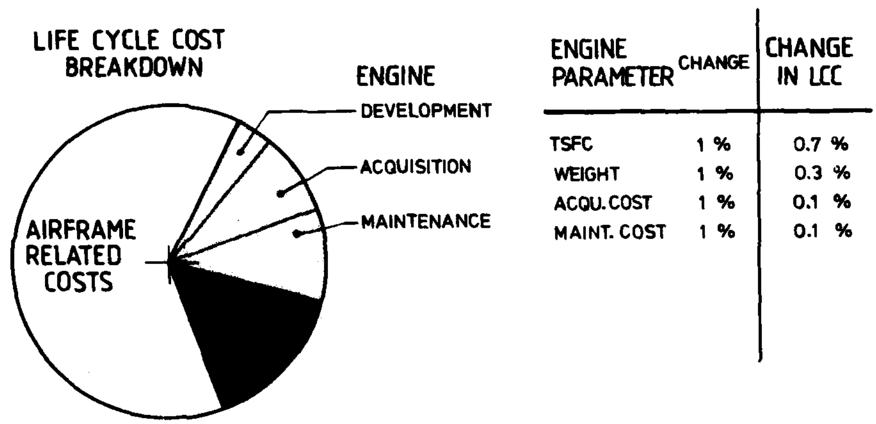

Fig. 2   Influence of engine parameters on life cycle cost of a high performance fighter aircraft

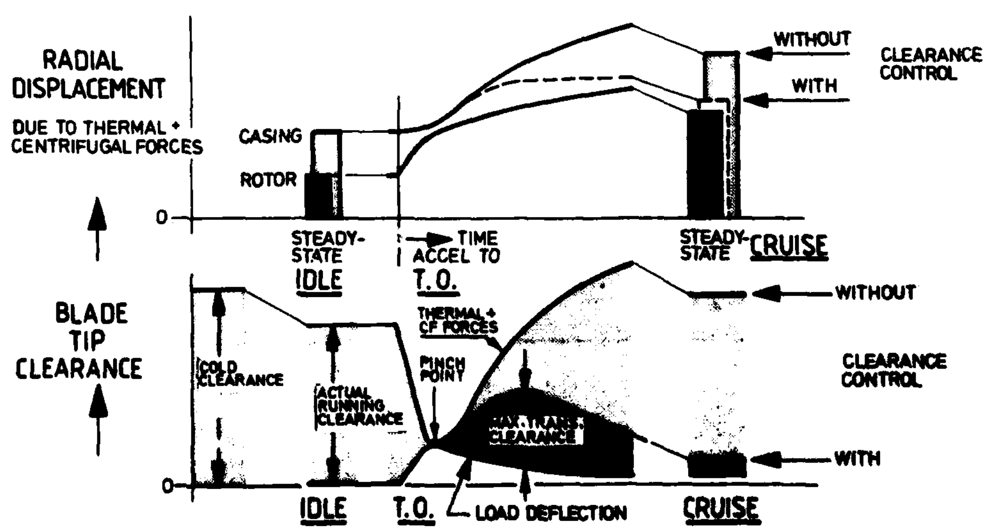

Fig. 3   Rotor and casing radial displacement pattern in an aero-engine

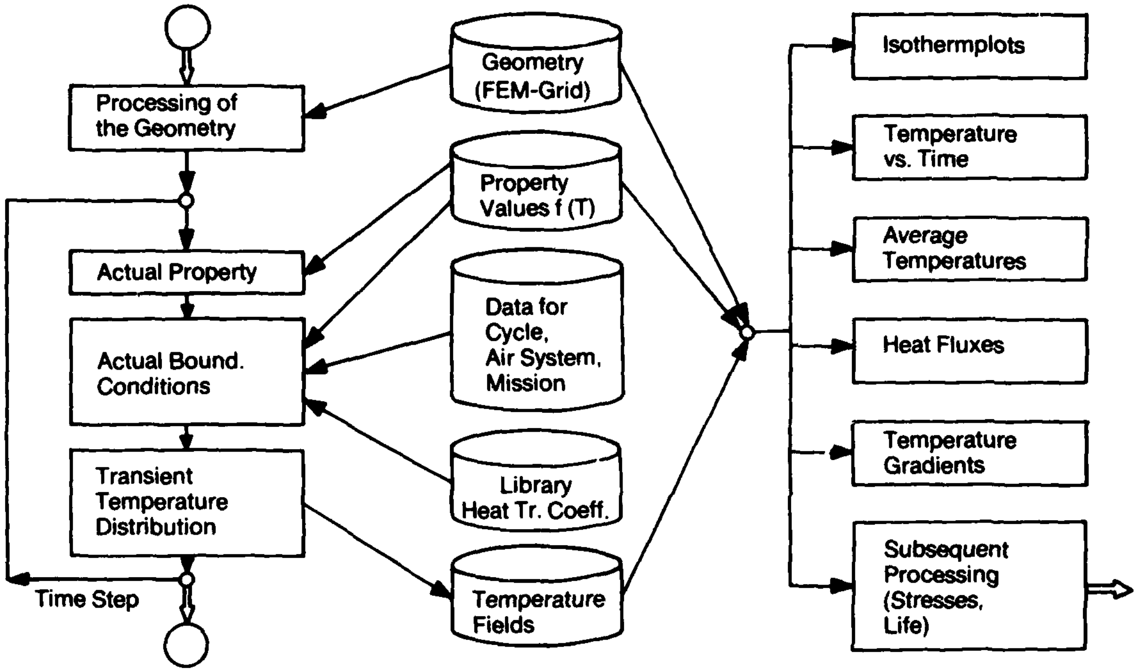

Fig. 4 Structure of the computer program system for thermal analyses

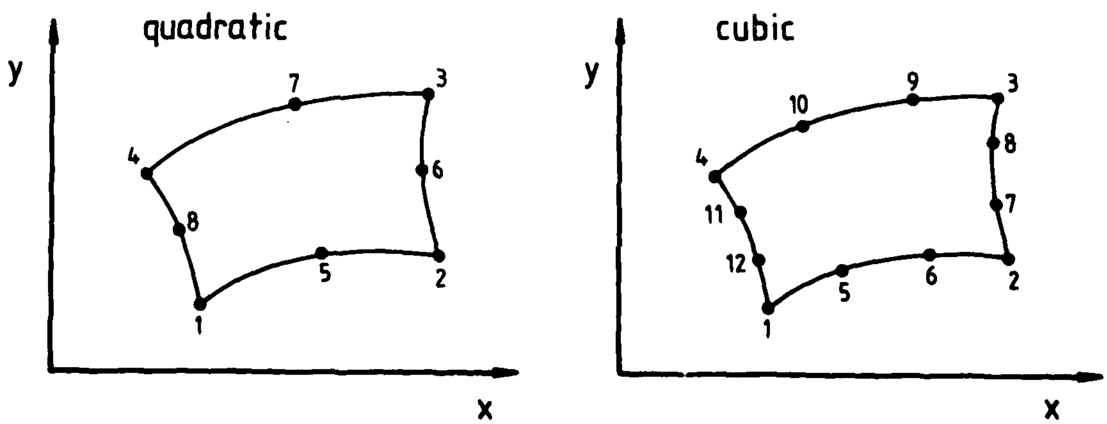

Fig. 5 Typical finite elements shapes

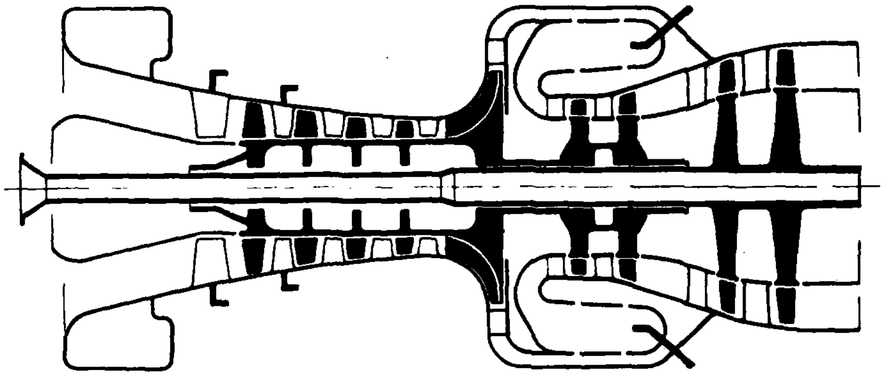

Fig. 6   Turboshaft engine

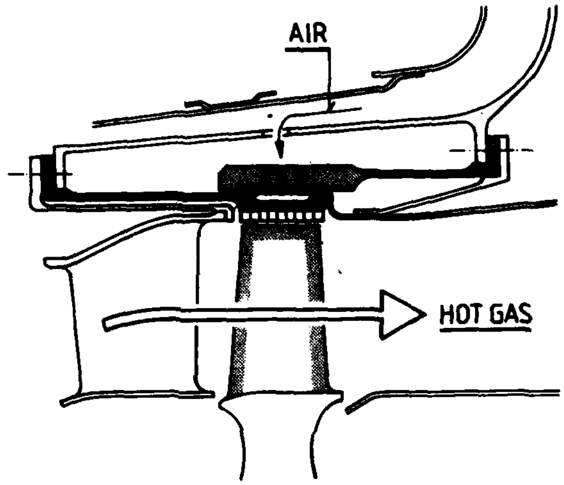

Fig. 7   Liner arrangement in a turboshaft engine

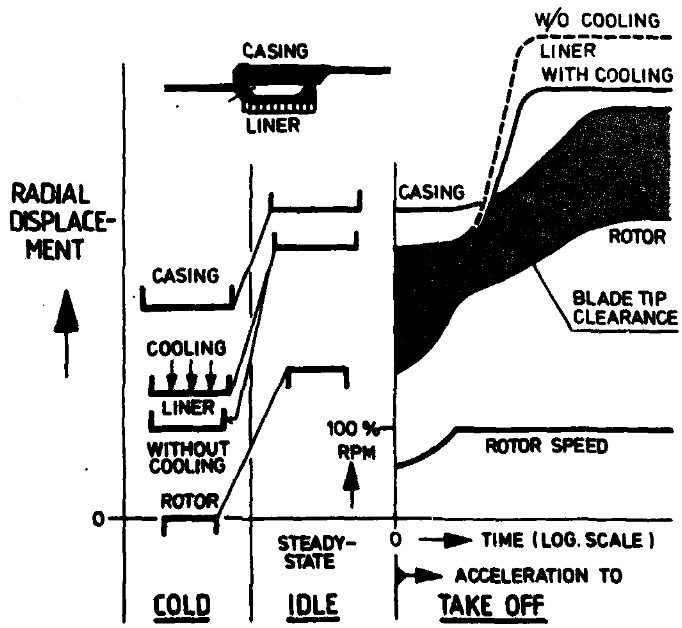

Fig. 8   Passive clearance control for a gasgenerator turbine

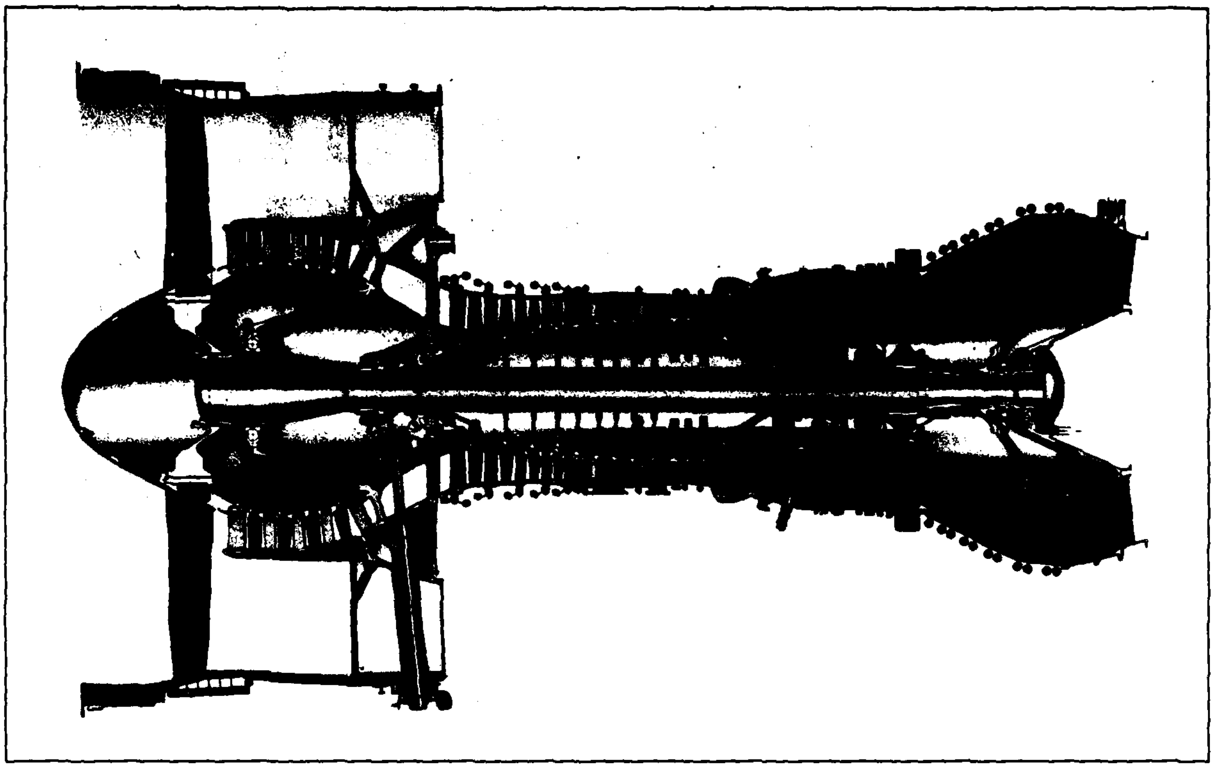

Fig. 9 Modern high bypass ratio commercial engine - The PW 2037

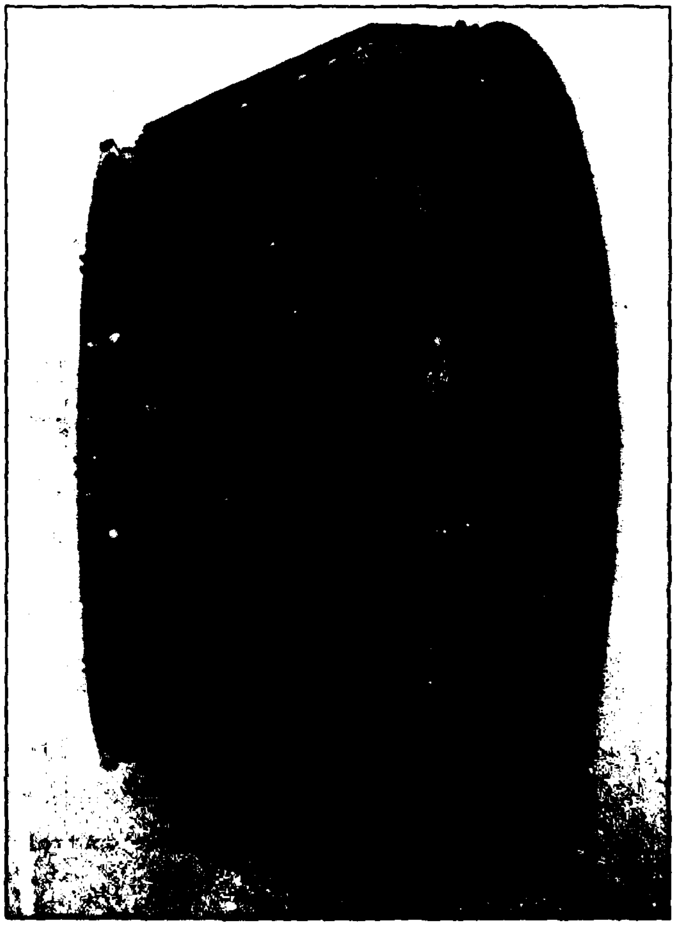

Fig. 10

The LP turbine
active clearance control system

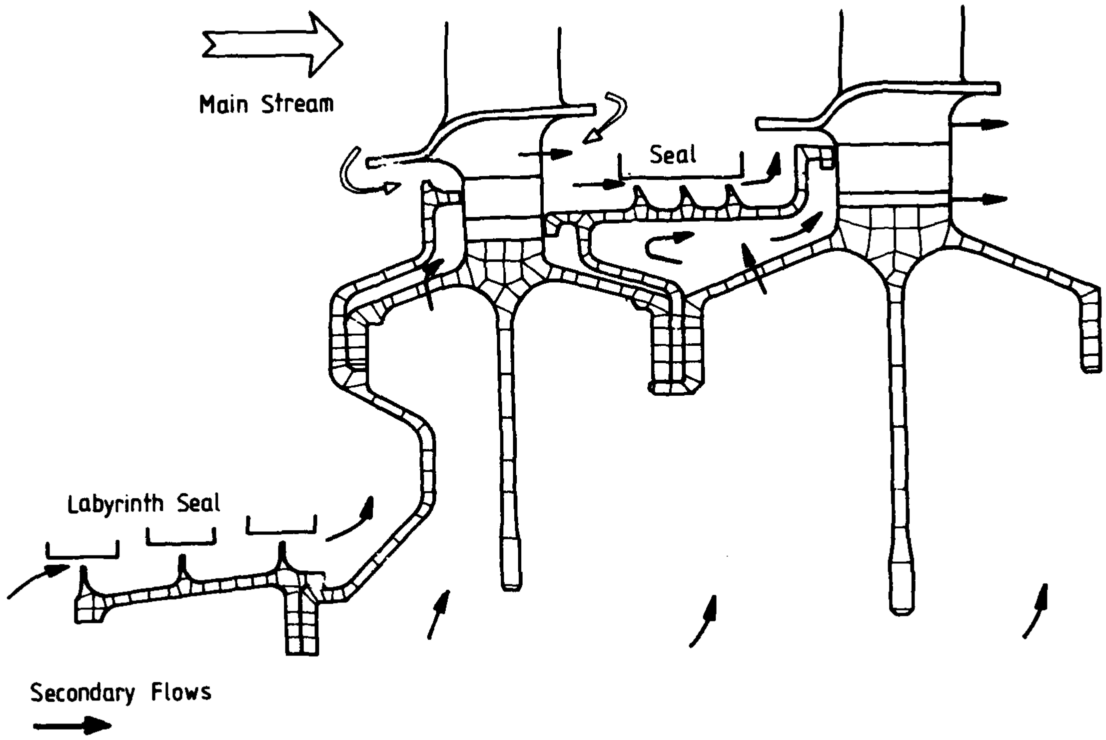

Fig. 11 Part of LP turbine rotor - Finite element grid and air system

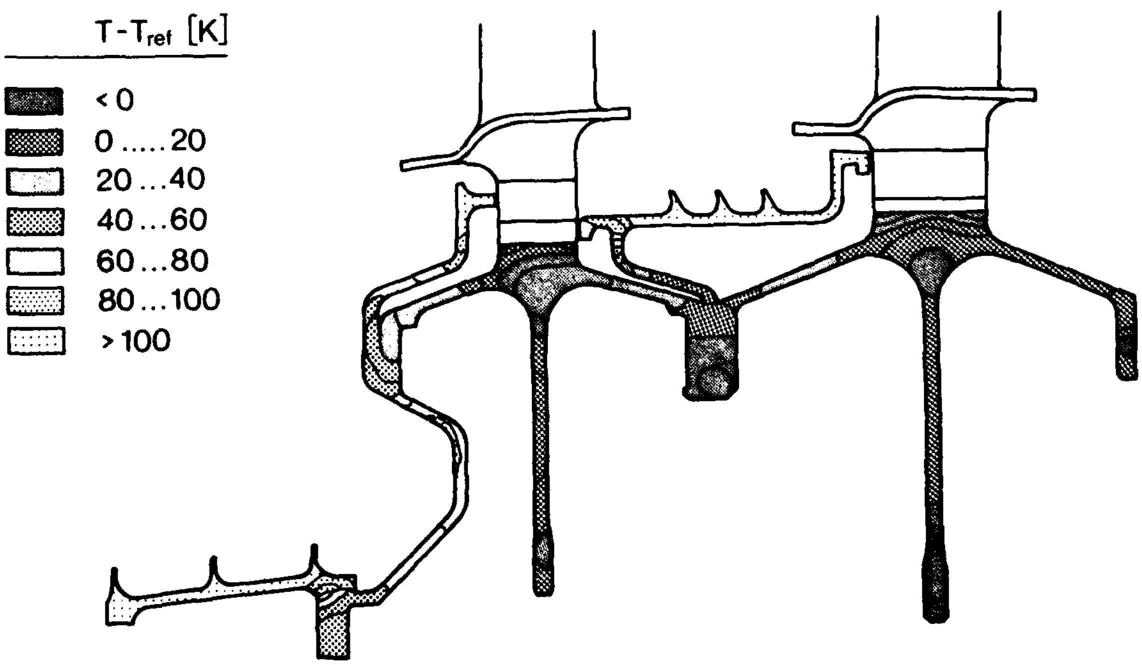

Fig. 12 Isotherm plot of a transient temperature field (40 s after acceleration from idle to take-off)

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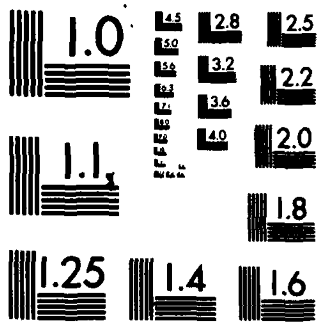

MICROCOPY RESOLUTION TEST CHART
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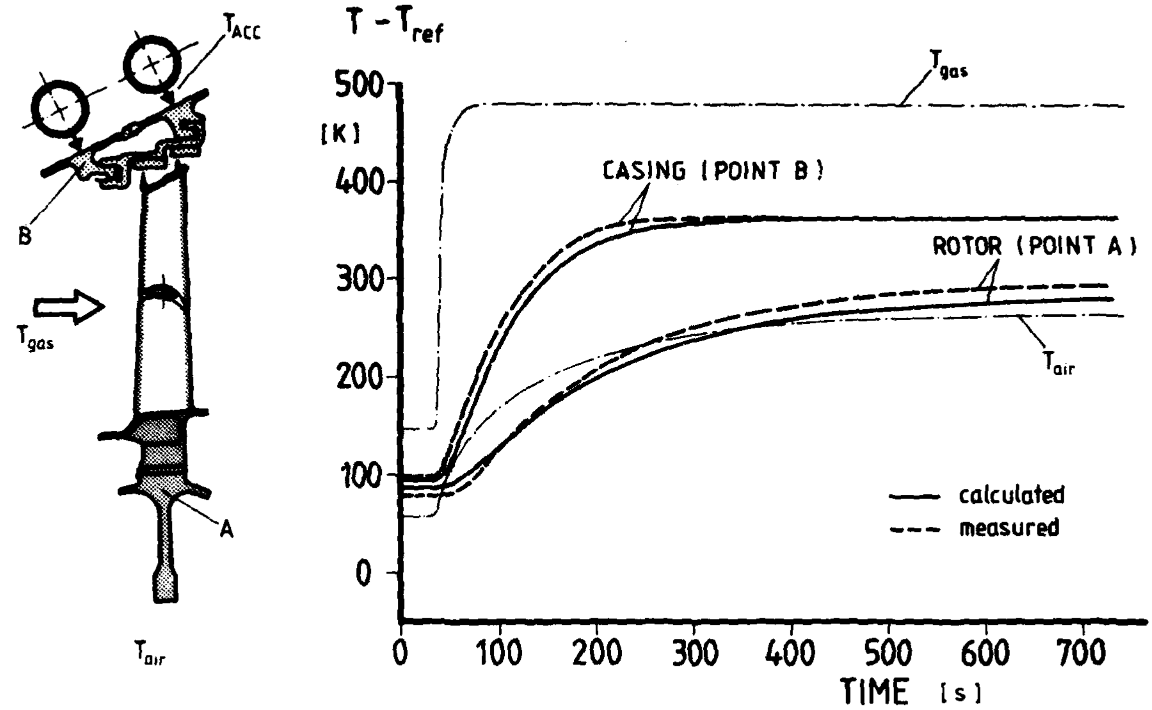

Fig. 13 Comparison between measured and calculated LP turbine temperatures

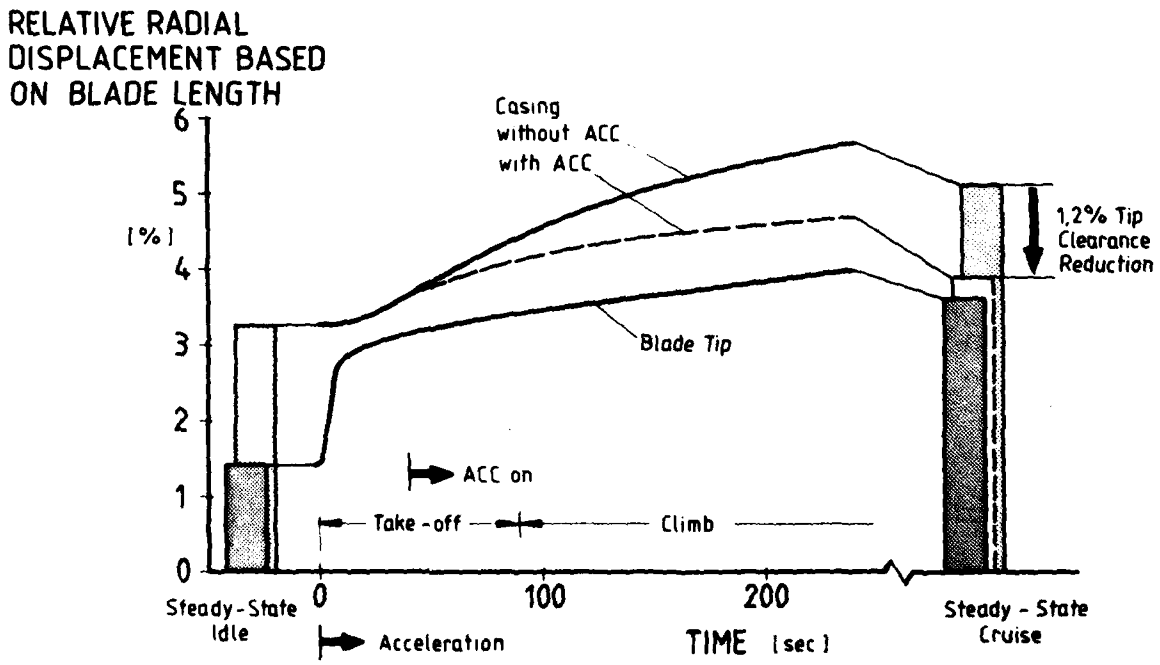

Fig. 14 LP turbine stage 1 - Achieved rotor and casing radial displacement

DISCUSSION

J. Dunham, UK

Please would you explain how the active clearance control system works – what form of control is used and how is it implemented?

Author's Reply

The amount of cooling air which is fed into the A/C-system can be varied between 0% and 100% by an electromagnetically operated valve. Thereby the casing temperature which controls the blade tip clearances can be influenced in such a way that under non-steady-state operating conditions with high thrust and manoeuvre loads, blade tip clearances are relatively large whereas under steady-state operating conditions where fuel consumption is of importance – such as cruise – blade tip clearances are near zero.

P.F. Neal, UK

(1) Do you take account of suspension link local deflections in the PW2037 LPT Casings when designing your clearance control system to prevent induced out-of-roundness?

(2) For an unshrouded turbine blade in the PW2037 do you have any means of axial control of the blade tip to avoid the possibility of axial movement at the same time as rubbing conditions occur which can lead to LE and TE cracking of the aerofoil tip?

Author's Reply

(1) Yes we do take into account thrust, manoeuvre and g-load induced casing deflections and try to minimize the adverse effects by proper ACC design.

(2) The PW2037 LP-turbine has only shrouded blades. In the case of unshrouded turbine blades – as for the small gas generator turbine shown – we run somewhat larger tip clearances so that blade tip rubbing is avoided during normal operating conditions and may occur only under high manoeuvre loads for very short times, where axial rel. movement between rotor and stator is negligible.

J.P. Langrange, Fr.

Comment fonctionne le système de controle des jeux que vous venez de decrise pendant les phases de décélération et de réaccélération du moteur?

Author's Reply

During non-steady-state operating conditions of the engine – take-off (rapid acceleration), approach and landing (rapid deceleration and reacceleration) – the cooling air supply to the ACC-system will be reduced or shut off so that blade tip clearances are increased and the danger of severe rubbing at the various "pinch points" is minimized.

J. Cocheteux, Fr.

What kind of clearance control would you consider for a fighter aircraft engine?

Author's Reply

The kind of tip clearance control – active or passive – to be considered for an aircraft engine depends on engine configuration and engine application. If compressor and turbine casings are directly exposed to the bypass stream and thereby cooled externally, and if long steady-state operating conditions when tip clearances should be kept near zero to save fuel do not exist in the aircraft's mission profile, a passive clearance control system will be sufficient.

W.G. Steenken, US

Was your active clearance control scheduled on speed or rate of change of speed?

Author's Reply

During the tests referred to in the paper the ACC system was turned on 40 seconds after acceleration to take-off power as indicated in Fig. 14. The schedule to be used in the production engines will be defined when more experience during the development phase has been gained. One could imagine that speed and altitude are suitable parameters for ACC scheduling.

Ph. Ramette, Fr.

In the case of the impingement cooling against the outside of the casing which you have shown for the low pressure turbine of the PW2037, are the holes round and are they perpendicular to the wall?

Author's Reply

Yes the impingement holes in the cooling air manifolds which surround the casing are round and perpendicular to the casing outer wall.

K. Robinson, UK

The lecture did not make clear whether the claimed improvement in efficiency when using the described form of active clearance control included a debit for the increase in fan losses which must occur when supplying cooling air.

If not, is the overall improvement in efficiency significant?

Author's Reply

The improvement in aerodynamic efficiency claimed for the first stage of the LP-turbine as a result of tip clearance control does not take into account the performance loss of the engine due to the air off-take from the bypass. A trade-off between this loss and the gain in efficiency by tip clearance control for the complete 5-stage LP-turbine, however, reveals that the overall improvement in TSFC is still significant enough to justify the extra expenditure for an ACC-System.

MODELS FOR PREDICTING TIP CLEARANCE CHANGES IN GAS TURBINES

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SUMMARY

Clearances at compressor and turbine blade tips and seals alter during and following transients. These changes affect the performance of the components of the engine.

In the present paper a model has been developed to predict these clearance changes. The model divides the stage into the casing, blade and disc sections, the disc being split into three elements. A simplified version has been produced for inclusion in engine transient performance programs.

As an illustration the models have been applied to the H.P. Compressor of a two-spool by-pass engine, and to two seals controlling bleed and cooling flows. The results indicate that the effect of compressor blade tip clearance is small. More significant are the effects of the seal clearance changes.

List of Symbols

D	Diameter
Gr	Grashof Number
L	axial length of stage
Nu	Nusselt Number
Pr	Prandtl Number
Re	Reynolds Number

Subscripts

av	average
i	hub of blade
o	casing
r	at a radius

1. INTRODUCTION

It is desirable to develop accurate methods for predicting the transient behaviour of gas turbines. For example, reliable predictions are needed for the speed response and thrust response of an aero gas turbine when a given acceleration fuel schedule is applied.

The earliest programs for the prediction of the transient behaviour used equilibrium characteristics for the components, and ignored heat transfer effects. However these simple procedures had weaknesses, for example they seriously underestimated the times required for the speed and thrust responses - Thomson (Ref. 1) quotes underpredictions of 20 to 30% for acceleration times.

The influences of the direct heat transfer effects on these predictions of responses have subsequently been studied for a typical single-spool (Ref. 2) and a typical two-spool by-pass engine (Ref. 3). Alterations in tip clearances cause changes in component efficiencies. This can be regarded as an indirect effect of heat transfer, although tip clearances are also influenced by centrifugal and pressure effects. It has been shown (Ref. 3) that predicted acceleration rates are very sensitive to changes in component efficiency - a loss of 1% in H.P. turbine efficiency increases the time for acceleration of a typical two-spool by-pass engine by 17%. A further indirect effect of heat transfer is the response of seals which control cooling air flows. It is therefore appropriate to consider whether engine transient programs should include allowance both for the direct heat transfer effects (Ref. 2) and for the indirect effects of tip clearance and seal response.

The object of the present paper is to find models from which tip clearance, and hence efficiency, changes can be predicted, and also models for seal clearance changes. The aim is to incorporate these models, if required, in the engine transient programs. The models can of course be used to predict the transients most likely to cause "rubs", thereby assisting design.

2. PROCEDURE FOR ESTIMATION OF BLADE TIP CLEARANCE EFFECT

The H.P. Compressor of a typical two-spool by-pass engine, of overall compression ratio 20 and having mixed exhausts, has been selected to demonstrate how the representation of a multi-stage turbomachine can be simplified, still retaining reasonable accuracy.

2.1 MODEL FOR BLADE TIP MOVEMENT

The movement of the blade tip depends on the responses of the disc and of the blade. Studies using the finite element program of Ref. 4 have indicated that the complex shape of the disc can be broken down into three components - a thick hub portion, a thin diaphragm and an outer section or rim (Fig. 1). With regard to transient heat transfers to, or from, these disc elements, some faces are rotating adjacent to stationary faces while other faces form walls of what are effectively rotating chambers. For rotating faces adjacent to stationary faces, the correlation used is (Ref. 5) -

$$Nu_r = 0.0253(Re_r)^{0.8} \tag{1}$$

For rotating faces which form walls of rotating chambers the heat transfer mechanism is effectively natural convection in a high gravity field, the value of the gravitational acceleration being a function of both the local radius and the rotational speed and the correlation is -

$$Nu_r = 0.12(Gr.Pr)^{0.33} \tag{2}$$

For heat transfer to turbine blades, a suitable correlation is that given by Halls (Ref. 6) -

$$Nu_{av} = 0.235(Re)^{0.64} \tag{3}$$

For compressor blades, one approach has been to adopt a weighted average between laminar and turbulent boundary layers developing on flat plates (Ref. 7). The results of this method have been found to be within 5% of the results obtained by applying Hall's turbine correlation to the compressor blades. Therefore for convenience in the present work, which is intended to produce a model applicable to both compressor and turbine tip movements, Hall's correlation has been used for all blades.

Centrifugal effects have also been accounted for. The centrifugal growth for the simplified "three element" disc was calculated ensuring continuity of radius dimension at the interfaces between the elements and a multiplying factor of 1.3 was introduced to line up with growths calculated by finite element analysis.

Disc distortion in the angular direction during acceleration has been calculated, but the resulting radial movement is found to be negligible.

2.2 MODEL FOR CASING MOVEMENT

The casing structure of the H.P. Compressor is subjected to internal pressure from the core air and to external pressure from the by-pass air, and also exchanges heat with these two air flows. In order to estimate the heat transfer coefficient at the internal surface of the casing, one approach is to regard this as a cylinder in which a smaller cylindrical shape is rotating, the heat transfer coefficient in this case being given by (Ref. 8) -

$$Nu = 0.015(1 + 2.3(D_o-D_i)/L)(D_o/D_i)^{0.45} (Re)^{0.8} (Pr)^{0.33} \tag{4}$$

In applying the above equation to the present work, the linear axial dimension, L, used was the axial length of the blade pair, it being assumed that the end-wall boundary layer is effectively restarted at each blade pair. For the outer surface of the casing, the expression for a developing turbulent boundary layer on a flat plate was used.

2.3 CLEARANCE MOVEMENTS

The methods described in the preceding sections have been used to predict the clearance movements during and following an acceleration for each of the 12 stages of the H.P. Compressor of the two-spool by-pass engine previously referred to. The engine was stationary, at sea level, and the speed transient was completed in 11 seconds. The predicted results for stages 1, 5 and 11 are shown in Figs. 2, 3 and 4 respectively. In each there is a rapid reduction in tip clearance during the speed transient, due to centrifugal growth and comparatively rapid thermal growth of the blades. The pressure movement of the casing is small, amounting to about 1% of the blade tip movement. The thermal response of the casing is less rapid than that of the blades. Much of this takes place after the speed transient is completed. Finally the disc thermal response is achieved, which may take up to 360 s or longer for completion. To indicate the relative rates of the thermal responses of the various components, the time constants for Stage 5 at three instants in the transient are given in Table 1.

Time in transient	2s	6s	10s
Disc hub	108s	66s	37s
Disc diaphragm	38s	17.8s	9.6s
Disc outer section	12.7s	9.0s	5.5s
Blade	2.3s	1.6s	1.0s
Casing	10.4s	7.6s	4.8s

Table 1. Stage 5: Time Constants during Acceleration

2.4 EQUIVALENT STAGE AND EFFICIENCY CHANGES

It would be excessively cumbersome to have to include each individual blade row of each compressor and turbine into the transient program for the engine. The use of single "equivalent" stages would be highly desirable. A single equivalent stage can give satisfactory heat transfer rates (Ref. 2). A single equivalent stage has been developed, based on the averaged dimensions of the 12 stages and using averaged properties of specific heat and thermal expansion coefficient, to represent tip clearance and associated efficiency changes. (Ref. 9).

The efficiency changes, relative to zero tip clearances, have been calculated during the acceleration transient using the efficiency alterations in each of the 12 stages calculated individually. The results are shown in Fig. 5. The efficiency changes predicted using the single equivalent stage were next calculated, and are also shown in Fig. 5. It is seen that the efficiency changes predicted by the equivalent method are in satisfactory agreement with the calculation based on all 12 stages, discrepancies never exceeding 0.7% of efficiency.

A further simplification has been introduced to reduce the property data required. The simplification involves the continuity of radius at the interfaces between the elements forming the disc, the movement of the outer edge of the platform being taken as the sum of the relative radial growths across the three elements. The efficiency changes predicted with this simplified analysis are shown by the solid line on Fig. 5. There is satisfactory agreement between these predictions and those based on the 12 stages. This simplified model has been used in the engine transient program and sample results are discussed in the next paragraph.

2.5 EFFECTS OF EFFICIENCY CHANGES IN ACCELERATIONS

The methods described above have been used to illustrate the effect of accounting for efficiency changes resulting from tip clearance alterations on predictions of acceleration rates. The transient considered was a sea-level acceleration of the two-spool by-pass engine previously considered. The effects in the H.P. Compressor only are illustrated here. At each time increment in the transient the tip clearance was determined and hence the efficiency change relative to zero tip clearance. The tip clearance for steady running at that speed and flow condition was also determined, and the corresponding efficiency change from zero clearance. Thus the alteration in compressor efficiency at each time increment during the transient as compared to the equivalent steady running condition was found. These alterations in compressor efficiency are shown in Fig. 6, where it is seen that the changes are very small, less than 0.15%, and generally beneficial. The reason for the smallness of the effect is that the partially incomplete expansion of the disc is compensated by the partially incomplete expansion of the casing.

The effect of these alterations in compressor efficiency on the predicted acceleration rate has been estimated using the engine transient program, and has been found to be very small, indeed too small to be illustrated.

3. PROCEDURE FOR ESTIMATION OF SEAL CLEARANCE MOVEMENTS

The methods and model used for tip clearance can be adapted for making estimates of seal clearance movements during transients. In the engine considered in the present study, two relevant seals are the H.P. Compressor 12th Stage Outer Seal and the H.P Cooling Air Seal on the H.P.1 Turbine Disc.

3.1 H.P. COMPRESSOR 12th STAGE OUTER SEAL

The movements of this seal have been studied by Lim (Ref. 10) using finite difference methods. The predictions of Lim and of the present much simpler method are compared on Fig. 7. The agreement is sufficiently close to allow the present models to be used for seal clearance predictions during transients. It is seen that during most of the acceleration speed transient, seal clearances exceed their maximum speed stabilised values by up to 30%.

3.2 H.P. COOLING AIR SEAL ON H.P.1 TURBINE DISC

The movements of this seal have been studied previously by finite difference methods (Ref. 5). The seal clearances during the acceleration speed transient exceed the maximum speed stabilised values by more than 100%. (Ref. 8)

3.3 EFFECTS OF SEAL CLEARANCE MOVEMENTS IN ACCELERATIONS

In the early programs for predicting the acceleration or deceleration rates of gas turbines it would probably be assumed that cooling and bleed air flows remained a constant fraction of the core air flow during the transient. However it has been illustrated above that seal clearances during the speed change period of a typical acceleration can be very much higher than the maximum speed stabilised or "design" clearances. Consequently these cooling and bleed flows, expressed as fractions, will exceed the "design" fractions.

Allowance for the movements of the H.P. Compressor 12th Stage Outer Seal and of the H.P. Cooling Air Seal on the H.P.1 Turbine Disc have been included in the engine transient program, and the results, for the sea-level acceleration, are illustrated in Fig. 9. The predicted acceleration rate is slowed by about 5% due to inclusion of these effects.

4. CONCLUSIONS

A simple model can be used to predict tip clearances of individual stages during a transient. An equivalent stage can be develcped which will give an "averaged" tip clearance from which efficiency changes of the component - e.g. the compressor - can be calculated. During an acceleration of a typical two-spool engine, in the H.P. Compressor, the tip clearance changes, and efficiency alterations, as compared to steady running, are small and tend to be favourable.

The techniques for tip clearance prediction can be applied to seals. Some seals are seen to have large clearances during accelerations. Allowance for this makes predicted acceleration rates significantly slower than if this effect had been ignored.

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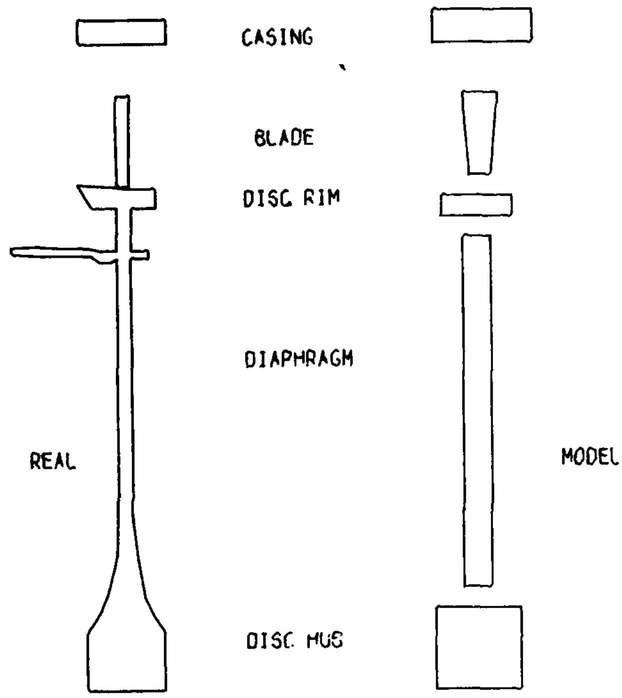

FIG. 1

REPRESENTATION OF A TYPICAL STAGE.

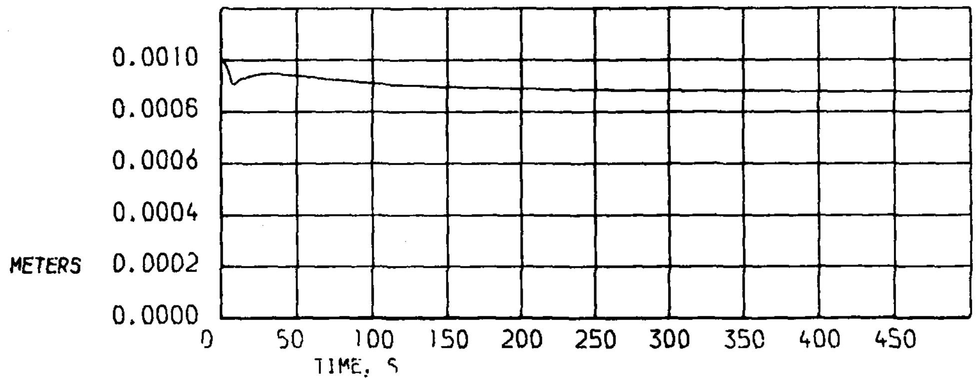

FIG. 2 STAGE 1

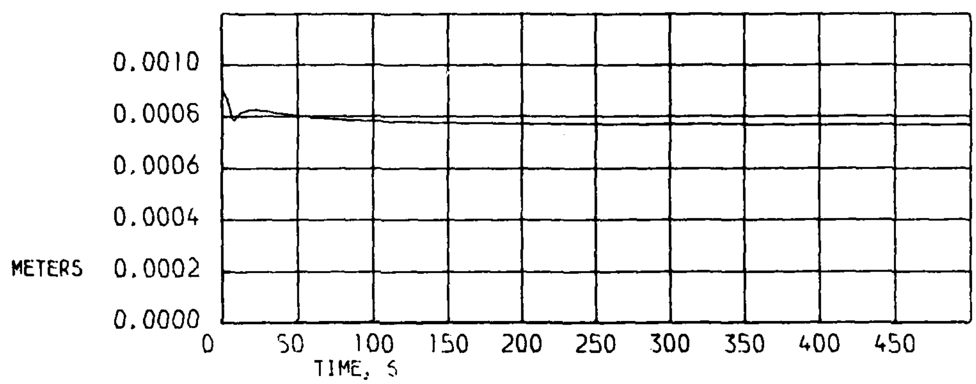

FIG. 3 STAGE 5

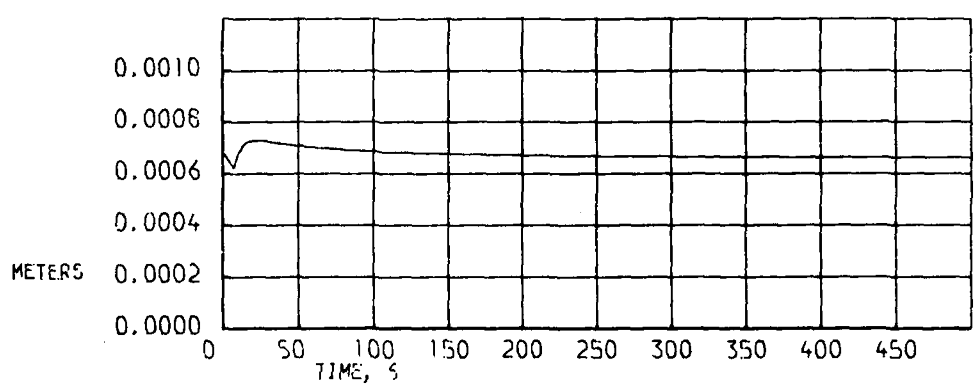

FIG. 4 STAGE 11

PREDICTED TIP CLEARANCES DURING AND FOLLOWING SEA LEVEL ACCELERATION.

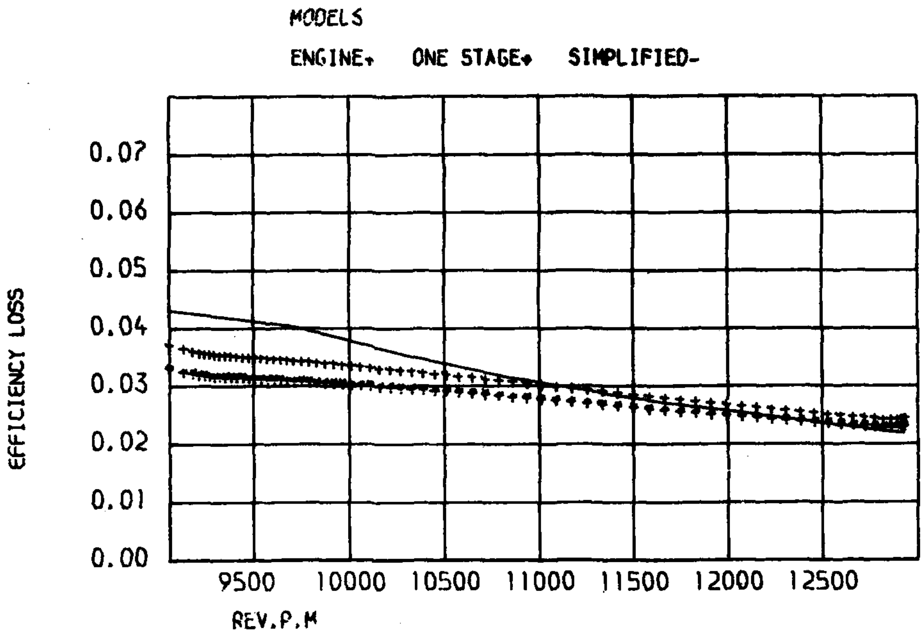

FIG. 5

EFFICIENCY LOSS COMPARED TO ZERO CLEARANCE

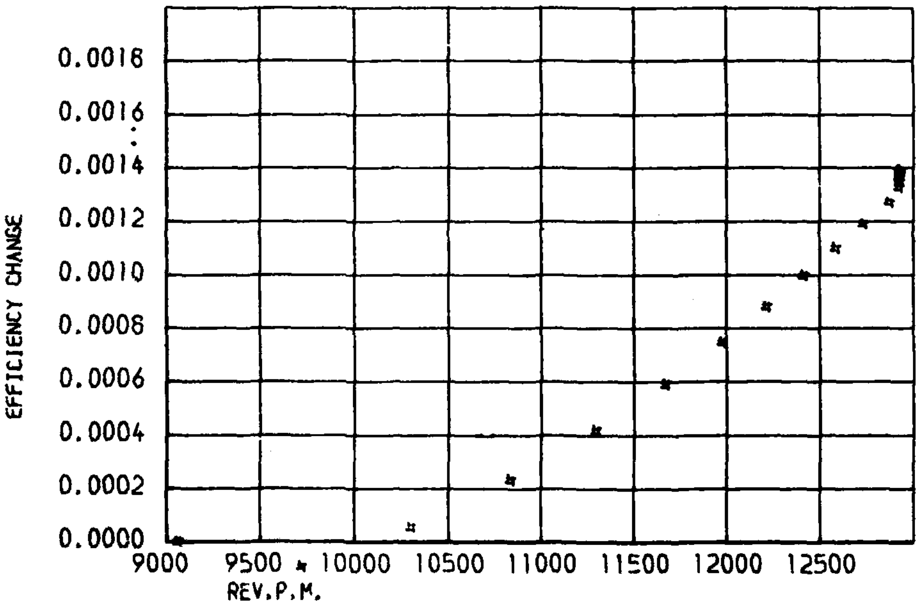

FIG. 6

EFFICIENCY CHANGE DUE TO DIFFERENCE BETWEEN TRANSIENT AND STABILISED CLEARANCE

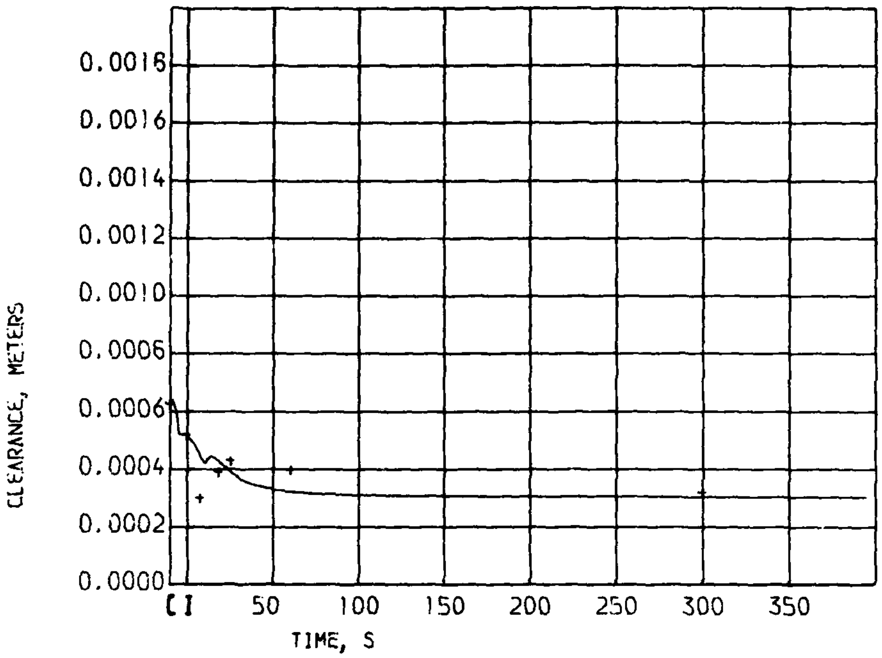

FIG. 7

PREDICTED HP COMPRESSOR OUTER REAR SEAL CLEARANCE DURING AND FOLLOWING SEA LEVEL ACCELERATION. PREDICTIONS FROM REF. 10 INDICATED BY +

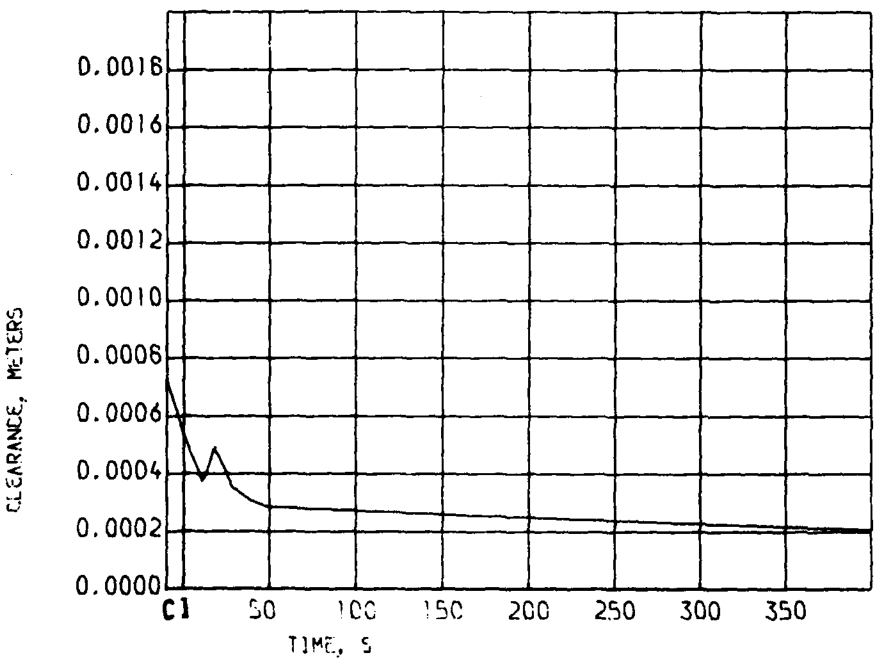

FIG. 8

PREDICTED HP TURBINE AIR SEAL CLEARANCE DURING AND FOLLOWING SEA LEVEL ACCELERATION. PREDICTIONS FROM REF. 5
C COLD BUILT I GROUND IDLE.

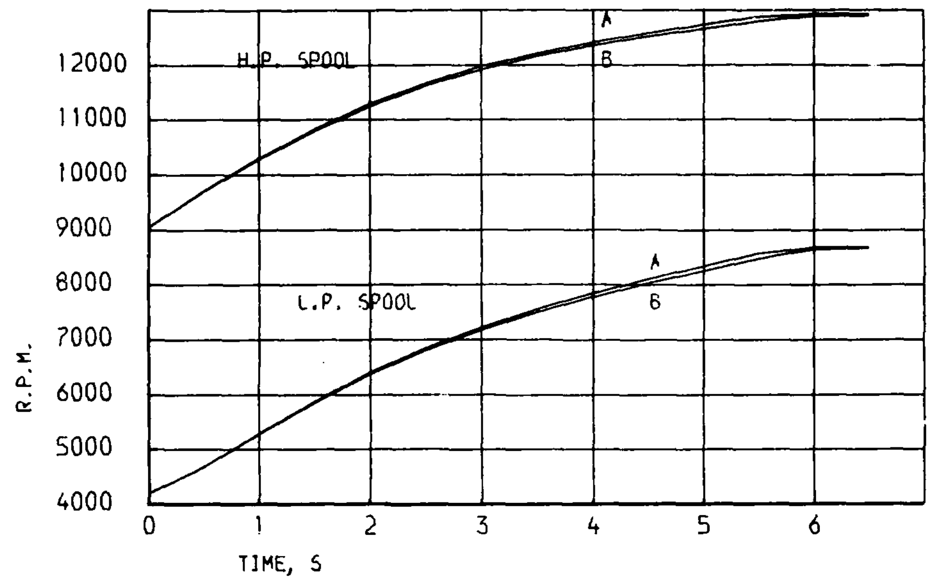

FIG. 9

EFFECT OF INCREASED SEAL OPENINGS IN THE H.P. TURBINE OF A TWO SPOOL ENGINE
A. ASSUMING CONSTANT SEAL OPENINGS
B. ALLOWING FOR SEAL OPENINGS

DISCUSSION

Ph. Ramette, Fr.

(a) In the Nusselt Number expressions in terms of Reynolds Numbers, what are the reference lengths which you have taken in your model, for a blade or for a disc?

(b) Is the equivalent stage you describe in your paper equivalent to the full high pressure compressor with 12 stages that you have studied?

Author's Reply

(a) The reference lengths used to obtain the Reynolds Number in the Nusselt Number expression are:

For the blades, blade-chords in both expressions for the disc outer section was the radial distance from the axis and for the casing the axial distance from the beginning of each stage.

(b) Yes. The idea of the equivalent stage is to represent the full compressor by one stage only. It was found that taking averages of the different dimensions and properties of the compressor gave results that were considered satisfactorily similar to the full compressor (Fig. 5).

DETERIORATION IN SERVICE OF ENGINE TRANSIENT BEHAVIOUR

by

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SUMMARY

In this paper a qualitative survey is given of conditions that may lead to stall and the measures that are designed to prevent it, provided the controls are rigged properly. A distinction is made between low corrected speed stall which is often rotating and stable, and rear stage stall which results in surge. Finally two case histories are given which describe some modifications of the standard diagnostic procedures to track down the reason for stall, as well as modified tolerances to improve stall margin.

ABBREVIATIONS AND SYMBOLS

AB	Afterburner
BLV	Bleed valve position
CDP	Compressor Discharge Pressure
CIT	Compressor Inlet (total) Temperature
CPR	Compressor Pressure Ratio
EGT	Exhaust Gas Temperature (between turbine and afterburner)
FJ	Thrust
IGV	Inlet Guide Vane
MAX	Maximum Afterburner Power Setting
MIL	Military Power Setting (maximum without AB)
N	Engine rotational speed
NPI	Nozzle Position Indication
PLA	Power Lever Angle
P3	Compressor Discharge Pressure
P5T	Total Pressure behind turbine
Q	Air Mass flow
RNI	Reynolds Number Index
SFC	Specific Fuel Consumption
TET	Turbine Entry Temperature
VEN	Variable Exhaust Nozzle
VPI	Vane Position Indication
WA	Air Mass flow
WF	Fuel Flow
WFE	Engine Fuel Flow (without AB fuel flow)
WFT	Total Fuel Flow (including AB fuel flow)
δ	Total Inlet Pressure divided by Standard Static Sea Level Pressure
θ	Total Inlet Temperature divided by Standard Static Sea Level Temperature

1. INTRODUCTION

In the good old days performance standards were low, development fast, and therefore fighters were obsolescent after 5 years; aircraft life rarely extended beyond 10 years. Nowadays an active squadron life of 15-20 years is no exception, and individual aircraft are still doing some jack-of-all-trades jobs at up to 30 years of age.

As jet engines are getting nearer to perfection, further improvements cost progressively more time and money; also parts are more expensive and therefore a tendency exists to more refurbishment instead of replacement. Nevertheless a 15 years old engine will have few original parts left - except for the proverbial nameplate - but instead consist of a mix of parts of all ages, often refurbished, sometimes stripped and recoated; all within the known tolerances no doubt, but the combination is not as good as new. This inevitably results in some degradation of performance and handling.

Usually thrust degradation can be kept small; SFC degradation will be a bit more but is not all that important for fighter engines. More difficult to keep in check, however, is degradation of stall margin which results in shrinking of the operational envelope, causing thrust disruption and requiring pilots attention, often at a crucial moment. And if thrust cannot be regained quick enough it can result in loss of the aircraft, and sometimes of the pilot as well.

Most tolerances, like FOD repair limits, are primarily intended to keep up structural integrity, and stall margin can deteriorate significantly before it is noticed on a post-repair or post-overhaul ground check and translated into additional tolerances. This means that the burden of stall margin acceptance is placed on the Functional Check Flight which - especially for single-engined aircraft - is an undesirable situation.

From an operator's standpoint it is important to know what types of stall may occur, what has been done against it, and therefore what may go wrong to cause stall anyway. This way he can direct his trouble shooting and find ways and means to avoid recurrence, mostly with the assistance of the manufacturer. Often though - especially when the stall occurs at the boundary of the operating envelope - it is a case of going from Scylla to Charybdis; any fix generates another problem ... A knowledgeable operator will be able to provide a feedback to the manufacturer, which will be to the advantage of both. It is hoped that the following notes will be beneficial to this cooperation.

2. TYPES OF STALL

The engine may cease functioning properly in a multitude of ways, some of which are externally induced (inflow disturbance) and some internally. In the latter case stall may be caused by a low stall line of the compressor, a high operating line or too much stall margin consumption by the controls during a transient. The afterburning bypass engine has aggravated augmentor-compressor interaction as a cause of stall, requiring special design measures. In all cases stall results from an unfavourable matching situation, which can arise between intake and engine (lip or diffusor stall), between fan and high compressor (splitter stall) or internally between the stages of one compressor. In the latter case two main types of stall can be distinguished: the low-corrected speed ($N/\sqrt{\theta}$) stall which is often stable and does not necessarily lead to surge, and the high-corrected speed - or rear stage-stall, which does.

Low Corrected Speed Stall. A highly loaded aero-gasturbine compressor has a very narrow RPM margin in which it can operate stable without configuration change - usually only about 10 % RPM. The blade height - or flow area - is for each stage matched to the design compression ratio (Fig. 1a). This means that at low RPM - i.e. low pressure ratio - the rear stage flow area will be relatively low, resulting in a high flow velocity and a low angle of attack in the rear stages. Consequently the rear stages will do less work, and shift compressor loading to the forward stages. There, this may lead to stall, which in first instance will be local, i.e. partial span and over some adjoining blades. The resulting blockage will reduce overall through-flow. In the rear stages this increases angle-of-attack and shifts the load off the forward stages. This type of stall is therefore self-stabilizing; the engine may continue running in a "second regime" with one or more stall cells, at a reduced through-flow and overall efficiency, and therefore reduced thrust and increased EGT. The stall cells rotate - usually at 0,4-0,6 of compressor RPM - which is explained in figure 1b. The flow divergence upstream of the blockage increases the angle of attack of the stator blades in A, which then tend to stall. In B the angle of attack decreases, recovering the blades there. The result is a movement of the stall cell to the right. In the rotors a similar effect occurs in the opposite direction as shown in the vector diagrams. Relative to the rotor blades the stall cell moves to the left. This results in the stall cell rotation with 0,4-0,6 RPM.

The internal efficiency can be so low that the engine is barely self-sustaining; an extra load in the form of a command to increase RPM will increase the extent of the stall cell - both spanwise and circumferentially - which may result in a hang-up, i.e. a refusal to accelerate (stagnation stall). On the other hand a reduction of the load will decrease the size of the stall cell, but sometimes the condition persists, requiring flame extinction for recovery.

High Corrected Speed Stall. At high corrected RPM the situation is the opposite from that discussed above; compressor loading shifts to the rear. The small blade height in the rear stages does not allow local stall, this therefore tends to be more widespread. Flow blockage now aggravates the situation: the front stages run choked and cannot accommodate less airflow and the rear stages are highly loaded to pass this flow. An increase of axial velocity through the non-stalled blades reduces loading below that required for passing the airflow which therefore becomes unstable and breaks down. Directly following breakdown the engine decelerates rapidly, and often flame-out occurs.

Compressor performance is governed by RPM, Compressor Inlet Temperature (CIT = T2) and loading; the above situation may obtain at low CIT (high RPM/$\sqrt{\text{CIT}}$ = N/$\sqrt{\theta}$) or at high loading, e.g. when the engine is commanded to increase RPM. In the latter case the Turbine Entry Temperature (TET = T4) must increase above the value needed for steady state operation. As the turbine operates near the choked condition, i.e. at constant inflow Mach number, throughflow velocity increases with $\sqrt{\text{TET}}$, while volume flow increases with TET. The result is that gas velocity in the combustion chamber and in the rear stages of the compressor is slowed down, forcing the rear stages to increase CPR proportional to $\sqrt{\text{TET}}$ to restore the turbine flow factor (WA$\sqrt{\text{TET}}$/CPR = ϕ). Too fast acceleration therefore can stall the rear stages of the compressor (Fig. 2). This can also happen in a low-corrected RPM condition when extensive rotating stall of the forward stages leaves the rear stages to do all the work. In the latter case the flow recovers again after breakdown; the cycle repeats with a frequency which is mainly governed by the volumes of compressor and combustion chamber - in the order of 10 cps (Fig. 3). This surge phenomenon can be violent on some types of engine; the stage which initiates surge can be recognized if high-response pressure transducers are installed on the compressor because surge causes a shock wave upstream and an expansion wave downstream of the origin.

3. DESIGN MEASURES TO PREVENT STALL

Only the very oldest and lowest compression ratio engines (CPR < 4) do not require any special measures to run through the whole of their RPM region. With higher compression ratio - required for better cycle efficiency - mechanically operated variable angle intake guide vanes or bleedvalves had to be introduced to unload the forward stages at low RPM by tuning the flow to RPM. In the early 50's the twin spool compressor was developed, which unloads the forward stages by allowing them to run at a progressively lower speed relative to the rear stages - tuning RPM to the flow. The modern high-compression engines (CPR > 20) require two or more of these measures for stable running.

The low-corrected RPM stall looms up again at the high (supersonic) speed side of the flight envelope because of high CIT (= T2). As closing the intake guide vanes lowers air mass flow and thrust, which is usually very much in demand in supersonic flight, another method is employed here which increases thrust, i.e. temporary increase of RPM (T2-reset). This is not practical for supersonic cruise vehicles, but it is for fighters. For very high flight speeds (M > 2,5) a (low) by-pass engine has potential because the by-pass ratio increases with decreasing N/$\sqrt{\theta}$, allowing a larger operating range.

At the low-speed/high altitude side of the flight envelope (low CIT) the high-corrected RPM stall may be a problem; the solution here is RPM cut-back. This can be done without undue thrust penalty as thrust is usually sufficient anyway at low CIT. Moreover the engine is operating in a region of diminishing returns as compressor efficiency is deteriorating rapidly with increasing N/$\sqrt{\theta}$ because of increasing shock losses in the first stage blading, so RPM cut-back does not hurt too much. Even though the engine is operating at constant RPM/$\sqrt{\text{CIT}}$ loading still increases with decreasing CIT if TET is kept constant, as the operating line curves upward when the first stage is getting choked (Fig. 4). Conservation of stall margin necessitates TET reduction proportional to CIT; this requires a variable area jet nozzle.

In order to prevent rear stage stall and surge during acceleration of the engine to a higher RPM it is necessary to limit the amount of fuel injected in excess of the value required for steady state operation. This is of course dependent on flight conditions and usually also on RPM - most engines have a region just above idle speed where the stall margin is minimum. A sophisticated acceleration control unit is required, with a fuel metering valve decoupled from the power lever so it will not open too fast when the pilot slams the power lever. Because the rear stage stall occurs without warning, the control must of necessity operate open-loop. Some degree of closed-loop control is usually achieved by scheduling not just fuel flow WF, but WF/P3, which in fact schedules Fuel/Air Ratio and therefore TET. This also provides a safeguard against overheating in case stall occurs because WF will be cut automatically as Compressor Discharge Pressure (CDP = P3) drops after stall.

If the engine is equipped with an afterburner - and therefore with a Variable Exhaust Nozzle -

acceleration can be greatly improved by opening this nozzle to reduce back pressure on the engine. The nozzle is usually also opened at idle RPM to reduce thrust, which otherwise may be too high for taxiing. On twin spool engines opening the nozzle also reduces back pressure which in this case - because of the increased low turbine pressure ratio - increases fan RPM and therefore airflow. This results in thrust being in first approximation independent of final nozzle area.

4. THE AFTERBURNER

The afterburner is in fact a second (ram) jet engine in series with the main engine, but with its own controls and requiring a Variable Exhaust Nozzle to accommodate the varying volume flow. At steady state running the main engine is unaffected by afterburner operation; the turbine which usually runs choked, isolates the engine from possible pressure fluctuation and combustion roughness in the afterburner. This is not true, however, for a by-pass engine, the fan of which may be unfavourably affected through the -subsonic- by-pass duct.

Pressure fluctuations during steady state operation of the afterburner can be of two types: transverse, high frequency (screech) which can be designed out by perforating the combustion liner to achieve sufficient damping, and longitudinal, low frequency (rumble or buzz) which cannot. In the latter case - which is associated with local areas of overrich fuel/air ratio due to uneven fuel and/or air distribution - it is possible to design a closed loop control which limites FAR to an allowed level of pressure fluctuation (buzz rider).

While steady state afterburner operation is relatively easy, this is far from true for light-up and shut-down, which require the greatest ingenuity in control design, and may require considerable attention in rigging and trimming in operation. The Exhaust Nozzle must open the moment the afterburner lights up to prevent back pressure building up which under unfavourable conditions could stall the compressor or the fan; on the other hand opening the Exhaust Nozzle too early will increase flow velocity in the flame holder region of the afterburner which under unfavourable conditions will prevent AB light-up.

Preferably the afterburner should light up on a small amount of fuel, then - once alight - it can be opened smartly to MAX. On the older engines the pilot operates it that way. Nowadays a delay - and sometimes a ramp function is built in which under normal conditions prevents maximum afterburner fuel flow at light-up. If ignition is delayed, however, a hard light results. Control would be greatly eased with an AB flame detector, which inhibits fuel flow increase till light-up has occurred. So far, however, reliability of such an installation has been a problem. On some engine types afterburner fuel flow is limited to an interim value till ignition has occurred, then it is increased automatically (closed loop) to the a priory commanded valve, which may be MAX. Under unfavourable conditions, however, the interim value can still cause stall. And if the controls are not rigged properly an AB hang-up may occur, i.e. AB fuel flow stays at the interim value and does not increase to MAX. Low combustion efficiency has a similar effect.

Afterburner shut-down requires fast controls to prevent disastrous overspeeding of either the compressor/turbine rotor on a single-shaft engine or the fan in a twin spool. Also in this case the metering valve is de-coupled from the power lever and damped to prevent it from closing too fast. Often the Variable Exhaust Nozzle is commanded by an electric control which keeps EGT constant and is also sensitive to RPM rate. EGT is sensed by thermocouples; to correct for the inherent lag - EGT rate may be 300 C/s - a lead circuit is incorporated. Proper afterburner operation is critically dependent on the dynamic response of Temperature Amplifier and VEN control. Sometimes a separate Overspeed Governor is employed.

5. CONDITIONS THAT CAN INDUCE STALL

Intake Flow Distortion. The prime outside condition that can cause stall is dynamic intake flow distortion, either pressure or temperature; the latter induced by gun or rocket firing, the former by combat maneuvering and/or tip vortex or jet wash ingestion. The modern fighters with advanced lifting vortex aerodynamics can attain angles of attack which make life for the intake and engine difficult, especially when the situation is compounded by simultaneous afterburner transients or use of the board weapons. To accommodate gun firing, engines are equipped with fuel dip or vane dip controls which act to give extra stall margin when the trigger is pressed; useful sometimes to clear an existing stall...

According to the "parallel compressor" theory (Ref. 1) stall may be caused in a sector of the compressor which experiences a local low in the total pressure at the intake. Because little overall tangential flow takes place in the compressor, and the outlet (static) pressure is uniform, this sector then experiences a high pressure ratio which can lead to stall. Another reason for stall can be longitudinal vorticity which accompanies the pressure distribution in the intake and can alter local angle of attack of intake guide vanes or rotor blades beyond the stall angle.

Low Compressor RPM. Steady state stall of either intake or compressor may result from an unmatch due to too low engine RPM or a compressor with deteriorated mass flow. This obtains usually at supersonic flight , speed; most engines incorporate separate measures to keep compressor or fan RPM - and therefore mass flow - high when retarding the Power Lever. The desired thrust reduction is achieved by keeping final nozzle flow area high and therefore EGT and back pressure low.

Compressor-AB interaction. A specific case of interaction stall on a by-pass engine is AB blow-out due to air starvation after compressor stall. After the blow-out the nozzle is still wide open, so the back pressure decreases and the compressor flow recovers; subsequent explosive re-ignition of the still hot afterburner triggers the next cycle until afterburner fuel flow has been shut off.

Reynolds number. Compressor stall margin is lowered with decreasing Reynolds Number Index - i.e. the ratio of Reynolds number under relevant flight conditions (based on total pressure and temperature) relative to sea level static ($RNI = \varphi\sqrt{\theta_2}/\delta_2$). With the usual approximation for the variation of viscosity with temperature: $RNI = 1{,}42\theta_2^2/\delta_2(\theta_2+0{,}42)$. Reynolds number governs the ratio of impulse and viscous forces and with that the secondary flow in the compressor and therefore effective flow area and stage match, which influence both the low and the high corrected RPM stall. For most engine parameters deterioration is linear with the inverse of RNI; this is assumed to hold for stall margin as well and some flight test results have shown this to be so. As different types of (control induced) stall consume different amounts of stall margin it is very difficult to obtain reproduceable results though, especially as the engine deteriorates as a result of the stall tests. Then the next time the engine will stall at a lower altitude (higher RNI), narrowing down the useful flight envelope.

Tip Clearances. The primary reason for stall in a compressor rotor is stage unmatch, which - as indicated above - can be caused by varying extent of the secondary flow. This is illustrated in figure 5 (from Ref. 5) which clearly shows the differences between the design axial velocity profiles and the measured velocity profiles in a compressor, especially at the hub and at the tip. The secondary flow is mainly governed by tip clearance, which therefore plays an all-important role in determining the compressor stall line. The boundary layer displacement area tends to grow with increasing loading, especially as the initial boundary layer is thicker (Ref. 3), giving a progressive effect at the stall point while at normal operating conditions performance may be close to design point. This was observed during mass flow measurements with a few stall prone engines (Fig. 6, Ref. 5). At a certain RPM the engine mass flow $Q\sqrt{\theta}/\delta$ was measured for several positions of the variable stator vane's (VPI). At low vane angles, close to the operating line, all engines had a mass flow which was the same as the value predicted by the manufacturer. But at high vane angles, near the stall line, the stall prone engines showed a mass flow which was much lower than the predicted value.

The effects of tip clearances and stage loading on the boundary layer displacement obtains both at the hub and at the casing; vane and blade tip clearance are equally important in securing stall margin, with the edge possibly being on the vane tips as they have to produce the pressure ratio at somewhat lower circumferential speed and therefore higher loading.

Tip clearance must be as small as possible, without giving rise to scoring, which may happen under unfavourable circumstances like high g-loading. The running clearance is usually smaller than the static clearance because of mechanical and thermal strain - the casing extends thermally and because of internal overpressure, the rotor thermally and because of centrifugal loading (Ref. 4). An important case is the transient; casing and blading have a low thermal lag while the buried rotor discs take longer to reach equilibrium temperature, resulting in clearance initially opening up when RPM is increased quickly. This lowers the stall line at a crucial moment; on the other hand stage unmatch due to metal-to-gas heat transfer may be favourable under this condition.

All together shows that it is extremely difficult to give a value to the stall margin, but it is clear that decreasing tip clearance will be beneficial in any case, as long as scuffing can be avoided, which would undo all the good. This even goes as far that evidently tip clearance is more important than airfoil shape at the tip, as is shown by the scalloped shape of so-called squealer tips (Ref. 3). These can run at a closer clearance than full tips: in case of scraping a minimum of material is gouged away out of the mating surface, leaving a better fit or rather giving less increase of clearance than after a scrape with full material tips. Also less heat is developed, minimizing the chances of a catastrophical seize-up.

Blade Shape. For the primary flow of course blade shape and airfoil are all-important. Blades tend to untwist under centrifugal loading, and possibly with increasing age may acquire a permanent set, but the net effect is probably minor. More important is possible leading edge deformation, through erosion or FOD. When reworking damaged blades the leading edge must be carefully blended because a blunt edge can significantly reduce the stall margin of that blade. Also surface smoothness is important, this can be affected by corrosion. Play in the mechanism, that rotates variable angle vanes, or in their bearings can also have an adverse effect.

The modern compressor employs the same tricks of the trade as modern wing aerodynamics to increase the operating range. The secondary flow is stabilized while passing each subsequent stage by "adding the wheel speed to the boundary layer flow" (Ref. 3). By introducing longitudinal vortices energy is injected in the boundary layer in order to keep flow attached. This requires considerable end twist in the blade roots; when, where and why are closely guarded company secrets, probably arrived at after lengthy testing and extensive small scale three-dimensional viscous flow calculations. Probably this kind of blading is less sensitive to tip clearance effects.

High Operating Line. Given a certain stall line - not uniquely defined because of the many types of stall - stall margin of course is lower with a higher operating line. This increases the probability of stall due to an outside disturbance. A deteriorated compressor will have lower efficiency and air mass flow, causing the turbine to run at somewhat higher inlet temperature when controlled to the same outlet temperature. This gives a higher operating line. If the engine is not rated at a constant EGT indication, but at a specified fuel flow, which in the average new engine gives a certain value of TET, then any reduction of airflow in a deteriorated compressor gives an increase of TET. This also gives a higher operating line. The rating is usually set at the test bed; in operation the engine is controlled to the resulting EGT indication. As the temperature profile in the turbine is often subject to pattern shift under other than test-bed conditions, this may result in a higher TET under certain operating conditions for the same indicated EGT.

Apart from that the possibility exists that a thermocouple - usually the hottest - burns out, causing the average output of the other thermocouples, which are connected in parallel, to drop whereupon the controls increase fuel flow and TET to restore the sensed EGT. The pilot need not know anything about all this as he usually sees an indicator which is fed by a different set of thermocouples and which is set to a standard rated EGT when operating at full power.

Controls. It was discussed before that the transient controls usually operate open loop, which renders the result sensitive to trimming and rigging operations. The schedule of any specific control is normally checked on its test bench in a few points over the range of operating conditions and then set on the engine at a reference condition. Flexibility and wear in the drive system can add to an already unfavourable stacking of tolerances to produce conditions that are too far out for the engine to cope with. This kind of control discrepancy is difficult to trace on a static test bed as the control schedule there can be quite different - for instance the WF/P3 accel schedule is usually governed by CIT which sets a three-dimensional cam. The presented contour is followed by a hydraulic servo and translated in fuel flow via a P3 servo. To detect possible discrepancies special tests have to be set up with false inputs into the controls, or the results must be measured in flight, which requires special instrumentation. Another source of discrepancies can be the CIT input, which often makes use of a capillary sensor filled with fluid. This kind of sensor has a time constant of several seconds, and -far- more than that if installed in an aspirator tube with insufficient flow. This may cause trouble during air combat maneuvering, as control actions like RPM cut-back may come in too late to prevent stall.

6. TWO CASE HISTORIES OF STALL EVALUATION

Low Corrected Speed Stall. On one engine type in use with the Royal Netherlands Air Force stall rate started to increase after a decade of faithful service. In several cases stall followed by hang-up occurred in low-level cruising flight, and more than one aircraft was lost because the pilot repeated the stall recovery procedure - stopcock/relight; unpopular on a single-engined aircraft but necessary to unload the compressor sufficiently - before it could have effect. It turned out that the flight simulator did not incorporate the usual delay before idle RPM was regained, getting the pilot used to instant thrust recovery. In flight therefore the pilot thought the procedure did not work, repeated it and in so doing prevented the engine to accelerate from the low RPM after the stall to idle. Thrust demand before idle RPM was regained worsened the situation as this schedules a more closed final nozzle which makes recovery less likely. So does activation of the emergency nozzle control, selected because a pilot may think from the wide open nozzle that the normal control has failed, while in fact it responds normally to the high EGT after a stall.

The post-overhaul acceptance test of this engine includes throttle slams on the test bed and in-flight at high altitude/low air speed, afterburner ignition and a run to the maximum-supersonic-Mach number. Also in this latter check stall rate was higher than normal; that could be traced to a mass flow deficiency at low corrected RPM. This was detected as a thrust loss when measuring acceleration performance of the aircraft. At sea level under equivalent conditions, however, no discrepancy could be found because the Variable Vane Position in supersonic flight at high CIT is scheduled further open than at sea level temperature at the same corrected RPM. The mass flow deficiency on a deteriorated engine is progressively worse at more open vane position (i.e. closer to the stall line at the same RPM), see figure 6. Also the ground stall check which can be performed on engines which have stalled in-flight did not show discrepancies because this check stops short of actual stall to obviate the need for a post-stall inspection and repair of possible damage every time an engine stalls. This reserve allowed a substantial reduction of stall margin to pass unnoticed.

The subject type of stall at quasi-steady low corrected RPM gives a distinct warning in the shape of pressure fluctuations, similar to pre-stall buffet on an airplane; probably originating from rotating stall cells. Detecting these with special instrumentation on the test bed while executing the stall check gives an indication of stall margin. Combined with mass flow determination at low corrected RPM with the Variable Vanes wide open, the engines with low stall margin could be weeded out. It was found that the stall margin could be restored by replacing the rear compressor casing and vanes. This is quite an expensive replacement but with the above method it could be limited to the engines that needed it, and without pulling any of those in a stall. Probably reason for this behaviour was permanent ovalisation of the casing, possibly due to high-g maneuvering.

In four out of every five deficient engines, replacing the rear casing worked like magic, air mass flow at the measuring condition increased by 4-6 %, the pressure fluctuation disappeared and the stall margin was as good as new (Ref. 5). On some of the remaining engines the rotor was checked and discrepancies were found in tip clearance (Fig. 7). Further analysis showed three causes. First, groups of cropped blades occurred as a result of inadequate FOD repair. Secondly, on some stages the overall tip clearance was large. Third, in several cases a sinusoidal variation was found, probably obtaining because the disc is first machined on a lathe, and then the blade root fittings are broached on a different machine, with sometimes a slight excentricity in the set-up. Replacing compressor discs also cured the rogues (see Fig. 7: new wheel).

It was tried to find out from pressure indications on an instrumented engine that stalled, which stage stalled first. This was partially successful, but yielded no further clues as it is by no means certain that the stage that stalls is itself deficient; it may be forced into stall by another stage upstream or downstream, which is deficient.

High Corrected Speed Stall. The high corrected RPM stall which often occurs on another RNLAF engine type is far more difficult to treat. The low Reynolds number and low CIT at which the steady state stall occurs cannot be reproduced on a static test bed and the latter cannot be simulated without exceeding engine limitations. The transient opens the door to a multitude of control ideosyncrasies which may cause stall under the aggravated conditions of the ground test but may be quite different from the cause of the high-altitude in-flight stall which prompted the investigation in the first place. Therefore what cures the ground test stall (often a control replacement) may not cure the flight case, or only temporarily.

During a transient so much happens and so fast, that registration is necessary for evaluation. Figure 8 gives an example of a registration of several parameters in an accel from IDLE to MAX AB. Once certain trends, which may occur on faulty controls, have been established, it may be possible to recognize them from closely watching the engine instruments, but any new trend or variation from the old trend will be missed. This makes stall trouble shooting on this engine type very much a hit-or-miss affair, unless backed up by registrations on the ground and preferably in-flight as well on the misbehaving engines. An example is the nozzle dip which occurs due to flexibility and wear in the nozzle control system and which may cause stall at altitude (Fig. 8a, Ref. 6).

To distinguish between a control malfunction and a low compressor stall line a standard slam accel check has been instituted for this type of engine. This check stalls a deficient compressor on the ground by the combined adverse effects of pressure maldistribution of the aircraft intake and a higher operating line achieved by decreasing nozzle area with a number of inserts. At one time during the slam accel to 100 % RPM, practically max accel fuel flow will occur together with minimum nozzle area and - while a good compressor will take it - a bad one will stall. Of course this test depends on a fine balance of well-tuned controls and it is necessary to check both the max accel fuel flow and the final nozzle area before the compressor can be found guilty - which starts an expensive refurbishment procedure. On the other hand, without a reasonably sensitive compressor quality check a lot of time and money can be wasted, changing controls which are within tolerance anyway.

Passing the above test is not proof that the compressor is good, however; the controls may act in a way which relieves a slightly more deteriorated compressor, allowing it to pass without stall. In that case the following sequence of events obtains: the deteriorated compressor will require more power to drive and therefore will accelerate slower. This can give the control time to open the nozzle to accel area at the crucial moment. A kind of limit cycling obtains which unloads the compressor so stall does not occur (Fig. 9). This limit cycling "double dip" is different in character from the behaviour which is termed "nozzle dance" which sometimes occurs on controls which are not quite compatible

Limit cycling, however, can also happen when turbine power is degraded, therefore it is not proof that the compressor is deficient. For separation of the two, Compressor Discharge Temperature must be determined, and this requires multiple probes to do with some degree of certainty, and these usually cannot be accommodated on a production engine. During steady state running turbine efficiency does not affect compressor loading, because the nozzle is opened till the engine runs at rated EGT. In a slam accel, however, the nozzle is momentarily closed to the normal minimum flow area; at that moment a reduced turbine efficiency will give an increased compressor loading which may stall it.

One thing is certain - apart from gross control malfunction - the engine will always benefit from an improved compressor stall line. Compressors with persistent stall problems have been checked for blade and vane tip clearances, and several discrepancies have been found which on correcting improved the behaviour of the matched rotor and stator set. One of these was scalloping of the 12 stator segments which make up a stage. Recurrence after correction could be avoided by giving the segments slightly more circumferential room to expand without bending. Another was excentricity of the stator casing, possibly because of high in-flight g-loading or a one-sided scrape after a stall - the stall is usually not symmetric and therefore may cause a radial load on the rotor and the compressor casing. This may also give a permanent set to the rear stage discs, which overhang the compressor rear bearing (usually detected from an increase in vibration).

Blade tip clearances of this small engine are subject to narrow limitation, with the result that blade replacement upon refurbishment is high, especially with recent further tightening of the limitation. Blades are installed oversize and then ground to limitations; this results in some cases in a typical B-distribution of blade lengths. If too many short blades are grouped together a low stall line will result despite the fact that minimum and average clearance are within tolerance (Fig. 10). This is especially so if successive stages exhibit this anomaly in the same sector - not unlikely after the above mentioned one sided scrape. Although possibly a good compressor would result if blade length was randomly mixed, it pays to replace more blades to avoid the above situation.

It is clear that factory-issued limitations can never be fixed once and for all; experience is gained all the time - not only by the factory representative but also by the operator and his consultants - and it is used to improve specification. On the other hand there is a great need for reliable tests in trouble shooting to avoid unnecessary control changes, which do not solve the problem, even though the stacking of the tolerances will be altered which may offer some temporary relief. Reliable diagnostics may be difficult to achieve but they will be cost-effective!

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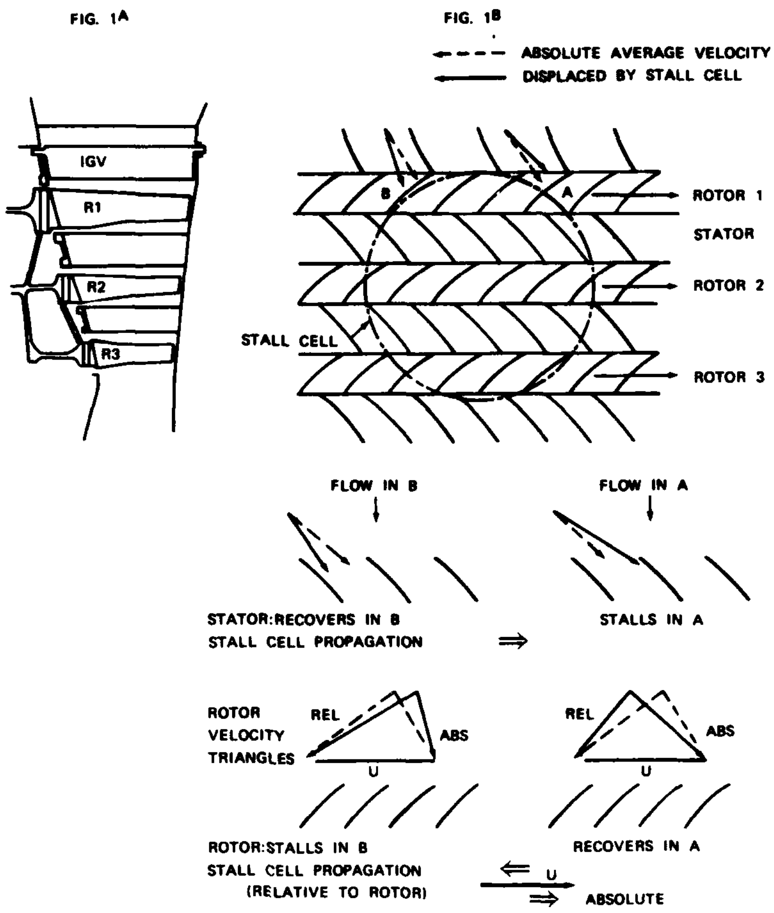

Fig. 1 Variation of blade length/stall cell propagation

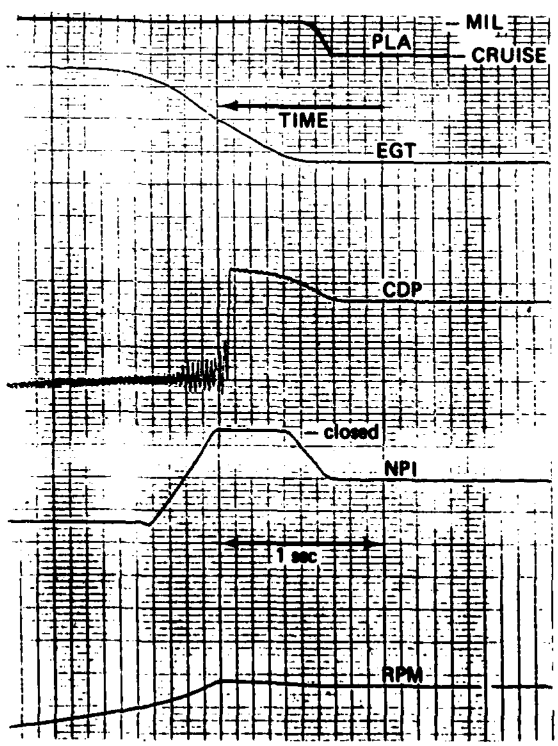

Fig. 2 Accel stall

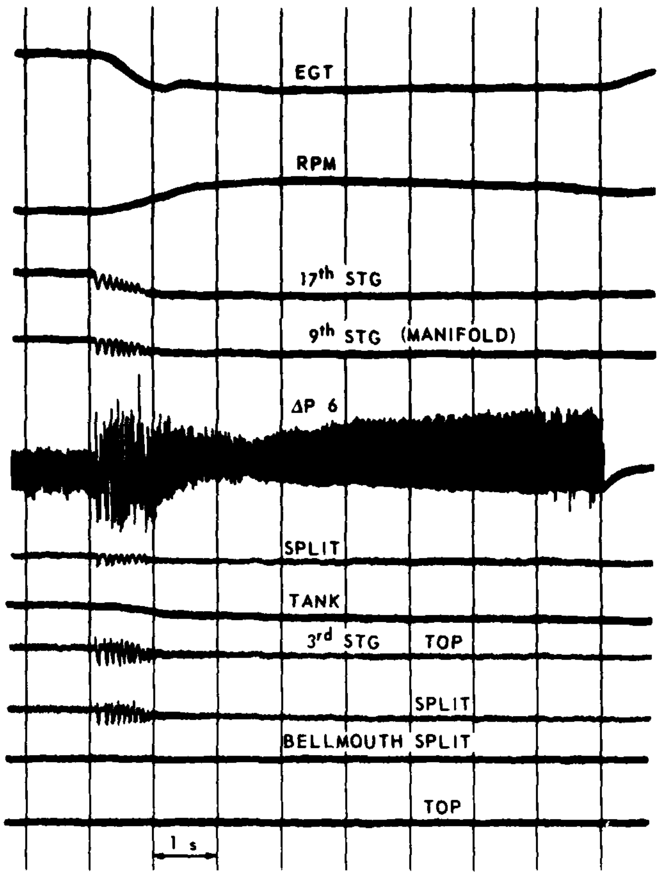

Fig. 3 Parameter variation during decel stall

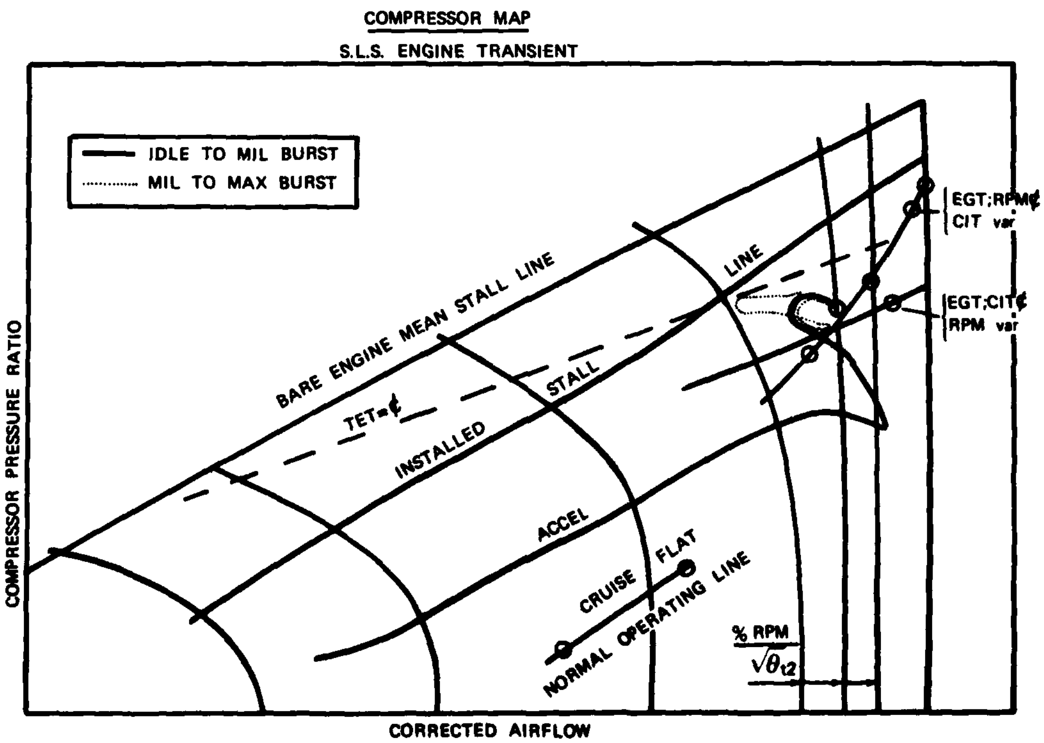

Fig. 4 Operating lines

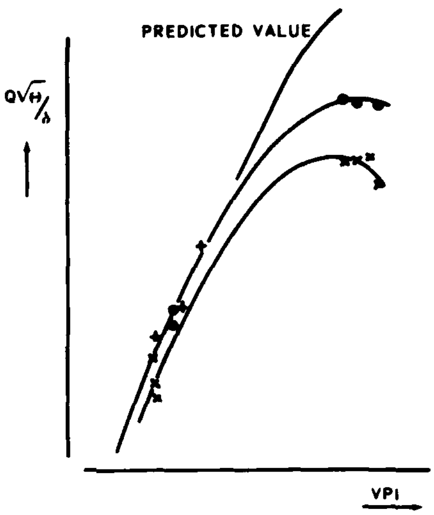

Fig. 5 Airflow vs vane position for constant corrected RPM

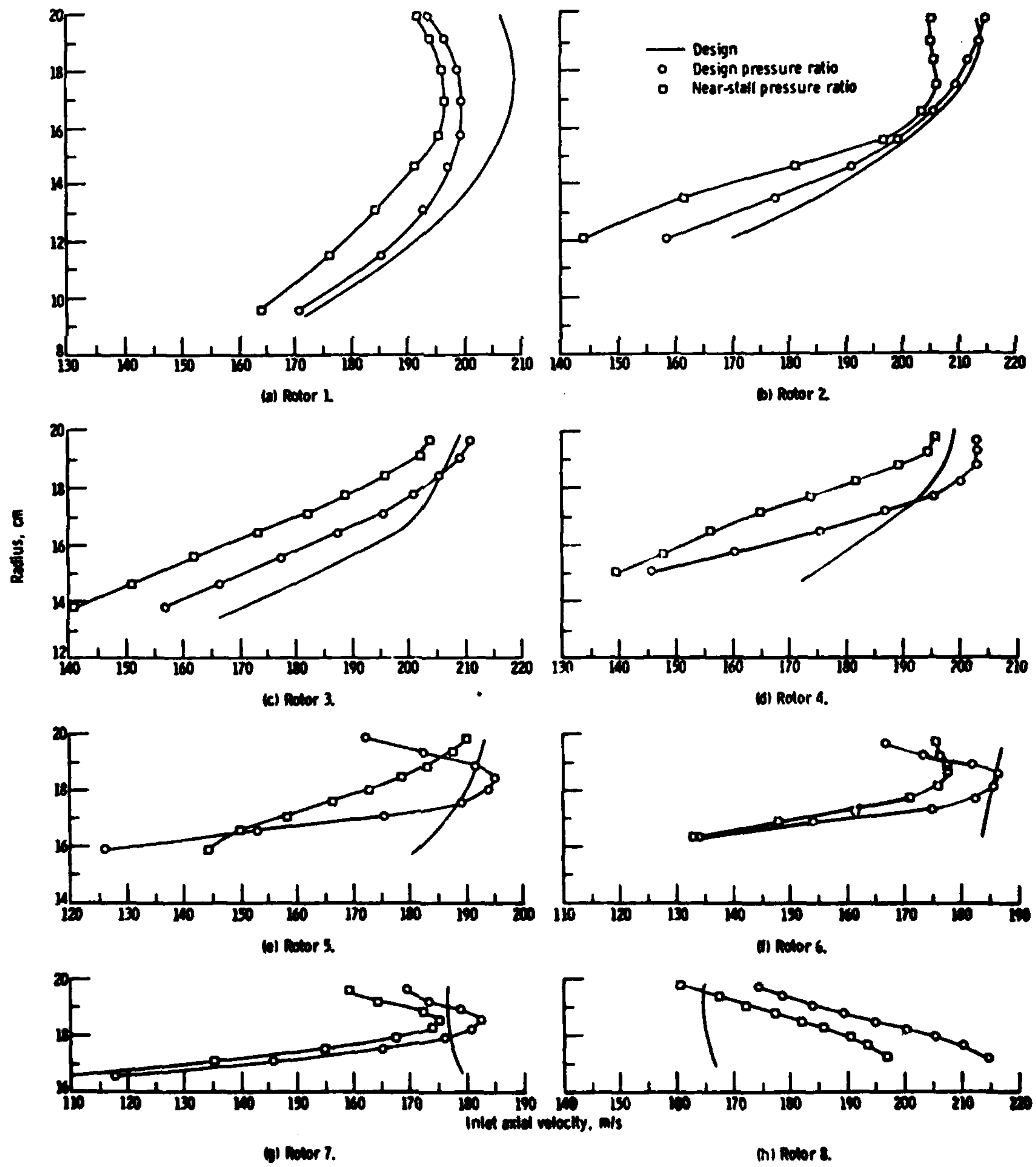

Fig. 6 Rotor inlet axial velocity profile for J85-13 engine at 100 per cent of design speed without casing treatment

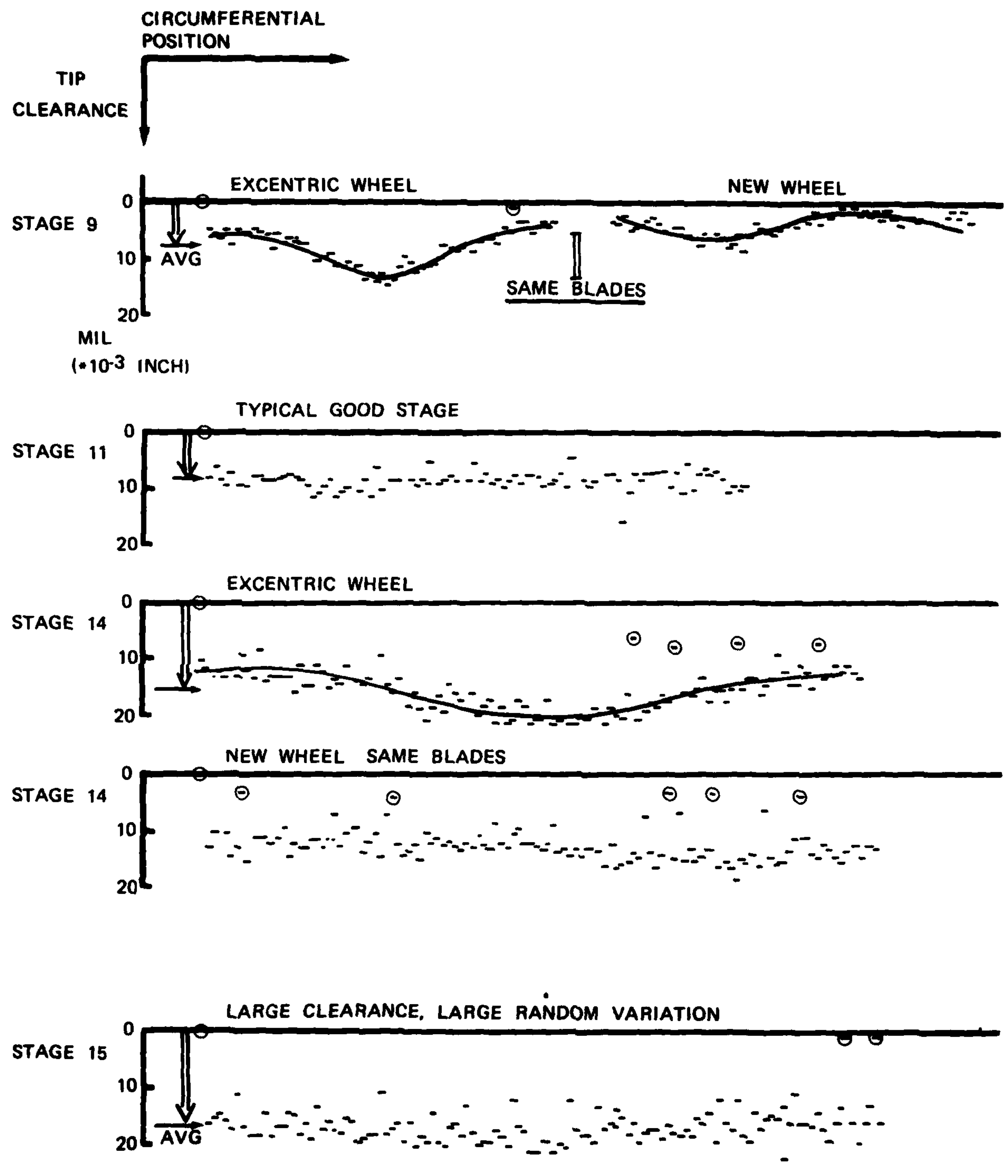

Fig. 7 Variation of tip clearance in deficient compressor

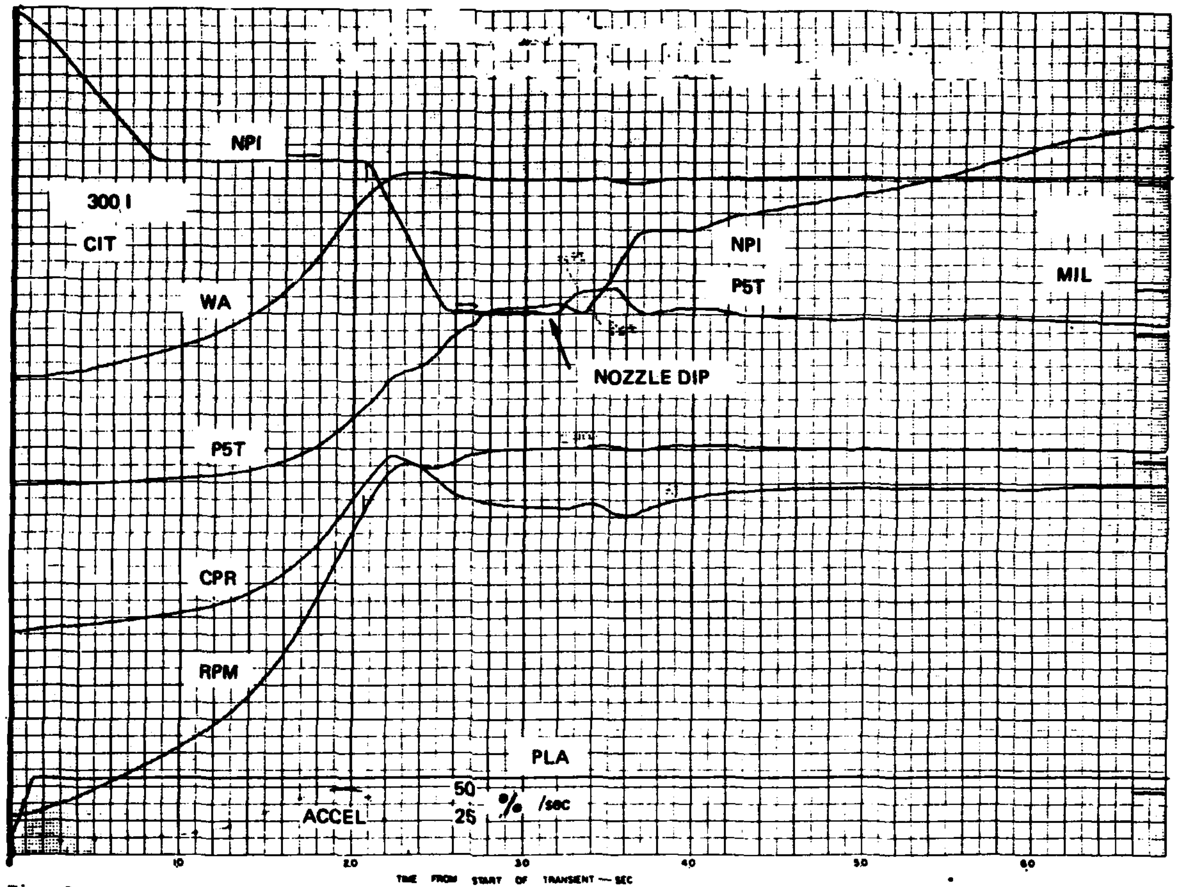

Fig. 8a

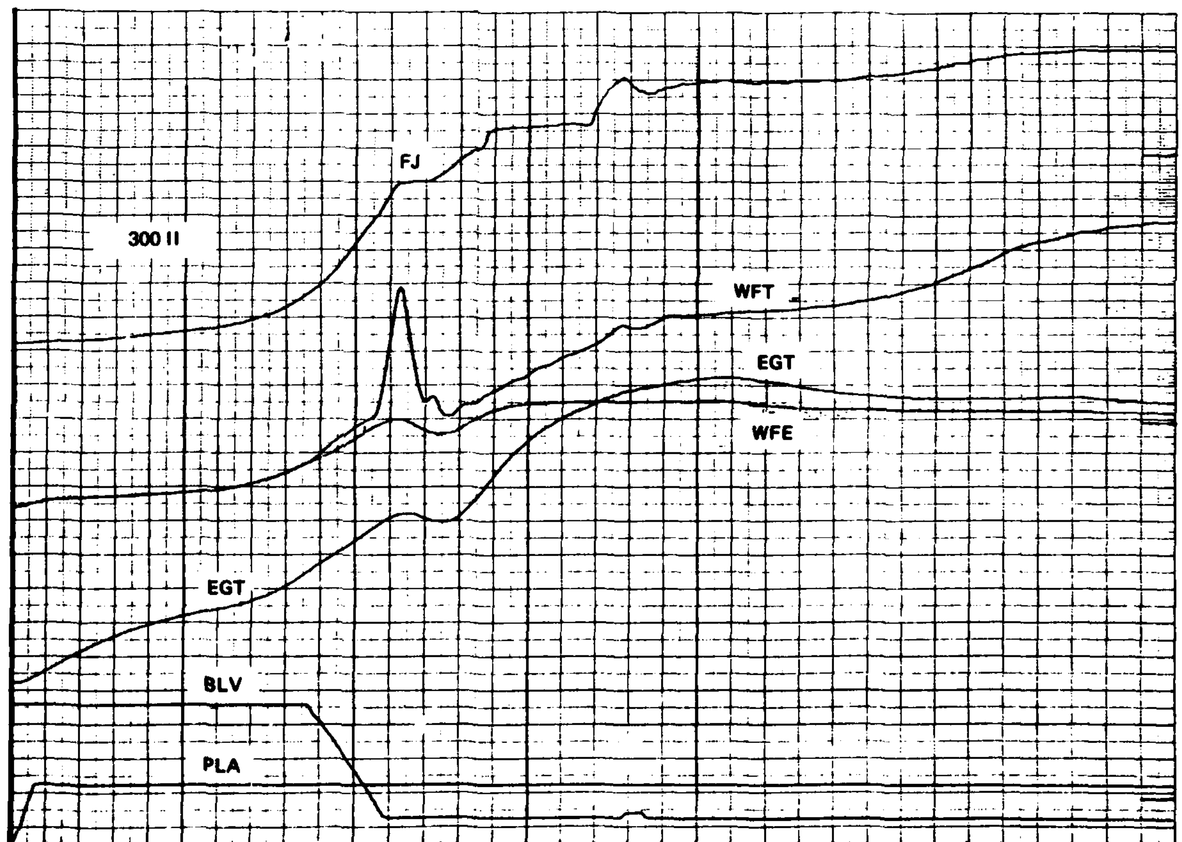

Fig. 8b

Fig. 8 Parameter variation during slam from idle to max AB

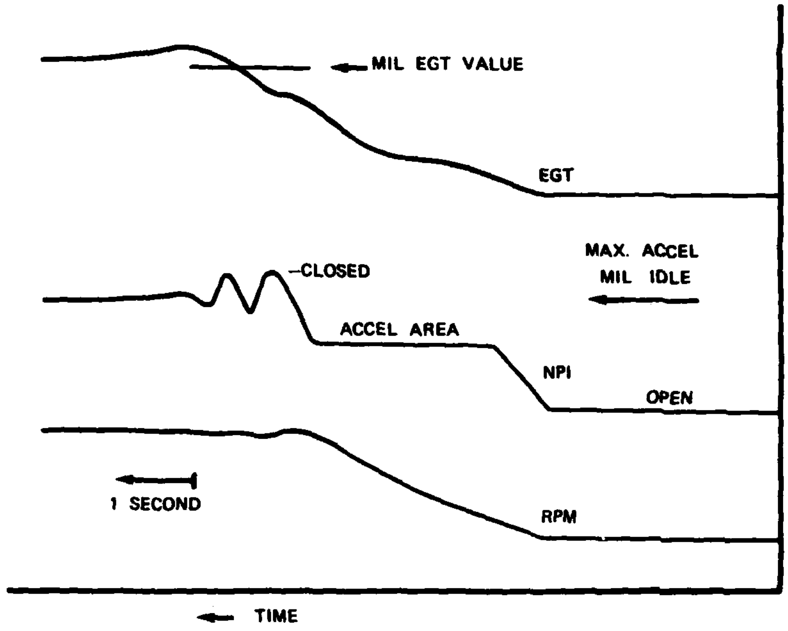

Fig. 9a Limit cycling of nozzle control

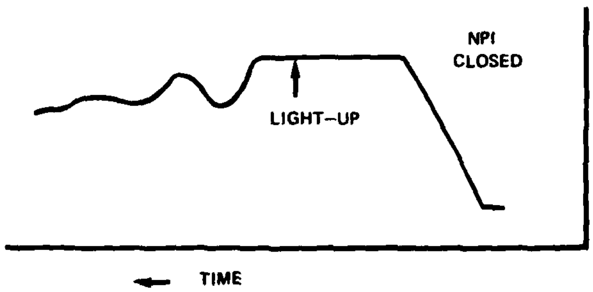

Fig. 9b Nozzle dance (AB initiation)

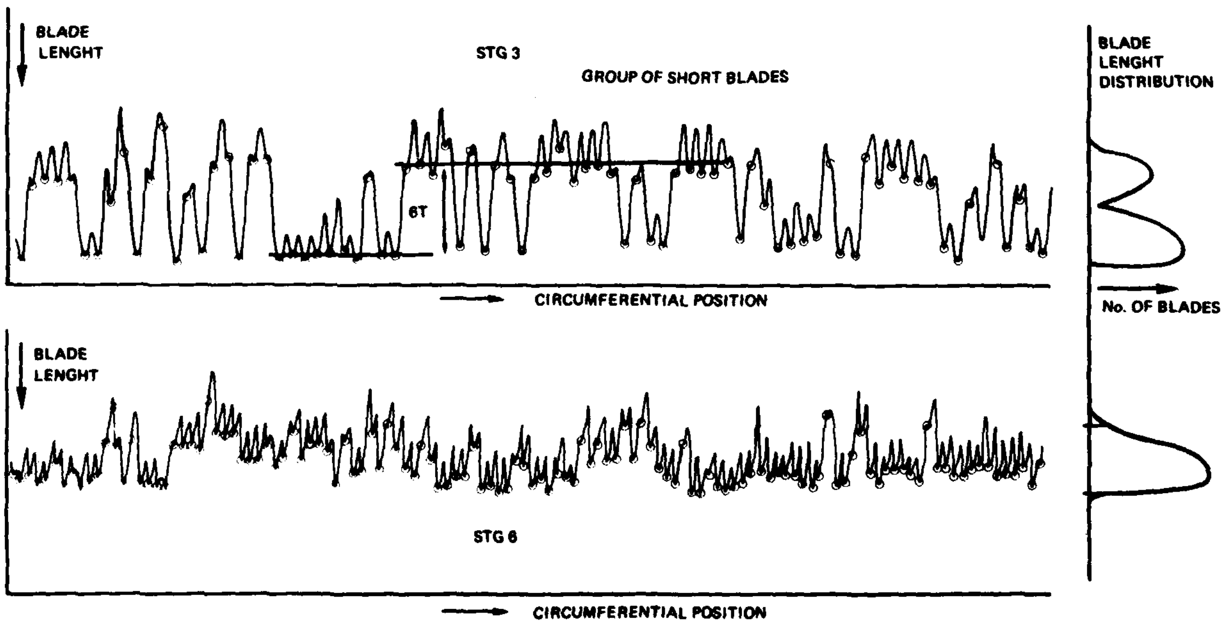

Fig. 10 Irregular tip clearance on deficient compressor

DISCUSSION

D.K. Hennecke, Ge.

How did you measure the tip clearances?

Author's Reply

On the J79 blade length was measured blade for blade by hand with a micrometer, and average case diameter in a number of points (only for a few selected engines).

On the J85 blade, length is measured per stage with a shadowgraph. The upper points are meaningless; the lower (marked) points give the gap relative to a master. The case inside diameter is measured in 12 points with a micrometer. This is the standard procedure at overhaul.

EFFECTS OF INTAKE FLOW DISTORTION ON ENGINE STABILITY

by

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SUMMARY

Current approaches to the intake/engine compatibility problem involve assessing the intake flow distortion in time-variant, spatial, total-pressure terms. Initial assessments of engine stability, conducted prior to engine ground tests, either with simulated total-pressure distortion or the intake, employ empirical or semi-empirical correlations of the surge-margin losses of individual compressors tested in isolation with simulated distortion. Component results are stacked to provide assessments of engine overall distortion tolerance in fixed and variable-throttle conditions, accounting for distortion transfer, defined in terms of total-pressure and temperature, determined from the isolated compressor tests.

In many instances this approach has proved successful in projecting powerplant stability limits. In several important cases however, it is proving inadequate as other forms of intake flow distortion, such as swirl, are recognised as significant, and the assessment of interactions between the intake and LP compressor, the compressors in a multi-spool engine, and other engine components, assumes greater importance. A problem of increasing importance to V/STOL aircraft is that of intake temperature distortion arising from re-ingestion of engine exhaust gas.

Notable advances are being made in understanding the problems, key features being the development of advanced mathematical models which, in their simple forms, provide physical insight into the complex flow processes involved, and in developed forms provide quantitative estimates of powerplant stability limits. The development of ubiquitous theoretical and semi-empirical methods is of increasing importance as more is learned, and the complexity and expense of experimental testing mounts.

This paper provides a broad review of current methodologies, as they apply to military installations, and outlines areas where future work is needed. Attention is restricted mainly to the circumferential distortion question. Topics covered encompass inlet/engine flow-field interactions and synthesis of time-variant total-pressure distortion using limited high-response pitot instrumentation with turbulent flow modelling. Fundamental aspects of the spool-coupling process are addressed using theoretical methods and extended semi-empirical mathematical models of the developed parallel-compressor type. Problems that need to be resolved in the area of compressor dynamic response are discussed. Future requirements for assessing the destabilising influence of temperature distortion are outlined.

LIST OF SYMBOLS

DC (Θ)	Interface circumferential total-pressure distortion parameter = ($\Delta P_\Theta / q$)
IAS	Indicated Airspeed
N_F	Fan rotational rpm
$\bar{P}$, P_{mean}	Overall mean total-pressure at the defined interface
P_Θ	Minimum mean-total-pressure in a Θ^o sector at the interface
P_{max}	Maximum overall mean total-pressure at the interface at an instant of time
ΔP_Θ	$\bar{P} - P_\Theta$
ΔPRS	Loss of compressor surge-pressure-ratio = $(1 - \frac{R_{dist}}{R_{clean}})$ (defined at constant mass flow)
q	Time-averaged dynamic pressure at the interface
R_{clean}	Compressor surge-pressure-ratio with clean inlet flow
R_{dist}	Compressor surge-pressure-ratio with inlet flow distortion
R_{op}	Compressor operating-pressure-ratio

$(\Delta PRS)_{avail}$	Available $\Delta PRS = (1 - \frac{R_{op}}{R_{clean}})$
$TC(\theta)$	Interface circumferential total-temperature distortion parameter = $\frac{\Delta T\theta}{\bar{T}}$
$\bar{T}$	Overall mean total-temperature at the interface
T_θ	Maximum mean total-temperature defined in a θ^o sector at the interface
ΔT_θ	$T_\theta - \bar{T}$
W	Engine air mass flow
WC	Engine corrected air mass flow = $(W \sqrt{\theta_T}/\delta T)$
α	Aircraft incidence angle
β	Aircraft sideslip angle
δT	Relative total pressure = (P/101.325), P KPa
θT	Relative total temperature = (T/288.16), T K
θ	Circumferential position (angle)
θ^-	Angular extent of distortion sector in which total-pressure is below average total-pressure

1 INTRODUCTION

The stability of the flow within a compressor may be affected - both adversely and favourably, by distortion of the flow at entry and exit from an ascribed standard (usually uniform, axial flow for an intake). Distortion may be expressed as the variation of a number of factors, such as total- and static-pressure, total-temperature, swirl, axial- and circumferential-velocities in space, time or space and time. If two or more of these factors, eg total pressure and total temperature at inlet or total pressure at inlet with static pressure at outlet, are involved, the relative positions of the distortions as well as their magnitudes need to be taken into account.

Clearly, the complete definition of the flow conditions at any relevant station such as the interface between the intake and the engine can be extremely complex and it is necessary to simplify to keep distortion assessments within manageable proportions. For example, it has become the practice to study engine-intake compatibility only in terms of spatial total-pressure distortion, usually in a time-variant form. However, it is important to ensure that no influential factors are overlooked in the process of simplification. Examples of such factors are the variation of mean total-pressure with time or in-phase components (Reference 1) and swirl (see later). The significance of the time-variant nature of spatial distortion, together with methods of its assessment, including low-pass filtering requirements, have been well established for the case of total pressure (Reference 2). Relatively little work has yet been done for the case of inlet total-temperature distortion or swirl.

This paper discusses the following items which arise during the study of inlet flow distortion effects on engine handling:

(1) The measurement of flow distortion

(2) The assessment of the tolerance of an engine to total-pressure distortion

(3) The use of mathematical models in the solution of inlet distortion questions

(4) The effect of inlet flow distortion other than total pressure

The data included in this note can be considered, in part, an update and an extension of some of the information of AGARD LS72 (Reference 11) which covers this subject in greater detail. Areas where further work is considered to be necessary form the main theme of the discussion.

2 DISTORTION PARAMETERS

Distortion parameters are used to express a complex flow pattern in terms of numerical quantities (indices) which can be related to compressor stability margin. The considerations involved in deriving intake total-pressure distortion parameters have been formulated by the SAE S16 committee and incorporated in Aerospace Recommended Practice ARP 1420 and Air Information Report AIR1419, which, at the time of writing, is at the committee final-approval stage. It is shown that the pattern should be expressed as a combination of a number of elements, which quantify features such as the extent and intensity of the low-pressure sector, radial distortion and multiple-per-rev segments on an instrumentation ring-by-ring basis. These elements can be manipulated to form parameters considered to be relevant to a particular installation. As an example of the use of the S16 methodology, values of the Rolls-Royce DC(Θ) parameter obtained by conventional analysis and those derived via the S16 elements are compared in Figure 1. Close agreement is obtained.

The practice of correlating engine stability loss with time-variant total-pressure distortion is well understood and will not be dwelt upon here. Corresponding consensus recommendations for forms of distortion other than total pressure have yet to be formulated, however. While a fair body of individual information exists - for example on the destabilising aspects of total-temperature distortion at the intake/engine aerodynamic interface plane (AIP), a totally satisfactory descriptor system is still needed. The problem is difficult in that this form of AIP distortion usually arises in physical circumstances when a high in-phase or planar temperature rise (or cycle) occurs in conjunction with spatially-distributed and locally high-rate temperature peaks. In this respect the AIP accounting problem differs considerably from that which arises in the interspool gap of a multi-spool engine as a consequence of total-pressure distortion attenuation. The associated problem of developing descriptors capable of accounting for high levels of total-pressure and total-temperature distortion, occurring simultaneously, is clearly a very complex one. An example of the latter problem arises in weapon-wake ingestion assessments.

A wide range of experience now exists to show that in high-incidence manoeuvre operations, when intake-lip flow separation can occur in curved inlet ducts, secondary flow and hence high, bulk- and local-swirl results. Distortion parameters may need to be developed to account for the destabilising aspects of unsteady swirl in combination with time-variant total-pressure distortion. Moreover, in poorly designed inlets, swirl can cause significant port/starboard propulsion system stability differences in twin-engine, high-performance, military aircraft.

Areas where short-term future work on the development of AIP distortion parameters would usefully be concentrated then encompass:

- Coupled spatial and in-phase total-pressure distortion
- Time-variant spatial total-temperature distortion coupled with the in-phase component
- Coupled spatial total-pressure distortion with swirl

3 INTAKE DISTORTION MEASUREMENT

A survey of techniques for measuring total-pressure distortion is given in Reference 3 and AIR1419. The following developments are now known to be significant:

- A major source of intake total-pressure distortion data is the test of a sub-scale model without engine representation. It is necessary to be clear on the relevance of such tests. Reference 4 illustrates that, for a simple pod intake, the engine-face flow distortion is markedly reduced if an engine is present (Figure 2). For long intake ducts, the differences between test results with and without an engine are small (Reference 5). Similar results to those of References 4 and 5 have been found in in-house research work. Such interaction effects also occur with distortion simulators, eg wire-mesh screens, and the engine.

- In full-scale tests, the presence of instrumentation, such as AIP pitot rakes, may affect engine behaviour - by reducing flow swirl, for example.

- The assessment of peak time-variant distortion levels and corresponding engine-face profiles by digital analysis of, say, 40 individual pressure-time histories is costly and time-consuming. During early phases of intake development, distortion synthesis methods (see References 6 and 7, for example) which require fewer and simpler time-variant outputs, may suffice. Figure 3 shows a comparison of peak DC(Θ) levels synthesised using only <u>one</u> RMS value on each of 8 rake-arms with those from a full digital analysis. Good agreement is achieved at lower distortion levels by assuming that the unsteady flow at each pitot has a normal, independent probability distribution ie, cross-correlation with all adjacent pitots is zero. At high distortion levels it is shown that cross-correlation becomes increasingly important.

- High-response, total-temperature distortion measurements and swirl measurements may be required in appropriate circumstances, eg V/STOL hot-gas reingestion and secondary flow situations, respectively.

Future work should take into consideration engine/inlet interaction effects and the need to develop simpler, time-variant total-pressure-distortion measurement procedures to ease data acquisition requirements and facilitate the measurement of additional parameters.

4 ENGINE DISTORTION TOLERANCE ASSESSMENT

One of the main objectives of assessments of the response of turbomachinery to AIP total-pressure distortion is the determination of the inlet distortion level at engine instability onset. By-and-large, the data used in current methods are derived from correlations of individual compressors tested in isolation. Inlet total-pressure distortion effects on the compressor characteristics, in particular on the surge line, are determined together with distortion transfer data. Individual spool results are stacked to determine the critical compression system component and hence the engine tolerance. The downstream (HP) compressor is affected by temperature as well as pressure distortion. For certain compressor designs with flatter, wider-spaced, constant-speed lines, the former effect can be dominant.

Prerequisite to the distortion assessment is a knowledge of component-compressor characteristics and the manner in which the compressors are matched in clean (undistorted) flow at specified propulsion-system operational points. The clean-flow compressor characteristics are affected by factors which act on surge and operating lines, and the operating lines themselves vary with engine running mode - fixed throttle or handling.

The major clean-flow factors include:

Surge Line Effects

a) Reynolds Number

b) Variable geometry, eg IGV and stator scheduling

c) Build and control tolerance and deterioration

d) Air bleed

e) Compressor back pressures, eg reheat light-up spikes

Surge line effect (a) can be accounted using either compressor test data or generalised correlations as contained, for example, in Reference 8. Test data on compressors or engines are necessary to quantify the effects (b), (c), (d) and (e).

Operating Line Effects

a) Reynolds Number

b) Customer air bleed and mechanical power offtake

c) Variable geometry

d) Throttle (PLA) movement

e) Fan and compressor back-pressure distortion: experienced, for example, with an assymmetric exhaust duct (Reference 9) or during reheat select/deselect (Reference 10)

f) Build and control tolerance

The operating line effects (a) to (d) can be obtained from engine cycle-deck calculations. In this context, the need for a transient performance deck, capable of providing operating lines for various throttle transients with representative control system inputs, cannot be overstated. Engine test experience is required to quantify the effects (e) and (f).

Figure 4 shows an example of a surge margin assessment for an HP compressor. It can be seen that, relative to the uninstalled, sea-level static case, the surge margin at altitude, with representative customer mechanical power and bleed offtake, is reduced by both a lowered surge line and a raised operating line. Slam throttle acceleration causes a further reduction of margin by raising the operating line.

Engine distortion tolerance can be estimated by dividing the available clean-flow surge margin by a 'sensitivity factor' which expresses the rate of loss of surge pressure ratio with an appropriate intake flow-distortion parameter. Distortion effects on the operating line as well as the surge line need to be taken into account.

The distortion sensitivities may be derived from compressor tests, from semi-empirical correlations or by theoretical methods validated by test experience. It has been found that further developments of the simple parallel-compressor theory (Reference 11) continue to provide useful predictions of the response of isolated compressors to both circumferential total-pressure and total-temperature distortion. Figure 5 shows that the predicted loss of surge pressure ratio due to total-pressure distortion for 3-, 4-, 5- and 6-stage compressors is well within 2% of the measured value - the accuracy suggested in ARP1420. To achieve this correspondence it is necessary to account properly for compressor dynamic response. Radial distortion effects need to be taken into account though some experience is that the loss of surge pressure ratio of LP compressors and fans from this source can be small

(Reference 12). Core compressors are not generally affected since radial profiles become identical to those with clean inlet flow after 3 compressor stages.

For multi-spool engines it is necessary to consider the interaction between the component compressors in terms of:

- the effect of the downstream (HP) unit on the LP compressor ('spool coupling')
- the transport of distortion through the upstream spool(s) in terms of total-pressure and total-temperature distortion, both of which have to be included in the sensitivity assessment. Figure 6 shows the transport of inlet total-pressure distortion across the hub of a 3-stage transonic fan. The consistent trends of total-pressure distortion attenuation and total-temperature distortion-generation ratios, at constant speed, were obtained by plotting against the difference between the test and surge pressure ratios. It is noted that the point of maximum pressure-distortion attenuation does not coincide with maximum temperature-distortion generation.

The effect of spool flow-field interactions, now known to be powerful in certain cases, is not always accounted for in assessment processes. Due to the difficulties of providing appropriate, flexible, spool-coupled rigs at reasonable cost there is an important need to develop prediction methods of a theoretical or semi-empirical nature to account properly the total compression-system stability. A prerequisite is a method for predicting individual (isolated) compressor response that is capable of being extended to the multi-spool case and account for other back-flow or outlet distortion effects. This subject is addressed further in Section 5.

Figure 7 shows a comparison between the distortion tolerance predicted for an engine at fixed-throttle conditions, compared with distortion levels obtained at flight-surge conditions when spool-coupling effects in a 2-spool engine have been properly accounted. Comparison of the predicted limits of distortion and levels obtained during surge-free slam accelerations are given in Figure 8.

5 TURBOMACHINERY MODELLING

Empirical or theoretical methods used to predict surge pressure ratio loss and exit-flow-distortion levels using isolated spool data, whilst providing results that have proven to be acceptable in essentially decoupled circumstances, are limited in the following respects:

- The component compressor clean-flow surge lines are assumed known
- The assessment models consider only steady, circumferential total-pressure distortion of the upstream compressor entry
- Little account is taken of the flow redistribution in the interspool gap. This results in an incorrect prediction of exit-flow distortion level, shape and angular location
- The effects of inlet swirl perturbations on compressor performance are not included
- The effects of interspool duct geometry (eg length) on spool-to-spool interactions are not usually included

More complex models are required to extend modelling capability to predict the effect of distortion on turbomachinery and to provide a more precise understanding of the flow fields. The flow may be modelled as (a) incompressible or compressible, (b) linear or non-linear, (c) steady or unsteady and (d) one-, two- or three-dimensional. The turbomachinery may be represented as an overall unit or a series of stages or blade-rows. Which of these options is necessary will be dictated by the nature of the problem being investigated. For example, an investigation of spool coupling will demand at least a two-dimensional model.

A discussion of some of the wide variety of models which have been produced may be found in References 13, 14 and 15.

Results from three typical in-house investigations are given below.

5.1 Distortion Transfer Through a Stage

Detailed turbomachinery response and flow-field descriptions can be obtained using a two-dimensional, linearised, compressible-flow representation, with semi-actuator discs modelling blade-row response. This allows gaps and blade chords to be included in the model. The model will provide a complete flow description at any axial station with arbitrary combinations of steady or unsteady inlet circumferential total-pressure, inlet total-temperature and exit static-pressure distortions. As an example, the model has been applied to the case of a single-stage with steady total-pressure distortion (Reference 16). Figure 9 shows good agreement between test and prediction, particularly for the (critical) low-pressure region. A non-linear version of this model may be used to predict stability limits with clean or distorted inlet-flow.

The prediction of the transfer of total-pressure and total-temperature distortion from this model and also from the multi-segment parallel compressor model (Reference 16) represents a significant improvement on early models using overall compressor characteristics only. A problem of the circumferentially-continuous and multi-segment models is that of establishing necessary and sufficient conditions for instability onset in distorted flow. Further work is required to establish usable circumferentially-lumped criteria based on stall cell growth/decay concepts.

5.2 Spool Coupling

In multi-spool engines, the response of an individual compressor to intake flow distortion will be affected by the presence of the other units. The modification of the level and form of the distortion transmitted through an upstream (LP) compressor is the most obvious form of such interaction. In the present context, 'spool-coupling' is taken to refer to the upstream interference caused by interspool static-pressure distortion created by the downstream (HP) compressor in the presence of HP-entry total-pressure and total-temperature distortion. With finite, non-zero, interspool-duct length, static-pressure distortion promoted at HPC entry persists in attenuated form up to the exit of the LPC. Reference 17 demonstrates that the distortion attenuation of a compressor can be changed significantly by the presence of an exit static-pressure distortion.

To illustrate the effect of spool coupling on compressor stability, the rate of loss of LPC surge-pressure-ratio due to inlet total-pressure distortion has been estimated with:

a) constant exit-static-pressure, ie, when there is zero spool coupling. This corresponds to an isolated compressor case.

b) a completely coupled HPC, ie, with effectively zero interspool-duct length. Static pressure is assumed constant at HPC exit.

Figure 10 shows that close-coupling the two compressors reduces the rate of loss of LPC surge pressure ratio by a factor of approximately 3. However, close-coupling also changes the distortion transfer which gives the HPC a different operating environment. Thus, for a given intake distortion level the required stability margin is reduced on the LPC and may be increased on the HPC, as spool-coupling is increased, depending on the total-pressure and temperature-distortion effects on the HPC.

For practical engine designs, the interspool duct lengths are finite and non-zero, ie, the compressors will be neither totally coupled nor totally decoupled. A two-dimensional, linear model has been used to investigate the effect of the interspool length. The governing equations can easily be solved analytically for the simple case of incompressible flow. Analysis shows that for two compressors with inlet guide vanes, represented by actuator discs separated by a constant-area, vaneless duct of ℓ mean compressor radii, the ratio of LPC exit-to-entry total pressure distortion ($R\ell$) may be expressed by:

$$\frac{R\ell}{R\infty} = \frac{\cosh \ell}{\cosh \ell - (1-(R\infty/RO))}$$

where $R\infty$ and Ro are the values of $R\ell$ with infinite and zero duct length, ie decoupled and completely coupled, respectively. From the above relationship a total-pressure 'Coupling Number' (CNP), which expresses the degree of coupling, may be defined as:

$$\text{CNP} = \frac{R\infty - R\ell}{R\infty - Ro}$$

The actual values of $R\ell$ and Ro will depend on the slopes of the speed lines of both the LP and HP compressors at their local matched operating point. $R\infty$ depends only on the local slope of the LPC.

A compressible linearised model of coupled compressors with IGVs fitted shows that there is a unique coupling number describing static-pressure, total-pressure and total-temperature distortions at LPC exit. The value differs from the incompressible value, which may be derived from the above equations as:

$$\text{CNP} = \frac{1}{1 + \frac{Ro}{R\infty}(\text{Cosh}\,\ell - 1)}$$

To investigate the compressible flow solution, including the effect of temperature distortion generated in the LPC, a numerical analysis has been performed on a two-spool arrangement having an HPC with no inlet guide vanes. In addition to the duct length effects this study also considered the influence of the slope of the HPC speed lines. The results have been presented in terms of:

- The pressure coupling number (CNP), as for the incompressible flow case

- The temperature coupling number (CNT), defined by:

$$CNT = \frac{RT_\infty - RT_\ell}{RT_\infty - RT_0}$$

where RT_ℓ is the ratio of LPC exit-temperature-distortion to inlet-pressure-distortion with duct length ℓ and RT_∞ and RT_0 are the values of RT_ℓ with infinite and zero duct lengths.

Figure 11 presents the values of CNP for the incompressible flow case and CNP and CNT from the compressible model plotted against duct length ℓ . The main points to be noted are as follows:

- For a given duct length the values of CNP and CNT obtained from compressible flow analysis are very similar but differ
- The slope of the HPC speed line has little effect on the degree of coupling in the range considered. It has a marked effect on the absolute values of $R\ell$ and $RT\ell$.
- The simple incompressible-flow model produces the correct trends although it overpredicts the actual value of CNP
- 'Half-coupling' (50%) exists with interspool duct lengths of the order of 0.5 mean radii which is in the range of current engine designs

A continuation of such analyses indicates that the addition of interspool vanes will increase the amount of spool coupling. The actual value will depend on the number, design and axial location of the vanes.

Spool-coupling will affect engine-compression system stability in distorted flow. To illustrate the distortion tolerance of an engine, a prediction, assuming zero coupling (when the LPC is the critical unit), 100% coupling and partial coupling (when the HPC is critical), was performed. The results are shown in Figure 12. These are compared with distortion levels occurring at surge, induced in flight. It can be seen that engine compressors were highly-coupled as was expected from the interspool duct geometry.

Estimation methods for spool-coupling in distorted flow are now available. Much work remains to be done to consolidate findings and improve prediction capability, eg for low hub/tip fans.

5.3 One-dimensional Pressure Distortion

In-phase, time-dependent distortion effects are predicted by one-dimensional, non-linear, compressible, 'lumped-volume' models in which the turbomachinery is represented usually by a series of stage performance characteristics. They are capable of handling time-variant perturbations of inlet total-pressure, inlet total-temperature and exit static-pressure. The basic models may also be used in a linear form to predict clean-flow surge onset by Eigenvalue Analysis. A satisfactory representation of the measured, clean surge line is a prerequisite to the use of the model.

A lumped, Continuity/Momentum/Energy model, applied to the case of a highly-loaded, high-Mach number, 7-stage compressor provides examples of the data which can be obtained. Figure 13 shows the predicted instability onset boundaries achieved by perturbing the inlet total-pressure sinusoidally at varying amplitudes and frequencies. Results are shown for two levels of surge margin. As expected, the amplitudes required to cause surge with low surge margin are much lower than for a high margin, although the differences become small at frequencies above 2 kHz. For the low surge margin case, a well-defined minimum surge amplitude occurred at approximately 400 Hz. As the margin increased (operating-pressure-ratio reduced) the minimum became less marked and occurred at higher frequency. Figure 14 shows the ratio of exit: entry amplitudes (gain) across the compressor due to small inlet perturbations. This indicates that the compressor behaves as a low-pass filter having a cut-off frequency of 150 Hz. The filter, which would be specified to analyse time-variant spatial distortion for this compressor has similar characteristics. This analysis also indicates there is a time-lag across the compressor varying from 2.1 msecs at 25 Hz to 1.4 msecs at 800 Hz.

Such models provide useful predictions of clean-flow surge lines and of the response of turbomachinery to one-dimensional inlet disturbances. Further work is needed to aid in the physical interpretation of the instability criteria used and to provide stage input data for the model covering operation well into the stage stall regions. In general, the models need to be extended to the multi-spool problem, bearing in mind the spool speed-match problem in clean flow.

6 OTHER FORMS OF DISTORTION

In Section 1 it was noted that it is usual to study intake/ engine compatibility in terms of spatial total-pressure distortion. There are other forms of flow distortion which can, in some circumstances, be significant. Two of the most important - temperature distortion and flow swirl, are discussed below.

6.1 Intake Temperature Distortion

Intake temperature distortion, which can arise from exhaust gas reingestion on V/STOL aircraft or during armament firing, has been studied experimentally in terms of steady-state spatial distortion, with and without spatial total-pressure distortion, and inlet temperature ramps. Correlations of the data from both types of test available in 1974 are given in Reference 11. Since that time a number of results (References 18, 19 and 20) and in-house studies have confirmed the earlier data. An updated correlation for steady-state distortion is given in Figure 15, which presents surge pressure loss, ΔPRS, correlated against the total-temperature distortion coefficient, TC(Θ).

Reference 19 contains the results of tests on an engine with inlet temperature ramps contained within sectors of 90°, 180° and 270°, as well as over the overall engine face. These tests, with ramps in limited sectors, have similarities to some important V/STOL aircraft reingestion cases, and can be viewed as time-variant increases in both inlet mean temperature and spatial distortion. The data for 90% rpm are shown plotted in Figure 16 as mean-rise-rate against mean-temperature-rise at surge (or maximum when surge does not occur). It can be seen that surges occur whenever the rise-rate exceeds a threshold value and persists for approximately 25 msecs. This trend is as given in Reference 11, although the time was then inferred as 20 msecs.

For the low sector angles the minimum rate to cause surge - 500°C/sec for the 90° sector, is much lower than for an overall rise. When the time-variant, spatial, distortion is considered, it can be shown that, for the 90°- and 180°-sectors, surge occurs when TC(Θ) reaches the level shown to cause surge during steady-state distortion tests (Reference 18). These results, shown in Figure 17, depend on the assumption that, as for time-variant pressure distortion, there is a time delay of approximately 15 msecs between the occurrence of the surge-inducing distortion level and the indication of surge. For the 270°-sector the spatial distortion is low (approx 40% of the critical level): the rise rates are very similar to that for the overall ramp.

Thus, while it is probable that both rise-rate and spatial distortion contribute to the instability, these tests would suggest that, when the rises are contained within a 180° sector, spatial distortion predominates. For sectors higher than 180°, the rise rate becomes increasingly important and the dominant consideration. There is however a clear need for further work to formulate precise guidelines for the treatment of time-variant temperature distortion. Analytical work needs to include gas-to-metal heat transfer in transient conditions. Experimentally, work is required to develop test facilities, data acquisition and data processing facilities to a level comparable with those currently available for treating the total-pressure distortion problem.

6.2 Swirl

An engine-entry mean or bulk swirl, which can be generated by a single-sided separation in an 'S'-shaped intake duct, can have a significant influence on intake-engine compatibility (References 21 and 22).

Tests with simulated swirl, both with and without total-pressure distortion, performed to quantify these effects, have shown that, of the various configurations considered, only the combination of swirl rotating counter to the fan with high total-pressure distortion had a significant adverse effect on surge. Figure 18 shows this result in terms of the nozzle area required to surge at a given engine corrected-speed.

Mathematical modelling of a compressor with swirl input suggests that the cause of this effect is an adverse change of the distortion transfer characteristics through the front stages of the LP compressor and hence a higher distortion at the inlet of the core compressor.

Theoretical flow-field calculations confirm that turbomachinery stability response to total-pressure distortion in swirling flows can differ markedly from that observed in axial flow. Circumferential phase issues become important, requiring a precise description of any forced flow-angle perturbation due to the inlet relative to the total-pressure distortion, when significant intake secondary flow exists. Such conditions may arise in poorly-designed curved inlets having flow separation at entry.

7 CONCLUDING REMARKS

Mathematical models and correlations of test data continue to improve the understanding of engine stability in intake distortion. Further theoretical and experimental work is required. Some suggested areas for work have been described in this paper. These and other topics, outside the scope of this paper, include:

- Development of improved interface distortion parameters to cover spatial total-pressure distortion in combination with in-phase total-pressure variations, total-temperature distortion and flow swirl.
- Improvement of measurement techniques to include engine/intake interactions and to develop simpler methods of time-variant total-pressure distortion acquisition
- Establishment of usable blade row or stage stability criteria, based on stall cell growth/decay considerations
- Consolidation and extension of spool-coupling analyses

- Interpretation of the physical meaning of numerical stability criteria and the provision of stage-performance data covering operation into the stage-stall region
- Formulation of guidelines for the treatment of time-variant total-temperature distortion
- Study of flow distortions involving changes in inlet flow angle as well as total-pressure variations, eg, due to ground vortices and intake-induced swirl
- Improvement of the treatment of three-dimensional (low hub/tip) flow models, in particular for compressors with split-stream flow
- Study of the behaviour of the engine following the onset of the instability. Whether an engine 'surge' is self-recovering or whether it is 'locked' or 'hung'. Some methods, currently available, are reviewed in Reference 14.

FIGURE 1
DISTORTION PARAMETER DERIVATION FROM ARP 1420 ELEMENTS

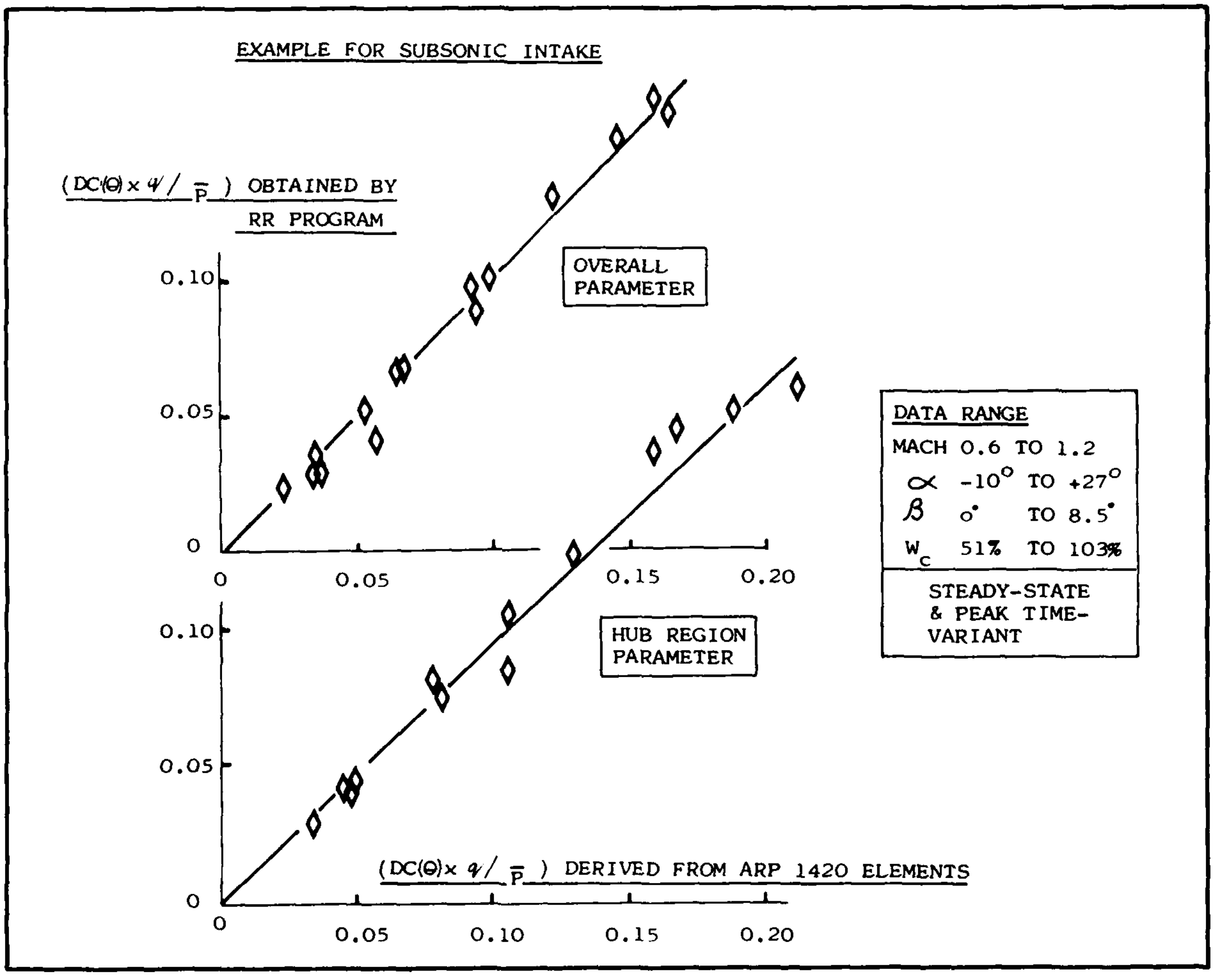


FIGURE 2
SHORT PITOT INTAKE - EFFECT OF PRESENCE OF AN ENGINE (FROM NASA CR-166136)

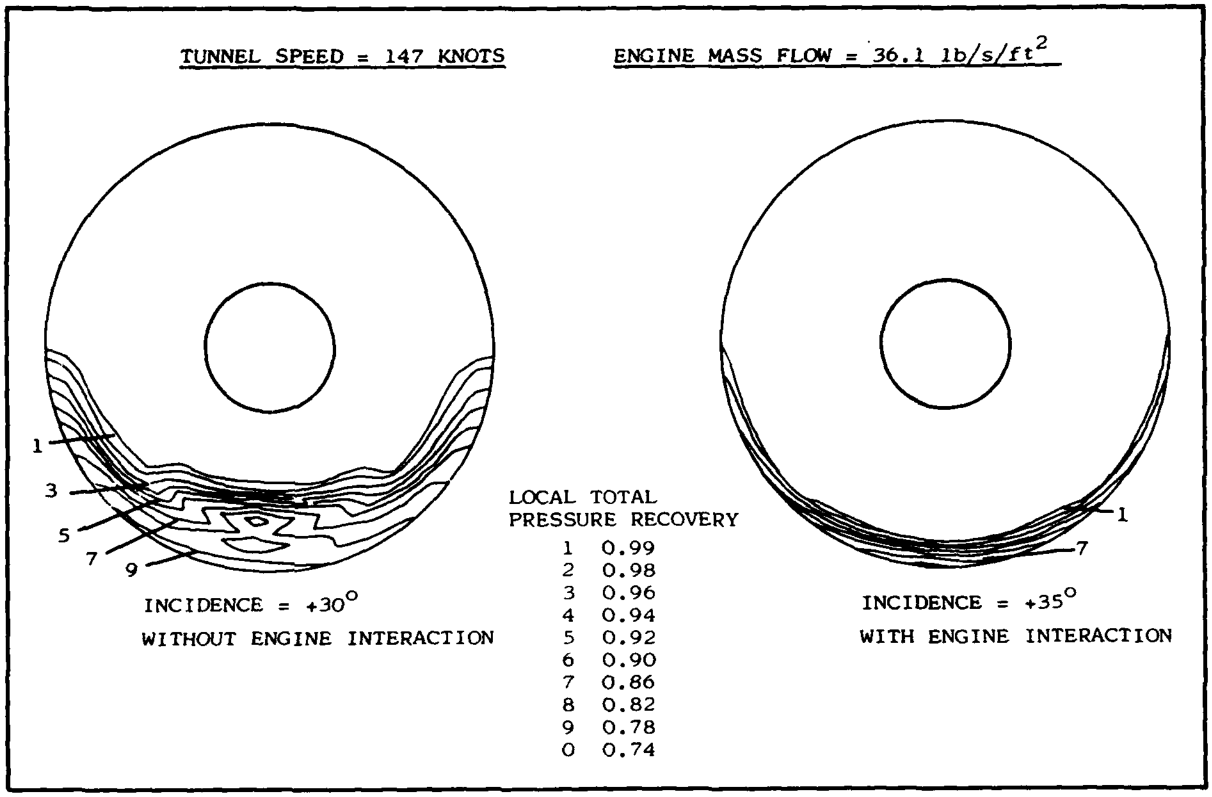


FIGURE 3
SYNTHESIS OF TIME-VARIANT INTAKE DISTORTION

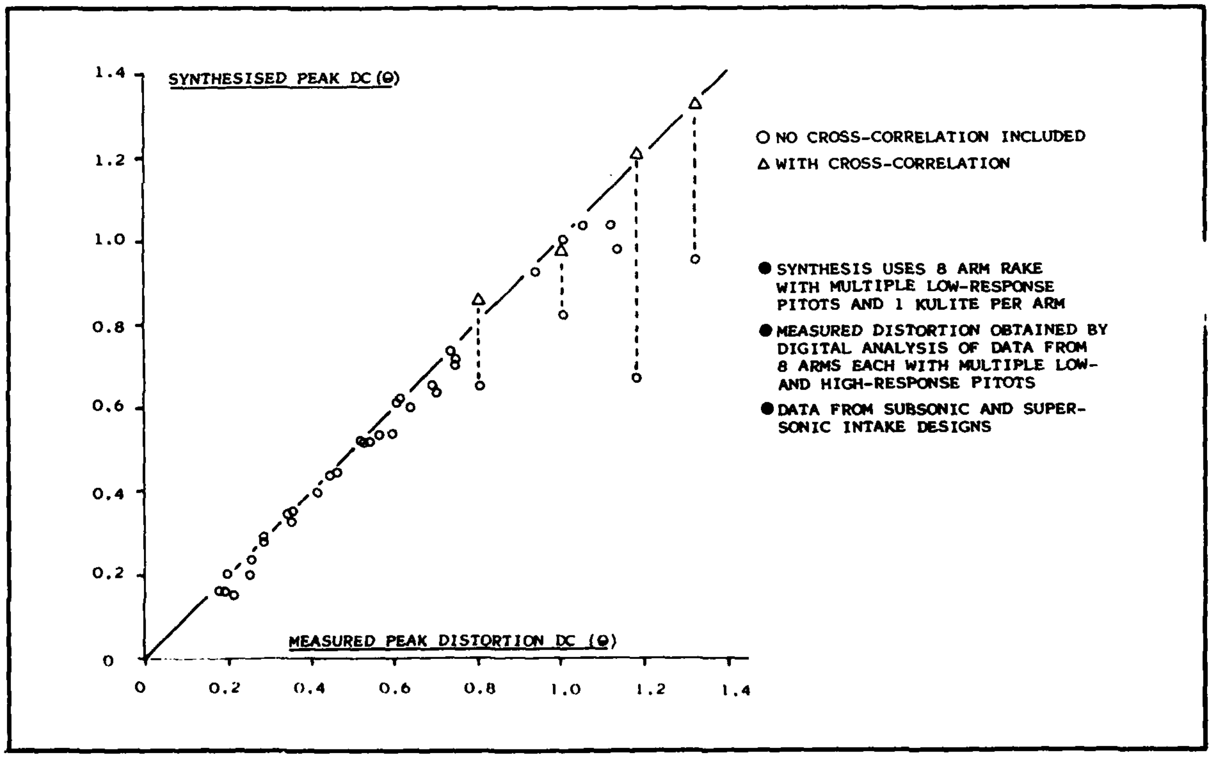


FIGURE 4
COMPRESSOR SURGE MARGIN ACCOUNTING - CORE COMPRESSOR

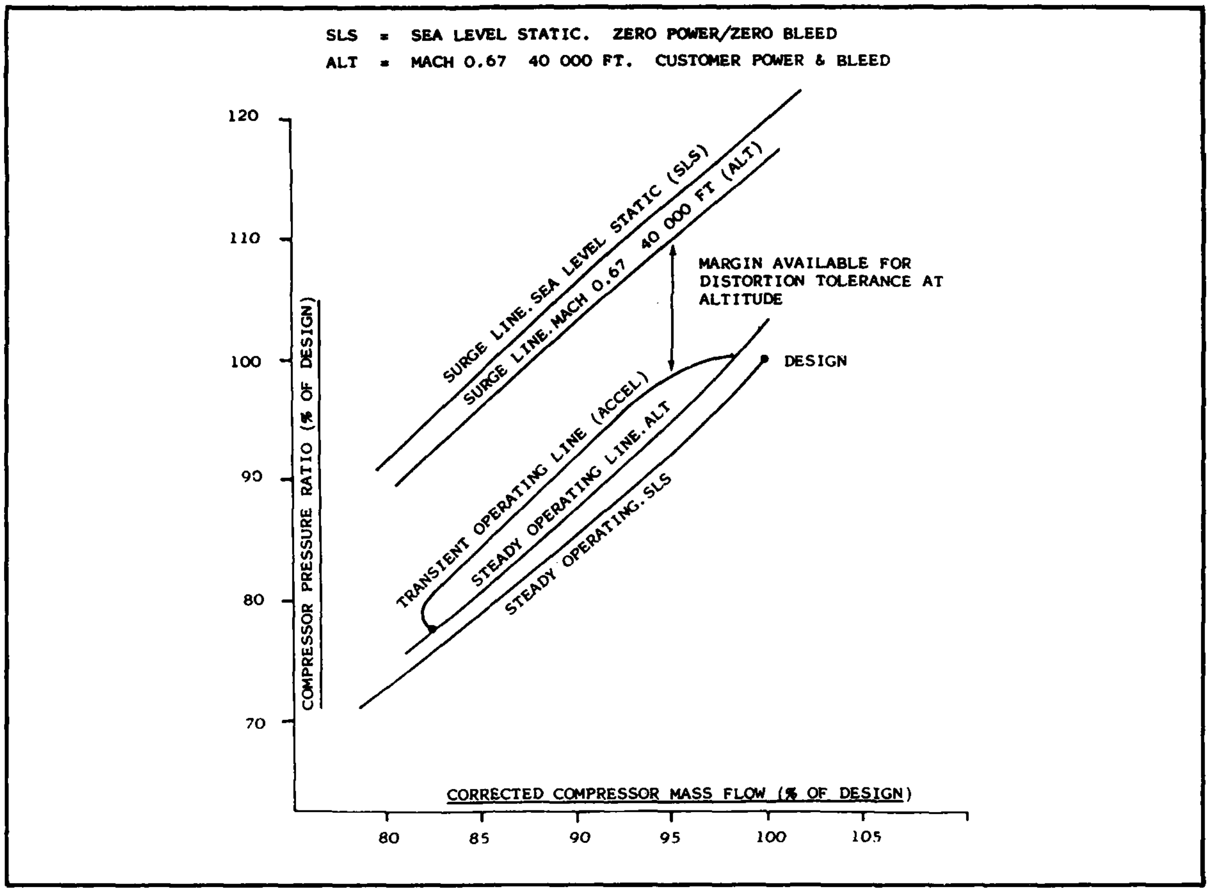


FIGURE 5
COMPARISON OF MEASURED AND PREDICTED LOSS OF COMPRESSOR SURGE PRESSURE RATIO

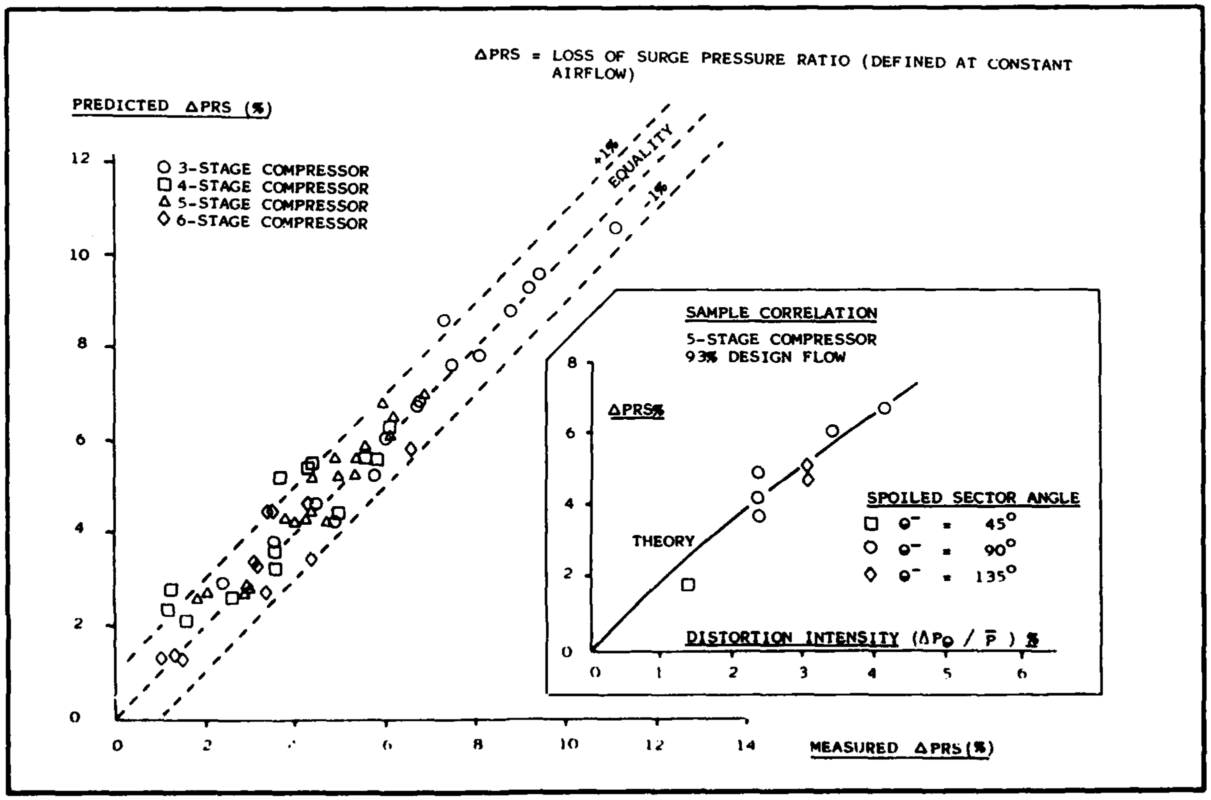


FIGURE 6
INTAKE TOTAL PRESSURE DISTORTION ACROSS THE HUB OF A 3-STAGE FAN

$$\frac{\Delta P\theta}{\overline{P}} = 1 - \left(\frac{\text{MIN TOTAL PRESSURE IN } \theta^{o} \text{ SECTOR}}{\text{OVERALL MEAN TOTAL PRESSURE}}\right)$$

$$\frac{\Delta T\theta}{\overline{T}} = \left(\frac{\text{MAX TOTAL TEMPERATURE IN } \theta^{o} \text{ SECTOR}}{\text{OVERALL MEAN TOTAL TEMPERATURE}}\right) - 1$$

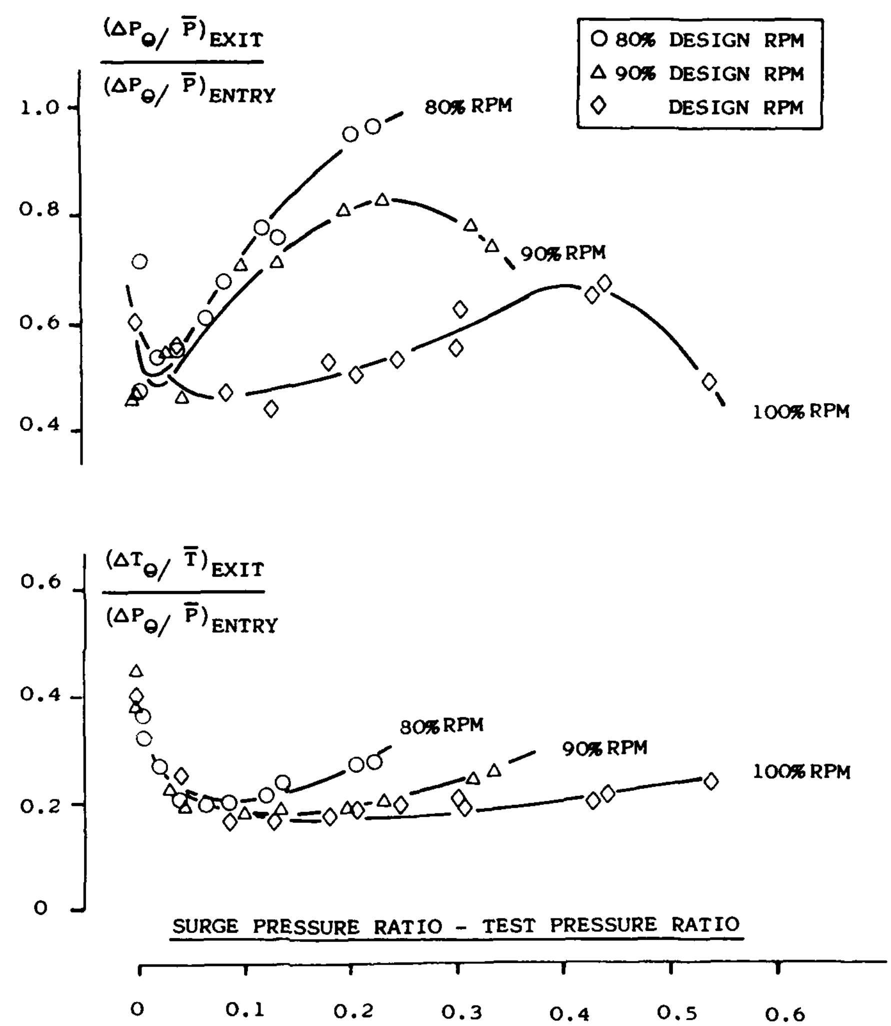


FIGURE 7
COMPARISON OF PREDICTED DISTORTION FOR SURGE WITH LEVELS AT SURGE

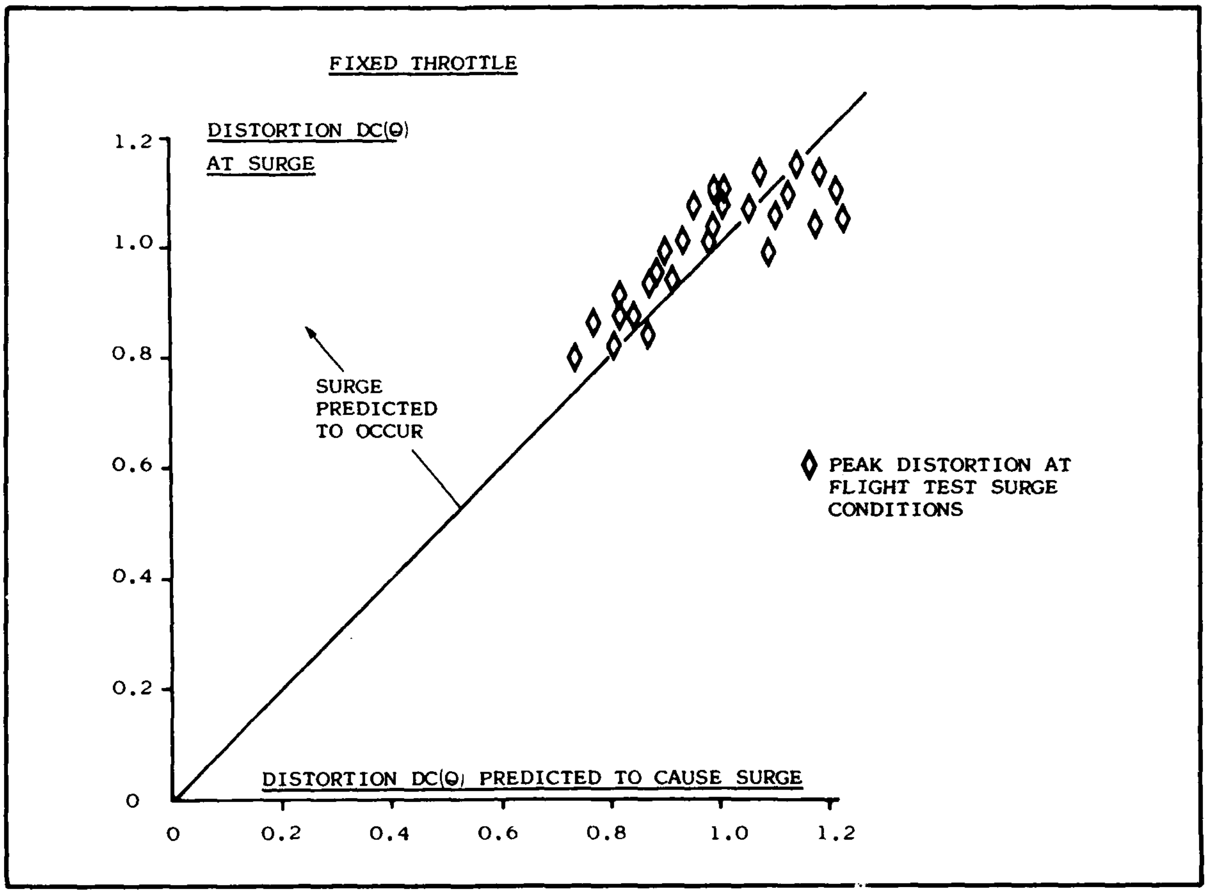


FIGURE 8
COMPARISON OF PREDICTED AND FLIGHT TEST RESULTS - SLAM THROTTLE TRANSIENTS

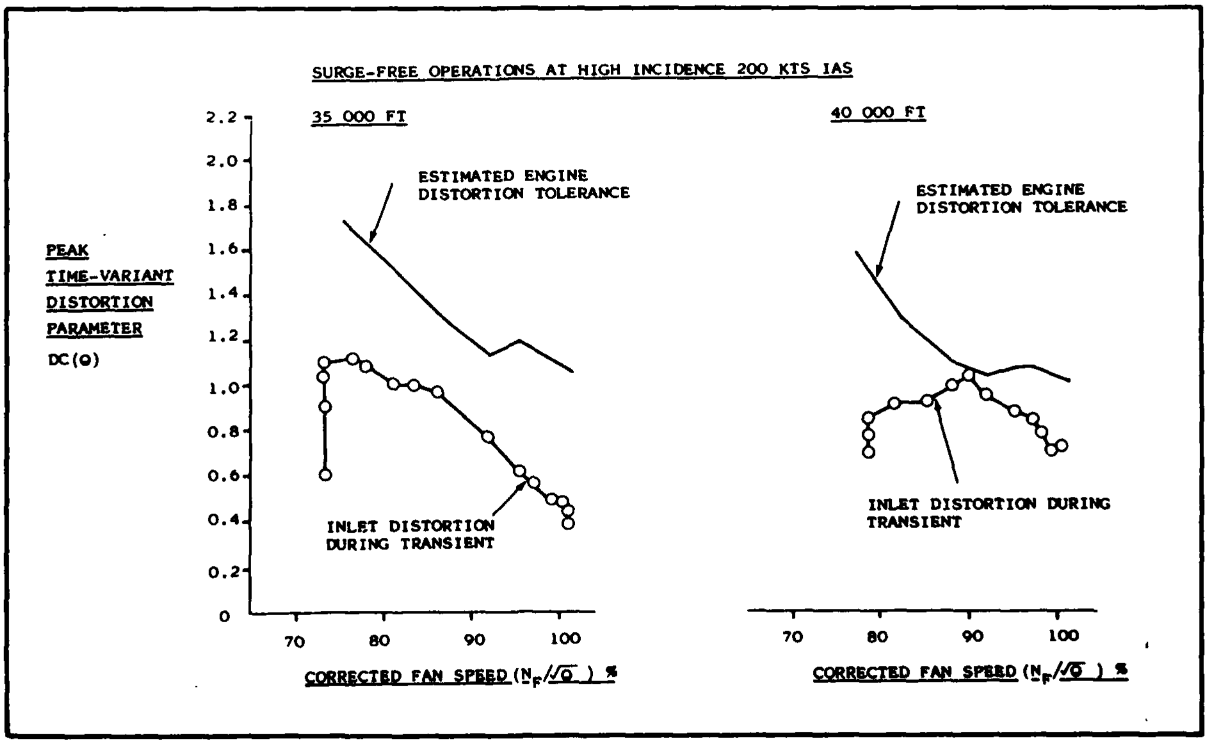


FIGURE 9
PREDICTION OF DISTORTION TRANSFER THROUGH SINGLE-STAGE

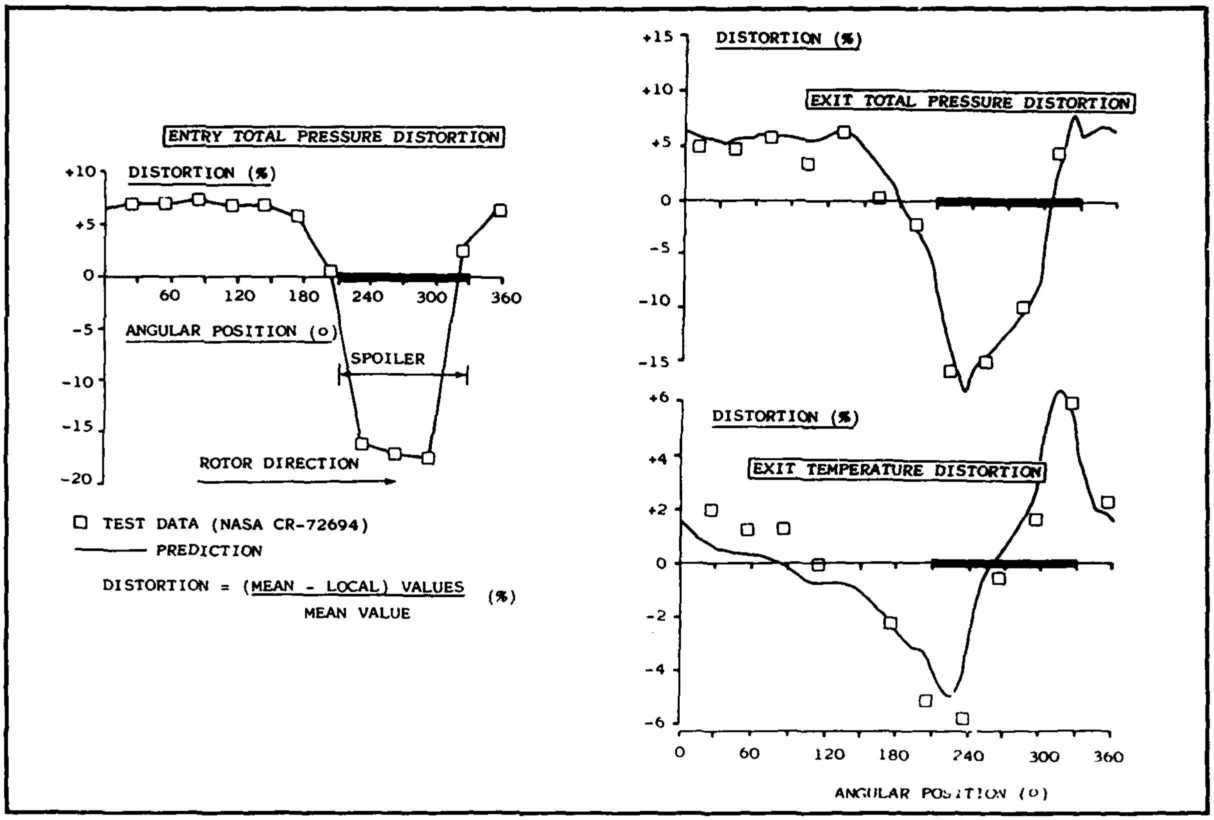


FIGURE 10
SPOOL INTERFERENCE CORE STABILITY PREDICTION - LPC SURGE SENSITIVITY

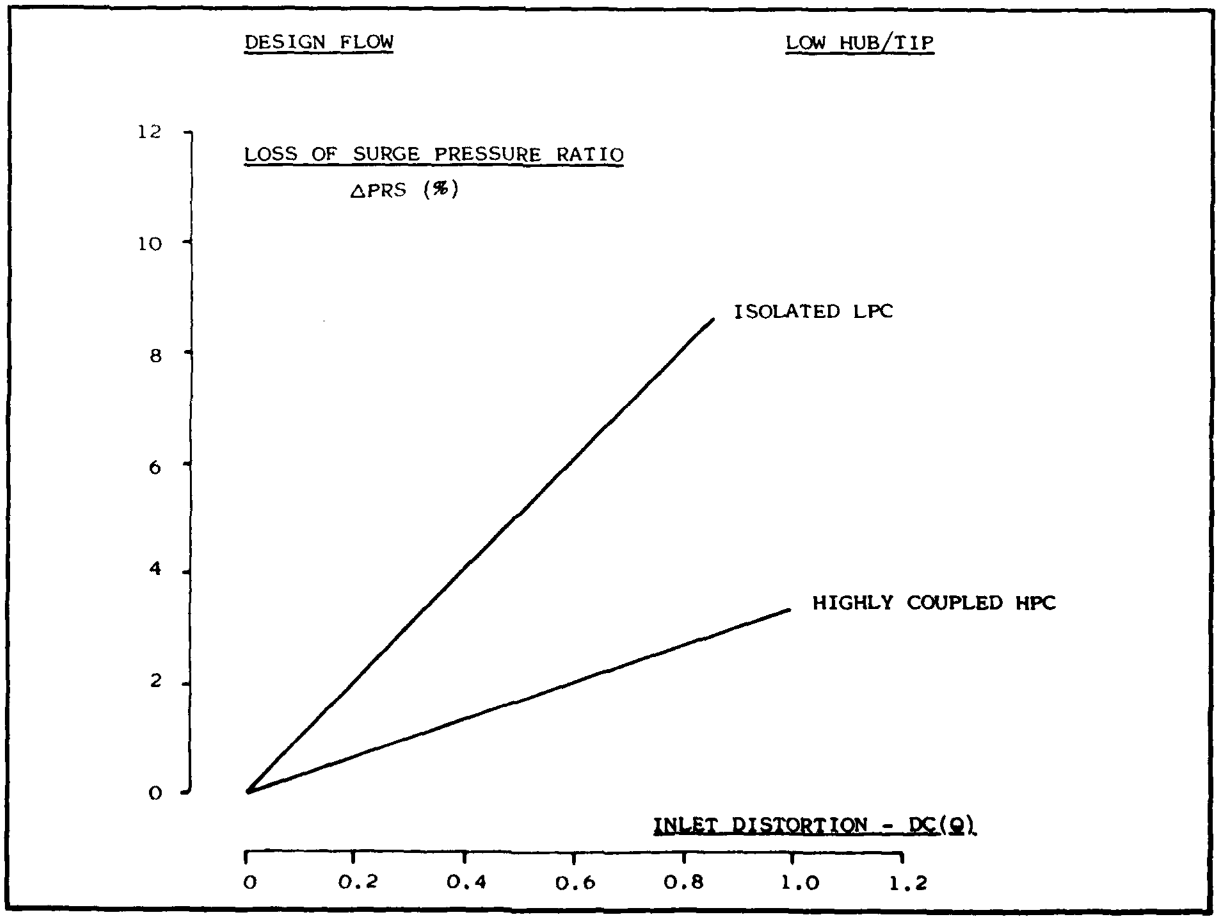


FIGURE 11
LP COMPRESSOR TOTAL PRESSURE DISTORTION ATTENUATION WITH SPOOL COUPLING - EFFECT OF DUCT LENGTH

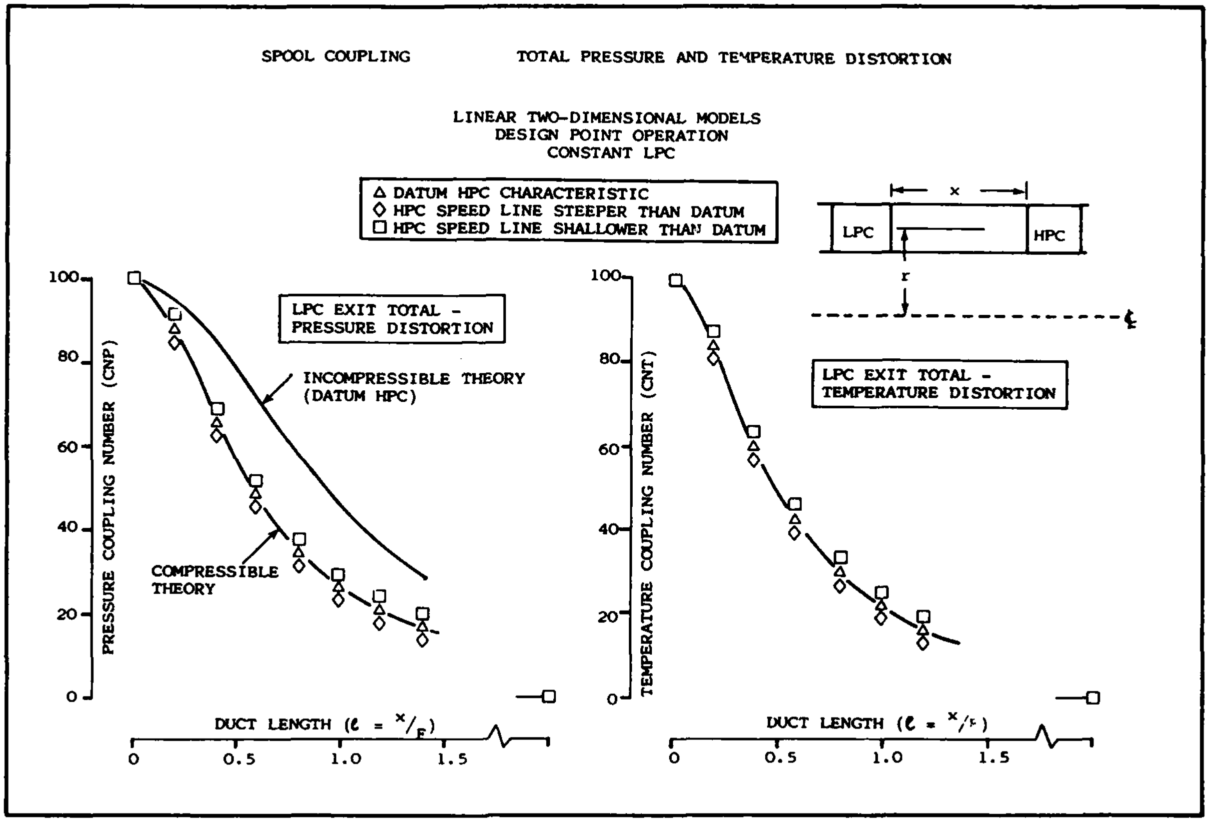


FIGURE 12
COMPARISON OF ENGINE DISTORTION TOLERANCE WITH ZERO, 100% AND PARTIAL SPOOL COUPLING

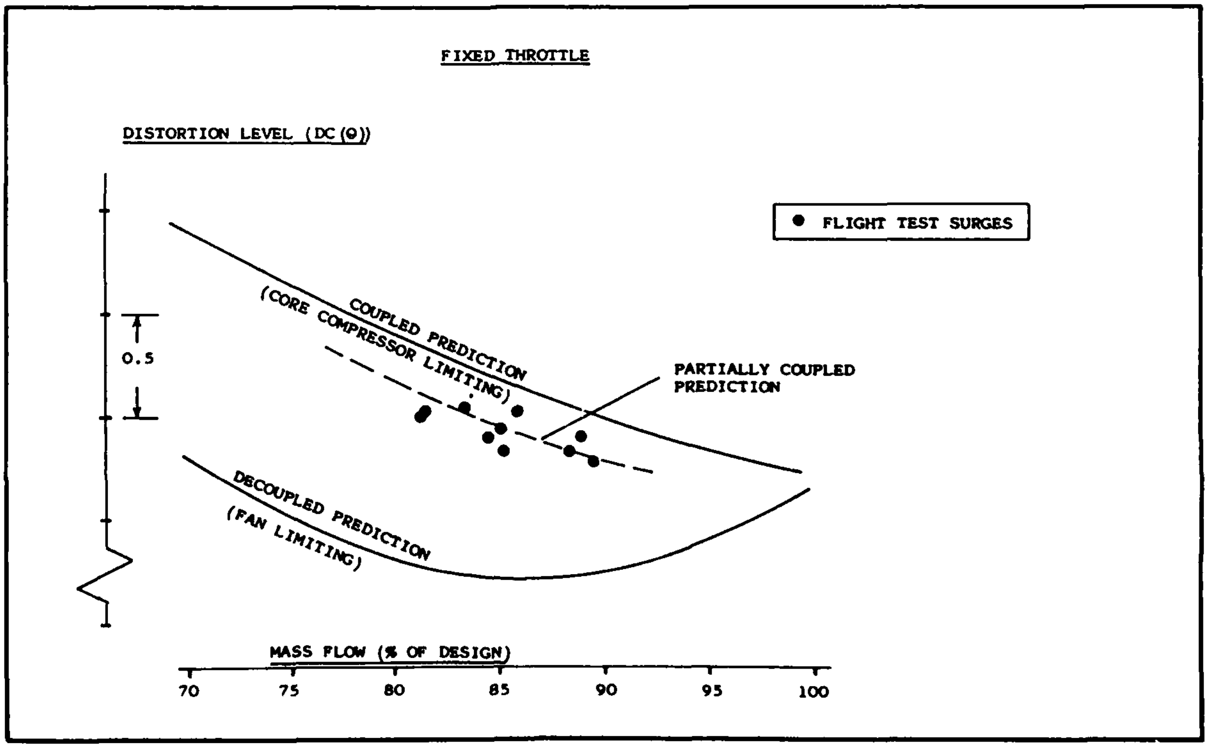


FIGURE 13
EFFECT OF VARIATION OF MEAN INLET TOTAL PRESSURE - PREDICTED INSTABILITY ONSET BOUNDARY FOR SINE-WAVE PERTURBATIONS WITH 7-STAGE HP COMPRESSOR

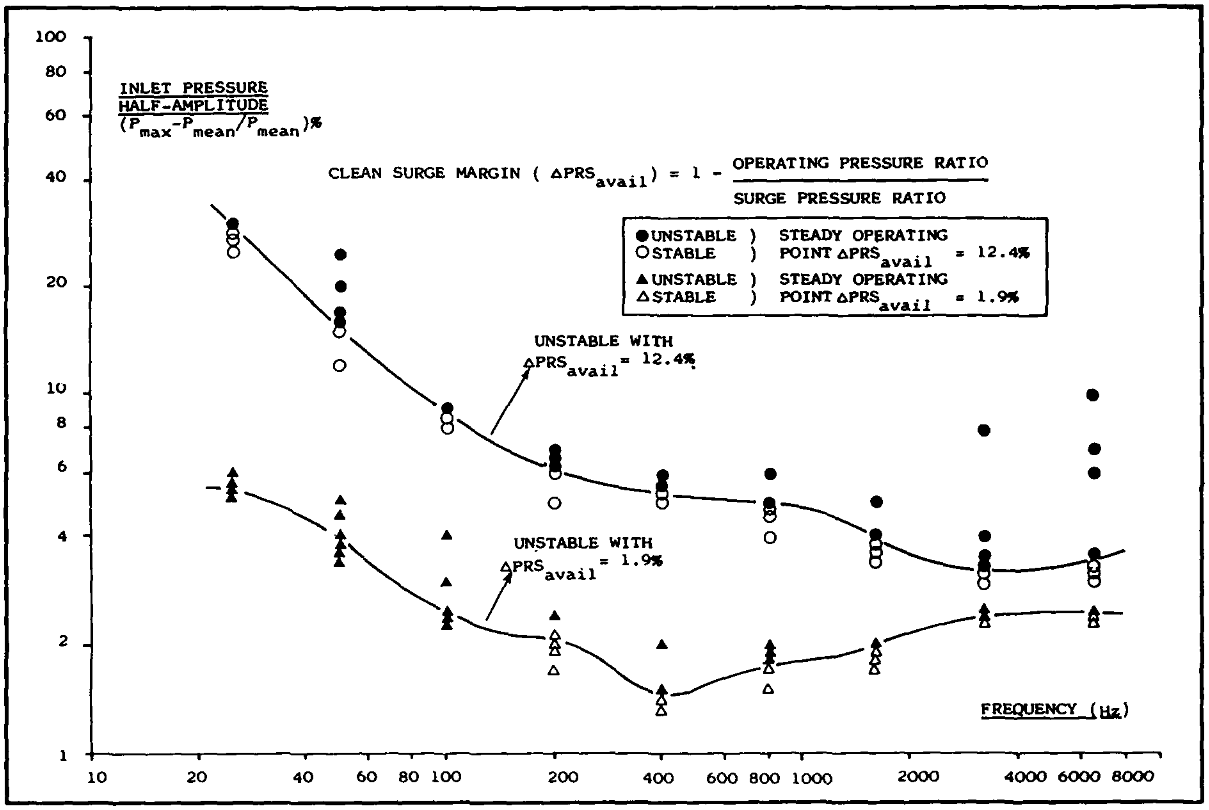


FIGURE 14
FREQUENCY RESPONSE OF 7-STAGE HP COMPRESSOR

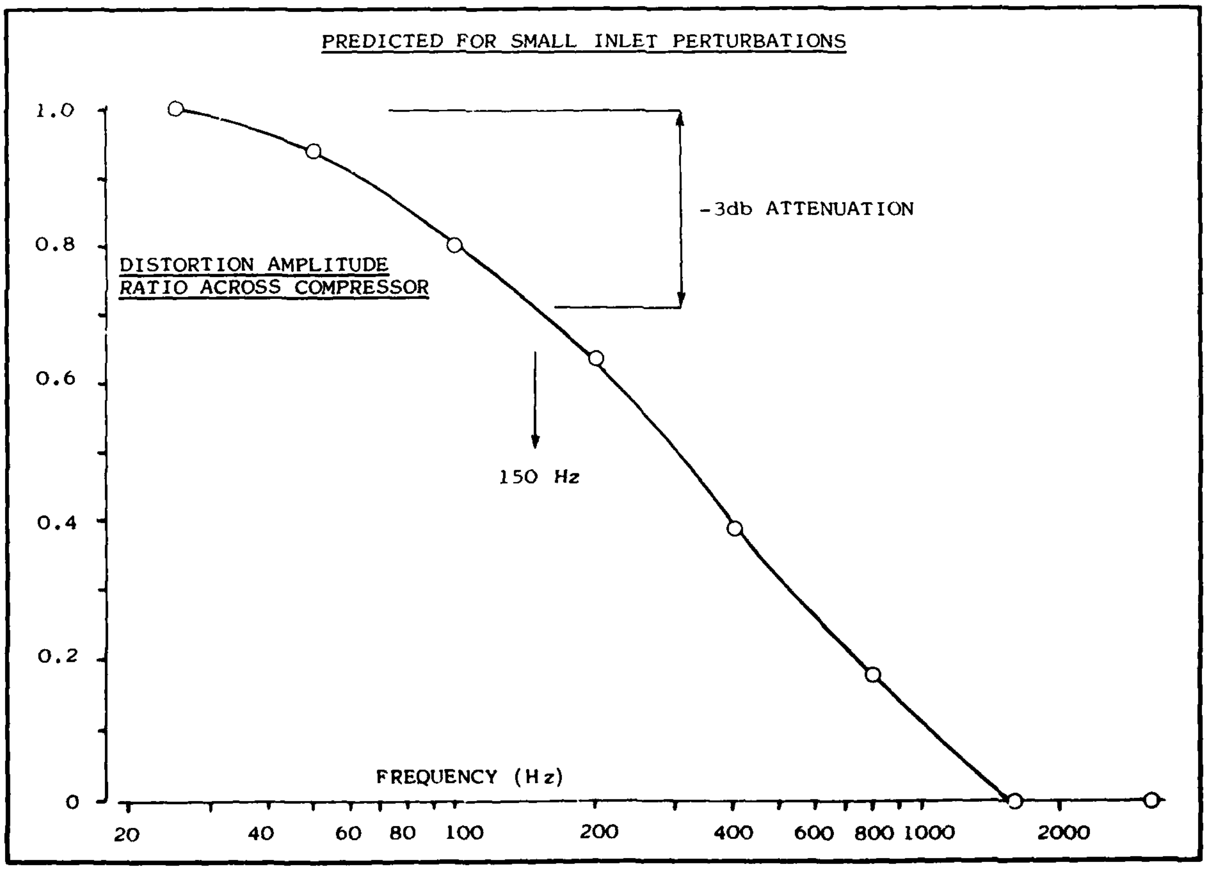


FIGURE 15
LOSS OF SURGE PRESSURE RATIO DUE TO INLET TEMPERATURE DISTORTION

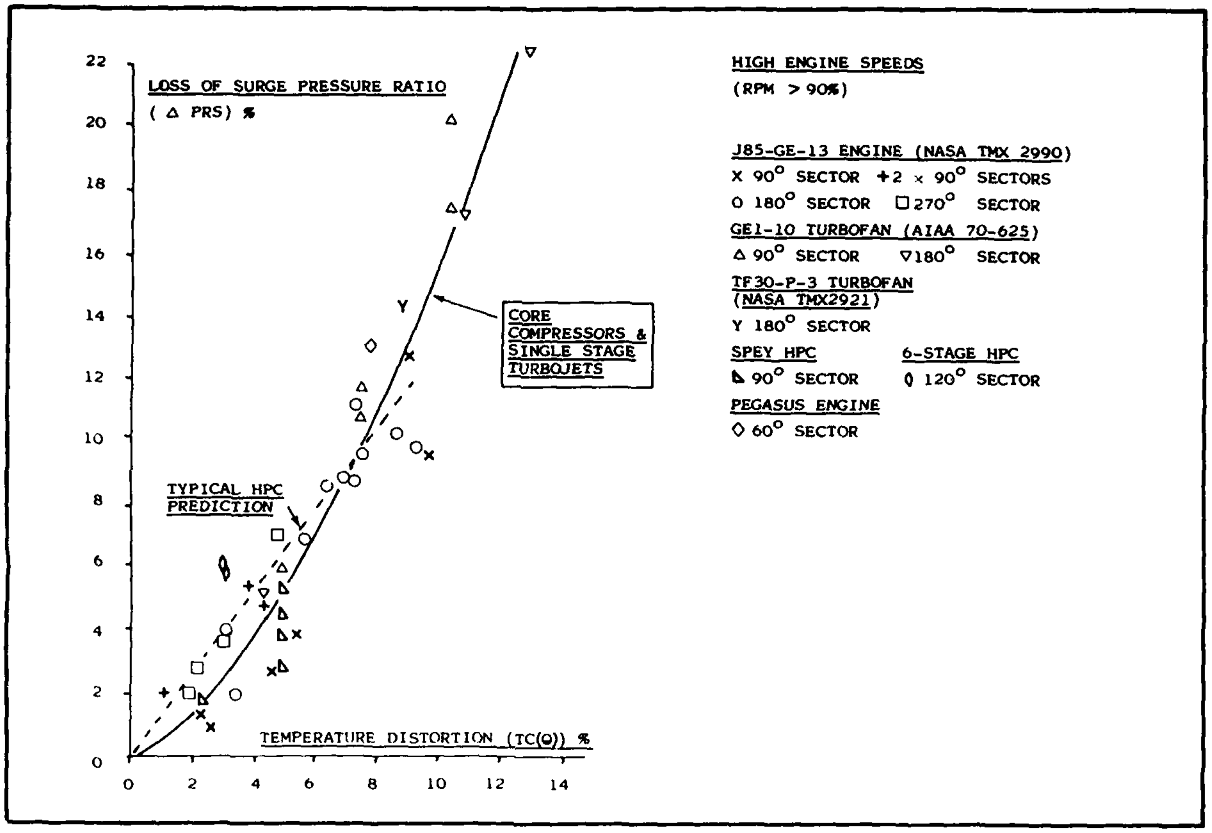


FIGURE 16
TURBOFAN ENGINE - MEAN INLET TEMPERATURE RISE RATE FOR SURGE
(FROM NASA TP1031 & TM X-2921)

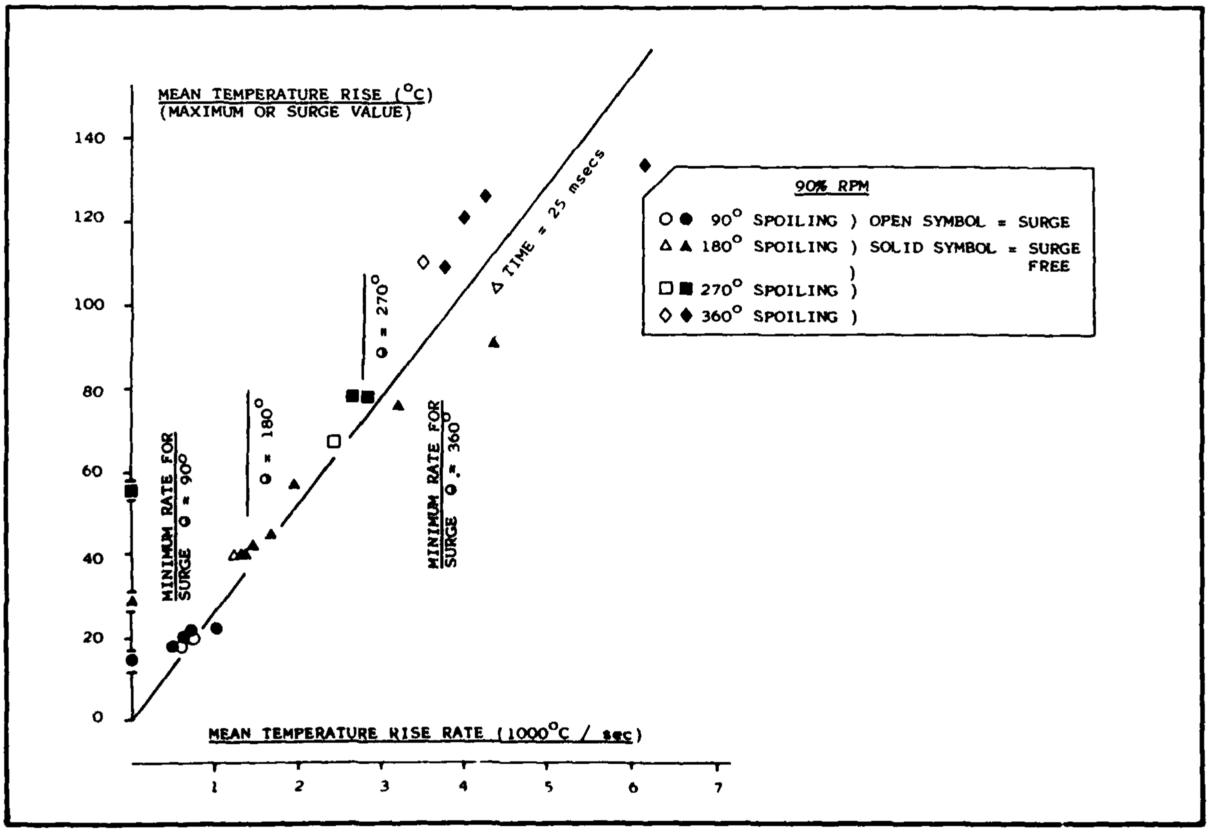


FIGURE 17
TURBOFAN ENGINE - TIME-VARIANT TEMPERATURE FOR SURGE

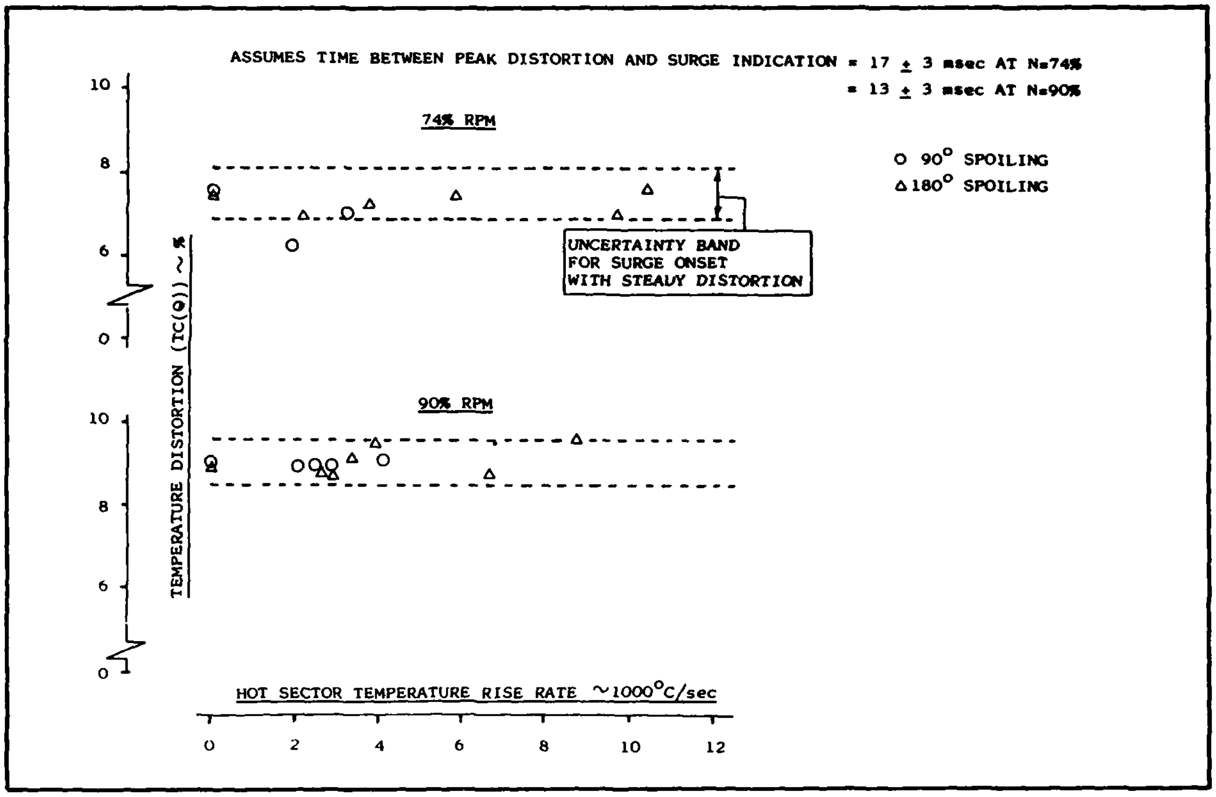
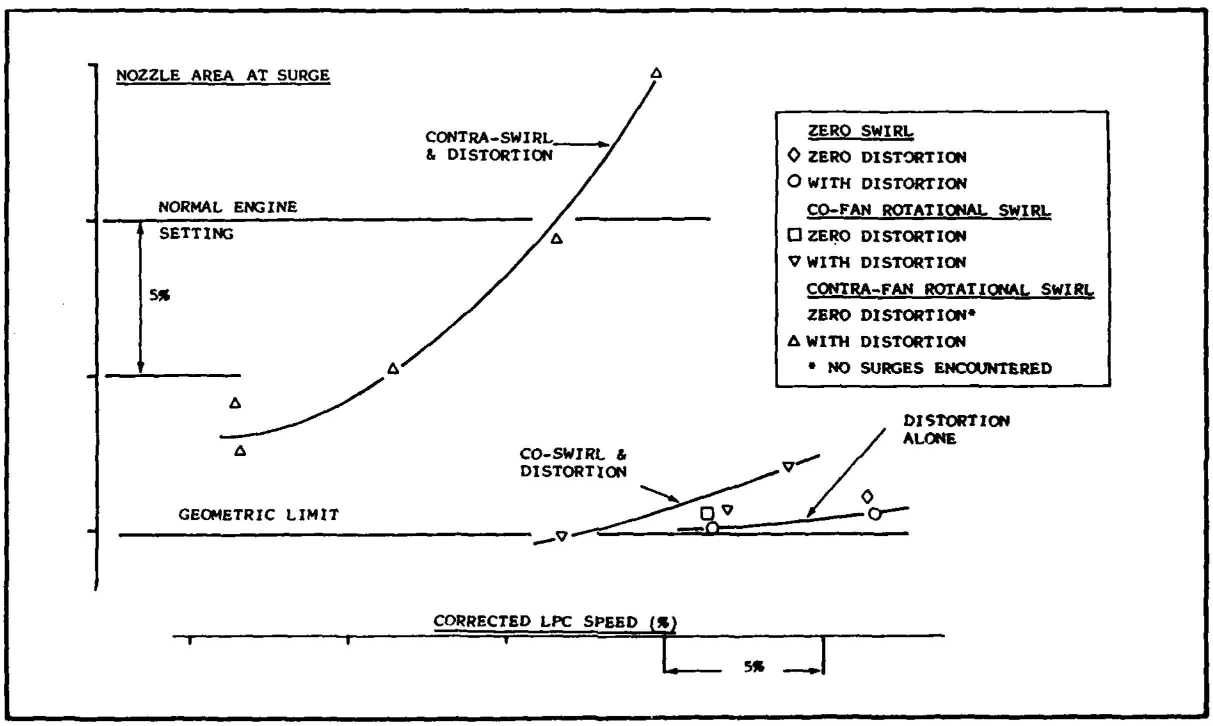


FIGURE 18
EFFECT OF INLET SWIRL & TOTAL PRESSURE DISTORTION ON ENGINE STABILITY



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DISCUSSION

J. Fabri, Fr.

Don't you think that relating the inlet pressure distortion level to the mean dynamic head would be more representative than to relate it to the mean inlet total pressure?

Author's Reply

Yes. In our DC (θ) parameter the distortion is relative to dynamic head. However, if we are studying distortion transfer across a compressor we do not use this form since changes in total pressure distortion can, to some extent, be masked by differences between the dynamic head at inlet and outlet.

W. Steenken, US

What is the corrected speed range of the data shown in Figure 5?

Author's Reply

The range of the data covers the flow required to cover the normal flight operating range of the compressor – say from 50 to 60% of design flow up to 105%.

de Richemont, Fr.

D'après votre expérience, quel est, ou quels sont, le (ou les) critère de distorsion à l'entrée d'un turbofan qui est le mieux corrélé à le perte de marge au pompage de ce turbofan?

Author's Reply

In our experience the most important influence of loss of surge margin of turbofan engines in STOL military aircraft is time-variant total pressure distortion with the circumferential form having a much greater effect than the forms of radial distortion encountered.

M. Berthier, Fr.

Avez-vous étudié un indice de distorsion intégrant le "swirl"?

Author's Reply

No, we do not have a swirl index which can be used to quantify loss of surge margin. As an aid to the intake designer we provide a limit of mean swirl angle.

S. Baghdadi, US

What, if any, frequency effects are incorporated in your parallel compressor model?

Author's Reply

We do need to account for the compressor dynamic response. This is achieved by the use of the critical sector angle.

BANC D'ESSAI COMPRESSEUR DESTINE A LA VERIFICATION DES MODELES DE REPONSE A LA DISTORSION

par

Jacques HUARD

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92320 CHATILLON (FRANCE)

Un banc d'essai d'étudeset de recherches fondamentales sur la réponse d'un compresseur subsonique mono-étage à une distorsion stationnaire ou instationnaire a été réalisé à l'ONERA suivant les spécifications suivantes :

- si l'on n'impose pas une distorsion, l'écoulement d'entrée doit être aussi uniforme que possible ;
- une longueur de veine suffisante doit être disponible à l'entrée du compresseur pour assurer l'établissement de la distorsion ;
- les essais et modifications doivent être aisés pour permettre l'expérimentation de configurations variées ;
- l'instrument de mesure doit permettre l'analyse tridimensionnelle fine de l'écoulement.

Le banc réalisé dans ces conditions est décrit et des exemples de résultats obtenus avec une distorsion stationnaire et une distorsion instationnaire sont présentés.

TEST FACILITY FOR BASIC RESEARCH ON DISTORSION

An experimental test facility designed for basic research on the transmission of an inlet distorsion through a single stage axial flow compressor has been realized.

The following conditions had to be met :

- with no distorsion imposed at the inlet, the flow has to be as uniform as possible ;
- the inlet channel must be as long as possible in order to insure a well defined distorsion pattern when a distorsion screen is used ;
- any change in compressor configuration must be easy ;
- the test instrumentation must give a detailed analysis of the flow in three dimensions.

The test facility that was realized with these requirements is described. Results obtained with a stationary and a non-stationary distorsion are presented.

1 - INTRODUCTION

Les avions de combat modernes sont amenés à évoluer dans un très large domaine de vitesse, altitude, incidence et dérapage, de telle sorte que l'alimentation des moteurs peut présenter des hétérogénéités très importantes avec pour résultat éventuel fâcheux le décrochage du compresseur, et le soufflage de la chambre de combustion. Ces problèmes sont loin d'être nouveaux, toutefois l'accroissement de la puissance des moteurs, et par là même du niveau de vitesse, ainsi que la recherche de performances de plus en plus élevées rendent caduques les solutions apportées jusqu'ici et nécessitent des analyses de plus en plus fines des phénomènes.

L'hétérogénéité de l'écoulement se traduit généralement par un déplacement de la ligne de pompage du compresseur par rapport au cas non perturbé. Cette zone du champ de fonctionnement se caractérise par des mécanismes fortement non linéaires que seules des analyses numériques puissantes sont en mesure de cerner sur le plan théorique. De telles études sont conduites dans divers laboratoires et notamment à l'ONERA [1]. Toutefois, il est évident que la prise en compte de tous les aspects du problème théorique dans le compresseur ne peut guère s'envisager qu'à très longue échéance ne serait-ce que par les limitations imposées par les ordinateurs. Même en se limitant au cas de la distorsion stationnaire, le calcul à la traversée des grilles ne peut faire usage de la condition de périodicité comme en alimentation uniforme, de telle sorte que l'écoulement doit être déterminé dans chaque canal du compresseur. Devant l'impossibilité d'un tel calcul, la solution classique et retenue à l'ONERA consiste à modéliser les grilles par des actuateurs à la traversée desquels les caractéristiques de l'écoulement subissent des discontinuités pouvant d'ailleurs être étalées dans l'espace. Les grandeurs de part et d'autre de la discontinuité sont alors reliées par des "lois de transfert" dérivées des équations générales de la Mécanique des Fluides.

Ce type de modélisation présente toutefois l'inconvénient de masquer les phénomènes interaubes complexes liés à l'existence de couches limites, de gradients de pression, de transfert d'énergie, etc... dans un volume fini soumis à des fluctuations instationnaires. Ces mécanismes non accessibles par le calcul dans leur généralité doivent être cependant nécessairement pris en compte dans les lois de transfert. Ils ne peuvent être appréhendés que par l'intermédiaire d'une analyse expérimentale détaillée de l'écoulement faisant appel à des techniques de mesure sophistiquées qui permettent d'accéder au caractère instationnaire des phénomènes. Ce type d'expérimentation fournit alors les informations recherchées. Le banc CERF de la Direction de l'Energétique (Compresseur d'Etudes et de Recherches Fondamentales) se prete bien à ce type d'essai, moyennant quelques aménagements. Après concertation entre les services officiels et les constructeurs de moteurs, les caractéristiques générales d'un banc compresseur destiné à la vérification des modèles de réponse à la distorsion ont été définies.

2 - CARACTERISTIQUES GENERALES DU BANC D'ESSAIS COMPRESSEUR

La photographie de la figure 1 montre l'allure générale du dispositif. L'organisation de celui-ci est schématisée sur la figure 2 tandis que certains détails sont précisés sur les vues photographiques de la figure 3.

Le banc se compose d'une cellule occupée par le dispositif d'essai et d'une salle de mesures. Une porte insonorisée fait communiquer les deux locaux et une baie vitrée permet l'accès visuel à partir de la salle de mesures.

L'air est aspiré à l'extérieur de la cellule à travers un filtre à poussières, passe dans le convergent qui délivre un écoulement uniforme en pression d'arrêt, puis traverse le compresseur. Le refoulement est effectué à l'extérieur de la cellule afin d'éliminer le recyclage de l'air. Le moteur d'entraînement de 30 kW à vitesse variable et régime nominal 3000 tr/min est également situé à l'extérieur. Un panneau latéral mobile permet d'introduire les appareils volumineux dans la cellule (filtre d'aspiration, etc...). En outre, le convergent est porté par un chariot monté sur rail permettant un accès aisé au palier avant et à la roue mobile.

L'arbre du rotor repose sur deux paliers à roulements situés, l'un à l'intérieur de la tuyère à l'amont du rotor et porté par trois bras profilés, l'autre à l'aval de la tuyère et fixé sur le support du moteur. Ces paliers sont lubrifiés par brouillard d'huile par l'intermédiaire de trous pratiqués dans deux des trois bras support en ce qui concerne le palier avant. Le troisième bras comporte également une perforation permettant l'acheminement des mesures brutes effectuées sur la roue mobile.

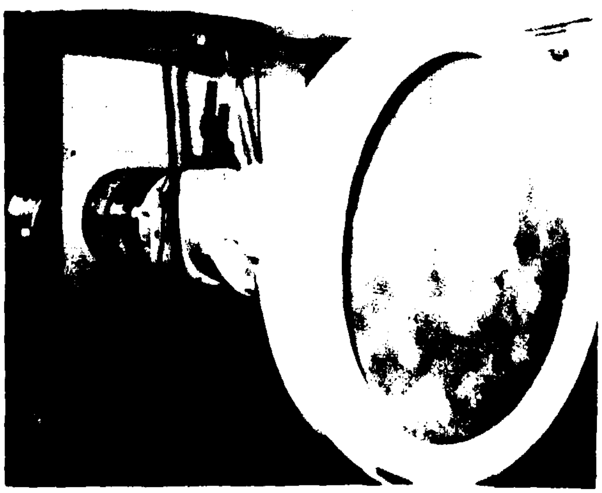


Fig. 1 – Vue générale du banc.

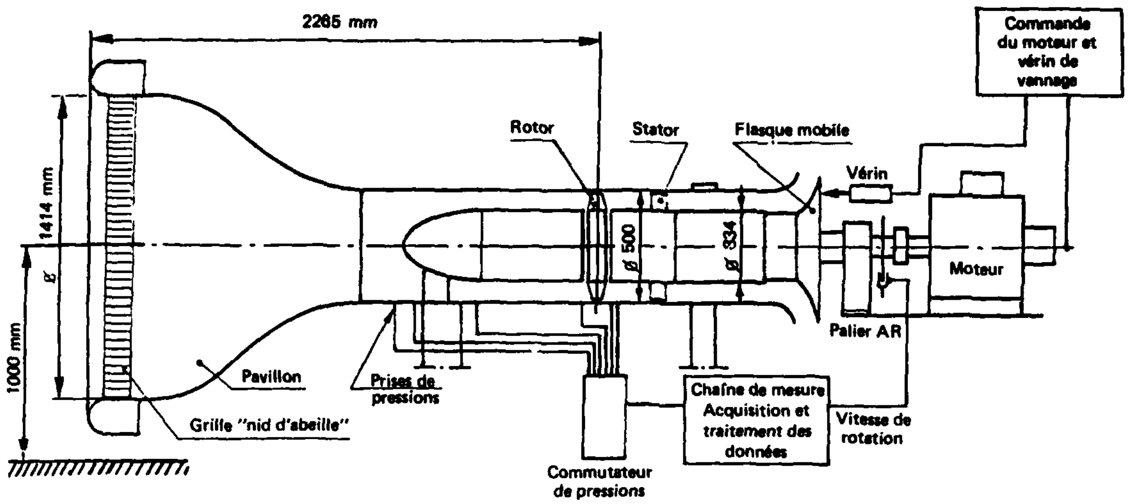


Fig. 2 – Schéma de l'installation.

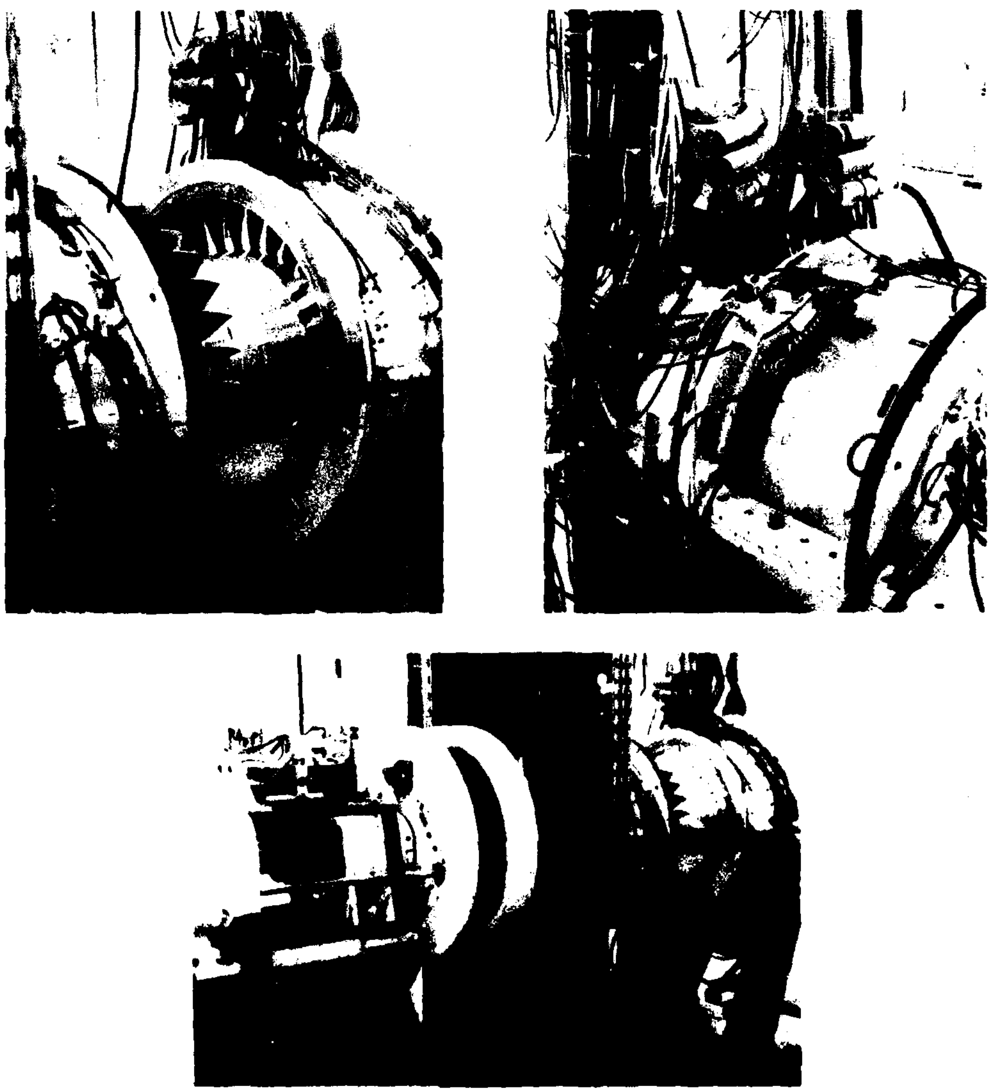


Fig. 3 – Détails du dispositif expérimental.

La roue est située dans une veine cylindrique de diamètre extérieur 500 mm et intérieur 334 mm. Le réglage du débit est effectué à l'aide d'un pavillon commandé à distance par l'intermédiaire d'électro-vannes . Le carénage interne à l'amont est constitué d'un nez creux abritant l'extrêmité libre de l'arbre qui peut recevoir un appendice tel un collecteur de mesures à mercure, un commutateur de pression ou tout dispositif tournant destiné, soit à la transmission de mesures sur la roue mobile, soit à l'adducteur de gaz ou liquide pour des expériences de visualisation.

Le Lanc dispose d'une importante chaîne d'acquisition et de traitement des données.

Outre l'installation d'un dispositif de vannage à commande pneumatique, deux réglettes porte-sondes motorisées en azimut ont été implantées sur le carter externe. Ces aménagements permettent d'accélérer le rythme d'acquisition des données.

3 - AMENAGEMENT DU BANC CERF DANS LE CADRE DES ETUDES D'INTERACTION ENTREE D'AIR-COMPRESSEUR

La réflexion ainsi que la concertation avec les différents participants aux études d'interactions entrée d'air-compresseur effectuées sous l'égide de la DRET, ont conduit à envisager l'aménagement de la veine du banc CERF d'une part et la redéfinition d'un étage de compression d'autre part.

3.1 - Justifications techniques

3.1.1 - Modifications de la veine

Compte tenu des difficultés de simulation,au banc,de distorsions de type instationnaire d'une part, et du caractère fondamental de l'étude d'autre part, l'écoulement sera perturbé dans un premier temps de manière classique à l'aide d'un écran à perméabilité variable placé à l'amont du rotor. Il reste toutefois que cette limitation n'a pas le moindre caractère définitif et que la prise en compte de distorsions réelles sera effectuée dès que les travaux de simulation de distorsion instationnaire engagés dans divers laboratoires et en particulier à l'ONERA seront suffisamment avancés.

a) Il est nécessaire de placer l'écran de distorsion dans la partie annulaire cylindrique de la veine, et non à l'amont du nez du compresseur, de manière à éviter le déplacement du point d'arrêt sur le nez et l'induction d'effets tridimensionnels indésirables dans le type de recherche engagé ici ;

b) il est nécessaire de placer l'écran de distorsion suffisamment loin à l'amont du rotor de manière à permettre la réorganisation naturelle de l'écoulement et la naissance d'écoulements circonférenciels entre l'écran et le rotor. Cette distance ℓ_e caractéristique est à relier directement à la présence du compresseur et à son effet de blocage de l'écoulement.

3.1.2 - Définition de l'étage de compression

L'étude théorique est orientée dans un premier temps vers le cas pseudo-bidimensionnel (évolution du rayon et de l'épaisseur de la nappe de courant) correspondant en bonne approximation à un compresseur HP. L'étude expérimentale des lois de transfert engagée sur le banc CERF doit être également réalisée sur une configuration d'écoulement bidimensionnel, les effets tridimensionnels doivent être limités à la traversée de l'étage de compression pour correspondre au cas théorique traité ; ceci impose en veine cylindrique la définition d'un étage de compression suivant une loi à circulation constante.

3.2 - Aménagements pratiques du banc

3.2.1 - Longueur de la veine amont

Compte tenu des impératifs dictés par les deux points du paragraphe 3.1.1, une veine annulaire de longueur 450 mm a été installée en amont du compresseur et l'écran de distorsion a été placé à l'entrée de celle-ci. Il est constitué d'un grillage rigide de support occupant toute la veine, sur lequel seront fixés des secteurs de grillage à perméabilité variable destinés à produire les pertes de charge. L'ensemble est monté sur un dispositif tournant permettant de déplacer la zone de perturbation sur toute la circonférence de manière à sonder l'écoulement sur 360° avec un instrument de mesure fixe.

La partie du carter interne située à l'aval de la roue mobile est constituée de modules interchangeables permettant de disposer le redresseur selon trois positions axiales, position "avancée" à environ 30 mm du bord de fuite des aubes mobiles (distance minimale pour l'introduction de sondes), position "intermédiaire" à environ 200 mm et position "reculée" très à l'aval du montage, soit environ à 470 mm, qui conduit en pratique à une configuration de rotor isolé.

3.2.2 - Définition de l'étage à circulation constante

Les caractéristiques globales de l'étage sont les suivantes :

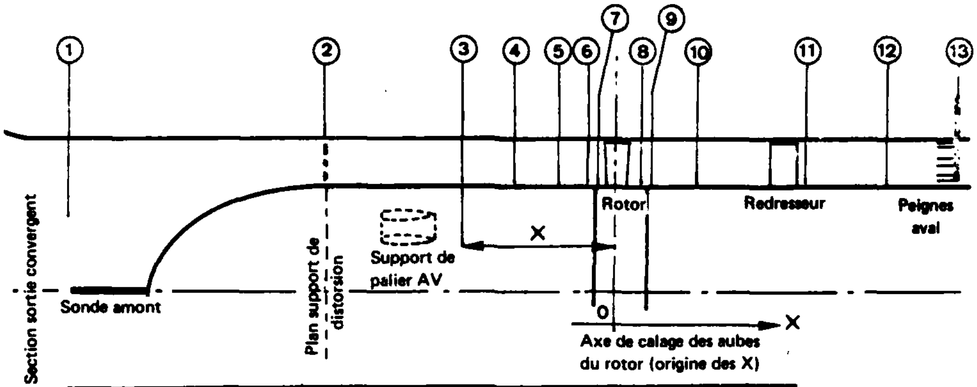
- accroissement moyen de pression : $\frac{2\Delta p_i}{\rho U^2} = 0,62$
- débit : Q = 5,97 kg/s
- rendement : η = 86%
- puissance absorbée : P = 11,8 kW (moteur 30 kW)
- vitesse de rotation : N = 3000 tr/mn.

La triangulation adoptée doit conduire à des ralentissements représentatifs d'une configuration réaliste sans prétendre toutefois représenter un compresseur chargé.

4 - IMPLANTATION DES MESURES

Etant donné le nombre considérable de mesures à effectuer dans les études de distorsion à venir, lié au caractère instationnaire des phénomènes à appréhender, un effort particulier a été consenti pour l'implantation et l'automatisation des dispositifs d'acquisition de mesures.

D'une manière générale, les mesures sont implantées selon 13 plans répartis le long de la machine, numérotés de 1 à 13 et repérés par leur abscisse axiale x à partir de l'axe de calage du rotor. La figure 4 permet de préciser leur disposition par rapport à l'écran de distorsion, au rotor et au redresseur. Le plan 1 de référence est situé à 80 mm à l'aval du plan de sortie du convergent et à 140 mm à l'amont du nez du compresseur.



AMONT	X/R	θ	AVAL	X/R	θ
Plan de sondage 1	-3,63	30.I50.270	Plan de sondage 8	+0,I5	75àI05
" " 2	-I,88		" " 9	+0,I9	" "
" " 3	-0,93	90	" " I0	+0,49	" "
" " 4	-0,66	90 - I20	" " II	+I,23	" "
" " 5	-0,38	70	" " I2	+I,74	" "
" " 6	-0,I4	75 à I05	" " I3	+2,I8	50.II0.I70
" " 7	-0,I0	" "	Peignes aval		230.290.350

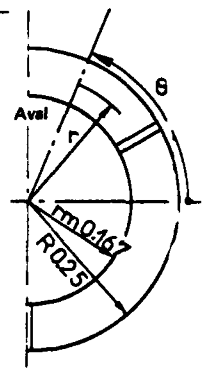


Fig. 4 – Implantation des plans de sondages.

Ces plans de mesure peuvent être classés selon quatre types en fonction de l'objectif :

- plan 1 infini amont = contrôle du débit,
- plan 2 = position de l'écran de distorsion,
- plans 3, 4, 5 et 6 = analyse de l'influence du compresseur sur l'écoulement incident,
- plans 6, 7, 8 et 9 = analyse de la réponse des grilles aux distorsions,
- plans 10, 11, 12 et 13 = analyse de l'écoulement à l'aval ; recherche des conditions infini aval pouvant être introduites dans le calcul théorique.

La chaîne d'acquisition et de traitement des données permet la scrutation de 200 voies de mesures.

Par ses caractéristiques générales, ce dispositif d'essai se trouve particulièrement bien adapté aux études à caractère fondamental entreprises à l'ONERA sur le thème des interactions entrée d'air-moteur. Des aménagements importants tenant à la nature des travaux envisagés ont dû être effectués sur la veine et sur l'étage de compression.

5 - CARACTERISTIQUES AERODYNAMIQUES DE L'ECOULEMENT EN ALIMENTATION HOMOGENE

Le nouvel étage de compression n'étant pas disponible, les essais de réception du banc ont été effectués à l'aide d'un rotor disponible de 20 pales de profils de la série C4. La loi de circulation adoptée est intermédiaire entre la circulation constante ($rV = C^{te}$) et le "solid body" ($V = Kr$). Nous disposons de deux positions de calage des aubes du rotor. Deux redresseurs à profil circulaire réalisés en tôle mince et adaptés pour chaque calage ramènent le fluide dans la direction axiale. A l'amont du rotor différentes mesures ont été effectués :

5.1 - Sortie du convergent d'entrée

- Sondage radial et azimutal à la sortie du convergent d'entrée à l'aide d'un peigne de pression tournant monté sur le nez de la machine. Ce peigne constitué de onze tubes pitot permet de mesurer la pression d'arrêt $p_i(r)$ ainsi que la pression statique $p(r)$ de l'écoulement.

La répartition des vitesses à l'entrée de la veine cylindrique peut être ainsi obtenuegrâce à la relation de Bernouilli en supposant le fluide incompressible, l'écoulement axial et permanent.

$$W(r,\theta) = \sqrt{\frac{2}{\rho}\left[p_i(r,\theta) - p(r,\theta)\right]} \tag{1}$$

La figure 5 représente pour les deux débits extrêmes obtenus dans la configuration de calage 1, les évolutions azimutales de la vitesse débitante sur trois rayons différents.

Cette courbe montre que l'écoulement est sensiblement axisymétrique à la sortie du convergent avec des fluctuations de vitesse inférieure au pour cent. Ces évolutions azimutales sont ensuite intégrées numériquement pour obtenir la valeur moyenne à chaque rayon :

$$\overline{W}(r) = \frac{1}{2\pi}\int_0^{2\pi} W(r,\theta)\, d\theta \tag{2}$$

L'évolution radiale de cette vitesse débitante moyenne est présentée sur la figure 6 pour la configuration de calage 1 et pour différentes positions de la vanne au refoulement.

Le déficit de vitesse qui apparaît sur l'axe est dû à la présence du nez du compresseur. Le débit qui traverse le compresseur est directement déduit de ces évolutions radiales.

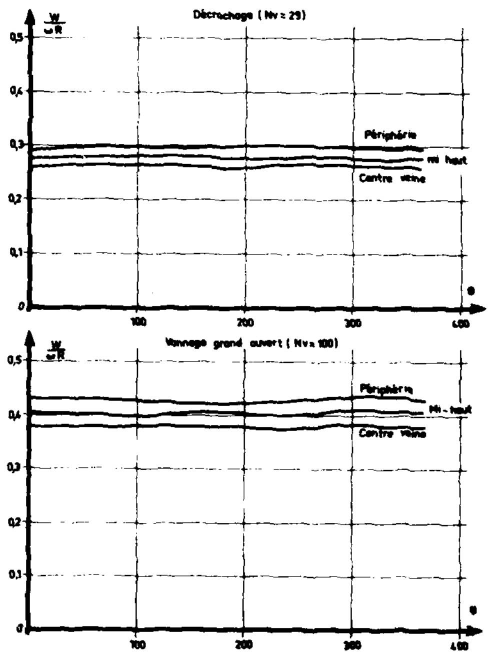


Fig. 5 – Répartition azimutale de la vitesse débitante à la sortie du convergent (calage 1).

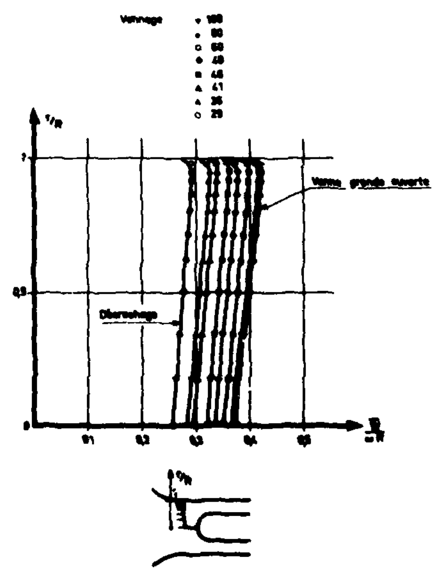


Fig. 6 – Evolutions radiales de la vitesse débitante moyenne à la sortie du convergent pour différents vannages (calage 1).

- Sondage radial effectué dans le plan de sondage 5 $X/R = -0{,}38$ à l'aide d'une sonde directionnelle du type NACA, permettant d'accéder aux caractéristiques locales de l'écoulement $p_i(r)$, $p(r)$, $\alpha(r)$ desquelles est déduite la répartition de vitesse débitante

$$W(r) = \sqrt{\frac{2}{\rho}\left[p_i(r) - p(r)\right]} \cdot \cos\alpha(r) \tag{3}$$

dont les évolutions sont présentées sur la figure 7. Cette courbe fait apparaître que la vitesse débitante en amont du rotor est sensiblement constante en dehors des couches limites pariétales.

5.2 - En aval du rotor

Les conditions aérodynamiques de l'écoulement en aval du rotor ont été obtenues dans le plan de sondage 10 $(X/R = +0{,}49)$ par un sondage radial effectué à l'aide d'une sonde directionnelle du type NACA. Parmi les caractéristiques de l'écoulement nous avons choisi à titre d'exemple l'évolution radiale de la vitesse débitante obtenue par la relation (3). La figure 8 qui présente ces évolutions fait apparaître une forte progression des effets secondaires qui affectent les coupes de têtes.

Néanmoins lorsque l'on étudie les évolutions radiales de la fonction de courant :

$$\Psi(r) = \int_{r_m}^{r} r\, W(r)\, dr \tag{4}$$

en amont et en aval du rotor, celles-ci ne font pas apparaître de déplacement significatif des lignes de courant (figure 9) sauf pour les coupes de têtes juste avant le décrochage (N_V = 38). L'écoulement reste sensiblement bidimensionnel à mi-hauteur de veine sur toute la plage de fonctionnement du rotor.

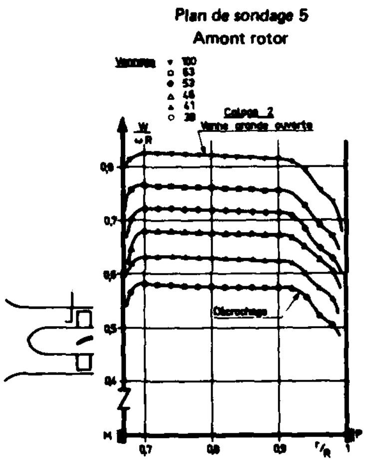


Fig. 7 – Evolution radiale de la vitesse débitante à l'amont du rotor.

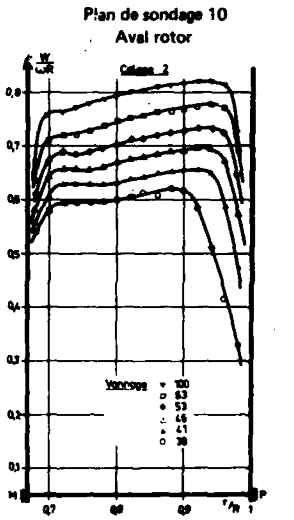


Fig. 8 – Evolution radiale de la vitesse débitante en aval du rotor pour différents vannages.

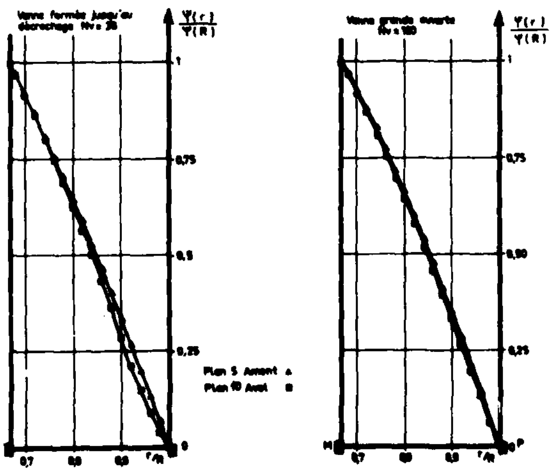


Fig. 9 – Comparaison des répartition radiales des fonctions de courant amont et aval rotor (calage 2).

6 - REPONSE D'UN COMPRESSEUR AXIAL SUBSONIQUE A UN ECOULEMENT INCIDENT HETEROGENE INSTATIONNAIRE PUIS STATIONNAIRE DE MEMES EFFETS MOYENS

L'étage de compresseur utilisé pour les essais de qualification du banc présente une bonne axisymétrie en écoulement homogène. Il était donc possible sans attendre de disposer de l'outil complet (nouvel étage à circulation constante) d'étudier la réponse du compresseur en alimentation hétérogène.

La réponse du compresseur aux distorsions étudiée au banc est habituellement obtenue en perturbant l'écoulement amont à l'aide de grillages. Toutefois, dans certaines configurations de vol, les perturbations sont de nature fortement instationnaire. Il est donc apparu intéressant de tenter de reproduire en valeurs stationnaires moyennes, une configuration instationnaire à l'aide de grillages et d'étudier de manière comparative l'effet sur le compresseur de ces deux types de distorsion afin de mettre en évidence l'influence de l'instationnarité sur le comportement de la machine.

Une lunule (figure 10) obstruant partiellement la veine en amont du compresseur (plan de sortie du convergent) permet de créer un écoulement instationnaire (figure 11) dont on étudie la propagation dans le compresseur. Un écoulement moyen équivalent est reproduit à l'entrée du compresseur à l'aide d'un écran à porosité variable (figure 12) délivrant une distorsion de type stationnaire.

Une comparaison détaillée des performances de l'étage est effectuée dans les deux configurations.

Fig. 10 – Lunule.

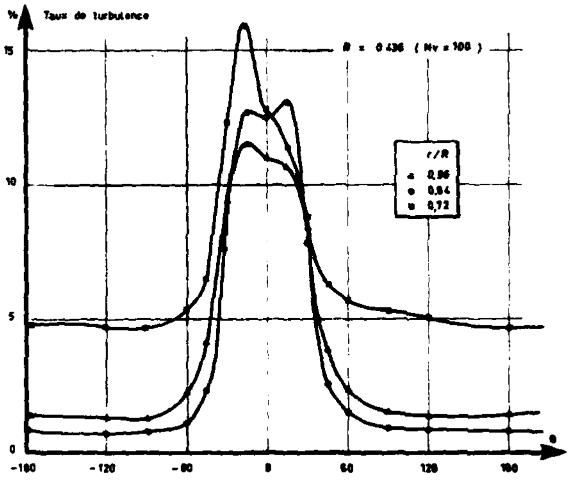


Fig. 11 – Répartition azimutale du taux de turbulence en amont du rotor pour la vanne au refoulement grande ouverte.

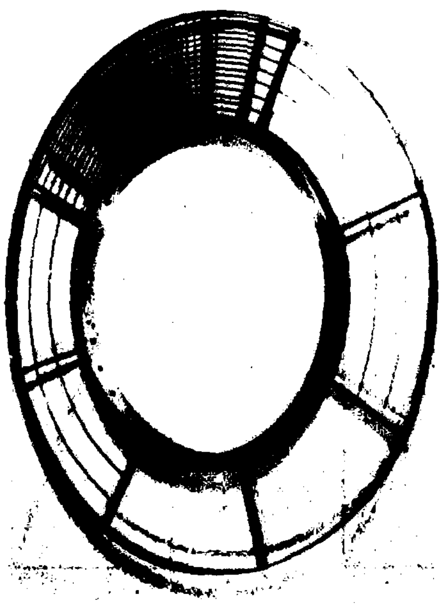


Fig. 12 - Ecran de distorsion type grillage

6.1 - Caractéristiques aérodynamiques de l'écoulement obtenues en amont du rotor avec lunule et grillage

La figure 13 montre les évolutions azimutales obtenues avec lunule et grillage de la pression d'arrêt, pression statique et direction de l'écoulement à mi-hauteur de veine lorsque la vanne au refoulement est grande ouverte (N_V = 100). Cette figure fait apparaître une dissymétrie sur la répartition de pression d'arrêt, dans le cas des essais avec grillage. Ceci peut être attribué en toute vraisemblance à un léger défaut de positionnement de l'ossature grillage ; il est probable que la sonde devait être située lors de cet essai juste derrière un clinquant support. Les résultats présentés ultérieurement montrent que ce phénomène ne se retrouve pas aux autres débits. Par contre la répartition obtenue avec lunule est parfaitement symétrique.

L'évolution azimutale de la pression statique met en évidence la naissance d'écoulements circonférenciels ainsi qu'une légère dissymétrie de l'écoulement confirmée par la direction de l'écoulement.

Le caractère instationnaire de l'écoulement issu de la lunule engendre des gradients azimutaux de pressions statiques plus importants que ceux observés avec la distorsion type grillage ; ceci se manifeste également au niveau de la direction absolue de l'écoulement avec des variations d'incidence sur rotor plus importantes avec la lunule qu'avec le grillage. Celles-ci sont toutefois modérées à ce débit (les angles sont comptés négativement dans le sens de rotation du rotor).

Les légères différences observées se compensent pour donner une répartition identique de vitesse débitante au rayon moyen (figure 14).

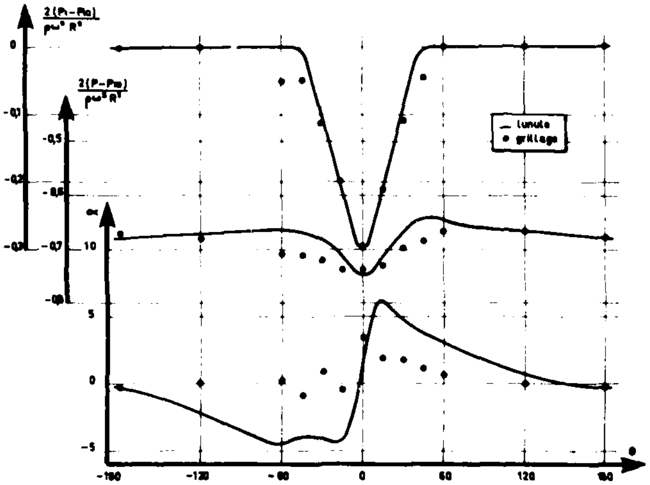


Fig. 13 – Comparaison des évolutions azimutales à mi-hauteur de veine de la pression d'arrêt, statique et de la direction absolue de l'écoulement en amont du rotor (vanne grande ouverte Nv = 100).

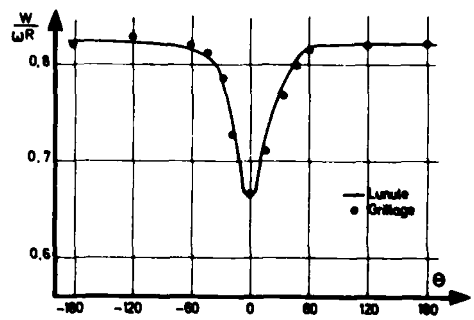


Fig. 14 – Evolution azimutale de la vitesse débitante en amont du rotor (vanne grande ouverte Nv = 100) à mi-hauteur de veine.

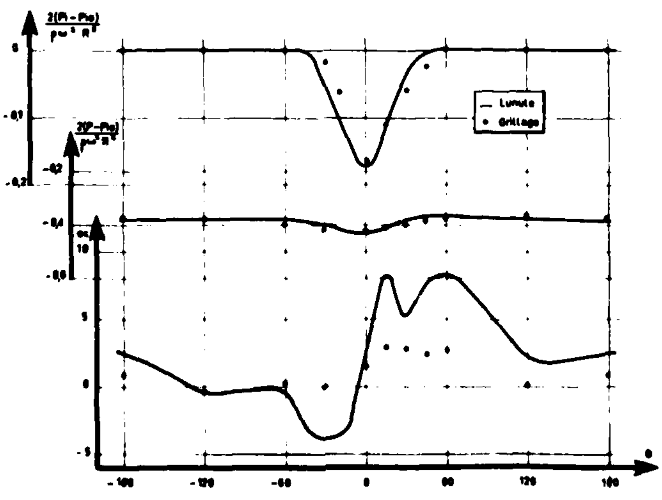


Fig. 15 – Comparaison des évolutions azimutales à mi-hauteur de veine de la pression d'arrêt, statique et de la direction absolue de l'écoulement en amont du rotor (Nv = 39).

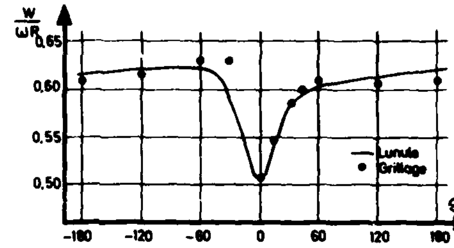


Fig. 16 – Evolution azimutale de la vitesse débitante en amont du rotor à la limite du décrochage (Nv = 39) à mi-hauteur de la veine.

Les sondages effectués juste avant le décrochage du compresseur (N_v = 39) ne font pas apparaître de phénomènes nouveaux au niveau des caractéristiques aérodynamiques de l'écoulement (figure 15).

Toutefois lorsque l'on combine les relevés de pression d'arrêt et les relevés de pression statique pour calculer la vitesse (figure 16) on relève une dissymétrie de l'écoulement et l'apparition d'un gradient azimutal supplémentaire de vitesse débitante.

Pour conclure ces remarques sur les résultats obtenus en amont du rotor, il importe de noter qu'un excellent recoupement existe au niveau des valeurs moyennes à mi-hauteur de veine dans les deux configurations. On notera également qu'entre les deux configurations existent des écarts sur la direction absolue de la vitesse pouvant aller jusqu'à cinq degrés au décrochage (N_v = 39).

6.2 - Caractéristiques aérodynamiques de l'écoulement obtenues en aval du rotor avec lunule et grillage

Les répartitions azimutales des grandeurs aérodynamiques obtenues en aval du rotor à vanne grande ouverte pour les deux configurations sont comparées sur la figure 17 à mi-hauteur de veine.

La répartition de pression d'arrêt montre que la perturbation s'est nettement atténuée à la traversée de la roue mobile. Elle met également en évidence une bonne symétrie de l'écoulement en azimut pour la distorsion de type grillage, tandis qu'apparaît pour la lunule un gradient azimutal important dans la zone non perturbée de l'écoulement.

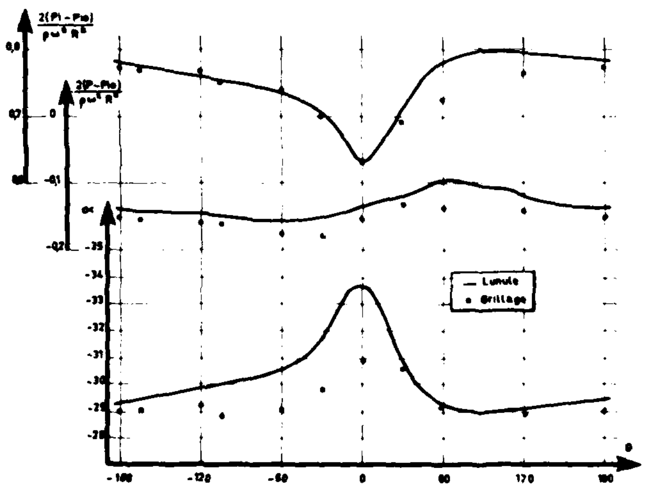
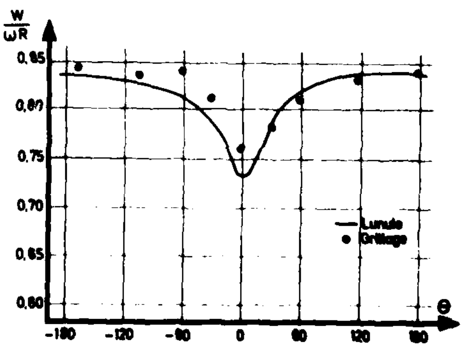


Fig. 17 – Comparaison des évolutions azimutales à mi-hauteur de veine de la pression d'arrêt, statique et de la direction absolue de l'écoulement en aval du rotor (vanne grande ouverte Nv = 100).

Ceci explique les écarts importants entre les deux configurations au niveau de l'accroissement de pression d'arrêt à la traversée du rotor dans la partie positive de l'azimut. Il est clair que pour la lunule la dissymétrie d'incidence impose une déflexion différente dans le rotor. La figure 18 montre l'évolution azimutale de la vitesse débitante et confirme les remarques précédentes. Les résultats obtenus juste avant le décrochage du compresseur sont présentés sur la figure 19. Ils n'appellent pas de commentaires supplémentaires.

Fig. 18 – Evolution azimutale de la vitesse débitante en aval du rotor (vanne grande ouverte Nv = 100) à mi-hauteur de veine.



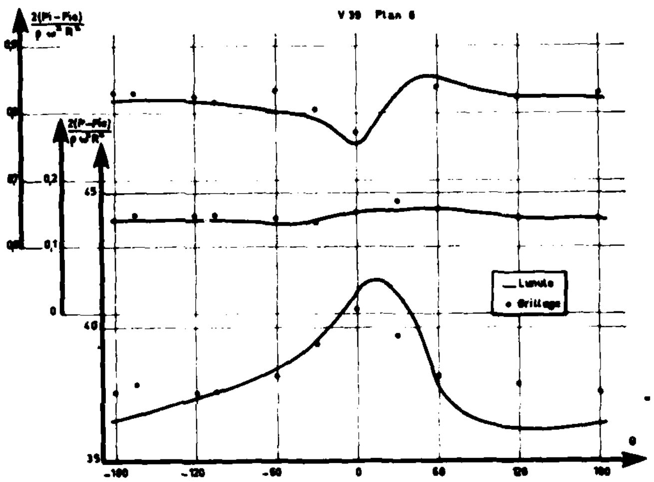


Fig. 19 – Comparaison des évolutions azimutales à mi-hauteur de veine de la pression d'arrêt, statique et de la direction absolue de l'écoulement en aval du rotor juste avant le décrochage (Nv = 39).

Sur la base des sondages effectués en aval du rotor, il apparaît que les deux configurations étudiées mettent en jeu des phénomènes physiques similaires. Les écarts azimutaux relevés à l'amont du rotor sur la direction absolue de l'écoulement perturbé avec la lunule conduisent cependant à des déflexions dans la roue, variables selon l'azimut, qui induisent un gradient azimutal d'accroissement d'enthalpie à la traversée de la roue. Il apparaît donc un gradient azimutal supplémentaire de pression d'arrêt en aval de la roue dans le cas de l'écoulement perturbé avec la lunule. De ceci découle un écart sensible avec les valeurs obtenues pour l'écoulement perturbé à l'aide du grillage dans la partie positive de l'azimut, dans le sens de rotation du rotor.

6.3 - Caractéristiques aérodynamiques de l'écoulement obtenues en aval du redresseur avec lunule et grillage

Dans le plan de sondage en aval du redresseur, la mesure des grandeurs aérodynamiques de l'écoulement a été obtenue grâce à 6 peignes de pressions disposés d'une manière régulière azimutalement. Trois peignes étaient équipés pour mesurer la pression d'arrêt (simples tubes de Pitot disposés radialement) les trois autres étant pourvus d'une prise de pression statique. Ainsi pour chaque position azimutale de la distorsion amont trois sondages radiaux de pression d'arrêt et de pression statique sont effectués instantanément pour 6 azimuts différents de sorte qu'après avoir effectué une rotation complète du générateur de distorsion le champ complet a été décrit et la mesure obtenue en chaque point se trouve être une moyenne de trois mesures effectuées pour des positions du peigne différente en azimut mais identique par rapport à la perturbation. Cette procédure a l'avantage de minimiser les éventuels défauts d'axisymétrie inhérents au banc d'essais. A vanne au refoulement grande ouverte, l'évolution azimutale de la pression d'arrêt (figure 20) montre que les écarts relevés directement à l'aval du rotor entre les deux configurations ont été atténués. Toutefois, le gradient azimutal de pression d'arrêt subsiste encore dans la partie non perturbée de l'écoulement avec lunule. Le gradient azimutal de pression statique est nul en aval de l'étage. C'est pour cette raison que l'évolution azimutale de la vitesse débitante n'apporte rien par rapport à la pression d'arrêt (figure 20) lorsque l'on ferme la vanne du refoulement jusqu'au décrochage. Les comparaisons effectuées ne font pas apparaître de différences fondamentales (figure 21).

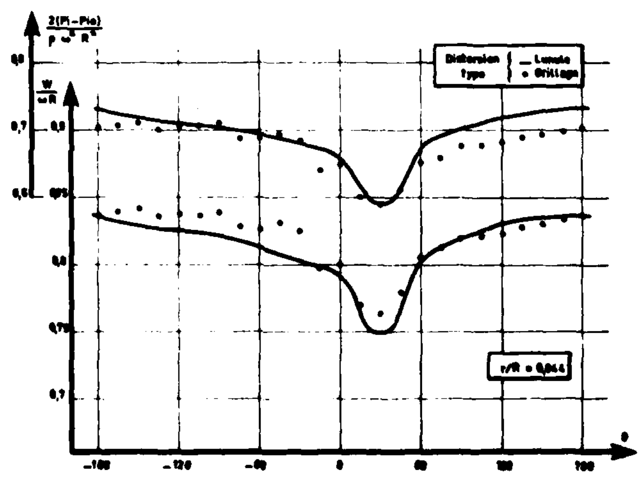


Fig. 20 – Comparaison des évolutions azimutales de la pression d'arrêt ainsi que de la vitesse débitante à mi-hauteur de veine en aval du redresseur (vanne grande ouverte Nv = 100).

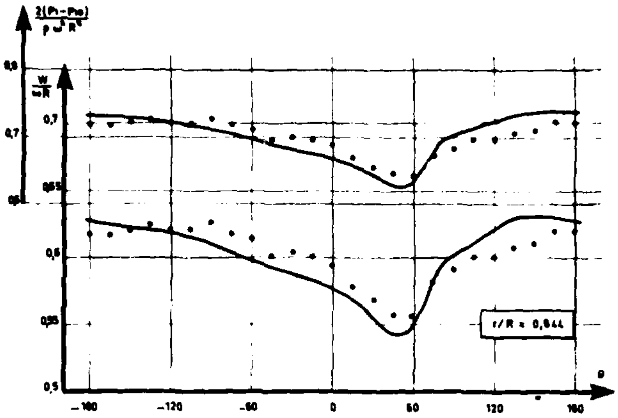


Fig. 21 – Comparaison des évolutions azimutales de la pression d'arrêt et de la vitesse débitante à mi-hauteur de veine en aval du redresseur (juste avant le décrochage Nv = 39).

6.4 - Comparaison des performances du compresseur muni d'un générateur de distorsion aux performances de base

Les deux générateurs de distorsion, la lunule et le grillage ayant donné des répartitions de vitesse, moyennées dans le temps, très semblables, il a fallu vérifier si leur effet sur les performances globales du compresseur était également semblable.

Dans le but de déterminer l'influence des deux types de distorsion expérimentés sur les performances globales du compresseur, il faut définir d'abord des valeurs moyennes. La caractéristique du compresseur sera représentée par le couple de valeur : coefficient d'accroissement de pression moyen de l'étage $\overline{\overline{\Psi_i}}$, coefficient de débit moyen ϕ ; pour définir ces deux paramètres, il faut commencer par calculer les valeurs moyennes à chaque rayon de la pression d'arrêt $\overline{P_i}(r)$ et de la vitesse débitante $\overline{W}(r)$. Ces valeurs moyennes sont déterminées par intégration numérique à partir des répartitions azimutales présentées. Dans chaque plan de sondage, le coefficient de débit moyen ϕ est calculé ainsi que la pression d'arrêt moyenne $\overline{\overline{P_i}}$ obtenue en pondérant par les sections les valeurs moyennes obtenues à chaque rayon ; le coefficient d'accroissement de pression moyen de l'étage est défini par :

$$\overline{\overline{\Psi_i}} = \frac{2\left(\overline{\overline{P_i}}_{aval\,redresseur} - \overline{\overline{P_i}}_{amont\,rotor}\right)}{\rho(\omega R)^2} \tag{8}$$

Le tableau I regroupe les coefficients de débit moyen ϕ, pour chaque type de distorsion étudié dans les différents plans de sondages obtenus par double intégration en θ et en r . Il apparaît qu'en amont du rotor, l'écoulement étant perturbé soit par la lunule, soit par le grillage, le débit qui traverse le compresseur est identique. En aval du rotor, le recoupement des mesures est bon, néanmoins quelques écarts très modérés peuvent être observés. En aval du redresseur les valeurs obtenues sont très proches de celles obtenues en amont de la roue mobile. La technique employée (peignes de pressions), avec le dépouillement développé pour cette campagne, se trouve pleinement justifiée et bien adaptée pour les essais entrepris en alimentation hétérogène.

Afin de montrer l'influence de la distorsion sur le fonctionnement du compresseur, l'évolution du coefficient de débit ϕ en fonction du vannage N_V a été portée sur la figure 22 qui met ainsi en évidence le déficit de débit en présence de distorsion.

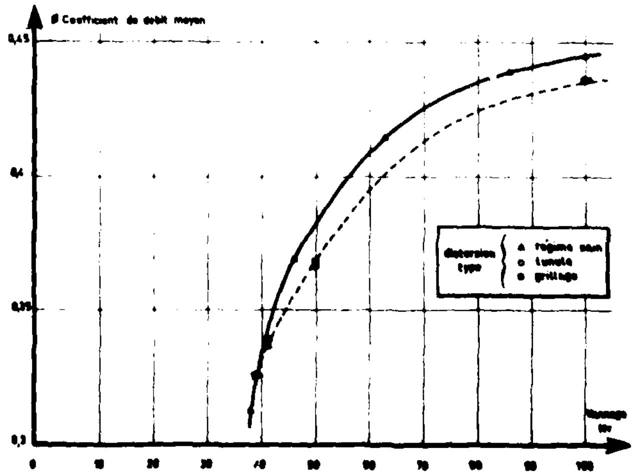


Fig. 22 – Influence d'une distorsion azimutale sur le coefficient de débit global du compresseur corf à vannage donné.

Repère de vannage \ Type de distorsion	Absence de distorsion	PLAN DE SONDAGE					
		AMONT ROTOR		AVAL ROTOR		AVAL REDRESSEUR	
		Lunule	Grillage	Lunule	Grillage	Lunule	Grillage
100	0,445	0,436	0,436	0,429	0,442	0,432	0,429
50	0,383	0,367	0,368	0,371	0,368	0,364	0,366
41	0,343	0,339	0,337	0,340	0,345	0,335	0,336
39	0,312	0,326	0,326	0,323	0,327	0,324	0,323

TABLEAU I

COEFFICIENT DE DEBIT MOYEN Ø DEDUIT DES MESURES

Chaque couple de valeurs $\overline{\overline{\Psi_i}}$, Ø obtenu en alimentation homogène et perturbée, représente un point de fonctionnement du compresseur pour une position donnée de la vanne au refoulement. Par fermeture de celle-ci, il est possible de déterminer les trois caractéristiques de l'étage, qui sont présentées sur la figure 23. La limite de décrochage en alimentation perturbée apparaît pour un débit supérieur de 4% au débit de décrochage en régime sain tant avec la distorsion stationnaire qu'instationnaire. En ce qui concerne l'accroissement de pression d'arrêt, on note qu'il se trouve fortement atténué dans la zone de fonctionnement proche du décrochage.

A la limite de décrochage, en écoulement perturbé, les performances de l'étage sont pratiquement identiques, alors que dans la zone de fonctionnement qui précède, les performances obtenues avec la lunule sont nettement moins bonnes. Ceci est certainement dû à l'interaction entre l'écoulement instationnaire délivré par la lunule et les effets secondaires déjà mentionnés au paragraphe 4. Pour les débits intermédiaires il apparaît une faible amélioration en régime perturbé par rapport au fonctionnement en régime homogène. A vanne au refoulement grande ouverte, le point de fonctionnement obtenu en alimentation hétérogène instationnaire se positionne sur la caractéristique saine alors que les performances obtenues en fonctionnement hétérogène stationnaire se trouvent nettement dégradées ; ces diverses observations n'ont pas encore trouvé une interprétation satisfaisante qui nécessite vraisemblablement un dépouillement plus fin au niveau du transfert des écoulements hétérogènes dans la roue.

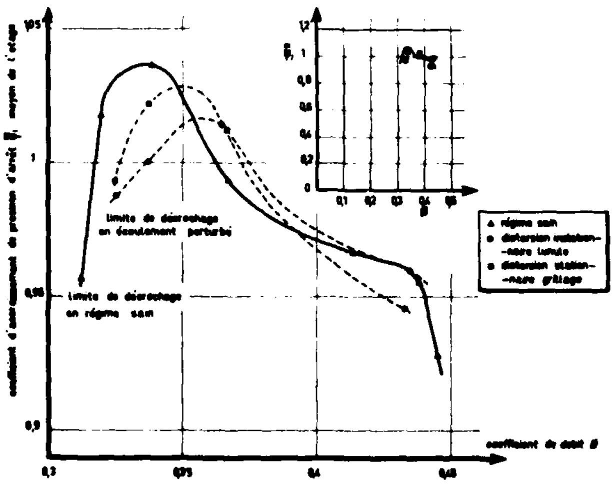


Fig. 23 – Caractéristique de fonctionnement de l'étage en alimentation homogène et perturbée.

7 - ANALYSE PAR UNE METHODE DE PETITES PERTURBATIONS DES RESULTATS EXPERIMENTAUX OBTENUS

Bien que centrés sur la représentation au moyen d'un générateur de distorsion stationnaire (grillage) des perturbations apportées dans l'écoulement dans un compresseur axial par un générateur de distorsions instationnaires (lunule), les essais effectués sur le banc CERF peuvent également servir pour analyser l'amplification ou l'amortissement d'une distorsion dans un compresseur axial.

Ne possédant pas encore de code de calcul tridimensionnel, en cours d'élaboration, et qui permettrait de représenter tout l'écoulement avec ses variations de rayon, nous allons analyser les résultats obtenus au rayon moyen en assimilant les surfaces de courant à des cylindres coaxiaux mais en tenant compte de la variation de la vitesse débitante à la traversée du compresseur.

Nous admettrons dans ces conditions qu'aux grandeurs moyennes, vitesse débitante W , vitesse tangentielle V (V est nulle en amont de la roue mobile et en aval du redresseur), pression statique P et pression d'arrêt P_i , se superposent des perturbations w , v , p et p_i de faible intensité (w^2, v^2 et wv négligeables devant W^2). Dans la théorie de petites perturbations que nous allons adapter à notre schéma, les équations régissant le mouvement du fluide (supposé à masse volumique constant) et l'équation de continuité s'écrivent [2] :

$$
(9)\quad \left\{
\begin{aligned}
W_j \frac{\partial w_j}{\partial z} + V_j \frac{\partial w_j}{r\partial\theta} + \frac{\partial p_j}{\rho\partial z} &= 0\\
W_j \frac{\partial v_j}{\partial z} + V_j \frac{\partial v_j}{r\partial\theta} + \frac{\partial p_j}{\rho r\partial\theta} &= 0\\
\frac{\partial w_j}{\partial z} + \frac{\partial v_j}{r\partial\theta} &= 0
\end{aligned}
\right.
$$

où j = 1 en amont de la roue mobile,
j = 2 entre roue mobile et redresseur,
j = 3 en aval du redresseur.

Par suite de la périodicité azimutale de l'écoulement w_j , v_j et p_j peuvent être développés en séries de Fourier de θ que pour la commodité de l'écriture ; nous allons considérer comme parties réelles de séries complexes :

$$
(10)\quad \left\{
\begin{aligned}
w_j &= \mathcal{R} \sum_{n=1}^{\infty} \left(a_n^j e^{\frac{nz}{r}} + b_n^j e^{-\frac{nz}{r}} + c_n^j e^{-\frac{inV_j}{W_j}\frac{z}{r}} \right) e^{in\theta}\\
v_j &= \mathcal{R} \sum_{n=1}^{\infty} \left(i a_n^j e^{\frac{nz}{r}} - i b_n^j e^{-\frac{nz}{r}} + \frac{V_j}{W_j} c_n^j e^{-\frac{inV_j}{W_j}\frac{z}{r}} \right) e^{in\theta}\\
\frac{p_j}{\rho} &= \mathcal{R} \sum_{n=1}^{\infty} \left(-(W_j + iV_j) a_n^j e^{\frac{nz}{r}} - (W_j - iV_j) b_n^j e^{-\frac{nz}{r}} \right) e^{in\theta}
\end{aligned}
\right.
$$

où les constantes a_n^j, b_n^j et c_n^j se déduisent des conditions de compatibilité aux limites amont et aval du domaine (j).

Si nous faisons encore l'hypothèse supplémentaire commode pour simplifier le calcul (justifié dans le cas du banc CERF) que le redresseur est suffisamment éloigné de la roue mobile pour que le couplage des deux ne se fasse que par l'intermédiaire du terme rotationnel de la solution (10) comportant le paramètre, c_n^j nous aurons les expressions suivantes pour les composantes de la perturbation :

$$
(11)\quad \left\{
\begin{array}{ll}
\underline{\text{amont de la roue mobile}} & \underline{\text{aval de la roue mobile}}\\[1ex]
w_1 = \displaystyle\sum_{n=1}^{\infty} \left(a_n^{(1)} e^{\frac{nz}{r}} + c_n^{(1)} \right) e^{in\theta} & w_2 = \displaystyle\sum_{n=1}^{\infty} \left(b_n^{(2)} e^{-\frac{nz}{r}} + c_n^{(2)} e^{-\frac{inV_2}{W_2}\frac{z}{r}} \right) e^{in\theta}\\[2ex]
v_1 = i \displaystyle\sum_{n=1}^{\infty} a_n^{(1)} e^{\frac{nz}{r} + in\theta} & v_2 = \displaystyle\sum_{n=1}^{\infty} \left(-i b_n^{(2)} e^{-\frac{nz}{r}} + \frac{V_2}{W_2} c_n^{(2)} e^{-\frac{inV_2}{W_2}\frac{z}{r}} \right) e^{in\theta}\\[2ex]
\dfrac{p_1}{\rho} = -W_1 \displaystyle\sum_{n=1}^{\infty} a_n^{(1)} e^{\frac{nz}{r} + in\theta} & \dfrac{p_2}{\rho} = -(W_2 - iV_2) \displaystyle\sum_{n=1}^{\infty} b_n^{(2)} e^{-\frac{nz}{r} + in\theta}
\end{array}
\right.
$$

Les conditions de transfert à travers la roue mobile s'explicitent en écrivant :

- la continuité du débit,
- le triangle des vitesses à la sortie de la roue mobile en utilisant une équation du type relaxation pour l'angle de sortie en axes mobiles,
- l'accroissement de pression statique qui doit tenir compte de la contribution des irréversibilités.

Elles nous permettent de déduire de ces équations les coefficients a_n, b_n, c_n.

Pour le redresseur nous tiendrons tout de suite compte de ce que la vitesse de sortie est axiale, d'où les composantes de perturbation de part et d'autre de cette grille.

amont redresseur — aval redresseur

(12)
$$\begin{cases} W_2 = \sum_{n=1}^{\infty} \left(a_n^{(2)} e^{\frac{n(z-l)}{h}} + c_n^{(2)} e^{-\frac{inV_2}{W_2}\frac{z}{h}} \right) e^{in\theta} & w_3 = \sum_{n=1}^{\infty} c_n^{(3)} e^{in\theta} \\ v_2 = \sum_{n=1}^{\infty} \left(i a_n^{(2)} e^{\frac{n(z-l)}{h}} + \frac{V_2}{W_2} c_n^{(2)} e^{-\frac{inV_2}{W_2}\frac{z}{h}} \right) e^{in\theta} & v_3 = 0 \\ \frac{p_2}{\rho} = -(W_2 - iV_2) \sum_{n=1}^{\infty} d_n^{(2)} e^{\frac{n(z-l)}{h} + in\theta} & \frac{p_3}{\rho} = 0 \end{cases}$$

en explicitant la continuité du débit ainsi que la perte de pression d'arrêt dans le redresseur les coefficients a_n, c_n se trouvent définis.

Nous supposerons dans nos calculs qu'à débit moyen donné l'accroissement de pression moyen peut être calculé à partir des données relevées au cours des essais stationnaires et qu'il en est de même des autres paramètres intervenant dans les relations que nous avons établies (figure 23). Dans la comparaison théorie-expérience nous représenterons la perturbation de pression par la série de Fourier

(13)
$$\frac{2(p_i - p_{i0})}{\rho} = \frac{12\,\overline{\omega}}{\pi^2} w_1^{2} \sum_{n=1}^{\infty} \left(\frac{\sin(n\pi/6)}{n} \right)^2 \cos n\theta$$

avec dans ces calculs : n = 100 et $\overline{\omega}$ = 0,22.

Nous admettrons donc dans nos comparaisons théorie-expérience que dans le cas des essais sur le compresseur la réponse de la roue mobile correspond à une succession d'états stationnaires.

Nous pouvons alors effectuer une comparaison systématique entre calculs et essais. Les figures 24 et 25 correspondent au rayon moyen r/R = 0,84 et aux deux vannages étudiés (N_V = 39, figure 25 ; N_V = 100, figure 24).

Rayon moyen $\frac{r}{R}$ = 0,84 . vanne grande ouverte Nv = 100

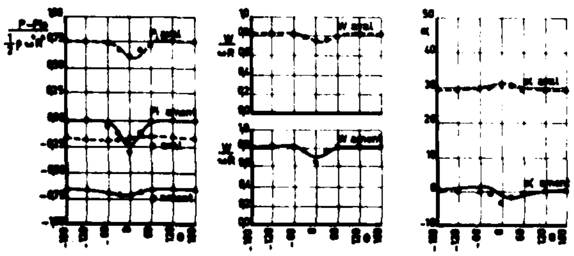


Fig. 24 – Comparaison théorie-expérience rayon moyen r/R = 0,84 ; vanne grande ouverte Nv = 100.

Rayon moyen $\frac{r}{R}$ = 0,84 ; voisinage du décrochage Nv = 0,39

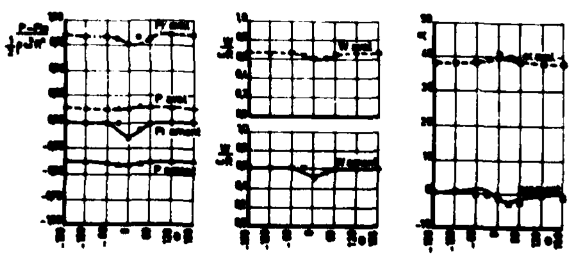


Fig. 25 – Comparaison théorie-expérience rayon moyen r/R = 0,84 ; voisinage du décrochage Nv = 0,39.

Dans les deux cas, nous avons comparé aux variations azimutales de la pression d'arrêt, de la pression statique, de la vitesse débitante et de l'angle de l'écoulement, les relevés expérimentaux effectués en amont (plan 5) et en aval (plan 10 de la roue mobile).

Pour le redresseur la comparaison théorie-expérience ne peut se faire que sur la vitesse débitante ou la pression d'arrêt, ces deux grandeurs étant d'ailleurs liées, puisque la pression statique aval est constante (l'expérience vérifie cette conclusion du calcul).

L'examen des résultats expérimentaux obtenus en l'absence de distorsion ayant montré qu'au rayon moyen (r/R = 0,84) la perte de pression statique est très faible, dans le cas du compresseur CERF la transmission de la perturbation à travers le redresseur doit se traduire par une simple translation azimutale du signal combinant l'effet de dérapage des lignes de courant entre le plan de mesure (6) et l'entrée du redresseur et l'effet de décalage angulaire induit par le redresseur lui-même.

Les figures 26 et 27 sur lesquelles nous avons porté les répartitions azimutales du coefficient de pression d'arrêt à la sortie de la roue mobile Ψ_{iM} , mesuré dans le plan 10, et à la sortie du redresseur Ψ_{iF} , mesuré dans le plan 13, pour les deux vannages extrêmes (figure 27, r/R = 0,84, N_V = 39 ; figure 26, r/R = 0,84, N_V = 100) montrent qu'effectivement ces répartitions se déduisent les unes des autres par une translation azimutale d'environ 40°, ce qui correspond assez bien aux résultats du calcul.

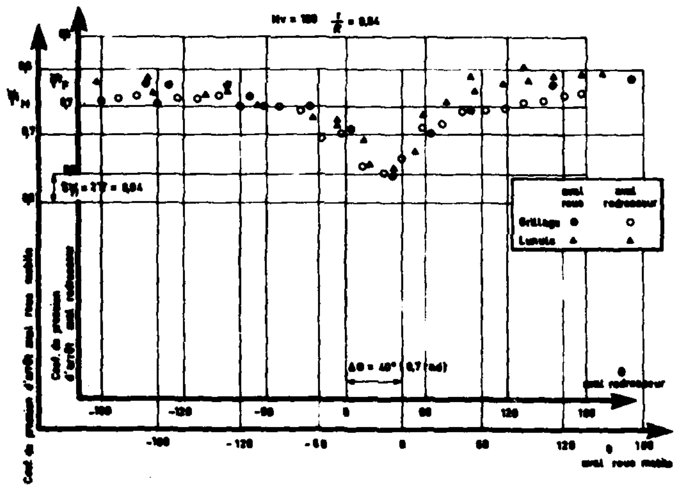


Fig. 26 – Déplacement azimutal de la pression d'arrêt entre les plans de mesure 10 et 13 Nv = 100 ; r/R = 0,84.

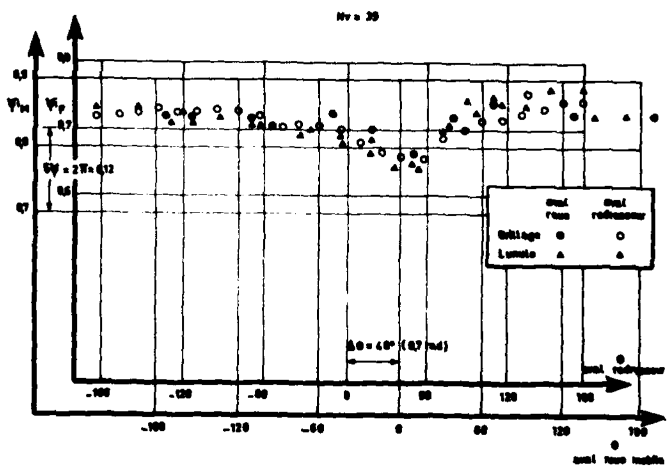


Fig. 27 – Déplacement azimutal de la pression d'arrêt entre les plans de mesure 10 et 13 Nv = 39 ; r/R = 0,84.

8 - CONCLUSIONS

Par ses caractéristiques générales, le banc d'essai de compresseur CERF décrit ci-dessus, est un dispositif particulièrement bien adapté aux études à caractère fondamental et plus spécialement aux recherches sur la distorsion.

Le banc, facile d'accès, ne consommant qu'une puissance très faible (30 kW), est doté d'importants moyens de mesure tant stationnaires qu'instationnaires. Un large effort a été consenti dans la motorisation et le repérage précis des dispositifs de mesure, compte tenu du volume et de la difficulté des travaux à réaliser.

Les essais de qualification du banc ont montré d'une façon générale que l'écoulement est propre, axisymétrique sur une hauteur suffisante de la veine. Ils ont également permis de comparer la distorsion instationnaire induite par une lunule opaque placée à l'entrée du canal à la distorsion stationnaire créée par des grillages correctement agencés.

Ce banc doit maintenant permettre de remonter aux caractéristiques transitoires de l'écoulement à la traversée du compresseur et de là aux équations de transfert régissant son fonctionnement.

REFERENCES

Il existe dans la littérature diverses publications se rapportant au sujet traité, nous avons porté dans les références les seules publications spécifiques à notre contribution.

[1] - P. BRY, P. LAVAL and G. BILLET
Distorted flow field in compressor inlet channels.
A.S.M.E. Paper n°82-GT-125.

[2] - J. FABRI
Amplification of distorsions in an axial flow compressor stage.
12ème CIMAC Tokyo, 1977

DISCUSSION

J.C. Corde

Envisagez-vous d'utiliser d'autres dispositifs – que la lunule – de génération de distorsion instationnaire?

Author's Reply

Oui, des études ont été menées pour la direction de l'aérodynamique dans les souffleries de Modane pour simuler les distorsions instationnaires.

Il ressort de ces essais qu'un dispositif en forme de sifflet reproduise correctement des configurations réalistes et la SNECMA souhaite que nous utilisions un tel dispositif.

Y. Rouanet, Fr.

Avez-vous pu calculer ou mesurer l'effet sur les rendements dûs à la distorsion?

Author's Reply

L'effet sur les rendements est directement calculable à partir des valeurs de rapports de pression obtenus avec et sans distorsion. L'écart des températures est très faible. Le compresseur fonctionne avec un taux de pression faible.

AXIAL COMPRESSOR CHARACTERISTICS DURING TRANSIENTS

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SUMMARY

The anticipated effects of heat transfer on the characteristics of an axial-flow air compressor have been estimated using two different prediction methods. The second method, which incorporates end-wall boundary layers, predicted somewhat smaller changes of characteristics than those indicated by the earlier method. As the second method uses a model of the flow which is probably more realistic, for the present it is considered that its predictions on heat transfer effects on characteristics are more accurate. In an altitude deceleration of a typical 12-stage compressor it is therefore estimated that surge margins might be reduced by about 20 or 25%. In a sea-level acceleration there are similar increases in surge margins, as compared to adiabatic characteristics.

Decreasing the stagger angle increases slightly the susceptibility to characteristics being altered due to heat transfer effects in transients.

List of symbols

c_p	specific heat at constant pressure
C_h	enthalpy-equivalent static-pressure-rise coefficient
e	deflection
g	staggered spacing between blades
h	annulus height
i	incidence
$\dot{m}$	mass flow rate
N	rotational speed
Q	heat flux to air in compressor
T, ΔT	temperature, temperature difference
α	air angle
δ	displacement thickness
ε	tip clearance
ν	kinematic viscosity
λ	fractional effective area
$\Delta\phi$	change in angle of projected displacement thickness line

Subscripts

ht	heat transfer
∞	core
2	exit

Superscript

*	design

1. INTRODUCTION

The behaviour of aircraft gas turbines during rapid accelerations and decelerations is of considerable interest. It is important that these transients can be carried out rapidly, without any malfunctioning of the engine - in particular without encountering a serious stall or surge in the compressor(s).

It appears that the steady-running characteristics of the engine and its components are not always applicable in transients. For example it has been found that the margin of compressor pressure ratio available between the working line and the surge line (the surge margin) is influenced by the immediately preceding history of the engine - in one case Warne (Ref. 1) reported that the "overfuelling margin" for an acceleration was halved if the engine had immediately previously been allowed to "soak" at a high speed.

The purpose of the present paper is to consider the factors arising in transients which might affect the performance characteristics of axial-flow air compressors. The magnitudes of these factors are estimated, and the factors are incorporated in two alternative procedures for predicting the performance of the compressors, the influence of these factors on the compressors' characteristics thus being indicated.

2. PREDICTION METHODS - ADIABATIC FLOW

The flow in an axial-flow air compressor, particularly as surge is approached, is influenced by three-dimensional effects. Ideally, the prediction method used should be based on blade-element data, with radial integration, followed by axial stacking. Such a procedure is complex, and some of the information required is as yet incomplete. In the present work, which is regarded as exploratory rather than definitive, a simple two-dimensional procedure has been adopted, using calculations at the pitch-line and modified in two different ways. The modification used in the first method allows the possibility of the existence of stall cells. In the second method the modification takes account of the growth of the end-wall boundary layers on the casing and hub of the annular flow passage. These methods are now described, referred to as Methods 1 and 2 respectively.

2.1 METHOD 1: PITCH-LINE WITH STALL CELLS

This method has been used by Maccallum and Grant (Ref. 2) and is described in fuller detail in that paper. Briefly, the method uses row by row calculations at the mean blade height, based on the two-dimensional cascade data published in Ref. 3. The flow is compressible and Mach Number effects (Ref. 4) are allowed for. Axial velocity variation within a row is incorporated into the exit velocity triangles and energy transfer calculations, although its effect on deviation (Ref. 5) is not included.

When the incidence of the flow into a particular row becomes too great, it is assumed that a stall cell is then formed in part of the annular area, and in the remaining area normal flow is restored. The quantitative definition adopted for this stalling of a blade row is that the profile drag coefficient rises to double its minimum value, and this occurs when the numerical value of the dimensionless relative incidence group $(i-i^*)/e^*$ reaches 0.4. When the flow through a particular row is reduced below that giving this stalling value of the incidence group, it is assumed that the flow redistributes itself in the annulus so that part of the annulus operates with a flow giving exactly this stalling value of the incidence group, and the remainder of the annulus is occupied by a stall cell with no significant flow.

Inherent in this type of pitch-line prediction method is the need to insert a work-done factor to account for the reduced energy transfer resulting from the development of the axial velocity profiles. Alternatively a blockage factor can be used. In the present study, work-done factors were used.

The application of this prediction method is illustrated later in Section 4.1.

2.2 METHOD 2: PITCH-LINE WITH END-WALL BOUNDARY LAYER

It has been found by Koch (Ref. 6) that the limiting pressure rise potential of axial-flow compressor stages can be correlated with the peak pressure recovery data (Ref. 7) of two-dimensional diffusers having a 9% blockage in the inlet flow profile. In this correlation, illustrated in Fig. 1, a parameter representing the stalling effective static pressure rise is plotted against the ratio of the diffusion length to the exit width (staggered) of the diffusion passages. The parameter representing the stalling effective pressure rise of a stage is based on the static pressure rise coefficient, C_h, normalised by the pitch-line free stream dynamic head. This coefficient is adjusted for tip clearance, axial spacing, Reynolds Number and for an "effective dynamic pressure factor" to account for the proneness to stall of low stagger stages.

As the flow through a stage is throttled so that the limiting, or stalling, pressure rise condition referred to above is approached, the end-wall boundary layers thicken, as shown by Smith (Ref. 8). The effective area ratio, at which the stalling pressure rise occurred was found to be a function of the tip clearance, staggered gap and blade height (Ref. 6)

$$\lambda = 1 - (0.34 + 2\,\overline{\varepsilon/g}\,)\,(\overline{g/h}) \tag{1}$$

At lower pressure loadings the end-wall boundary layers are thinner and the effective area ratio is higher. The relationship is complex and is illustrated in Fig. 2 by plots, given by Smith (Ref. 8), of the normalised displacement thickness of the casing boundary layer as a function of the loading and of the tip clearance.

This data of Smith's on the relationship between the displacement thicknesses and the pressure loading allows a means of calculating the blockage factor for loadings lower than the limiting value. This, combined with the stalling pressure rise factors provided by Koch (Ref. 6), was the basis of the second prediction method used in the present work. The flow is assumed to be concentrated in the central core of the annulus with blocked regions of thicknesses equal to the displacement thicknesses adjacent to the casing and hub walls. For the flow in the core, it is assumed that the two-dimensional procedure described in Section 2.1 above can be used. In a typical calculation procedure for a stage, first the limiting pressure rise is determined from Koch's correlation, the flow velocities used in the denominator of the pressure rise coefficient being those corresponding to the core flow when the boundary layer blockages are those occurring at the stalling pressure condition (Eqn. (1)). A test blockage is then tried, giving core velocities, and angles through the stage. The pressure rise predicted from the two-dimensional procedures is then compared with the limiting, or stalling, value. The ratio of these two pressure rises then gives displacement thicknesses from Fig. 2, and its equivalent at the hub which give a new blockage for that stage. This procedure is repeated until convergence is obtained. This calculation is carried out through all stages of the compressor in succession.

The loss in efficiency to be expected from the effects of blockage has been incorporated using the results of Smith (Ref. 8) who found, on average, that the "tangential force thickness" was about half the displacement thickness i.e. the reduction in torque due to the low mass flow in the boundary layer was only half the reduction there should have been had the blockage area been passing no flow at all.

Koch's correlation was developed essentially for low speed compressors. For high speed compressors Koch found the stalling pressure rises of the stages tended to fall below the correlation. To allow for this effect a factor was introduced into the present prediction program to scale down the correlation values of limiting pressure rise.

The use of this second prediction method is illustrated later in Section 4.2.

3. EFFECTS IN TRANSIENTS - GENERAL DESCRIPTION

Several effects occur in transient operations of gas turbines which can alter the characteristics of the compressor. These have been listed for an axial-flow compressor (Ref. 9) and partially studied (Ref. 9,2). These effects are now reviewed.

3.1 DENSITY CHANGES RESULTING FROM HEAT TRANSFER

During an acceleration of an initially "cold" engine, there will be heat transfer from the air as it passes through the compressor to the blades, platforms and casings. This will make the air more dense, and will lower the axial component of velocity at a particular stage in the compressor as compared to the steady-running state at that particular non-dimensional rotational speed, based on inlet temperature. This alters the stage matching. Similar effects, but opposite in direction, occur during decelerations.

The temperature distributions shown by Smith (Ref. 8) for a 12-stage compressor, under steady-running conditions, indicate that the energy dissipated into heat due to losses adjacent to the end-walls tends to remain within the end-wall boundary layers. Consequently it appears that heat transfers to, or from, platforms and casings will have little influence on the temperature, hence density, of the flow at the pitch-line. However the heat transfers to, or from, the aerofoils of the blades will immediately influence pitch-line density. Thus the procedure that has been adopted in the pitch-line calculations described in Methods 1 and 2 in the previous section is that only the heat transfers to, or from, the aerofoils of the blades alter the density as compared to adiabatic running.

3.2 BOUNDARY LAYER DEVELOPMENT ON BLADE AEROFOILS

It has been shown (Ref. 10) that the separation from a convex curved surface in an adverse pressure gradient can occur earlier if the surface is at a temperature above the temperature of the flowing air. These results, on a large curved surface, have been used to validate a prediction method which has subsequently been applied to flows over typical compressor blades (Ref. 2). For conditions when the flow on the suction surface of the blade is near to separation, if the blade is then made hot, compared to the air, the displacement thickness increases much more rapidly near the trailing edge. There is no significant effect on the pressure surface. The behaviour of the boundary layers is shown qualitatively in Fig. 3. If the projected suction surface displacement thickness line with the hot blade is compared with the corresponding adiabatic line, it has been moved through an angle $\Delta\phi$. Since the boundary layer on the pressure surface has not been affected, it has been assumed that the flow leaving the blade is, on average, displaced through an angle $\Delta\phi/2$. Thus the flow through the compressor blading of that row receives a reduced deflection. The correlation found in Ref. 2 for this reduced deflection, or increased exit angle, $\Delta\alpha_2$ is

$$\Delta\alpha_2/e^* = \Delta T(0.0005 + 0.00084(i-i^*)/e^*) \tag{2}$$

It is also likely that the wake from the hot blade will be wider, and there will be higher profile drag losses associated with this. The assumption has been made (Ref. 2) that the drag coefficient is that corresponding to the pseudo incidence angle which would give the "revised" exit angle.

3.3 BOUNDARY LAYERS ON END-WALLS

The effect of heat transfer on the thickness of the boundary layers on the end-walls, and hence on the blockage, is uncertain. One simple approximation is to assume that the effect is similar to the effect on the thickness of turbulent boundary layers on flat plates (Ref. 11).

$$\delta_{ht}/\delta = (\nu_{wall}/\nu_\infty)^{0.2} \tag{3}$$

This assumption has been used in Method 2, described in Section 2.2.

3.4 TIP CLEARANCE CHANGES

Due to the different rates of response of discs, blades and casing to temperature changes resulting from pressure ratio changes, tip clearances will differ from those occurring in steady running. These clearance changes will alter efficiencies and hence characteristics. The changes in tip clearances can be fed directly as input into prediction Method 2. Prediction Method 1 requires this information expressed as changes in efficiency.

Tip clearance changes in transients, and their relation to efficiency changes are considered in Paper 17 to this meeting (Ref. 12)

4. PREDICTIONS IN TRANSIENTS

The effects during transients of the factors of density change due to heat transfer, boundary layer development on the aerofoils and end-wall boundary layer development have been considered with application to the 12-stage axial flow H.P. compressor of a two-spool by-pass engine having mixed exhausts and of overall compression ratio 20. However the effects on the characteristics resulting from tip clearance changes (Section 3.4) have not been included.

The method of calculating the average heat transfer coefficient over the blades has been described previously (Ref. 2). Briefly, a weighted average coefficient between that for a laminar boundary layer and that for a turbulent boundary layer starting on a flat plate has been used, increased by 50% as an approximate allowance for the high turbulence levels occurring in compressors.

4.1 PREDICTIONS BY METHOD 1: PITCH-LINE WITH STALL CELLS

This method has previously been applied to the conditions occurring in the compressor at the end of a rapid acceleration at 12,200 m altitude (40,000 ft.) from maximum speed to the flight idle speed. It had been found, for a range of engines, that an immediate re-acceleration of the engine from the flight idle condition represented a severe test of the engine's ability to avoid surge - hence the interest in this transient.

The results are shown in Fig. 4, the adiabatic constant speed characteristic being represented by the solid line. In order to obtain satisfactory agreement between the adiabatic predictions and rig results, work-done factors in the range 0.81 to 0.71 and an efficiency factor of 0.92 were used. The prediction of where surge will occur on this constant speed characteristic is not easy. Some workers (e.g. Ref. 13) have suggested the maximum pressure-ratio point, while others (e.g. Ref. 14) have suggested an averaged stage loading. In the present work, for the adiabatic characteristic, the maximum pressure-ratio point has been used. For the non-adiabatic situations, the point has been selected where the group of four stages most heavily loaded have an average axial velocity to blade speed ratio equal to that in the same group of four stages when at "surge" on the adiabatic characteristic. Thus the characteristics shown in Fig. 4 were obtained. This method predicts very considerable reductions in surge margin, reduced by between 50 and 60% due to the combined effects of density change caused by heat-transfer and aerofoil boundary layer changes, each of these individually causing a reduction of 25 to 30%.

It was explained in Section 3.1 that it has been assumed that only the heat transferred from the aerofoils contributed to a density change of the air at the pitch-line. This representes about one third of the total amount of the heat being transferred from the complete compressor. The remaining heat, from platforms and casings, is regarded as being held in the end-wall boundary layer flows.

It has been suggested (Ref. 15) that cl e movements of the constant-speed characteristics are equivalent to the compressor operating at a new "effective" speed. Correlations for these changes have been given for a 16-stage compressor of a single-spool engine (Ref. 15) and for the 12-stage compressor being studied in the present work (Ref. 16). The latter correlations for this compressor are

$$(\Delta N/N)_{\text{boundary layer}} = -0.15 \frac{\Delta T_{ave}}{T_{air,\ ave}} \tag{4}$$

and

$$(\Delta N/N)_{\text{density change}} = -0.15 \frac{Q}{T_{air,\ ave}} \tag{5}$$

4.2 PREDICTIONS BY METHOD 2: PITCH-LINE WITH END-WALL BOUNDARY LAYER

This Method has been applied to the transient condition described above.

In order to line up the prediction of the adiabatic characteristics, work-done factors in the range 0.78 to 0.72, a factor on efficiency of 0.95 and a factor of 0.70 on the correlation of Koch (Ref. 6) were used. The adiabatic surge point was taken as the point of maximum pressure ratio, and for the non-adiabatic cases it was the condition giving the same averaged axial velocity to blade speed ratio as at adiabatic surge for the group of four most heavily loaded stages - i.e. the same criteria as used in Section 4.1. The results are shown in Fig. 5.

It is seen that this method predicts a less severe reduction in surge margin, and a somewhat reduced change in effective speed than had been predicted by Method 1. Part of the reason for the reduced effects is that, considering the effects on the boundary layer on the aerofoil suction surface, with this end-wall program which incorporates blockage, the indicences to the blades in the core flow are reduced, and the flow on the suction surface is less liable to separate - consequently the effects of heat transfer from the blade are less noticeable.

This Method has been applied to an instant during a sea-level acceleration of the same engine, and the results are shown in Fig. 6. Movements of the constant speed line and an improvement in the surge margin, as compared to adiabatic, are predicted.

It is thought that the predictions of this Method 2 will be more accurate than the predictions of Method 1, due to the better flow description used.

5. INFLUENCE OF STAGGER ANGLE ON SUSCEPTIBILITY TO HEAT TRANSFER EFFECTS

Using Method 2, the susceptibility to heat transfer effects has been studied for a compressor having altered stagger angles. The compressor essentially had the geometry of the compressor considered in Section 4, but with stagger angles decreased by 3^{o} throughout and then increased by 3^{o} throughout compared to the orginal. The predicted changes in characteristics at the conclusion of the altitude deceleration are shown in Figs. 7 and 8 respectively. It appears that the changes occurring when the stagger angles are increased by 3^{o} are similar to those in the reference compressor. When the stagger angles are reduced by 3^{o} the effects of heat transfer are slightly more marked than in the reference case.

6. CONCLUSIONS

Compressor characteristics are altered during transients due to the effects of heat transfer.

Of the two prediction methods used, it is considered that the results from Method 2, which incorporates end-wall boundary layers, are more accurate. This method indicates reductions in surge margins of 20 to 25% in altitude decelerations, and increases of similar magnitudes in accelerations.

Movement of constant speed lines can be quantified in terms of "effective" speeds.

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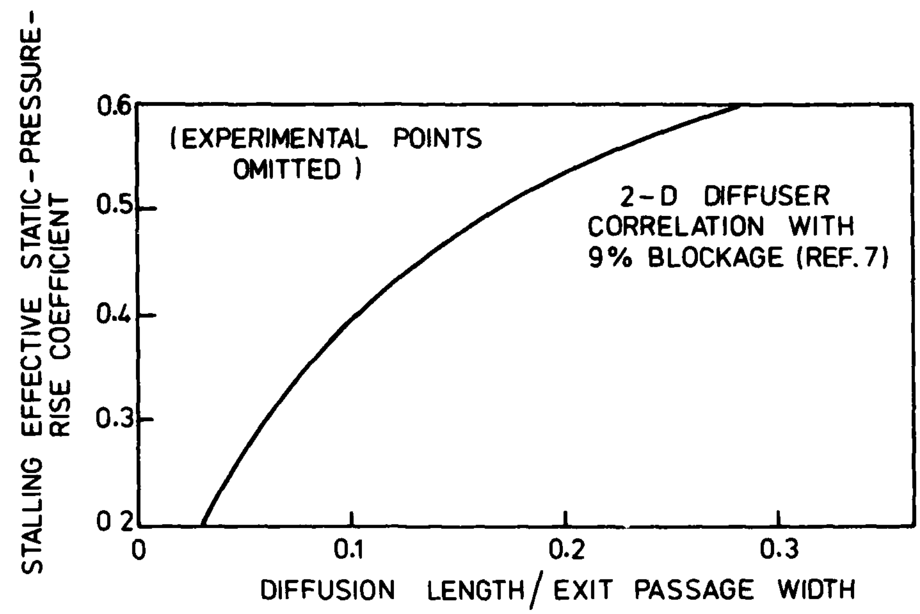

FIG. 1 CORRELATION OF STALLING EFFECTIVE STATIC PRESSURE RISE COEFFICIENTS (REF. 6)

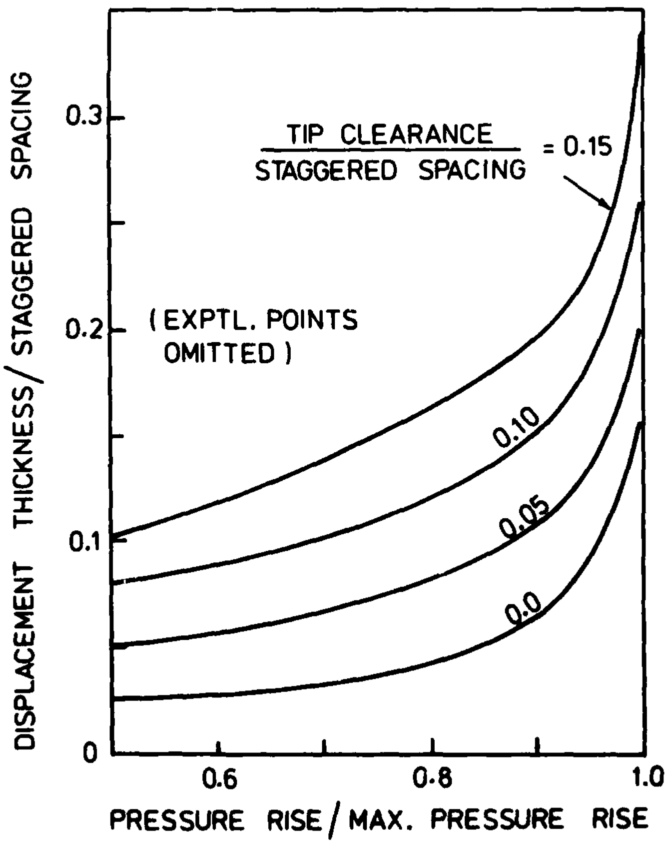

FIG. 2 DISPLACEMENT THICKNESS OF CASING BOUNDARY LAYERS (REF. 8)

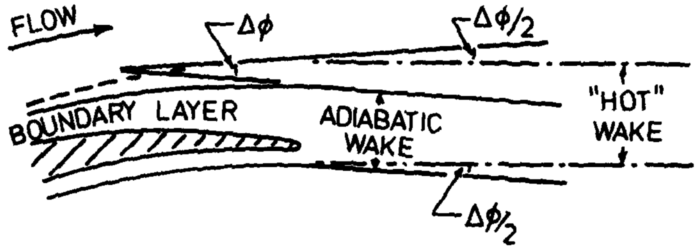

FIG.3 THE EFFECT OF HEAT TRANSFER ON WAKE DEVELOPMENT (REF. 2)

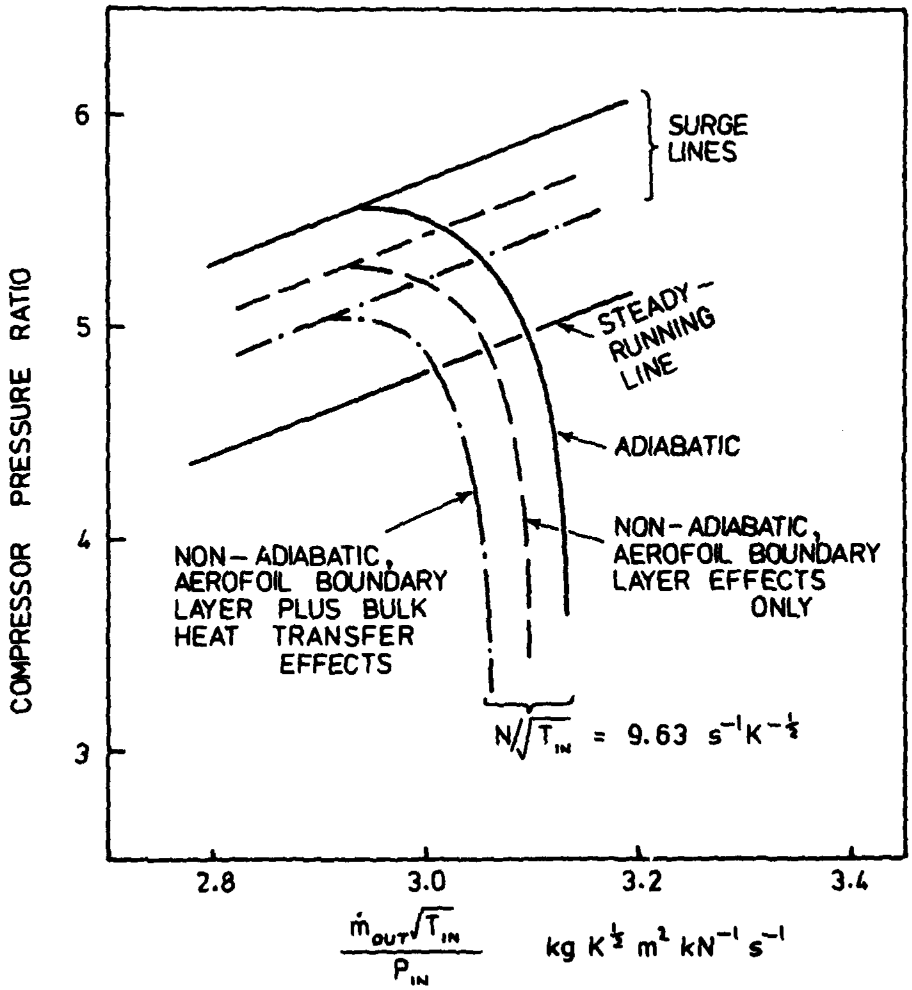

FIG.4 EFFECTS PREDICTED BY METHOD 1 ON H.P. COMPRESSOR CHARACTERISTICS AT END OF ALTITUDE DECELERATION

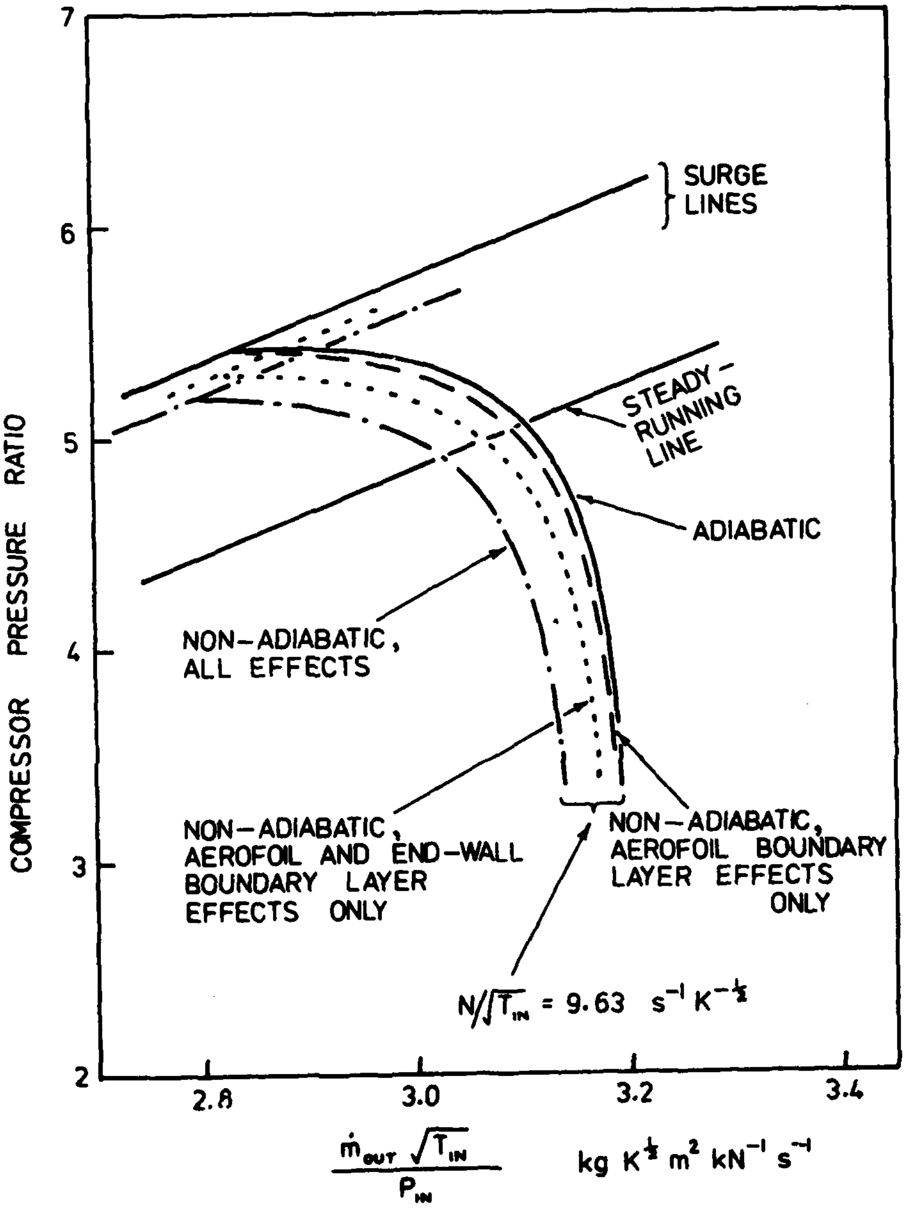

FIG.5 EFFECTS PREDICTED BY METHOD 2 ON H.P. COMPRESSOR CHARACTERISTICS AT END OF ALTITUDE DECELERATION

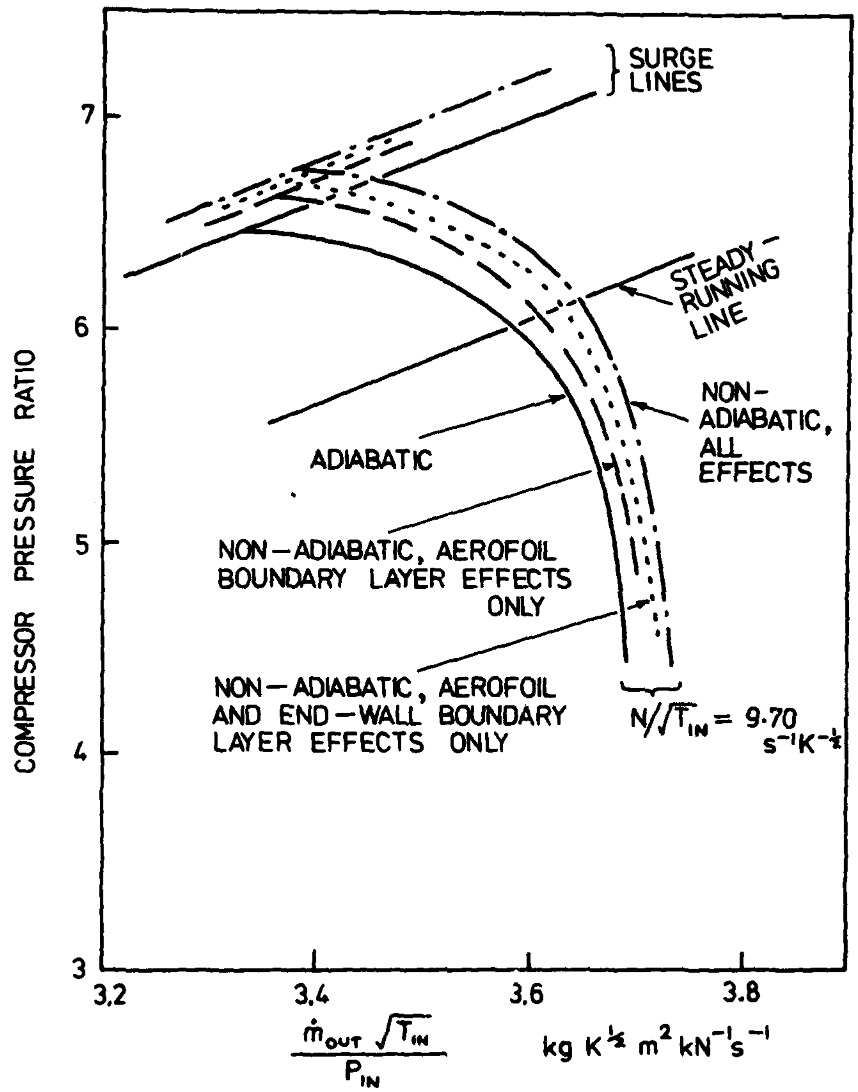

FIG. 6 EFFECTS PREDICTED BY METHOD 2 ON H.P. COMPRESSOR CHARACTERISTICS AT 6 SEC. IN SEA-LEVEL ACCELERATION

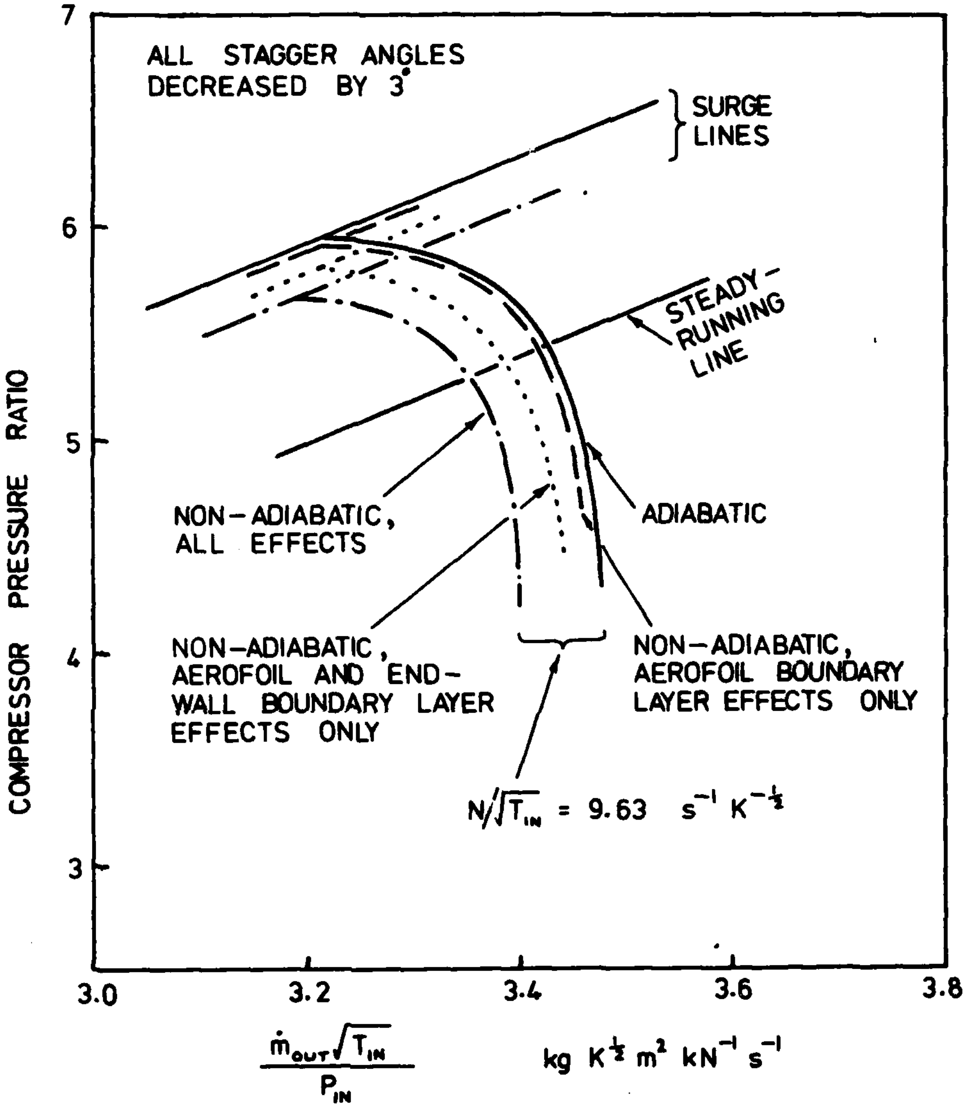

FIG.7 EFFECTS PREDICTED BY METHOD 2 ON H.P. COMPRESSOR CHARACTERISTICS AT END OF ALTITUDE DECELERATION – ALL STAGGER ANGLES DECREASED BY 3°

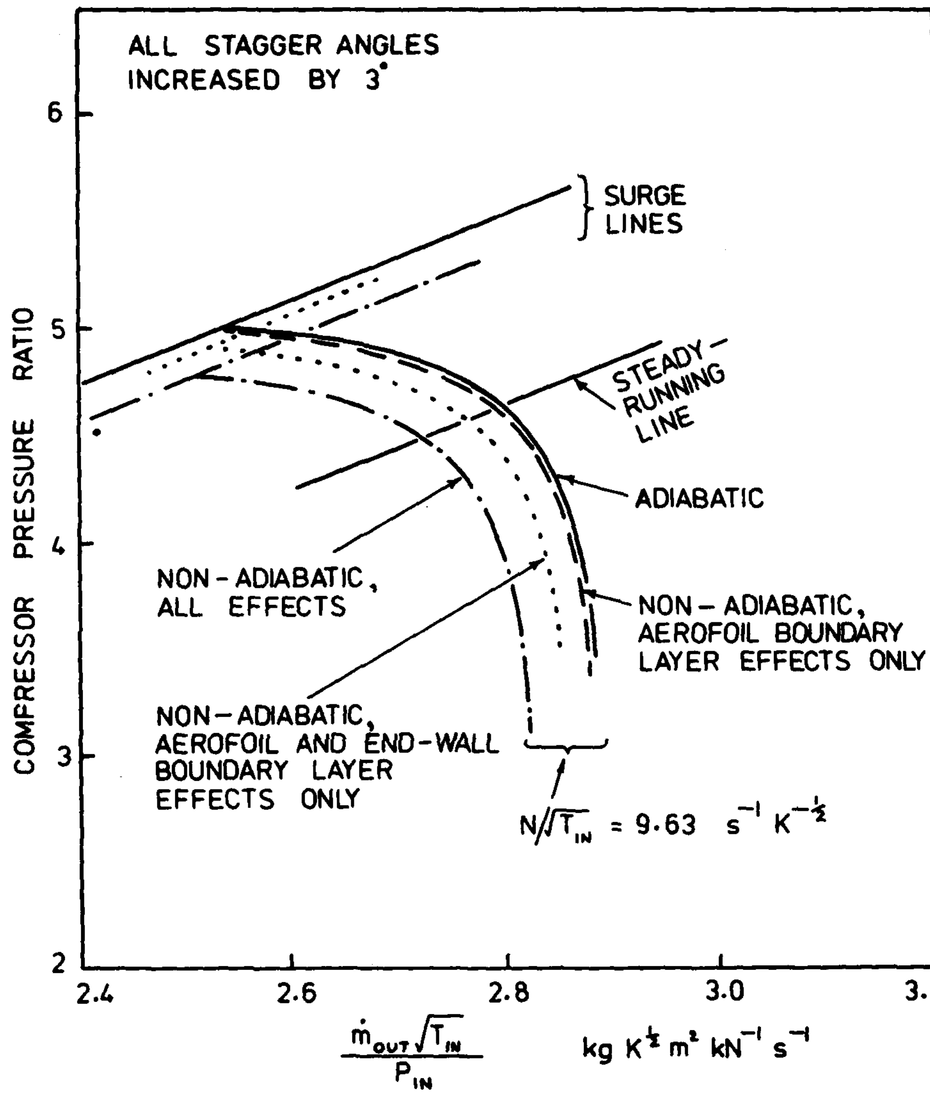

FIG. 8 EFFECTS PREDICTED BY METHOD 2 ON H.P. COMPRESSOR CHARACTERISTICS AT END OF ALTITUDE DECELERATION — ALL STAGGER ANGLES INCREASED BY 3°

DISCUSSION

D.K. Hennecke, Ge.

The heat transfer in a compressor during an acceleration or deceleration is transien[illegible] erefore, I would like to know if the compressor characteristics you have calculated are valid for one instan[illegible] me.

Author's Reply

The compressor characteristics shown are in each case valid only for one instant in a particular transient.

J. Hourmouziadis, Ge.

MTU has had some experience with surging compressors during transients, however we were not able to identify any heat transfer effects. What is your estimate of the relationship in magnitude between such effects and those of changing clearances at the blade tip?

Author's Reply

The heat transfer effects must exist and the predictions in the paper indicate they are not insignificant. I would estimate that the changes in characteristics due to heat transfer effects might approach in magnitude the changes due to tip clearances.

P.F. Neal, UK

In your work on the effect of heat transfer on surge margin you have shown a significant effect with measurements at steady conditions. Dr Hennecke has already asked about the transient heat transfer effects on surge but have you done any work on the difference in surge margin between say a cold accel and a hot reslam considering the dynamic effect that any mass that is accelerated when its density reduces basically produces an oscillatory (or unstable) motion?

Author's Reply

I have not considered the dynamic effect in accelerating a flow that you describe. The present method does predict that the surge margin when accelerating a hot engine will be less than when accelerating a cold engine from the same initial speed. The effect you mention could accentuate this reduction.

K. Bauerfeind, Ge.

From some work carried out at MTU in the late 60's we found an additional effect, i.e. a change of the efficiency due to what one could term "interstage cooling" during an accel with a compressor. Could you comment on the magnitude of this effect?

Author's Reply

The procedures described in the present paper recognise this effect. It has been assumed that the small-stage efficiencies of the steps between the interstage coolings are the same as they would have been in adiabatic flow at the same values of the stage axial velocity ratios and stage pressure ratios. The method is described in Reference 9.

In the H.P. Compressor considered in the present paper, during a sea level acceleration the sum of the interstage cooling effects, i.e. the temperature drop due to heat transfer, peaks at about 7 per cent of the temperature rise due to compression. This effect gives beneficial reductions in the powers absorbed by the compressors. However, there is a detrimental loss due to transient heat transfers in the turbines during the acceleration which more than balances the benefit in the compressors, and predicted accelerations are slowed 5 to 6 per cent when these effects are taken into account (Refs. 15, 16).

"Applications of a Compression System Stability Model"

Samy Baghdadi
Detroit Diesel Allison Division
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Summary:

A mathematical model for compression system aerodynamic stability is presented and discussed. The model is applied to simulate several situations in which the operational stability of aircraft gas turbine engines is compromised, including radial and circumferential inlet pressure and/or temperature distortion, planar inlet pressure pulsations, hot gas ingestion resulting from armament firing, sudden variable geometry resets, and compressor stalling behavior. The theoretical results are validated by comparison with relevant test data.

Introduction

The effective operating range of gas turbine engine compressors is limited by aerodynamic instabilities variously known as stall, surge, or rotating stall.

These flow instabilities typically manifest themselves in one of two modes--intermittent flow reversal throughout the machine (surge), or localized low flow regions that move in the direction of compressor rotation (rotating stall). Both of these instability modes result in potentially destructive oscillations of internal engine pressures and temperatures; aircraft powerplants must, therefore, be designed with substantial compressor "surge margins" to minimize the possibility of compressor operation in the unstable zone.

Background

The aerodynamic instabilities arise from a complex interaction between the volume dynamics and the pressure/flow characteristics of the entire compression system (i.e., the compressor inlet and exhaust ducting, and upstream or downstream components, as well as the compressor itself). In general, it is found that the location of the instability limit on a compressor map is negligibly affected by the inlet configuration of the compressor or by the configuration of components and ducting beyond a choking point or a large plenum downstream of the compressor. The instability mode (i.e., surge versus rotating stall), however, is strongly affected by these parameters [1].

The compressor baseline "surge margin" is used in a variety of ways, depending on the engine installation and usage. Typical factors that utilize all or part of an engine's baseline surge margin include engine power transients (including after-burner transients), altitude operation (Reynolds number effects), inlet flow distortion (steady or unsteady), engine wear, and engine manufacturing tolerances. Engine manufacturers have typically used empirical data to assess the impact each of these factors has on the engine stability [2].

Mathematical models of compression systems have also been used to analyze engine stability. Some of these models are applicable to steady inlet distortion only [3,4], while others can be applied to unsteady flow pulsations as well [5,6] and even to compressor post-stall behavior [1]. Most recent mathematical models are sophisticated versions of the parallel compressor model [3, 4] and/or unsteady one-dimensional volume dynamics [1, 5, 6]. All the theories mentioned above make use of a "compressor dynamic response" correlation to relate the dynamic behavior of rotor airfoils in distorted flow conditions to their steady-state behavior.

The dynamic response correlations are intended to express the fact that, as the rotors sweep into and out of the distorted regions, the flow relative to the rotors is unsteady; the longer the period of time the rotor spends in the distorted region--i.e., the greater the circumferential extent of the distorted region--the closer the airfoil response will be to its steady-state characteristic. For short rotor immersion times, the airfoil can exceed its steady-state stall incidence limit without triggering a global compressor instability. Thus, small sector distortions are less destabilizing than large sector distortions of the same intensity [7]. [3] treats this aspect of the flow entirely empirically, by modifying the simple parallel-compressor-predicted loss in a compressor surge margin by a sensitivity that is a function of the compressor airfoils' average reduced frequency. [5] and [6] have adapted the theoretical work of [8] and [9] to calculate an "effective incidence" for each rotor airfoil that is lower than the actual instantaneous incidence by an amount proportional to the airfoil's reduced frequency. However, [6] notes that, while experimental cascade work [10] generally supports the theoretical trends, measured values of lift coefficient are 10% to 20% higher than the analytical results.

Unsteady Flow Theory

The Dynamic Compressor Analysis (DCA) presented here is an unsteady volume dynamic stability analysis developed to apply to compressors operating with a wide variety of inlet or exhaust conditions--steady-state or dynamic, and with total pressure and/or total temperature radial and circumferential inlet distortion patterns.

The DCA uses an unsteady Method of Characteristics technique to solve equations for the conservation of mass, momentum, and energy in the blade rows and volumes of a turbomachine. The turning and loss characteristics of the various airfoil rows are generally deduced from steady-state test data. No empirical "dynamic response" correlations are used in the analysis. The important aspect of the air-

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foil's dynamic behavior--i.e., the capability of stable operation beyond the steady-state stall incidence--is inherent in the unsteady, volume dynamic analysis. In this analysis, the flow reversal process within an airfoil row takes time to develop, so that disturbances of short duration may force momentary airfoil operation beyond the steady-state stall point without triggering a global instability of the compression system. It should be noted that the airfoil rows' loss and turning characteristics must usually be extrapolated beyond data-deduced values to account for momentary airfoil operation beyond the system's steady-state surge line. In general, a linear extrapolation of the airfoil characteristics is used in the analysis; however, it is probable that the development of a more physically realistic model for airfoil characteristics in the stalling flow region would significantly enhance the applicability of the analysis. Details of the Dynamic Compressor Analysis are provided in [11] and [12]. Examples of applications of the analysis follow.

Response of a Fan to Planar Inlet Disturbances

The Air Force/General Electric "Planar Pressure Pulse Generator" (P^3G) program provides a set of experimental data for a modern, high-speed, two-stage fan (F101 fan) subjected to planar inlet pulsations of various frequencies.

The test program involved a determination of the steady-state surge line of the fan as well as the surge line with the inlet total pressure being pulsed sinusoidally at frequencies ranging from 40 Hz to 600 Hz. Data were obtained at 80% and 100% speeds.

A DCA model of the P^3G system was constructed to validate the frequency response fidelity of the analysis. The fan geometry and blading characteristics were modeled as shown in Figure 1. The fan area distribution, inlet and outlet blading angles, and steady-state stage characteristics were obtained from [13].

The steady-state surge line was determined analytically by running the dynamic compressor calculation at gradually decreasing flows at each of the two speeds. The analytical surge points were in excellent agreement with the test data, as shown in Figure 2. Surge was analytically determined by the indicated breakdown of the flow properties within the machine; typically, the instability exhibits itself as an oscillation of increasing amplitude terminated by the computation of a negative axial velocity. Figure 3 shows the variation of the calculated total pressures within the fan with time at design speed for a stable operating point very near the surge line and for a point on the surge line.

DCA results were also compared with test data obtained with dynamic inlet conditions at 80% and 100% speeds. The data are presented in Figures 4 and 5 as fan dynamic distortion sensitivity versus disturbance frequency. The fan dynamic distortion sensitivity [13] is defined in Figure 6 as the fractional loss in surge pressure ratio at a flow divided by the fractional inlet total pressure pulse half amplitude. In Figures 4 and 5, the DCA results are compared with the test data of [13]. The DCA results are seen to fall within the test data band at both 80% and 100% speeds.

These results indicate that the use of blade row characteristics obtained from steady-state experimental data allows a realistic assessment of the dynamic response of, at least, the F101 fan. The experimentally observed decrease in dynamic distortion sensitivity with increasing frequency is analytically predicted by the DCA model without resorting to empirical modifications of the steady-state blade loss characteristics to account for unsteady effects. The DCA results suggest that the insensitivity of the fan blading to high-frequency disturbances is principally a volume dynamic effect rather than an unsteady blade response effect. The analysis indicates that a blade row can be driven well beyond its steady-state stall incidence limit without causing compressor surging, provided the airfoils are operating beyond stall incidence for a short enough time. (See Figure 7 for the instantaneous operating point of the fan at a stable point and Figure 8 for the total pressure distribution through the fan at an unstable point with high-frequency dynamic distortion.) Thus, the higher the disturbance frequency, the further beyond the steady-state stall incidence limit the airfoil may be driven. This does not mean that airfoil stall does not occur under these conditions but, rather, that compressor surging will not occur because surge is a volume dynamic effect that takes time to develop.

VSTOL Fan Response to Variable Geometry Changes

The Dynamic Compressor Analysis was used to study the response of a single-stage, variable inlet guide vane (IGV) fan to the rapid IGV transients required for thrust modulation in a V/STOL aircraft application. The fan was modeled with aircraft-type inlet and exhaust ducts and also with the very long inlet and exhaust ducts of a typical test cell installation. The fan instantaneous operating points for the two cases are presented in Figure 9. Figure 10 shows the total pressure distributions within the fan and ducting for the test cell installation. The figure indicates that the response of the fan installed in an aircraft having a relatively short inlet duct is quite different from the response of the same fan in a test cell. The test cell installation does not allow the fan airflow to change as rapidly as in the aircraft installation, so that the instantaneous fan operating point trajectory rises substantially beyond that of the aircraft installation. Clearly, a stability assessment of the fan's variable geometry features cannot be realistically accomplished in a typical test cell.

Hot Gas Ingestion

The Dynamic Compressor Analysis has been used to assess the response of an aircraft engine to the hot gas ingestion that might result from the firing of missiles or other armaments. The aircraft inlet duct was assumed to be seven feet long, and the turbine nozzle was assumed to remain choked throughout the transients.

The hot gas was assumed to have the properties of air, and the inlet conditions were specified as total temperature pulses of various durations and amplitudes. Figure 11 depicts the temperature pulse shape, the compressor/duct geometry, and several computed operating point trajectories for compressor operation at 87% speed. Although hot gas ingestion test data do not exist for this specific configuration, the computed results are in fair agreement with the test results and correlation of Hercock and Williams, [14], as shown in Figure 12.

Application of the Theory to Distorted Flows

The theory can be applied to compressor flows described as single-streamtube or multi-streamtube, depending on the flow resolution desired. Flows in which radial gradients are not dominant, such as clean inlet flows, unsteady flows with planar inlet or exhaust pulsations, and some compressor post-stall flows, can be modeled realistically using a single streamtube. Flows that include substantial radial inlet or exhaust distortions should be modeled as multistreamtube flows to allow for radial flow shifts. Circumferentially distorted flows are modeled relative to an observer fixed to the compressor rotor system, so that the spatially nonuniform, steady flow is seen as unsteady but spatially uniform. A single-streamtube model is used for purely circumferential distortion, and a multi-streamtube model is used for complex patterns involving both radial and circumferential distortions. The compressor exit static pressure can be specified as a function of time, or can be set at the choke flow value, depending on the exhaust receiver and valving arrangement and the transient under consideration.

Figure 13 shows how a complex flow having a tip circumferential and a hub circumferential pressure distortion pattern can be simulated in a three-streamtube mathematical model. The inlet total temperature and pressure are specified as functions of time (i.e., rotor circumferential position) for each streamtube, and the exhaust back pressure is held constant for all streamtubes.

A series of steady-state inlet distortion tests has been performed at DDA. The tests were conducted using the High Flow Compressor test rig and DDA's Air Jet Distorter system. Figure 14 is a schematic of the test rig. The Air Jet Distorter nozzles were arranged to provide the classical inlet distortion patterns shown in Figure 15. The total pressure distortion pattern was generated by injecting high pressure air directly counter to the incoming airstream, while total temperature distortion patterns were generated by injecting hot, high pressure air through nozzles at an angle to the incoming airstream. The combined temperature and pressure patterns was generated by injecting hot, high pressure air counter to the airstream.

The High Flow Compressor test vehicle is a five-stage transonic compressor with variable inlet guide vanes and stators, a very high specific annulus flow rate, and highly loaded, low aspect ratio blading.

The compressor rig was comprehensively instrumented with total pressure, total temperature, and static pressure probes. The large number and the spanwise and circumferential distribution of the probes allowed "tracking" of the inlet distortion patterns throughout the compressor.

A mathematical model of the High Flow Compressor was constructed using the techniques described above. Single-streamtube models were used for the clean inlet and circumferential distortion patterns, and a three-streamtube model was used for the radial distortion pattern. The compressor blading characteristics used in the Dynamic Compressor Analysis model were derived from the clean inlet testing of the machine. For the multi-streamtube model, the clean inlet test data were first reduced along streamlines using DDA's Axial Compressor Design System (ACDS), which is a detailed radial equilibrium throughflow analysis [5]. The ACDS was also used in conjunction with the clean inlet blading characteristics to obtain the spanwise flow distribution through the compressor under radially distorted inlet conditions. The streamtube areas and blading characteristics thus obtained were then used in the Dynamic Compressor Analysis (DCA), ensuring that radial flow migration due to the distortions was properly taken into account in the stability analysis.[1] The streamtubes were coupled in that the sum of their areas was constrained to equal that of the compressor annulus.

Figure 16 compares the analytical and experimental compressor maps for clean inlet conditions. The test surge line is seen to be very closely duplicated by the DCA mathematical model. Figures 17, 18, and 19 compare DCA results to test results for the cases of the 1/revolution total pressure, total temperature, and aligned total temperature and total pressure circumferential inlet distortion patterns. In each case, the measured circumferential variations of both inlet temperature and pressure were converted to temporal variations and were used as the inlet boundary conditions in the DCA model. Figures 17, 18, and 19 show that the test and analytical results are generally in good agreement for the compressor surge line. The compressor's flow at a given speed, however, is less accurately estimated by the DCA. This is due to the fact that, under distorted conditions, the compressor rotor stages operate well beyond their steady-state data limits; the linear extrapolation beyond the clean inlet data assumed in the DCA tends to underestimate the high loading losses, particularly within the front stages of the machine. The appropriate representation of the airfoil characteristics beyond existing test data is (and is expected to continue to be) a major source of difficulty in compressor modeling work.

Figure 20 shows measured and computed variations of compressor total pressure and total temperature at the inlet, exit, and each stage of the compressor for the near surge point at 80% corrected speed with aligned 1/revolution pressure and temperature distortion. The static pressure variations at the inlet and outlet of the compressor are also shown. These comparisons show that the attenuation and migration of the distortion patterns are quite well predicted by the analysis, although the prediction is generally "smoother" than the test data.

Results of the multi-streamtube radial distortion modeling are shown in Figures 21 and 22. Figure 21 compares the spanwise variation of total temperature and pressure measured with tip radial distortion to that predicted by the ACDS radial equilibrium calculation using only the clean inlet blading charac-

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[1]This approach implicitly assumes that the blade element characteristics are not affected by the distortion patterns. In actuality, some secondary flow effects may be aggravated by the presence of the inlet distortion; however, even in the case of the low aspect ratio machine discussed in this paper, these effects appear to be minor at least up to the fourth stage.

teristics. The predicted spanwise distributions are seen to be generally in good agreement with the test data, implying that the use of clean inlet blading characteristics provides a reasonable representation of compressor operation with radial inlet distortion.

Figure 22 compares the measured and DCA-computed 95% speed line and compressor stability limit. Further comparisons of analytical and test results may be found in [2].

Compressor Stalling Behavior

Recent testing of an advanced DDA dual stage centrifugal compression system showed that the machine has a dual characteristic, i.e., that two stable operating regimes exist along a speed line. These are separated by an unstable region that manifests itself as a "surge" during clean inlet testing and as a "drift stall" region when inlet distortion is applied.

Analytical modeling of the first stage of the compressor indicates that it may also have two stable regions. Figure 23 shows the rotor and diffuser (stator) loss and turning characteristics assumed in the analytical model. These characteristics were modeled from the two-stage test results. Post-stall behavior of the diffuser is of particular interest; the distortion tests showed that the second-stage diffuser exhibits an abrupt increase in loss at the initial stall point, followed by a constant loss in the stall region. The first-stage diffuser was modeled with a similar abrupt loss increase, and also with a smaller post-stall loss increase. Figure 24 shows high and low diffuser loss cases for the first stage at design speed, as predicted with the DCA model. For the high diffuser loss case, the analysis indicates that the compressor surges at the initial instability. For the low loss case, the compressor drifts from its critical operating point to a lower, more stable operating point. Figures 25 and 26 show the first-stage transient operating trajectories and interstage static pressure distributions for the low and high diffuser loss cases as predicted with the DCA analysis.

The above results indicate that a "drift stall" occurs at the first instability if the stalled diffuser loss level is relatively low, while a "hard surge" instability occurs when the stalled diffuser loss level is high. The latter condition is more realistic for the transonic diffuser of the first stage. However, when the compressor operates with a circumferentially distorted inlet condition, the diffuser is only stalled over a portion of its periphery, so that the lower loss level might be considered a more realistic average of the unstalled and stalled diffuser performance. One might therefore anticipate that, even with realistically high diffuser stall losses, the distorted compressor will drift through the first instability, whereas the undistorted compressor would surge at the first instability.

Analytic modeling of the distorted first stage (with high diffuser stall loss) does indeed confirm that a drift stall can occur at the first instability. Figure 27 shows the compressor mean operating point trajectory as the exhaust throttle is closed for a 180 deg extent, 5% amplitude square wave inlet total pressure distortion. It is noted that the distortion pattern input is rotor-relative, i.e., as a time varying signal having a period of one rotor revolution. Figure 28 shows the static pressure distribution predicted through the first instability. The figure indicates that the distortion attenuation is dramatically reduced as the compressor is loaded beyond the first instability. This change in attenuation is due to the periodically stalled and unstalled operation of the diffuser, with the attendant major changes in diffuser losses and flow capacities.

Conclusion

This paper presents several applications of a mathematical model for compression system stability; reasonably accurate assessments of the impact of destabilizing influences on engine operability can be performed using such a mathematical model. It should be noted that the accuracy of the analysis is highly dependent on the accuracy with which the compressor airfoil loss and turning characteristics are represented in the analytical model.

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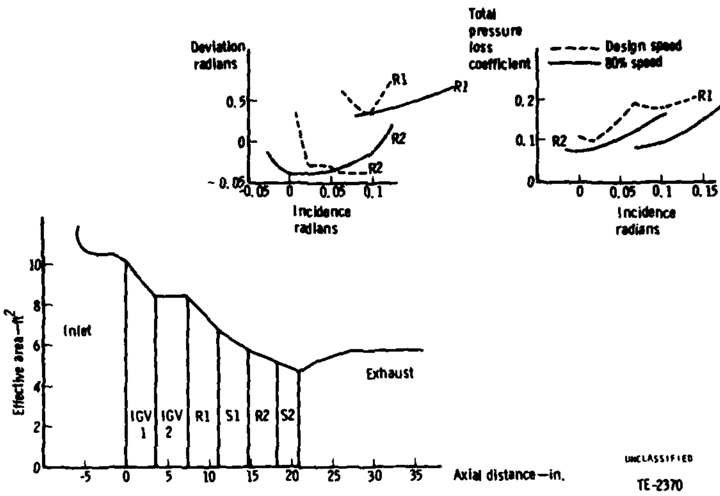


Figure 1. F101 fan geometry and stage characteristics [13].

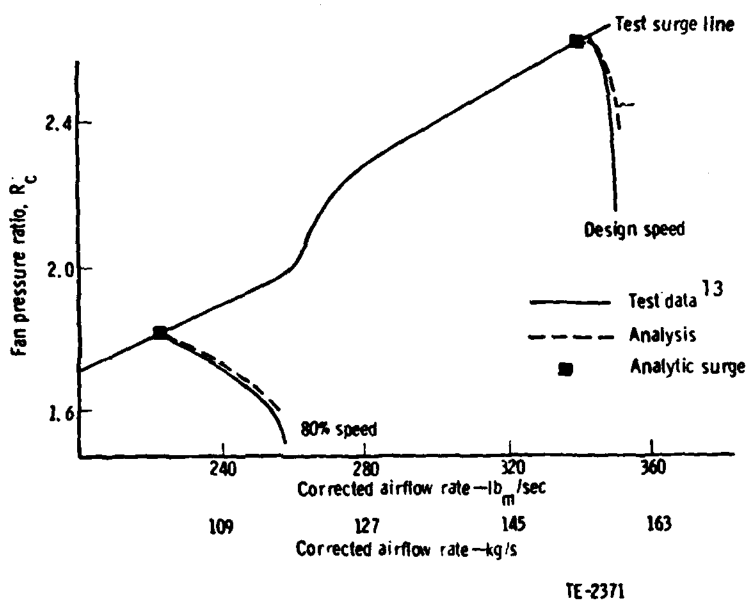


Figure 2. F101 fan clean inlet map--comparison of analysis and test data.

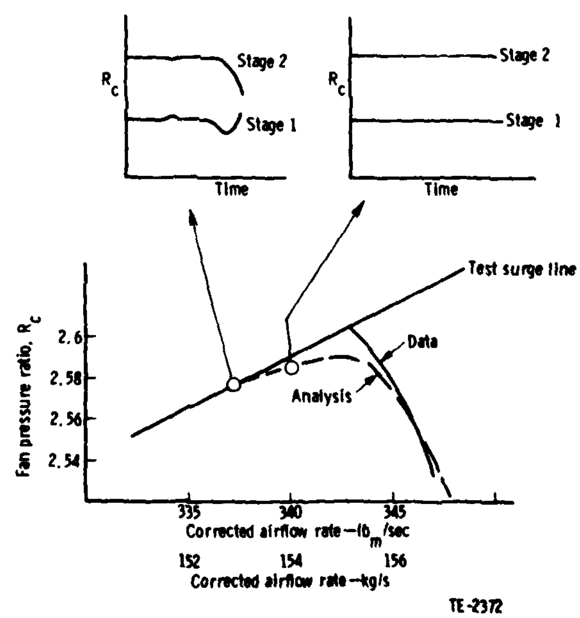


Figure 3. F101 fan analytical surge prediction at design speed.

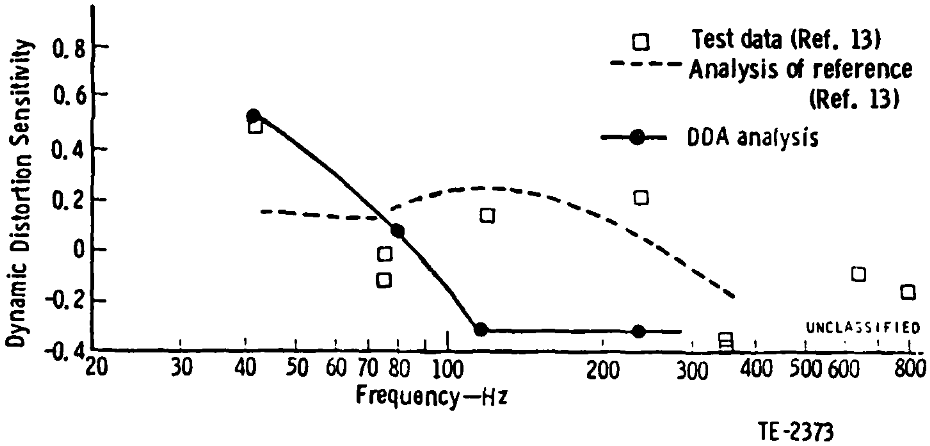


Figure 4. F101 fan planar wave distortion sensitivity at 80% speed [13].

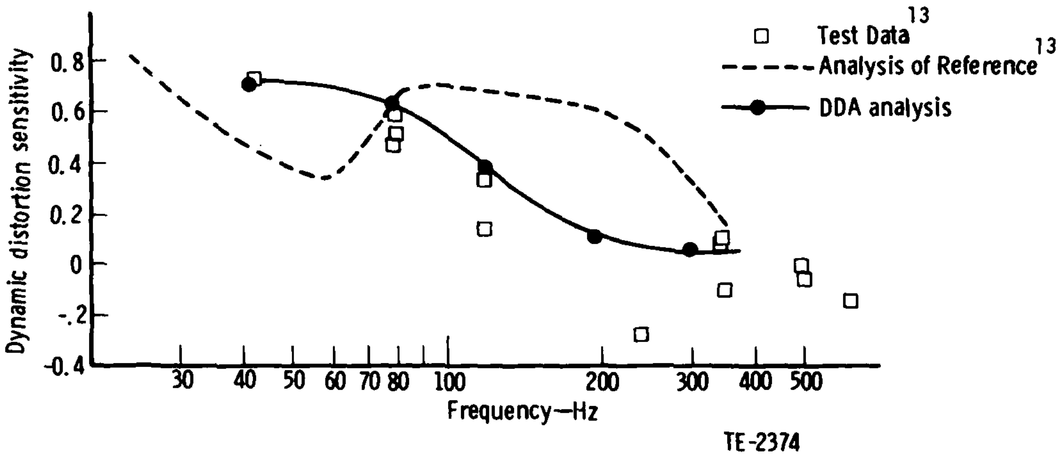


Figure 5. F101 fan planar wave distortion sensitivity at 100% speed [13].

$$\Delta PRS = \left(\frac{(P_r)_{\text{surge undistorted}} - (P_r)_{\text{surge distorted}}}{(P_r)_{\text{surge undistorted}}} \right)_{Wc\ =\ \text{constant}}$$

$$\text{Dynamic distortion sensitivity} = \Delta PRS \Big/ \left(\Delta P_{T_{IGV}} \Big/ P_{T_{IGV}} \right)$$

Where, $\Delta P_{T_{IGV}}$ = total pressure half amplitude at the fan IGV inlet

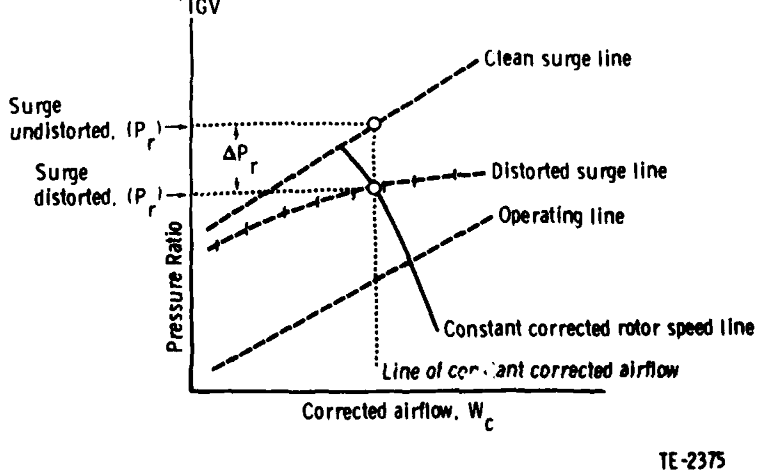


Figure 6. Definition of dynamic distortion sensitivity [13].

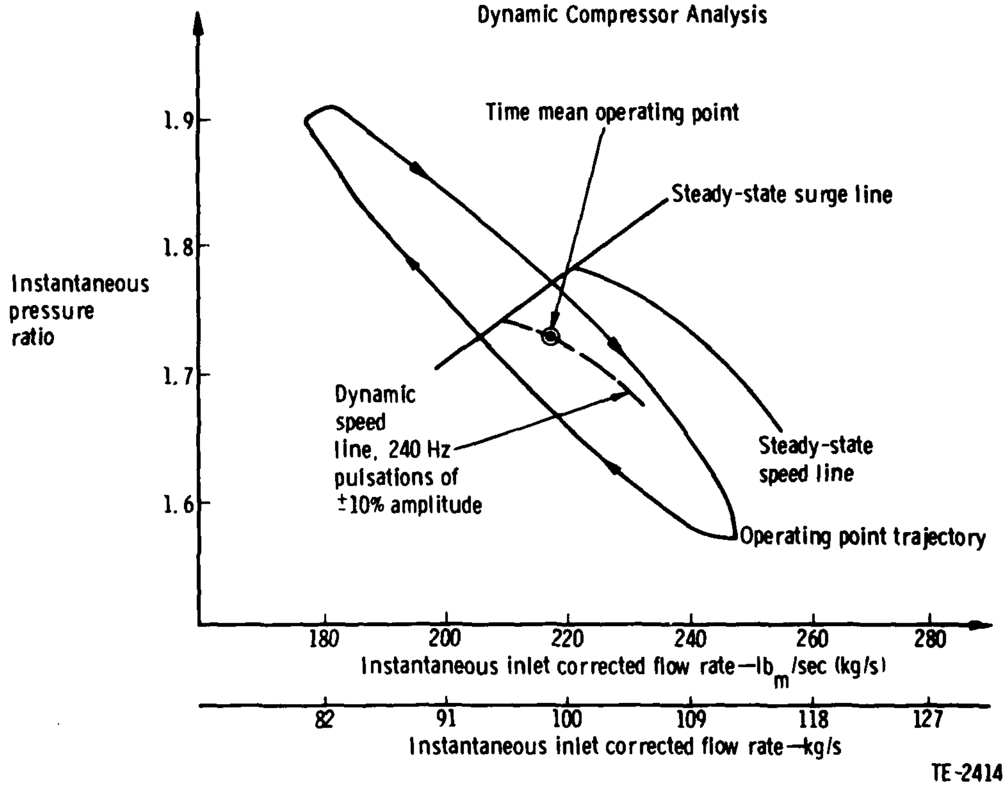


Figure 7. F101 fan instantaneous operating point at 80% speed with ±10% inlet pressure pulsations at 240 Hz, stable point.

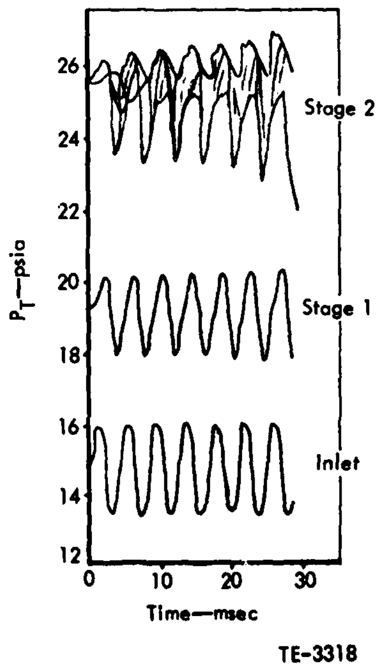


Figure 8. F101 fan total pressure distribution versus time at 80% speed with ±10% inlet pressure pulsations at 240 Hz, surge point.

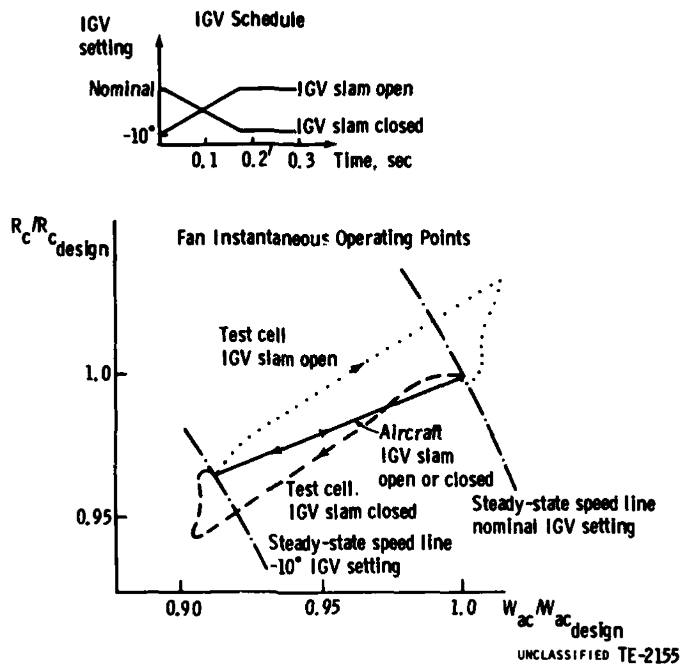


Figure 9. Inlet guide vane reset transient for a V/STOL fan.

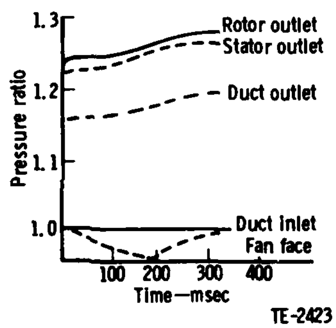


Figure 10. Total pressure ratio distribution for a test cell installation of a V/STOL fan undergoing an inlet guide vane transient (IGV slam open).

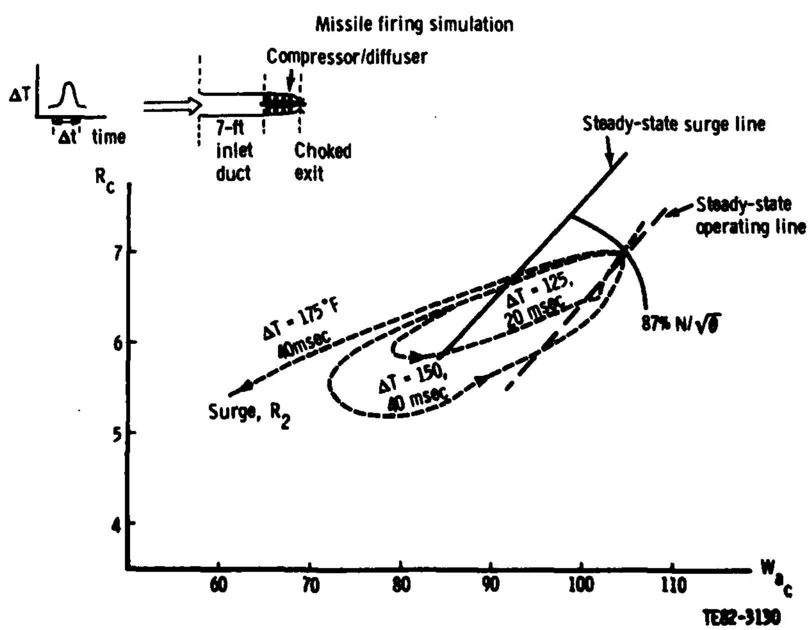


Figure 11. Compressor operating point trajectories.

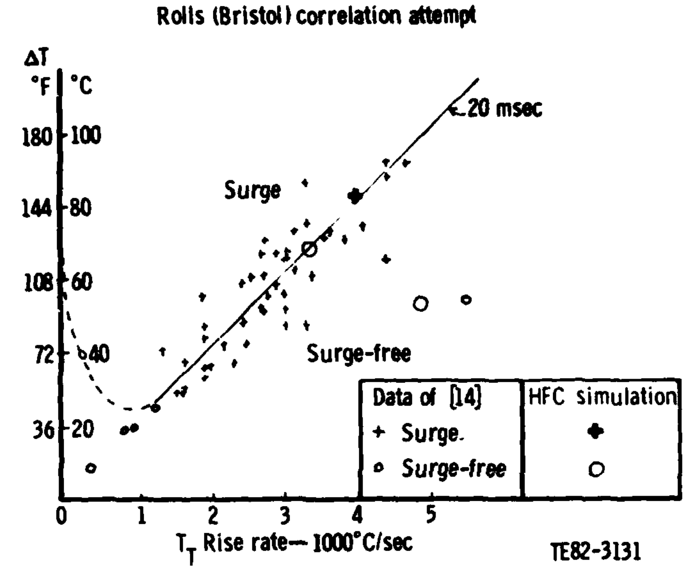
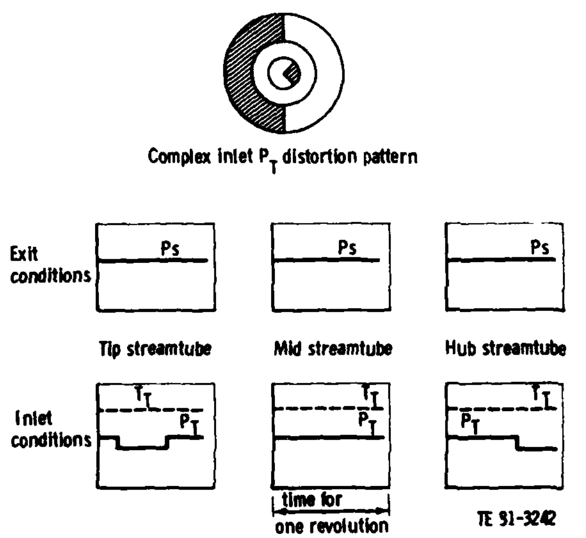
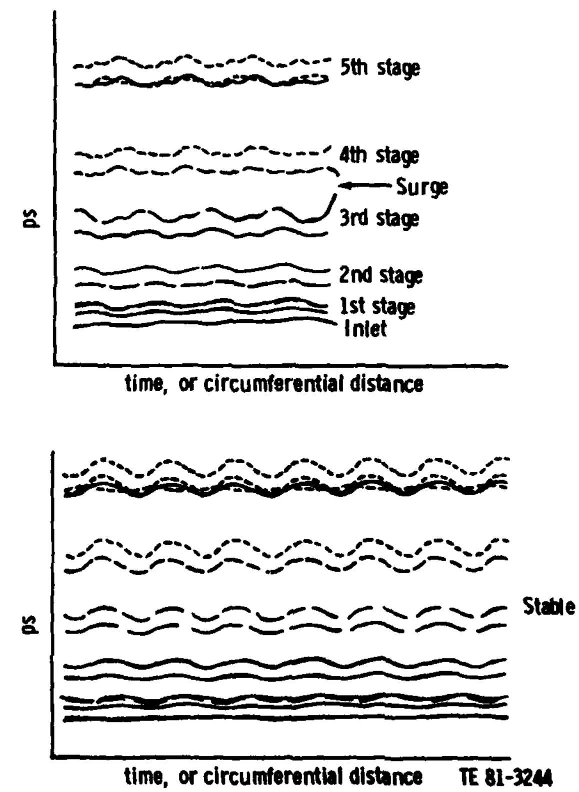
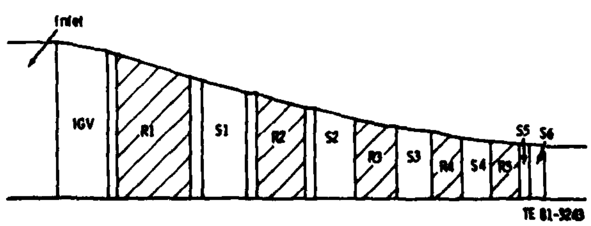


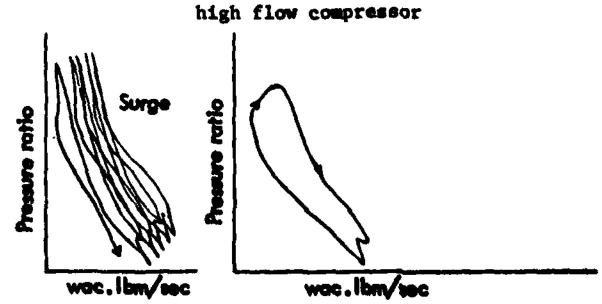
Figure 12. Analytic results compared with data of Hercock and Williams [14].





(b) Static pressure variation through the high flow compressor





(a) High flow compressor single streamtube model

(c) Operating point excursions of the high flow compressor

Figure 13. Analytical model of a complex flow.

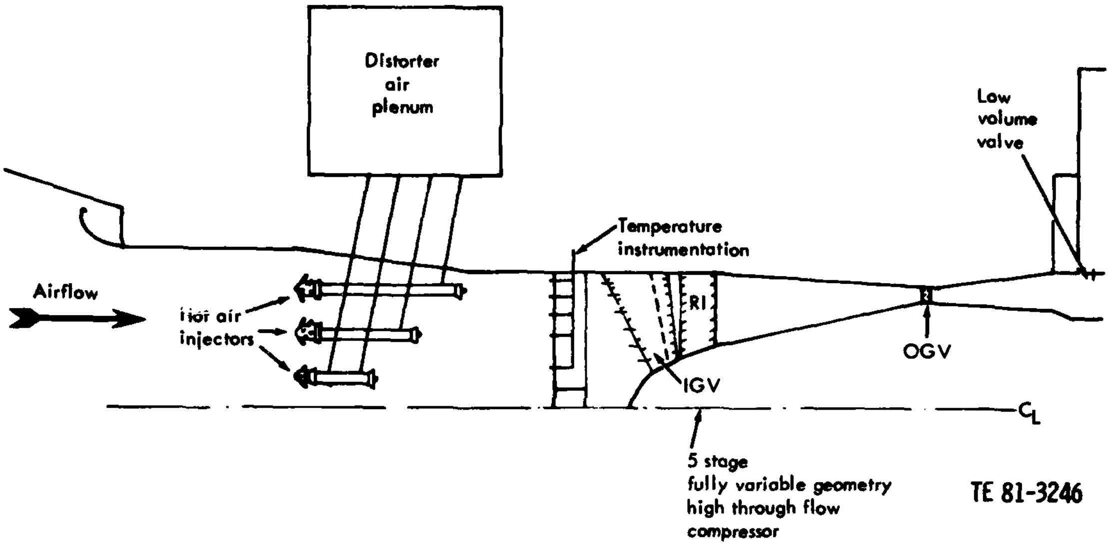


Figure 14. Compressor inlet distortion test configuration.

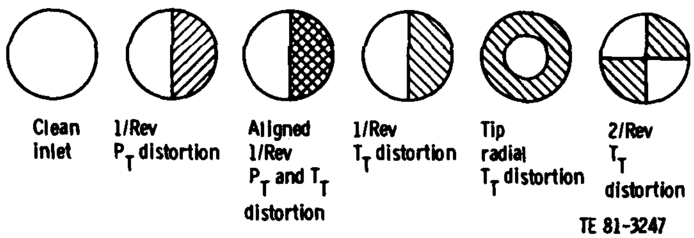


Figure 15. Inlet distortion patterns.

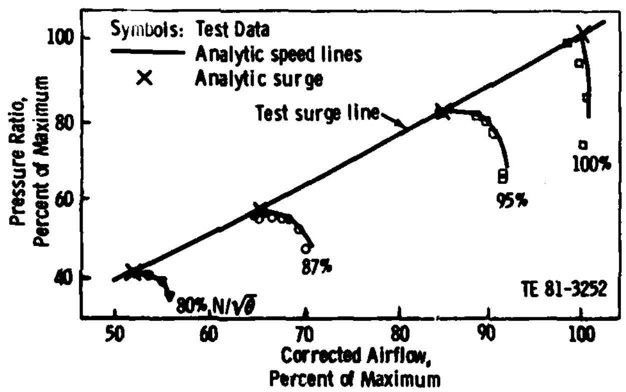


Figure 16. Clean inlet map (data uncertainty within symbols).

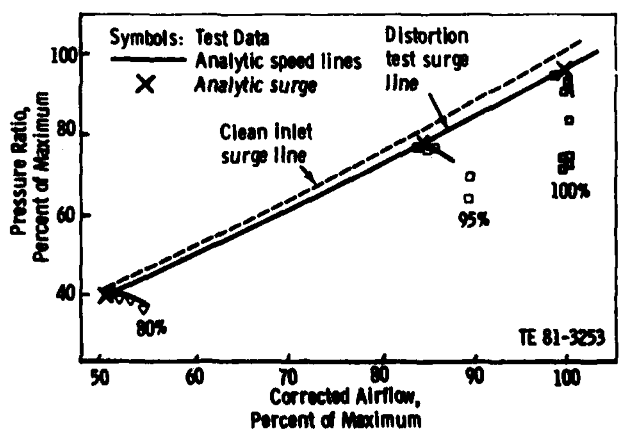


Figure 17. 1/rev P_T distortion (data uncertainty within symbols).

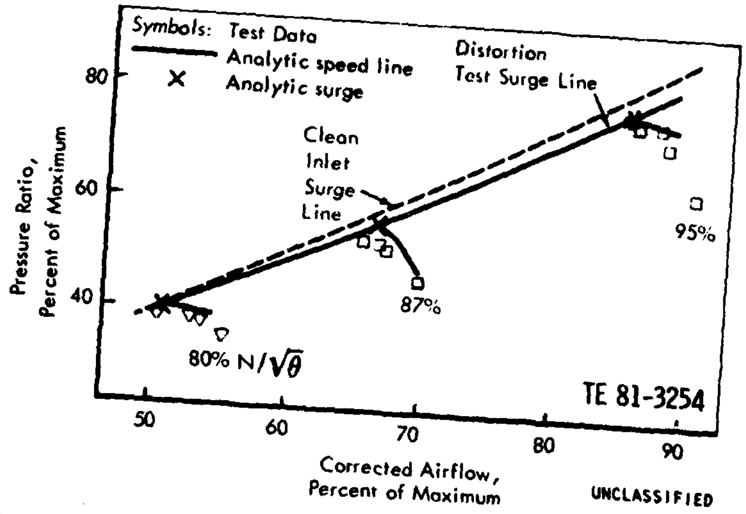


Figure 18. 1/rev T_T distortion (data uncertainty within symbols).

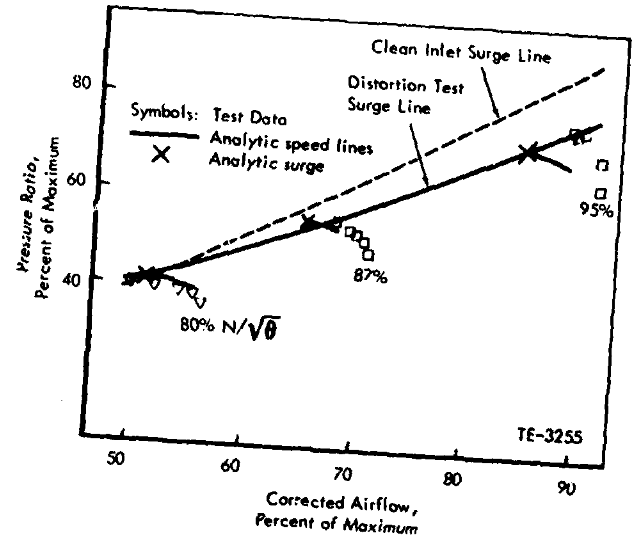


Figure 19. Aligned P_T and T_T distortion (data uncertainty within symbols).

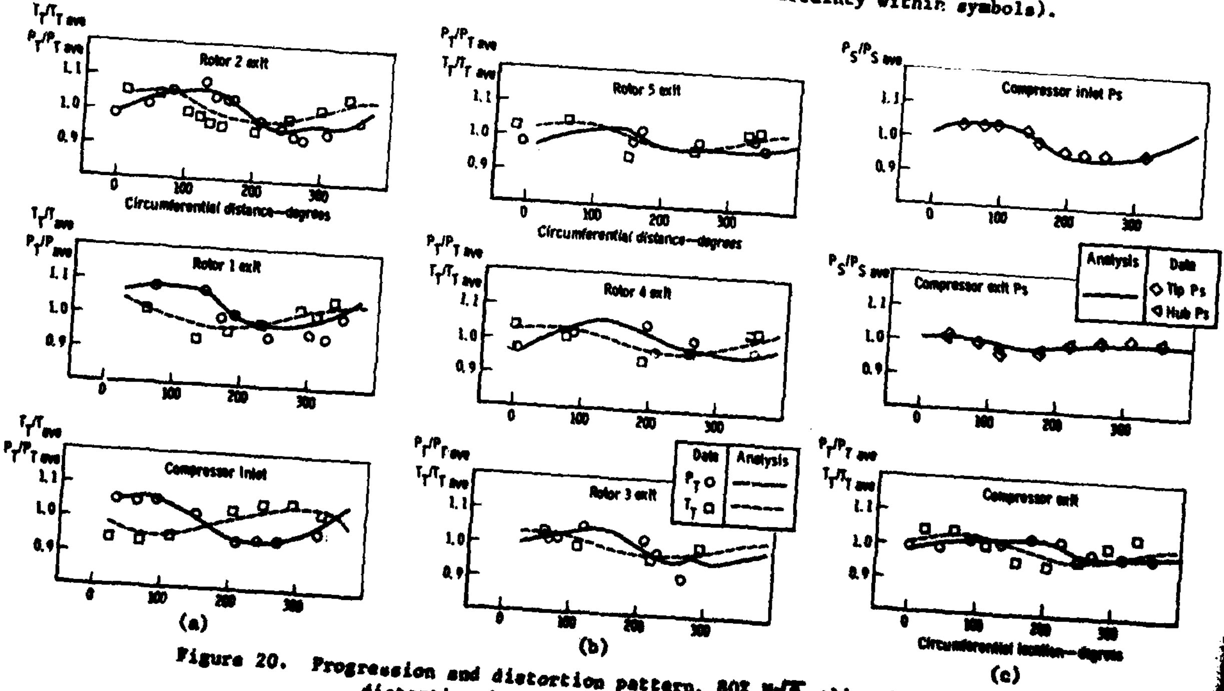


Figure 20. Progression and distortion pattern, 80% $N/\sqrt{\theta}$, aligned P_T and T_T distortion (data uncertainty within symbols).

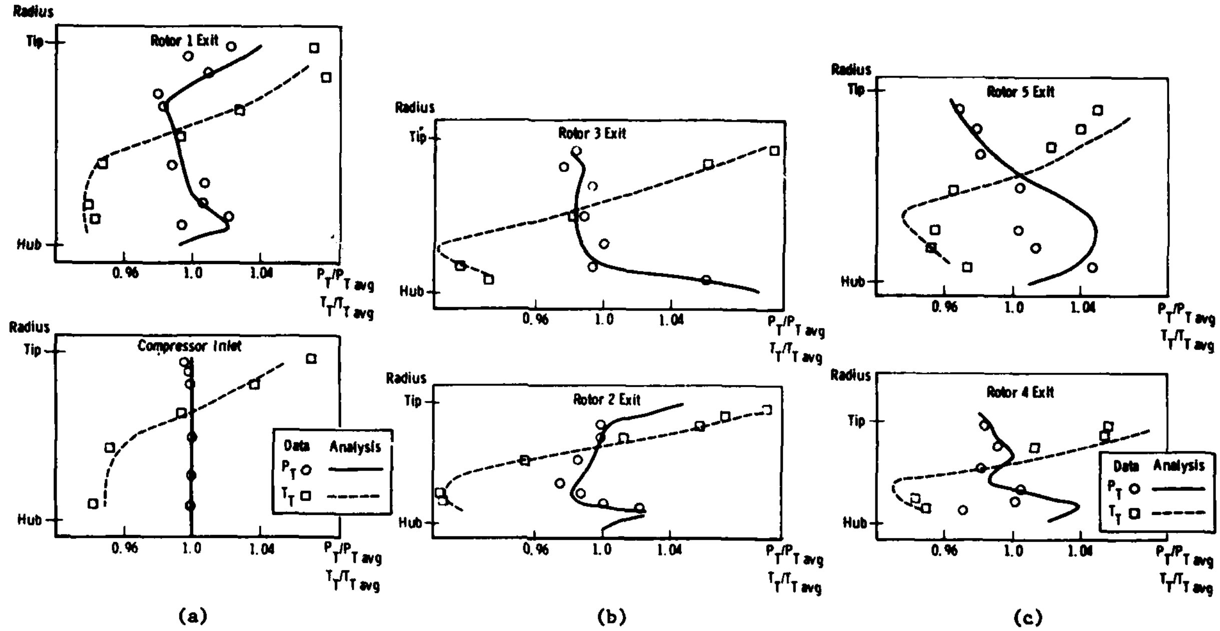


Figure 21. Radial temperature and pressure distribution, 95% $N\sqrt{\theta}$ (data uncertainty within symbols).

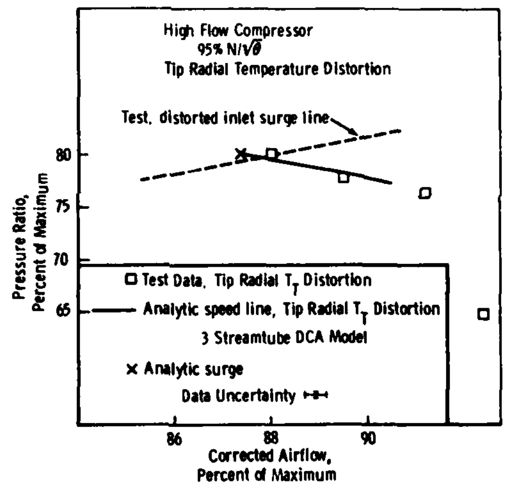


Figure 22. Predicted and measured speed line and stability limit, tip radial T_T distortion, 95% $N\sqrt{\theta}$.

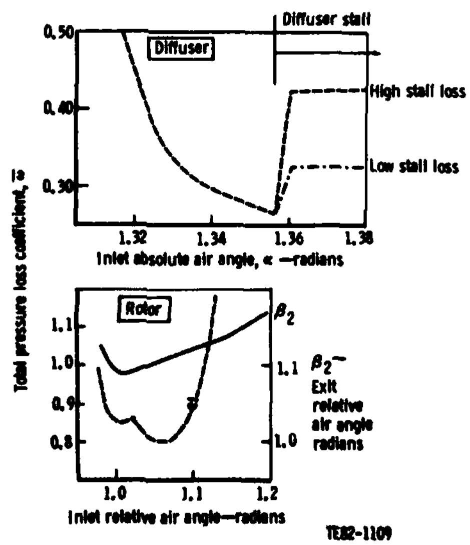


Figure 23. CX38 first-stage characteristics.

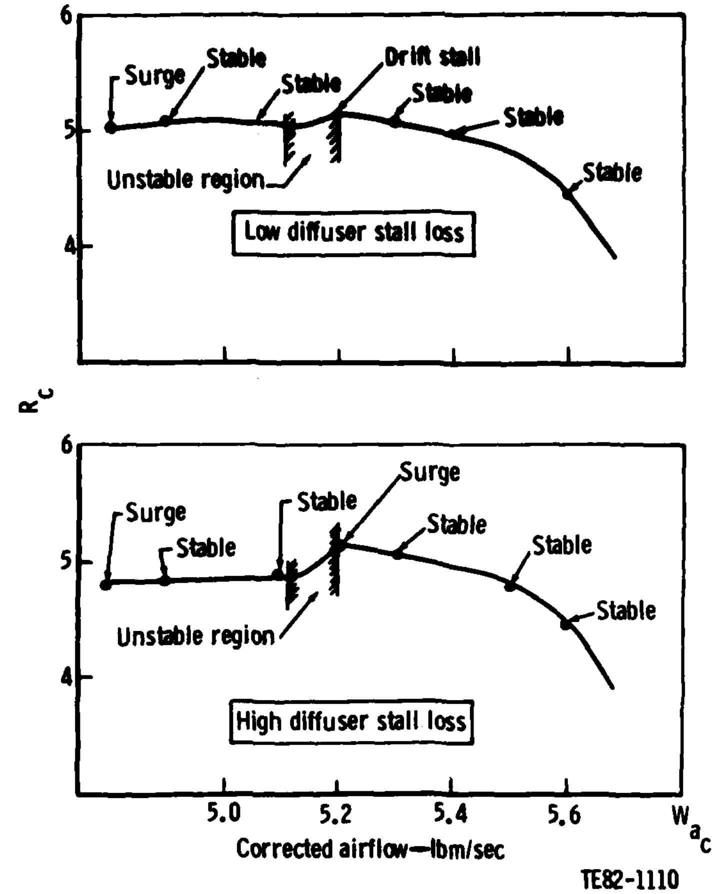


Figure 24. CX38 first-stage stability limits.

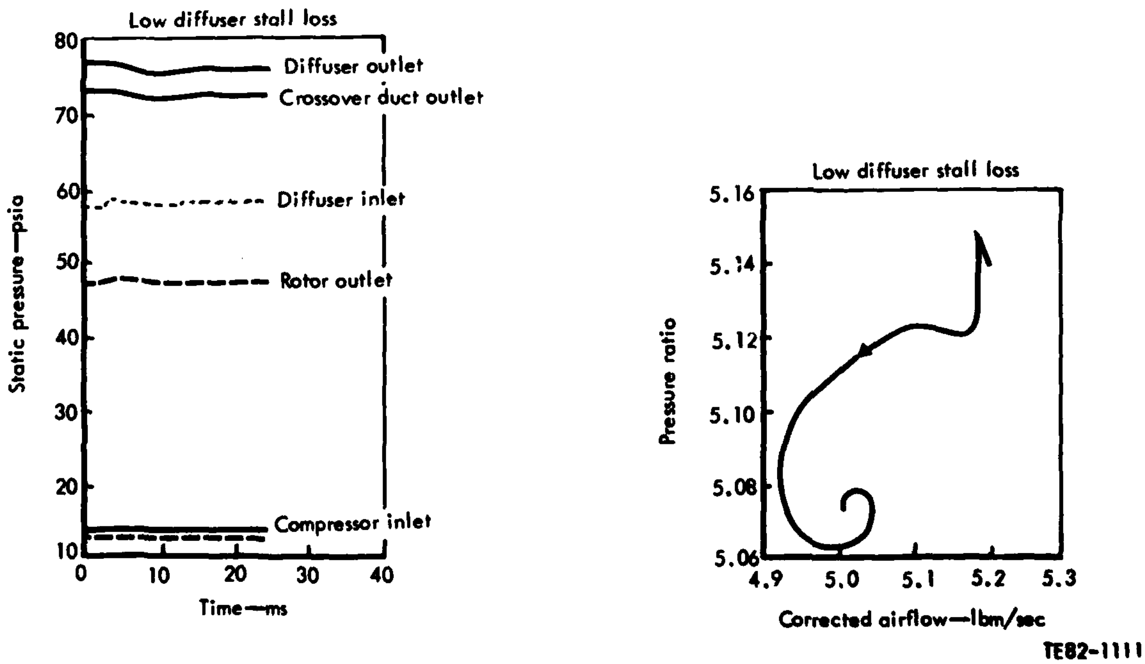


Figure 25. Predicted interstage static pressure distribution (left) and transient operating point (right) for CX38 first stage--low diffuser post-stall loss.

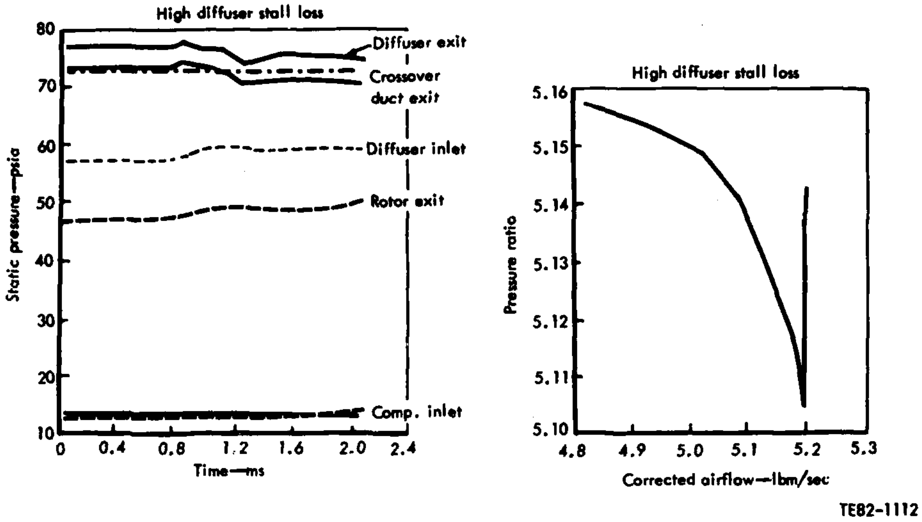


Figure 26. Predicted interstage static pressure distribution (left) and transient operating point (right) for CX38 first stage--high diffuser post-stall loss.

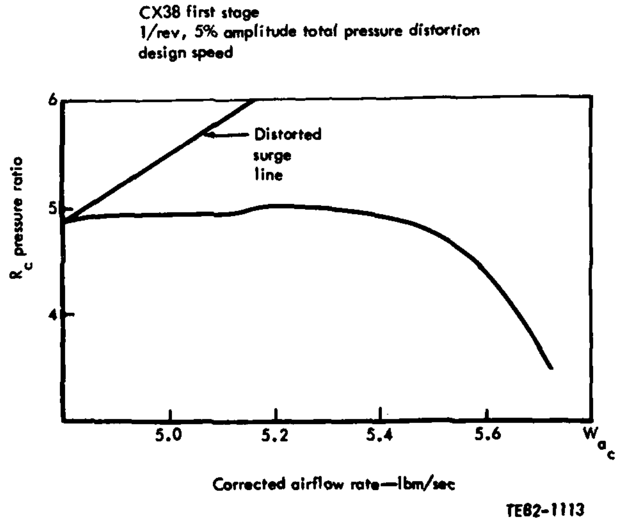


Figure 27. Predicted stability of CX38 first-stage centrifugal compressor with 1/rev circumferential inlet distortion.

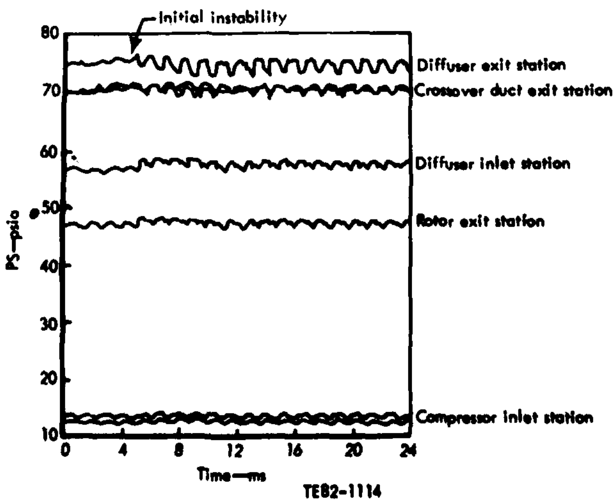


Figure 28. Predicted interstage static pressure distribution for first stage as a compressor is loaded beyond the initial instability.

MODELLING COMPRESSION COMPONENT STABILITY CHARACTERISTICS

- EFFECTS OF INLET DISTORTION AND FAN BYPASS DUCT DISTURBANCES -

by

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SUMMARY

This paper presents two extensions to a quasi-one-dimensional, time-dependent-aerodynamic-stability, digital-computer model. These extensions permit studying turbofan compression systems where, in one case, radial variations are important and, in the other, where circumferential variations are important. The basic equations used in the radial flow-redistribution model and in the circumferential flow-redistribution model are presented. The features of this computer model which lead to inherent numerical stability are discussed. The turbofan system which is being modelled in each case is described, and the manner in which a solution is obtained is presented. The capabilities of the radial flow-redistribution model are illustrated by examining the propagation characteristics of a pulse such as might be initiated by augmentor "hard lights." Also, the effect of splitter location on pulse propagation is illustrated. The capabilities of the circumferential flow-redistribution model are illustrated by coupling it with a parallel compressor model of a fan component to obtain an estimate of the loss in surge pressure ratio due to a 180° inlet total-pressure distortion. The estimates obtained using both a parallel-compressor model and the circumferential flow-redistribution parallel-compressor model are compared with test data.

SYMBOLS

A	Area
C	Velocity
L	Length
M_t	Mach Number, $M_t = 2\pi Nr_2/a'_{t1}$ where a'_{t1} is the blade row inlet speed of sound evaluated with the relative total temperature.
P_d	Dynamic Pressure, $P_T - P$
P_T	Total Pressure
S	Specific Entropy
T	Temperature
V	Volume
W	Mass Flow Rate
g_o	Gravitational Constant
r	Radius
q	Kinetic Pressure, $q = 1/2\ \rho C^2$
t	Time
ρ	Density

Subscripts

r	Radial Direction
z	Axial Direction
θ	Tangential Direction

Superscripts

'	Relative Frame of Reference
——	Volume Averaged Quantity
i	Axial Face of Volume
j	Tangential Face of Volume
k	Radial Face of Volume

1. INTRODUCTION

During the last decade, significant effort has been expended to develop computer simulations which model some of the destabilizing influences on turbine-engine, axial-flow, compression systems. Destabilizing influences, such as inlet distortion during aircraft maneuvers and pressure pulses during afterburner lights, can severely impact the handling characteristics and transient behavior of engines installed in modern combat aircraft.

Various approaches have been taken to modelling these destabilizing influences (References 1-5). The results reported in this paper are based on logical extensions to the one-dimensional, time-marching method of Reference 5 and result in options to the General Electric Aerodynamic Stability Program (AEROSTAP) that are conceptually simple, easy to use, and especially economical for studies involving high frequencies (up to 200 Hz, Reference 6). In this paper, two recent options that have been added to AEROSTAP are discussed.

The first option permits radial redistribution of axisymmetric flows upstream and downstream of compression components. This option is particularly useful for analyzing the time-dependent behavior of flows in large, high-bypass-ratio turbofan engines in which the blade characteristics vary significantly along the span of a fan blade. One case of special interest is the study of augmentor-induced pulses and the manner in which they propagate upstream through the fan duct, through the splitter/gooseneck region, and through the fan hub and tip regions and the high-pressure-ratio compressor causing concomitant losses in surge pressure ratio.

The second option allows for circumferential redistribution of flow in volumes upstream and downstream of a compression component and in volumes within a compression component that are large. One particular use of this model is to couple it with a parallel compressor model for improved estimates of losses in surge margin due to the effect of circumferential distortions on such parameters as inlet total pressure, inlet total temperature, exit static pressure, or any combination thereof.

2. MATHEMATICAL FORMULATION OF MODELS

The governing equations for the circumferential and radial flow-redistribution models are derived from the mass, momentum, and energy conservation equations that have been integrated once over space to provide a set of equations applicable to lumped volumes and suited for studying time-dependent flows in compression systems. Conceptually, the compression system is divided axially into volumes such that each blade row (rotor or stator) is contained within a volume. The blade-free volumes upstream and downstream of a compressor or large blade-free volumes contained within a compression component are divided such that no volume is longer than the longest blade row. In this way, the frequency-response capability of the model is maintained as high as possible and is governed by the longest blade row.

In the radial direction, the flow is divided only into as many annuli as there are derived blade characteristics to represent blade performance. Typically, this is either two or three annuli, that is, the hub and tip or the hub, midspan, and tip annuli.

Circumferentially, the flow annulus can be divided into as many sectors as desired. Typically, the compressor is divided into no more than 12 sectors because it is assumed that a compressor blade must reside in a distorted region for approximately 30° if it is to have an effect on the aerodynamic stability of a compressor.

While the conservation equations are necessary to effect a solution, they are not in themselves sufficient to effect a stable solution in normal operating regimes and to predict without artifices the occurrence of aerodynamic instabilities indicative of the presence of surge or rotating stall even when the thermal and caloric equations of state and appropriate boundary conditions are provided. Sufficiency is provided by introducing one of the thermodynamic TdS relationships. When this relationship is combined with the energy equation, the resulting relationship gives the balance between the entropy-storage term, entropy fluxes, and an entropy-production term. Detailed derivations of these equations are given in References 5 through 9.

The resulting equations are given below in the form they are used in each option to obtain solutions to compression-component stability problems:

Radial Redistribution

Continuity

$$\frac{\partial \bar{\rho}}{\partial t} = \frac{1}{V}\left[(W_z^i - W_z^{i+1}) + \left\{ W_r^k - W_r^{k+1} \right\} \right]$$

Axial Momentum

$$\frac{\partial \bar{W}_z}{\partial t} = \frac{g_o}{L_z}\left[\frac{W_z^i C_z^i}{g_o} - \frac{W_z^{i+1} C_z^{i+1}}{g_o} + \left\{ \frac{W_r^k C_z^k}{g_o} - \frac{W_r^{k+1} C_z^{k+1}}{g_o} \right\} + (P^i A^i - P^{i+1} A^{i+1}) - P_M (A^i - A^{i+1}) + F_z \right]$$

Radial Momentum

$$\frac{\partial \bar{W}_r}{\partial t} = \frac{g_o}{L_r}\left[\frac{W_r^k C_r^k}{g_o} - \frac{W_r^{k+1} C_r^{k+1}}{g_o} + \frac{W_z^i C_r^i}{g_o} - \frac{W_z^{i+1} C_r^{i+1}}{g_o} + \frac{2\pi \bar{W}_\theta \bar{C}_\theta}{g_o} + \pi (P^k - P^{k+1})(r^k + r^{k+1}) L_z \right]$$

Entropy Balance

$$\frac{\partial \overline{\rho S}}{\partial t} = \frac{1}{V}\left[W_z^i S^i - W_z^{i+1} S^{i+1} + \left\{ W_r^k S^k - W_r^{k+1} S^{k+1} \right\} + S_F \right]$$

where $\bar{W}_\theta$ can be evaluated from

$$\bar{W}_\theta = \bar{\rho}\, \bar{C}_\theta L_z L_r$$

Circumferential Redistribution

Continuity

$$\frac{\partial \bar{\rho}}{\partial t} = \frac{1}{V}\left[W_z^i - W_z^{i+1} + \left\{ W_\theta^j - W_\theta^{j+1} \right\} \right]$$

Axial Momentum

$$\frac{\partial \bar{W}_z}{\partial t} = \frac{g_o}{L_z}\left[\frac{W_z^i\, C_z^i}{g_o} - \frac{W_z^{i+1}\, C_z^{i+1}}{g_o} + \left\{ \frac{W_\theta^j\, C_z^j}{g_o} - \frac{W_\theta^{j+1}\, C_z^{j+1}}{g_o} \right\} + P^i A^i - P^{i+1} A^{i+1} - P_M (A^i - A^{i+1}) + F_z \right]$$

Circumferential Momentum

$$\frac{\partial \bar{W}_\theta}{\partial t} = \frac{g_o}{\bar{r} L_\theta}\left[\frac{r^i W_z^i C_\theta^i}{g_o} - \frac{r^{i+1}\, W_z^{i+1}\, C_\theta^{i+1}}{g_o} + \frac{r^j W_\theta^j C_\theta^j}{g_o} - \frac{r^{j+1}\, W_\theta^{j+1}\, C_\theta^{j+1}}{g_o} + \bar{r}\, (P^j - P^{j+1})\, A^j \right]$$

Entropy Balance

$$\frac{\partial \overline{\rho S}}{\partial t} = \frac{1}{V}\left[W_z^i\; S^i - W_z^{i+1}\, S^{i+1} + \left\{ W_\theta^j\, S^j - W_\theta^{j+1}\, S^{j+1} \right\} + S_F \right]$$

The appropriate set of the above equations is used to solve for the gross, two-dimensional, redistribution characteristics of the flow in blade-free volumes depending on the symmetry that can be assumed. In volumes that contain blades, the main features of the flow are assumed to be represented by considering only changes in the axial direction. The flow is considered to be quasi-one-dimensional, and the above equations are reduced to one-dimensional flow relationships by omitting the terms within the braces and omitting either the radial or the circumferential momentum equation.

In order to solve the above equations, additional relationships must be supplied for the mean pressure (P_M), the axial blade force acting on the fluid (F_Z), and the entropy production term (S_F). The mean pressure is the integrated pressure acting on the annulus walls of a control volume element. For volumes containing blades, the mean pressure is taken to be

$$P_M = \frac{1}{3}\left[P_L + 2\,P_H\right]$$

where L and H refer to the lower and higher values, respectively, of the volume inlet or exit static pressures. For volumes that do not contain blades and are swirl free, P_M can be obtained from isentropic relationships. For volumes that are blade free but where significant swirl is present, a correlation for P_M must be developed on an individual-compression-component basis. The axial blade force is obtained by resolving the forces acting on a fluid element and includes contributions from the energy-producing torque term and drag force term. The axial blade force (See Figure 1) is represented by the following relationship:

$$F_z = F_T \tan\beta_\infty - F_{Dz}$$

where the tangential force F_T is obtained from the generalized Euler Turbine Equation and is written as

$$F_T = \frac{2}{g_c}\left[\frac{r_2\,W_2\,C_{u2} - r_1\,W_1\,C_{u1}}{r_1 + r_2}\right] + V\,\frac{\partial(\overline{\rho C}_u)}{\partial t}$$

and the lift direction angle β_∞ *is defined as*

$$\beta_\infty = (\beta_1 + \beta_2)/2 + \beta_c$$

The lift-direction-correction angle β_c is introduced to accounted for the fact that the lift direction is not the simple average of the inlet and exit-air angles and to assure consistency between the steady-state solutions obtained from the energy and momentum equations.

The drag force F_{Dz} can be written as

$$F_{Dz} = F_D/\cos\beta_\infty$$

where

$$F_D = \bar{\omega}'\,\frac{P_1}{P_{t1}'}\,\frac{P_d'}{q_1'}\,A_{1\beta}\,q_1'$$

and the relative total-pressure loss coefficient is defined by the relation

$$\bar{\omega}' = \frac{\left.P_{t2}'\right)_{Ideal} - \left.P_{t2}'\right)_{Actual}}{P_{t1}' - P_1}$$

The entropy-production term S_F in the absence of heat transfer at the boundaries is written as

$$S_F = W\,R\,\ln\left[\frac{\left.P_{t2}'/P_{t1}'\right)\ Ideal}{\left.P_{t2}'/P_{t1}'\right)\ Actual}\right]$$

where

$$\left.\frac{P_{t2}'}{P_{t1}'}\right)_{Ideal} = \left\{1 + \frac{\gamma-1}{2}\,M_t^2\left[1 - \left(\frac{r_1}{r_2}\right)^2\right]\right\}^{\frac{\gamma}{\gamma-1}}$$

and

$$\left.\frac{P'_{t2}}{P'_{t1}}\right)_{Actual} = \left.\frac{P'_{t2}}{P'_{t1}}\right)_{Ideal} - \bar{\omega}' \left\{ 1 - \left[\frac{1}{1 + \frac{\gamma-1}{2} (M'_1)^2}\right]^{\frac{\gamma}{\gamma-1}} \right\}$$

As can be ascertained from the above equations, the deviation of the compressor from ideal behavior is described by the blade characteristics: the relative total-pressure loss coefficient, ($\bar{\omega}'$) and the deviation angle (δ). These are derived from undistorted inlet-flow data where the compression component has been throttled along speed lines to the point of instability. The relative total-pressure-loss coefficient and the deviation angle are input as functions of the incidence angle (i) for each speed line for which there are data. Other input to the model includes the inlet and exit metal angles, the inlet and exit flow areas, and the pitch line radii for each blade row.

3. SOLUTION TECHNIQUE

An explicit-solution technique is employed to solve the appropriate set of equations. The technique is based upon a Taylor series expansion correct to second order for the following state variables:

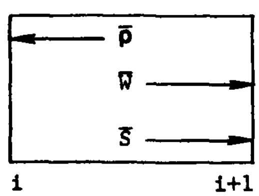
One-Dimension:	$\bar{\rho}$ $\bar{W}_z$ $\overline{\rho S}$
Circumferential Redistribution:	$\bar{\rho}$ $\bar{W}_z$ $\overline{\rho S}$ $\bar{W}_\theta$
Radial Redistribution:	$\bar{\rho}$ $\bar{W}_z$ $\overline{\rho S}$ $\bar{W}_r$

The solution technique is illustrated for one variable: the volume-averaged density. The other variables are handled in a similar manner.

$$\bar{\rho}\ (t+\Delta t) = \bar{\rho}\ (t) + \frac{d\bar{\rho}}{dt}\ \Delta t + \frac{d^2\bar{\rho}}{dt^2}\ \frac{(\Delta t)^2}{2}$$

The term $\bar{\rho}$ (t) can be obtained from initial conditions or the previous time step; $\frac{d\bar{\rho}}{dt}$ can be obtained from the equation describing continuity; $\frac{d^2\bar{\rho}}{dt^2}$ can be obtained by differentitating the equations for the derivatives of velocities with respect to time.

The solution technique is completed by providing a method for relating station values of a parameter to the volume-averaged values. While many schemes have been used during the course of developing the Aerodynamic Stability Program, only one method has always avoided numerical instabilities, especially when large gradients are involved. This has been dubbed the "Transmittal of Properities" technique and is illustrated by the following schematic for a control volume.



$\bar{\rho} = \rho_i$

$\bar{W} = W_{i+1}$

$\bar{S} = S_{i+1}$

The sets of equations and the solution technique have resulted in solutions that are numerically extremely stable for all types of steady-state and time-dependent boundary conditions. Numerical stability is attributed to three aspects of this modelling technique: (1) Entropy has been explicitly included through a Tds thermodynamic relationship, (2) the Transmittal of Properities interpolation technique, and (3) the second-order Taylor series time-marching solution. Error accumulation in this model is minimized due to the Taylor series solution technique. This feature has been demonstrated by allowing the model to sit at an equilibrium operating point on a speed line for many thousands of time steps. The operating point never significantly moves from the equilibrium operating point. Any small calculational perturbations cause the first and second derivatives of the state variables to become other than zero. These derivatives act like "restoring forces" and cause the model to return to the equilibrium operating point with the derivatives concomitantly attaining zero value once again.

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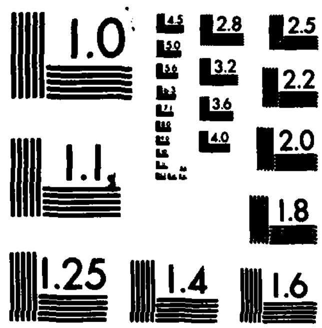

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

4. FAN BYPASS DUCT DISTURBANCES

The radial equations of Section 2 together with the solution technique of Section 3 were used to study the propagation of pressure pulses in a turbofan compression system and associated ducting. In particular, the dual-rotor compression-system model described in Reference 8 was used to study the propagation of large-amplitude pressure pulses such as might be created by "hard lights" resulting from poor augmentor control. This simulation (Reference 10), although of a nonaugmented turbofan compression system and fan duct, was used to illustrate the propagation characteristics of a pulse and the effect of splitter location on transmitted pulse amplitudes throughout the turbofan compression system.

The turbofan compression system and ducting that was modelled is shown schematically in Figure 2; the rotating blade rows are depicted by heavy shading. This compression system was divided into volumes as shown in Figure 3. It should be noted that no blade row is longer than the fan rotor blade row. Radial redistribution is allowed ahead of the fan, in the interaxial gap between the rotor and the stator, in the stator blade row, and between the fan stator and the leading edge of the splitter. No radial redistribution was allowed in the fan rotor blade row. The dividing surface is chosen approximately consistent with the stagnation streamline associated with the steady-state, initial-operating-point solution. From this point in time the surface is fixed, and all radial flows are calculated with respect to this surface. The tip and hub regions of the fan rotor were represented by undistorted inlet-flow-derived characteristics; undistorted, inlet-flow, pitch-line characteristics were used to represent the compressor. For boundary conditions, constant inlet total pressure, constant total temperature, and zero radial velocity were imposed while the fan duct exited to constant static pressure and the combustor-exit flow was assumed to be choked. No heat was added in the combustor although this is not a restriction. After initializing the time-dependent portion of the program using a steady-state solution at high-speed conditions ($NF/\sqrt{\theta_2} = 104.3\%$ and $NC/\sqrt{\theta_{2.5}} = 99.1\%$), the calculations were allowed to settle out prior to introducing an 8% amplitude, half-sine-wave, static-pressure pulse at the fan duct exit.

4.1 PULSE PROPAGATION CHARACTERISTICS

Figure 4 shows the propagation of the static-pressure pulse as a function of time at several locations in the compression system.

The ordinates were chosen as $\Delta P/P$ where this ratio is defined as the difference between the instantaneous static pressure and the local steady-state value of static pressure divided by the local steady-state value of static pressure. This choice of ordinate permits studying the amplification/attenuation characteristics of the compression system. The first column of graphs shows the pulse spreading in time and travelling up the duct with a propagation velocity equal to the acoustic velocity minus the local convective velocity. Determination of the velocity can best be accomplished by using the peaks of the pulses. Further, it is noticed that the pulse takes significantly longer to travel through the fan rotor than through a similar length of blade-free ducting. This is a result of work being done in the fluid resulting in significant acoustic impedance changes and resulting in acoustic-transmission characteristics that differ significantly from ducts with homentropic flow. The second column of graphs illustrate that the pulse takes a few tenths of a millisecond longer to reach the fan-hub discharge compared to the fan-tip discharge. The last column of graphs shows the pulse travelling downstream through the core compression system from the splitter leading edge (core inlet) to the compressor inlet guide vanes (IGV) and to the compressor discharge. At all stations, the amplitude of the pulse is attenuated to about one-half of the initial amplitude and to approximately one-sixteenth of the initial value by the time it reaches the discharge of the compressor.

4.2 SPLITTER PROXIMITY EFFECT

The location of the splitter was varied from its normal design position downstream of the fan stator ($L/L_D = 1.0$), ranging from the fan rotor exit ($L/L_D = -2.0$) to well downstream ($L/L_D = 3.0$) of its normal position. The locations that were studied are shown schematically in Figure 5.

The effect of the location of the splitter was quantified in terms of an amplitude ratio, ϕ. This amplitude ratio was defined as the maximum local value of $\Delta P/P$ ($\Delta P/P$ was defined in Section 4.1) divided by the maximum value of $\Delta P/P$ at the fan duct exit where the pulse was introduced. Plots of the amplitude ratio as a function of L/L_D are shown for the fan discharge, the fan inlet, and the compressor inlet in Figures 6, 7, and 8 respectively. Although the pulse is attenuated throughout the compression system ($\phi < 1.0$), dramatic reductions in the transmitted pulse amplitude at the fan-hub inlet and discharge are obtained by moving the splitter up to the stator leading edge. Similar results are obtained at the core inlet (retained in original position) and at the compressor IGV inlet. One finding of this study, although not unanticipated, was that the pulse attenuation was not as low at the compressor IGV as it was at the core inlet. This difference is purely a geometrical effect due to a slight flow-area decrease in the gooseneck region from the core inlet to the compressor IGV inlet. This result illustrates the necessity for following geometric details closely if a computer model is to give predictions that will simulate test data closely.

5. INLET DISTORTION CALCULATIONS

The parallel-compressor technique is a standard method for obtaining estimates of the loss of surge pressure ratio due to circumferential inlet total-pressure and total-temperature distortions. While the method gives reasonably accurate estimates of the loss of surge pressure ratio for speed lines that are steep over their entire flow range (such as for high-pressure-ratio compressors at high speed), the method consistently overestimates the loss of surge pressure ratio at low speeds where the constant-corrected-speed lines are much flatter. Examples of these types of prediction characteristics are clearly shown in Reference 7. Possible reasons for this overprediction of the loss in surge pressure ratio for the flatter speed lines are discussed in Reference 9. It was argued that since this overprediction of loss in surge pressure ratio could not be attributed to flow redistribution within modern, axial-flow compressors due to the low axial-gap-to-radius ratios, then it must be in large part due to circumferential redistribution of the flow upstream of the compressor and downstream of the compressor prior to attaining a circumferentially uniform static-pressure state.

The circumferential redistribution equations of Section 2 describing the blade-free volumes in the flow fields upstream and downstream of the compressor were coupled to a multisector, parallel-compressor model. The downstream circumferential flow field terminated in a uniform-static-pressure flow at some convenient point and became one-dimensional from that point on.

The results, allowing for upstream and downstream circumferential redistribution, showed little improvement in the ability to predict the loss in surge pressure ratio due to circumferential total-pressure distortion for the multistage compressor over the speed range, although the upstream and downstream circumferential flow-redistribution elements (axial flow, tangential velocity, and static pressure) appeared to be calculated quite properly. Two possible explanations were offered. One, the compressor being used for this study had an IGV that held Rotor 1 incidence angle constant regardless of the magnitude of any upstream tangential velocity components that were generated. The second explanation rested on the fact that the dynamic response of the rotors entering and leaving distorted regions had not been modelled. Although this is certainly a factor for distortions with angular extents considerably less than 180°, it should not be a particularly strong effect for 180° sector distortions.

The circumferential-redistribution/parallel-compressor model can give much improved results over the basic parallel-compressor model as subsequent studies have shown. The results from one of these studies are discussed in the following paragraphs. In particular the fan used in the Reference 8 studies was modelled using pitch-line characteristics and six sectors to predict the effect of a 180° total-pressure distortion where the (PT Max - PT Min)/PT Avg value was 0.2. This fan has no IGV and should show the greater effect of tangential velocity components at the rotor entrance. Circumferential redistribution was allowed upstream of the fan and downstream of the stator. Since frequency response was not an issue when throttling the fan in a quasi-steady manner, but economical computation was, the fan component system was divided into volumes as shown in Figure 9 with no volume being longer than seven inches (the fan rotor volume was three inches long). This representation provided a geometrically correct simulation of the component test rig from the inlet distortion screen to the flow-controlling discharge valves. Aerodynamic instability was assumed to occur when the flow in any 60° sector became inherently unstable (see References 7 and 11 for a discussion of aerodynamic and numerical stability using this model).

The results of various throttling studies conducted at $NF/\sqrt{\theta_2} = 100.0\%$ are shown in Figure 10 with the test data shown as background. This single-stage fan at this speed has a constant-corrected-speed characteristic that is considerably flatter with significant flow rollback compared to multistage fans and compressors at similar speeds. This speedline shape provides a significant test of the capability of the circumferential-redistribution model to provide improved estimates of the loss in surge pressure ratio due to distortion as compared to parallel-compressor estimates. The standard-parallel-compressor, average-operating point at instability obtained using two 180° sectors is shown as the solid circle. It is evident that the parallel-compressor model significantly overpredicts the loss in surge pressure ratio due to the imposed 180° circumferential total-pressure distortion.

The complete fan component test rig (Figure 9) was modelled using the circumferential-redistribution parallel-compressor model. The annulus was divided into six equal sectors with three of the sectors located within the low-total-pressure region of the 180° square-wave distortion; thus, the remaining three sectors were located in the high-total-pressure region. The average performance of the compressor at the predicted instability limit is shown as the "open triangle" on Figure 10. Clearly, use of the redistribution model significantly improves the accuracy of the surge pressure ratio loss prediction. This improvement occurs for two major reasons: (1) More realistic compressor entrance conditions in terms of entrance incidence angle are established and (2) each parallel-compressor sector exits to a more realistic static-pressure distribution.

In the course of accomplishing the combined circumferential-redistribution parallel-compressor model simulations, two interesting results were obtained. The first interesting result illustrated the effect of model configuration. If the discharge flow was modelled only to the normal-production-nozzle discharge plane (Figure 9), then results represented by the open square and labelled the "abbreviated redistribution model" were obtained. Although these results are an improvement relative to the test configuration-redistribution model results, it is not clear at this time why this result was obtained. That a difference occurred is, of course, due to the change in the discharge static-pressure field. The implications of this finding are that if one expects to reproduce test results, it is necessary to execute a geometrical replication of the test vehicle and that one should not necessarily expect to obtain test-vehicle values for surge pressure losses due to distortion in production configurations that are significantly different from the configuration tested. This is an extension to the findings of Greitzer (Reference 12) who showed that different results would be obtained depending on whether a compressor with distorted-inlet-flow conditions discharged to a nozzle, a constant-area duct, or a diffuser.

The second interesting result illustrated the importance of the manner in which the circumferential-redistribution-parallel compressor model is initialized. It was found that when the model was initialized using a steady-state, parallel-compressor solution, it caused the time-dependent model to find a stable solution near the parallel-compressor solution with the result that unrealistic discharge velocity components were established to satisfy the solution. Instead, it was found that a significantly different and improved solution was found when all sectors were initialized to a uniform inlet-flow condition and distortion was imposed by increasing it until the desired value was obtained. It is possible that the multistage compressor simulations (Reference 7) with distortion would be improved if the simulations were started from the uniform inlet condition rather than a standard, parallel-compressor solution. Time has not permitted investigation of this possibility.

6. CONCLUSIONS

Two analytical models have been described. Each has particular capabilities for estimating engine-handling characteristics due to external disturbances depending on the flow symmetry under consideration. The radial flow-redistribution model demonstrated the significant effect that splitter location can have upon the magnitude of an afterburner pulse being transmitted to the high-pressure compressor.

The circumferential flow-redistribution parallel-compressor model provides significantly more accurate estimates of the loss of surge pressure ratio due to inlet distortion as compared to the standard parallel compressor. This finding was demonstrated for a constant-corrected-speed line which has considerable flow rollback. These improved estimates of loss of surge pressure ratio are due to the more realistic inlet and exit flow conditions that are presented to the fan; that is, tangential velocities are allowed to develop in the inlet, and the simulated exit-static-pressure field more closely represents the test-condition aerodynamics.

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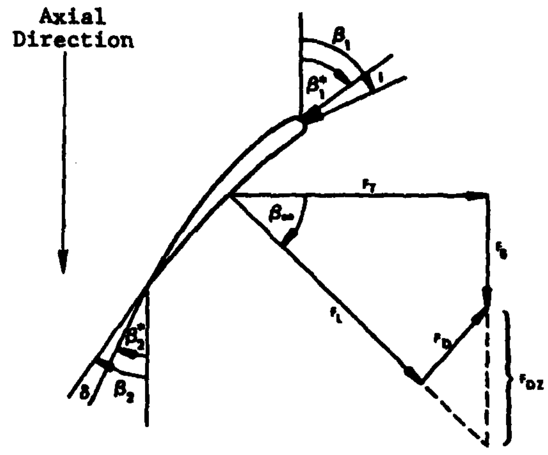

Figure 1. Blade Force and Blade Angle Relationships.

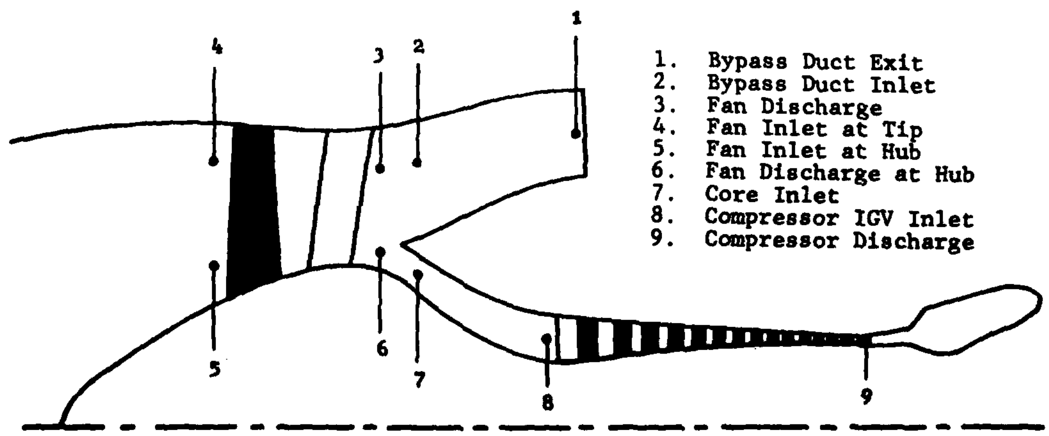

Figure 2. Schematic Cross-Section of Turbofan Compression System.

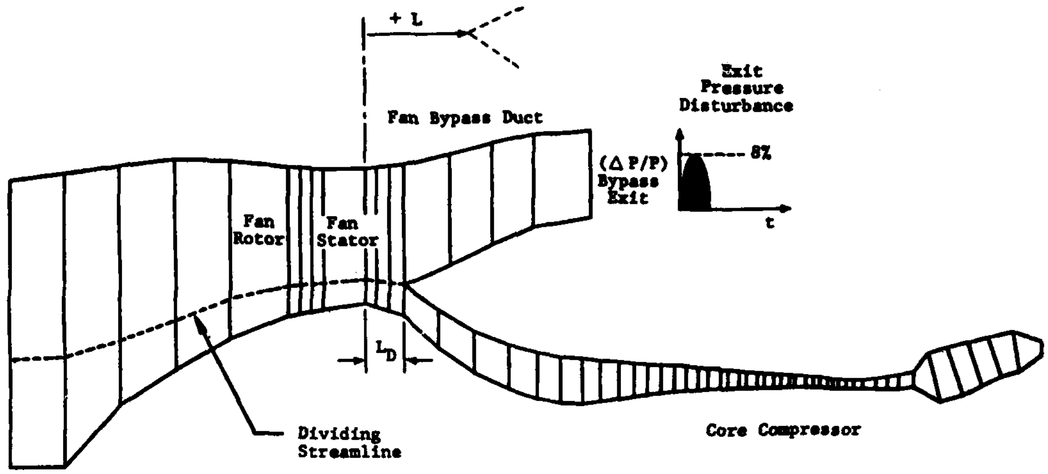

Figure 3. Schematic of Model Volumes.

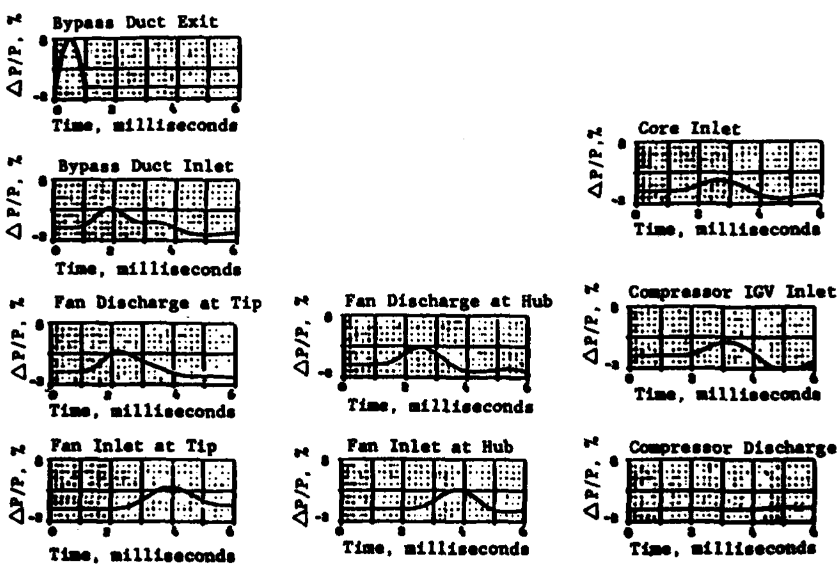

Figure 4. Pulse Propagation Characteristics in Turbofan Compression System and Ducting.

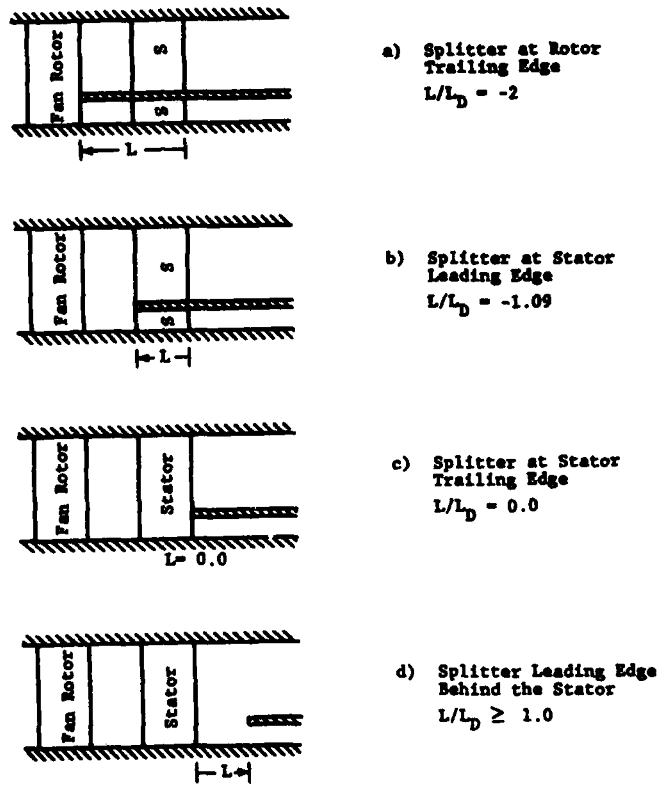

Figure 5. Schematic of Splitter Location Relative to the Fan.

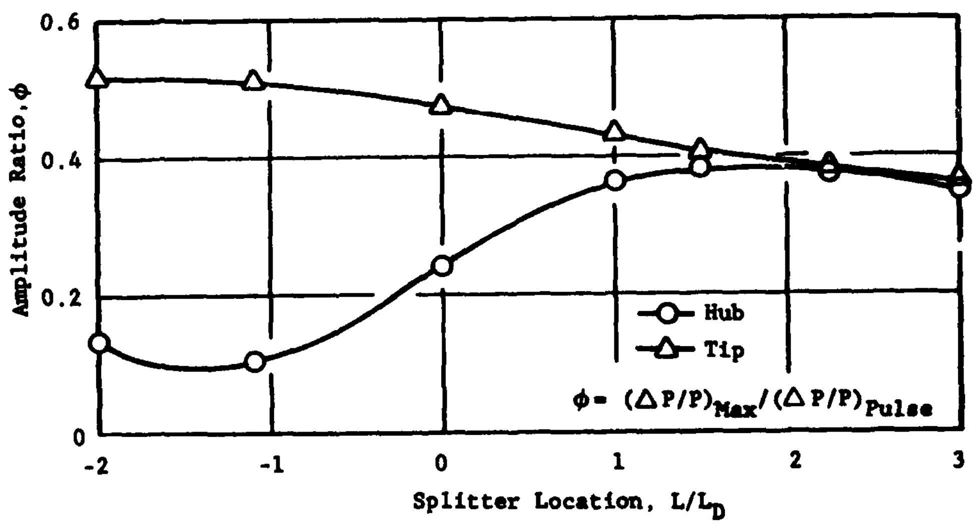

Figure 6. Amplitude Ratio at Fan Discharge.

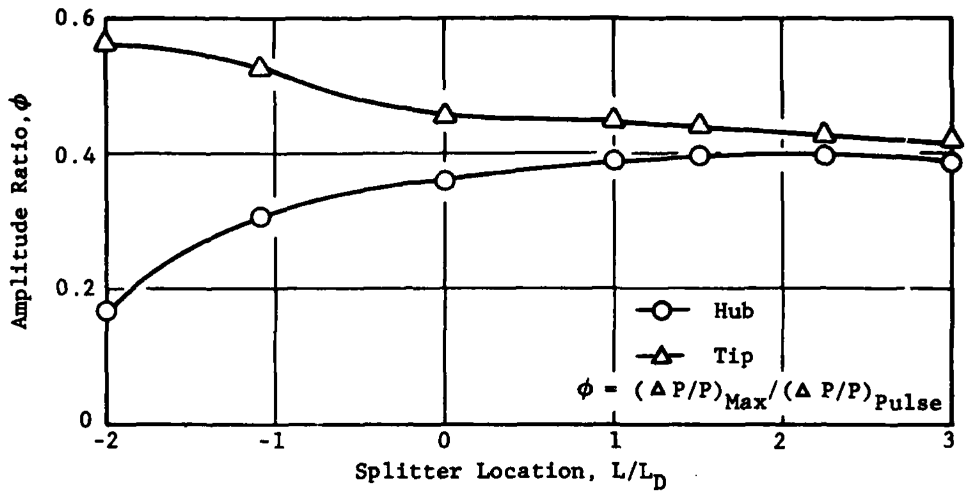

Figure 7. Amplitude Ratio at Fan Inlet.

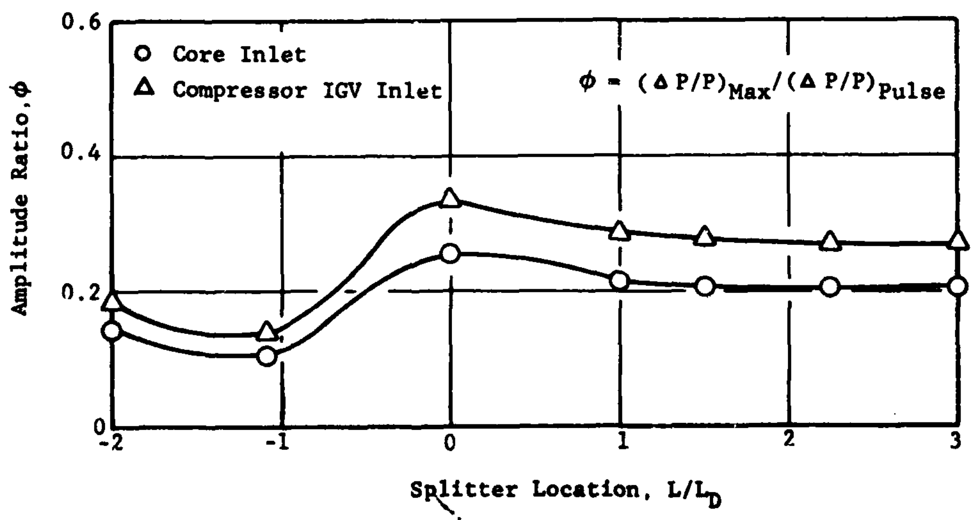

Figure 8. Amplitude Ratio at Different Locations in Core Inlet.

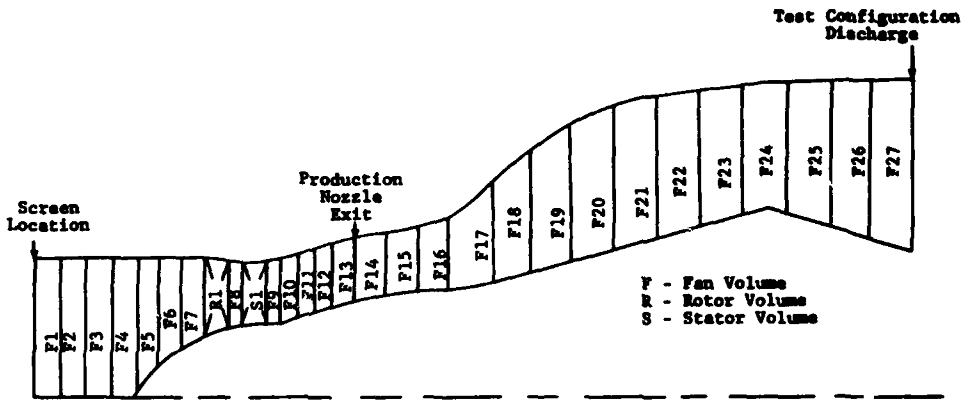

Figure 9. Schematic of Volumes For Circumfrential-Flow Redistribution Model.

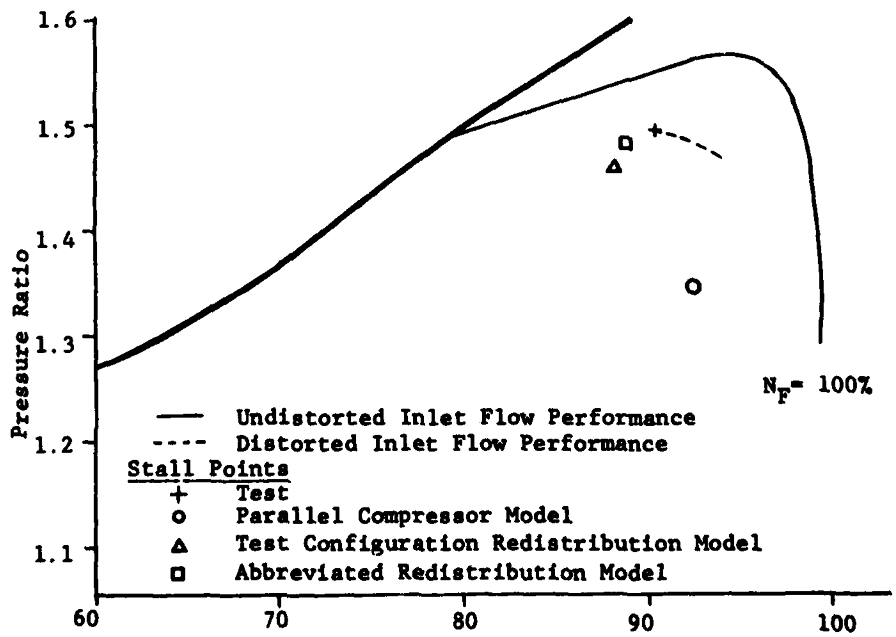

Figure 10. Inlet Flow Distortion Simulation Results.

DISCUSSION

de Richemont, Fr:

Based on your experience, what is the inlet flow distortion index which best correlates with the loss of fan stall margin?

Author's Reply:

Our analysis indicates that indices which describe both the circumferential and radial variations of inlet total-pressure distortion are required. These must then be combined using an appropriate supposition function if a good correlation is to be established with the measured loss of surge pressure ratios. Efforts have been made to standardize these indices for the industry and for further details I would refer you to Automotive Engineers Aerospace Recommended Practice ARP 1420 entitled "Gas Turbine Engine Inlet Flow Distortion Guidelines" and the soon to be published supporting document Society of Automotive Engineers Aerospace Information Report AIR 1419 entitled "Inlet Total Pressure-Distortion Considerations for Gas-Turbine Engines".

H. Stetson, US:

(1) Is your model adaptable to the low bypass ratio military engine and have you made similar predictions for such an engine?

(2) What experimental verification do you have to confirm your assumption of no radial flow shift through the rotating blade?

Author's Reply:

(1) The model is readily adaptable to low bypass military engines and has been used successfully in such cases. It should be noted, however, that we have not specifically carried out splitter location studies for such engines.

(2) It was purely an assumption and we would expect a radial flow shift through the blade as bypass ratio is changed. However, to some extent these effects are implicitly included since our clean-inlet-flow blade characteristics were obtained with proper bypass ratio at each point of each speed line for which we obtained throttling data.

J. Dunham, UK:

Please clarify your surge criterion. Was it

(a) the point at which the lowest-flow segment reached the undisturbed surge point or

(b) the point at which your model went unstable itself?

Did (a) and (b) agree?

Author's Reply:

The criterion for surge was the point at which throttling was terminated and an equilibrium operating point no longer would be maintained. The clean inlet flow model and the parallel compressor sector which went unstable agreed in pressure ratio and corrected flow at the stability limit point.

AN OVERVIEW OF ENGINE DYNAMIC RESPONSE
AND MATHEMATICAL MODELING CONCEPTS

by

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SUMMARY

The paper introduces the problem of the dynamic response of gas turbines based on fundamental concepts of aerothermodynamic matching of components, leading to a physical understanding of engine behaviour and the resulting control requirements. Although the basic principles are straightforward, implementation requires the use of mathematical models of varying degrees of sophistication. The development of advanced fighter aircraft requires careful integration of engine, control systems and aircraft installation and there must be free exchange of information between the various participants. The paper introduces the various approaches to modelling of engine response, considers the evolution of computing facilities and discusses the differing requirements of manufacturers and users.

SYMBOLS

m	air flow rate	I	polar moment of inertia
c_p	specific heat at constant pressure	$\dot{\omega}$	angular acceleration
T	temperature	N	rotational speed
η_m	mechanical efficiency	t	time
p	pressure	W_f	fuel flow
G	torque	τ	rotor time constant

INTRODUCTION

The problem of gas turbine response rates has been studied intensively for many years and specific requirements for civil aero engines are laid down by both FAA and the CAB. It is interesting to note that there are many applications of the gas turbine where response rates are not really an important problem, a good example being main propulsion units on gas turbine powered warships. The basic response rate of the engine is much faster than the response of the ship to power changes, and acceleration times from idle to maximum of 15 seconds are quite acceptable; significant reductions in acceleration time could be achieved, but only at the expense of excessive output torques and decreased engine life.

In the case of aircraft, however, rapid response rates are needed both for manoeuvrability and safety; the requirements for civil and military aircraft are quite different. In the case of civil aircraft the basic reason for specifying response rates is the requirement for baulked landing, but, apart from that, there is little reason for rapid throttle movements. Military aircraft, and fighters in particular, must be able to change power settings very rapidly in air combat situations and a large number of throttle excursions will be experienced during a typical flight. A further problem affecting fighters is the extremely rapid changes of altitude and Mach number which are encountered, leading to large changes in stagnation pressure and temperature at the engine inlet. During extremely rapid manoeuvres, intake distortion could result, causing a deterioration in the surge line which would adversely affect handling characteristics.

The manufacturer must be able to predict engine response rates early in the design program, and identify methods of controlling the engine to give adequate response without compromising engine or aircraft safety. No matter how extensive the development program, however, handling problems may not appear until the engine has been in service for some time; indeed, the same engine may encounter different problems in different aircraft due to changes in the overall propulsion systems.

Experimental and flight test investigations of handling problems are essential, but are very costly and potentially dangerous. The judicious use of mathematical models can give a deep insight into engine transient behaviour, and models of steadily increasing sophistication have become common in recent years. Properly used, mathematical models can significantly reduce the need for experimentation, but it must be clearly understood that they are a tool to help identify problems and cannot be seen as a substitute for engine/airframe testing.

This paper will restrict itself to giving an overview of the engine handling problem and offer some suggestions on the differences between the mathematical modelling requirements of the manufacturer and the user.

REQUIREMENTS FOR HIGH PERFORMANCE

Modern fighter engines require high specific thrust, good subsonic specific fuel consumption, thrust augmentation for take off, combat manoeuvring and supersonic operation and a high thrust to weight ratio. High specific thrust requires high values of turbine inlet temperature (TIT), while good subsonic fuel consumption requires a high pressure ratio. These requirements were orginally met using turbojets and large numbers are still in the NATO inventory, examples being the F104/J79 and F5/J85. New supersonic aircraft, however, are powered by low by-pass ratio turbofans, examples being the F15/F16/F100, F18/F404 and Tornado/RB199. The need for high pressure ratio is now met by the use of multi-spool configurations combined with variable stator vanes; the F100 and F404 are twin-spool and the RB199 is three-spool. The use of multiple spools reduces the pressure ratio required on each compressor, simplifying the aerodynamic design. The low by-pass turbofans with afterburning make use of a single exhaust; there is a direct flow path between the afterburner and the fan outlet and problems may be encountered during afterburner light up because of pressure pulses being transmitted upstream.

It should noted that the A10 ground attack aircraft uses a TF34 high by-pass turbofan, this configuration requiring a single stage fan plus booster stages and a very high pressure ratio single spool core using several stages of variable stators. Although the A10 is not subject to the same range of transients as a typical fighter, its engines must still be capable of very good throttle response.

Another example of a medium by-pass engine without afterburning in fighter aircraft is the Pegasus used in the Harrier. The handling characteristics are especially critical because of the VTOL nature of the Harrier, which requires considerable, and intermittent, bleed flows for aircraft stabilization.

BASIC PRINCIPLES

In order to study the transient behaviour of gas turbines, a prior requirement is the understanding of steady state behaviour. This must be determined by matching calculations, which make use of the compressor, turbine and nozzle characteristics. These methods are described in detail in (1) and will only be outlined here, for the simplest case of the single spool turbojet; station numbering used is shown in Fig 1. Typical component characteristics are shown in Fig 2.

Two basic requirements must be satisfied for equilibrium running; these being

(i) Compatibility of work, and
(ii) Compatibility of flow

For the gas generator of the jet engine, compatibility of work requires that the rate of doing work in the turbine must be equal to the work input for the compressor, thus

$$m_1 \, c_{p12} \, \Delta T_{12} = m_3 \, c_{p34} \, \Delta T_{34} \, \eta_m$$

To keep the analysis at its simplest, it will be assumed that $m_1 \approx m_3$ for an uncooled turbine. Because we are dealing with component characteristics which are presented in non-dimensional form, it is useful to express the compatibility of work in terms of non-dimensional groups to give

$$\frac{\Delta T_{12}}{T_1} \approx \frac{\Delta T_{34}}{T_3} \cdot \frac{T_3}{T_1} \cdot \frac{c_{p34}\eta_m}{c_{p12}} \tag{1}$$

Considering flow compatibility,

$$\frac{m_1\sqrt{T_1}}{P_1} = \frac{m_3\sqrt{T_3}}{P_3} \cdot \frac{m_1}{m_3} \cdot \frac{P_3}{P_2} \cdot \frac{P_2}{P_1} \cdot \sqrt{\frac{T_1}{T_3}} \tag{2}$$

It can be seen that both equations (1) and (2) include the temperature ratio T_3/T_1 and can be rewritten in the form

$$\frac{T_3}{T_1} = \frac{\dfrac{\Delta T_{12}}{T_1}}{\dfrac{\Delta T_{34}}{T_3}} \cdot \frac{c_{p12}}{c_{p34}\,\eta_m} \tag{1A}$$

$$\sqrt{\frac{T_3}{T_1}} = \frac{\frac{m\sqrt{T_3}}{P_3} \cdot \frac{P_3}{P_2} \cdot \frac{P_2}{P_1}}{\frac{m\sqrt{T_1}}{P_1}} \qquad (2A)$$

The method of solution is to find, by trial and error, a compressor operating point at a given rotational speed which gives the same value of T_3 from both equations (1A) and (2A); the compressor operating point obtained requires a particular value of turbine pressure ratio and temperature drop, which in turn fixes the turbine outlet flow function

$m\sqrt{T_4}/P_4$. The value of $m\sqrt{T_4}/P_4$ obtained must be in agreement with the value of $m\sqrt{T_4}/P_4$ at inlet to the downstream component, in this case the jet nozzle.

Having determined the operating point on the compressor characteristic, all thermodynamic parameters can now be determined, including fuel flow, thrust, compressor pressure ratio and engine pressure ratio. Repeating the process for a series of speeds, the well known steady running line is obtained and it becomes convenient to plot the significant parameters as functions of corrected speed as shown in Fig 3, with only fuel flow and TIT shown here for clarity of presentation.

The basic method used to predict steady state performance is the foundation for all predictions of transient performance. If a fuel flow greater than that required for steady running is supplied, the turbine inlet temperature will be raised, the net results being that the turbine output will exceed the compressor requirement and the excess torque will cause the rotor to increase speed. In its simplest form, it can be assumed that flow compatibility continues to be satisfied while work compatibility is not. The rotor angular acceleration can then be determined from

$$G_{excess} = I\,\dot{\omega}$$

or

$$G_{turbine} - G_{compressor} = \frac{2\pi}{60}\, I\, \frac{dN}{dt} \qquad (3)$$

The basic requirement of any mathematical model, then, is to continuously calculate the excess torque and integrate the resulting acceleration to determine the variation of rotor speed with time. It is clear that the model must be able to vary fuel flow in the same way as the control system if the simulation is to be useful.

It is instructive to consider, qualitatively, the effect of a sudden increase in fuel flow, with the compressor operating at a relatively low speed. Following the increase of fuel to the combustion system, the TIT will rise rapidly, causing a reduction in density at entry to the turbine; the increased volume flow rate discharging into a fixed area turbine nozzle results in an increase in pressure, forcing the compressor to operate at a higher pressure ratio and a reduced flow rate. These changes take place rapidly and it can be assumed that they occur before the rotor accelerates. In idealized form, then, the operating point follows the trajectory AB in Fig 4, and it can be seen that excessive overfueling can cause surge. Once the rotor starts to accelerate, a trajectory BC will be followed, and the turbine inlet temperature may be less than its limiting value. If the limiting turbine temperature is reached at C, the fuel control must be arranged to limit the temperature along CD until the maximum speed is reached. It can readily be seen that a trajectory A B' C' D defines the fastest possible acceleration between the two specified speeds and the acceleration capability is determined by the surge margin and over temperature capability. Thus rotor acceleration is determined primarily by the rate at which excess fuel can be supplied.

The behaviour of the high pressure spool of multi-spool gas turbines is the same as that of the simple jet engine, the explanation being that the HP spool can be considered to be equivalent to a simple jet engine subject to a ram pressure ratio and discharging into a fixed nozzle area, determined by the LP turbine stator area. The behaviour of the LP spool, however, is quite different and during transient operation the running line lies very close to the steady running line. Thus, during acceleration of a multi-spool engine the LP operating trajectory may be well removed from the surge line, an important consideration when intake distortion effects are likely to be serious. Conversely, when decelerations are considered the LP operating trajectory may run close to surge although the HP compressor is not troubled. Summing up, the multi-spool engine may encounter handling problems during either acceleration or deceleration; the only potential problem with decelerating a single spool jet engine is the possibility of flame out due to weak extinction.

THE PROPULSION SYSTEM OF A MODERN FIGHTER AIRCRAFT

The primary requirements of any modern fighter aircraft are speed and agility. Speeds are currently well into the supersonic flight regime and great strides have been made in the design of the airframe to reduce weight without compromising aircraft stability margins. No less attention has been paid to the engine itself and thrust-to-weight ratios have increased steadily over the past several decades.

The requirement to fly faster than the speed of sound has resulted in increasingly sophisticated inlet designs with more and more use of variable geometry. Similarily, since the need for supersonic speed is usually of quite short duration, the afterburner remains the only logical design approach; afterburners and variable nozzles have thus seen considerable development and the combination of inlet, engine and propelling nozzle represents a multi-variable propulsion system requiring very complex control systems to ensure efficient use of the system.

The problem of matching variable geometry inlets to the gas turbine has been discussed at length by many authors (2,3,4). Figure (5) is representative of a high performance inlet system and, as indicated, its primary purpose is to establish a shock system such that it minimizes total pressure loss. It is, thus, a sophisticated supersonic diffuser whose flow characteristics at minimum pressure loss may not match the swallowing capacity of the engine. If the engine cannot accept the flow, the shock system will naturally take up a different position with sufficient pressure loss to reduce the flow to an amount which can be handled by the engine. If the inlet is equipped with variable geometry and/or bypass ducting, it can be set up to minimize the pressure losses at the required airflow; the result is improved propulsion system performance and greater aircraft speed. The control system required to dynamically match the inlet to the engine over the entire flight envelope is both sophisticated and costly to develop, requiring information from both the airframe manufacturer and the engine manufacturer.

At first glance, the propelling nozzle of the supersonic fighter aircraft is somewhat less complex a problem than is the inlet; however, it is also a tricky problem which further complicates the interaction between the various elements of the propulsion system and the airframe. Figure (6) indicates the basic interaction between a supersonic nozzle and the airframe. As indicated, in the afterburning mode the nozzle is positioned to allow the expansion of the jet beyond sonic conditions. By so doing, the engine generates a rapidly expanding plume which is geometrically much larger than the afterbody of the aircraft. The plume deflects the supersonic free stream flow over the aircraft by establishing an oblique shock somewhere on the outer surface or "boat-tail" of the nozzle system. Whether or not the engine is to blame for the increased aircraft drag due to the large exhaust plume is academic. It is a major system interaction which involves the complete propulsion system and which clearly requires the best compromise to yield a fast, agile fighter aircraft.

The foregoing discussion was intended to indicate that any discussion of the problem of engine handling in a military aircraft is incomplete without consideration of the interaction of the engine with the airframe. It must be treated as a complete system with free exchange of information from all contributors if a viable aircraft is to be developed. The application of the same modelling principles used in the development of the engine are normally the concern of the airframe designer.

THE ROLE OF MODELLING IN DESIGN

From the previous section it is clear that the eventual role of mathematical modelling is to produce a comprehensive model of the overall power plant system, requiring participation by the engine manufacturer, the control designer and the airframe manufacturer. The overall model must be developed continuously as the engine and aircraft development programs mature.

Mathematical modelling may be used very early in the engine design process, once the basic cycle and engine dimensions have been determined. Estimates of the compressor, turbine and nozzle characteristics can be prepared and used to provide a basic mathematical model of the engine thermodynamics capable of operating over the complete envelope of the engine and aircraft system. With the moments of inertia of the rotors available, the accelerations due to torque imbalance can be calculated and integrated to give the change of speed from an initial condition. Fuel flows may be varied arbitrarily to establish over-fuelling limits, based either on surge or a specified temperature limit; thus mathematical modelling can be used to determine preliminary fuel schedules before the engine is run. Having determined the basic operating characteristics of the engine alone, it is now necessary to focus attention on the control system design.

Investigation of the overfuelling information provided by the mathematical model leads to consideration of the control strategies required to provide suitable acceleration fuel flows which will permit safe but rapid accelerations; deceleration fuel flows can be determined based on the need to prevent flame out. In earlier engines, fuel flow was scheduled, typically as W_f/P_2, and there was no closed loop control of TIT; an alternative arrangement was to schedule throttle valve opening rate, usually with this rate increasing as a function of speed. On some modern engines, acceleration is carried out in conjuntion with closed loop control of TIT, or even turbine blade temperature in the case of highly cooled blades. The availability of a highly flexible thermodynamic model is thus, of great benefit to the control designer, who can evaluate a variety of control strategies and sensor requirements without endangering a valuable engine. It is the responsibility of the engine manufacturer to provide such a model.

Once the mathematical model has been extended to combine the engine and control models, it must be validated against actual engine test data. The validated model can then be used in parallel with the engine development program to provide detailed insight into transient behaviour of the engine and engine/control system integration. The model can be especially useful in optimizing throttle response by examining effects such as modifications to the amount of overfuelling permitted, and the operation and scheduling of variable geometry devices such as variable stators, blow-off valves and variable nozzles. It is clear that, to operate in this mode, an extremely sophistacated thermodynamic model is absolutely essential, and it is only as a result of many years of intensive development of both modelling techinques and computing facilities that this is now possible.

The basic problems of engine durability and control philosophy must initially be investigated using sea level test beds; once this has been achieved the progression to altitude test facilities and flying test beds follows. While these can considerably extend the flight test envelope, they cannot reproduce actual flight conditions in the aircraft under development. Before aircraft flight testing commences, it is essential that the modelling move into its final phase, i.e. representation of the overall power plant and its various control systems. When handling problems arise during flight test, the model can make significant contributions to determining the required fixes and different strategies can readily be investigated without endangering either aircrew or aircraft.

The EVOLUTION OF MODELLING TECHNIQUES

It is the availability of extremely powerful computing facilities which have made possible the very advanced thermodynamic models in widespread use to-day. It is instructive , however, to examine the evolution of modelling techniques realizing that the basic investigations into the engine response problem began in the late 40's, when computers were in their infancy; at that period in time, control engineering was very much in the field of electrical engineering and there was a very considerable gulf between the disciplines of control and engine performance.

Although this pioneering work may now be regarded as somewhat elementary, it is important to realize that the early attempts to quantify engine dynamics were very successful in identifying and understanding the parameters which affect the response rate. Probably the earliest published work was that of NACA, where Gold and Rosenzweig (5) showed, in 1952, that the rotor of a turbojet responded to sudden changes in fuel flow as a first-order system which could be conveniently expressed in terms of a rotor time constant; their expression for time constant, however, appeared in terms of partial derivatives which were not readily available and also were rather difficult to interpret quantitatively. A major advance was made by the Lucas Company (6), whose analysis assumed that a sudden increase in fuel flow would cause an instantaneous increase in turbine torque but zero increase in compressor torque. As a result of this simplified analysis, the rotor time constant could be expressed in terms of thermodynamic parameters which were readily obtained from normal performance calculations. The expression obtained was

$$\tau = \frac{K\ I\ N}{\left(\frac{\Delta T_{34}}{T_3}\right)\left(\frac{dW_f}{dN}\right)}$$

The variation of τ as a function of N could thus be determined as soon as the off-design performance had been evaluated. Referring back to Fig 3, it can be seen that fuel flow changes relatively slowly at low speeds and then changes much more rapidly as speed increases. Fig 7 shows typical variations of $\Delta T_{34}/T_3$ and dW_f/dN for a simple turbojet, and it can be seen that both terms decrease with reducing speed causing a significant increase in time constant; it can readily be see that the dominant effect in determining τ is dW_f/dN. Thus, the simple theory yields the important practical result that response is much more sluggish at low speeds. Despite its simplifying assumptions, the Lucas method gives quite respectable results when compared with engine tests, as shown in Fig 8, and improvements on this method are still in use for preliminary investigations.

Another important deduction following from the simple expression for the time constant is that, at high altitudes, where dW_f/dN will be much lower because of the reduction in fuel flow, the time constant will be increased. It is possible (but hardly convincing!) to express the moment of inertia in non-dimensional form, but a much more reasonable explanation is to consider that the engine inertia remains fixed while the energy release decreases with altitude.

It is important to realize that it is quite fundamental that response from low values of compressor speed will be sluggish, and much sophisticated modelling for a wide variety of engines of differing complexity has shown that it is essential to keep the HP rotor speed as high as possible for good response rates.

The major disadvantage of the time constant approach, however, was that it was limited to small changes in speed (say ± 5%) and it did not give much information beyond the rotor speed response; it was primarily of use to control system designers, and was of little use to the engine or airframe designer.

It was clear that for mathematical modelling to be useful to the engine designer, models capable of operation over the complete running range were essential. The computations required for continuous calculation of engine dynamics were greatly in excess of the capacity of early digital computers, and attention was initially focussed on analog computers because of their capability of operating in real time. Notable work was carried out by Larrowe and Spencer (7), sponsored by the US Air Force, in the hope that, with a model operating in real time, the control hardware could be developed using a simulator rather than an actual engine. The anticipated advantages were not immediately realized, mainly because the difficulty of integrating hydromechanical control systems and their required sensors and actuators with analog computers. A further major problem was the use of conducting surface bi-variant function generators for compressor characteristics, as these introduced both inaccuracies and dynamic effects on their own. Larrowe and Spencer, however, are due credit for the first real time simulation based on component characteristics using the approach of the engineering thermodynamicist. The problem of bi-variant function generators was overcome by using three single function generators to generate a function of two variables, and Saravanamuttoo (8) used this to develop full range analog methods which were used to predict dynamic behaviour of the Orenda OT-4 at the design stage. The analog proved fully capable of carrying out dynamic performance investigations on the Olympus 593, a twin-spool turbojet with variable nozzle (9). While analog models have been extremely useful in predicting engine behaviour, their greatest use has been in providing an understanding of different control stategies.

Confidence in modelling techniques was by then increasing, and digital computers were developed with large enough capacity to handle the data storage and calculation, but computing times were very long. As computing speeds increased the required computing time was reduced, but real time digital simulation of a complex engine is still difficult to achieve. Admittedly, the computer power exists with large machines to permit real time computation, however, such powerful machines invariably have virtual memory, multi-user operating systems which from a practical viewpoint make real time simulations very difficult. Smaller dedicated machines have sufficient speed for real time provided the model complexity is not too great. A further disadvantage of the very large scale digital computer is the difficulty of providing interaction for the engineers involved in engine development, and 'hands on' simulation is unlikely. The hybrid computer offers the computing speed of the analog with the storage capacity and logic of the digital computer, while permitting 'hands on' operation.

Once full range models, based on known component characteristics, became established, major efforts were focussed on effects such as heat transfer to and from engine parts, changes in clearances during transients, transient combustion efficiencies and changes in component characteristics during transients. Bauerfeind (10) discussed many of these effects in an earlier PEP paper. It is clear that these can only be considered because of the enormous developments in computing capability, and also that they can lead to a vast increase in modelling complexity.

It is appropriate at this point to consider the requirements for a successful mathematical model, bearing in mind that the model should be kept as simple as possible consistent with the needs of the particular user. The prime requirement of any model is that it should faithfully and accurately represent the behaviour of the engine over its complete running range and flight evelope. Further important requirements include:

1. Flexibility: The simulation must be capable of handling all the obvious requirements, such as scheduled accelerations and the operation of variable geometry devices; it must also be capable of dealing with situations which were not anticipated initially. During the development of an engine compressors and turbines may be modified to improve their performance, and the simulation must be able to keep abreast of the latest developments.

2. Credibility: The simulation must be readily understandable to performance, development and management engineers who are not simulation specialists; for this reason, the simulation should produce results in a form similar to a real engine and should make use of commonly available data.

3. Availability: Once the simulation has been verified it must be capable of being rapidly brought into use whenever required without the necessity of lengthy setup times.

4. Reliability: A high degree of reliability and repeatability is clearly essential, and the simulation must be capable of easy checking to ensure that it is functioning correctly; this is especially important for complex engine simulations.

The relative merits of analog, digital and hybrid computer models for engine simulation were discussed by MacIsaac and Saravanamuttoo in 1972 (11); since that time, however, there have been major developments in computer technology.

THE EVOLUTION OF COMPUTER TECHNOLOGY

The development of more and more detailed and sophisticated models of aircraft propulsion system dynamics has been made possible by the singular advance of computer technology over the past three decades. As discussed previously, the models have evolved from simple quasi-linear models which could be solved by hand to fully detailed component models capable of investigating secondary effects such as interstage bleeds or blade tip clearnace effects. To include these effects in a dynamic model of a propulsion system which must first include all of the primary effects, requires very powerful computers. Furthermore, if control system development is the objective, special purpose computers are required in order to provide the ability to solve the required equations in real-time.

The early major advances in the understanding of engine dynamics have been through the study of component based models on analog computers (7,8). These models were based on the ability to represent the major components using analog computing elements and required a reasonably large analog computer in order to solve the equations. Even with the approximations required to "fit the problem to the computer", very considerable insight into engine behaviour was achieved.

The analog computer is a natural means of representing ordinary differential equations for the very simple reason that it is, by design, a parallel processor. Modern analog circuits have a frequency response of at least 10^6 hz; a computer based on this technology is therefore virtually unlimited with respect to frequency insofar as propulsion system control problems are concerned. Unfortunately, frequency response is not the only consideration in choosing a computer for engine modelling. The analog computer has several major weaknesses; its inability to represent non-linear algebra and its inability to store programs for future use are major and fundamental disadvantages which have limited its usefulness as a modelling tool.

The explosive development of digital computer technology has, in essence, paced the development of propulsion system modelling. The basic advantages of representing non-linear equations and the long term storage of programs allow continued, step-by-step development of digital programs which add progressively to the technology base of a given organization. The primary disadvantage of the digital computer is frequency response. The approximation of ordinary differential equations forced by the discrete, serial nature of the digital computer has characteristic destablizing effects which force a time step size which is an order of magnitude smaller than theory could suggest. For a given computer execution speed, therefore, there is a clear limit to the frequency content that can be described in real time. Developments in integration techniques have helped reduce this problem; the demands of the industry, however, have continued to push towards greater detail with correspondingly larger models.

The concept of combining the parallel, high computing speeds of the analog computer with the non-linear, stored programming features of the digital computer has led to the development of the hybrid computer. This computational tool has existed in a variety of forms for at least 20 years and has been the means by which large scale dynamic problems have been solved by the aerospace industry during the intervening time. Until quite recently, the ability to store programs has not been a fèature of the hybrid computer. This fact, together with the size limitation imposed on the system because of discrete analog computing elements, have limited the growth of this class of equipment.

The use of discrete computing elements makes the hybrid computer an expensive system to buy and to maintain. Little has changed in the world of analog electronics for the past two decades while the growth of digital electronics has been trememdous. The digital computer has become smaller, faster and cheaper to the point where small businesses can afford very powerful computer facilities and the large facilities may disappear except for the extremely large problems.

Application of modern digital technology to the field of dynamic modelling has recently produced new forms of computers commonly referred to as parallel processors (12,13). Basically, this approach is an adaptation of the old parallel analog computer in which special purpose processors were dedicated to a given task. Interestingly enough, these units were initially developed as adjunct processors for function generation on a hybrid computer; however, it was very quickly realized that they had the potential to solve much more complex problems than merely function generation. As a consequence, more parallel processors were developed to displace all of the other tasks of the conventional hybrid computer including integration. The result has been the evolution of a completely different form of simulation computer which is less than 20% of the cost of a conventional hybrid computer with about 5 times the capacity. In general, for larger problems such as a complete propulsion system simulation, it has a wider frequency response than a hybrid computer.

Perhaps the most important, single feature of the development of the parallel processor simulation computer is that is has retained the stored program feature completely. If offers the potential to modularize any problem and to develop software tools which allow more than a small team of highly skilled specialists to contribute to model development. It allows sharing of software from problem to problem without the need for rewrite or adaptation. In short, as more software tools are developed, engine specialists can participate in the modelling effort and there will be less and less reliance on computer specialists.

THE ROLE OF MODELLING FOR THE USER

Before discussing user requirements for mathematical modelling it is pertinent to ask the simple question "Who is the user?"

As discussed earlier, during engine development the ultimate user is the aircraft manufacturer, and the requirement is for a fully comprehensive model of the overall power plant/aircraft combination including all the secondary effects encountered by the engine. During this phase extensive computer facilities and highly specialized engineering staff will be available; the modelling effort will be expensive, especially in terms of manpower, but should yield significant savings in terms of flight development.

The picture changes, however, once the engine completes its development and goes into service, when the final user becomes the armed forces. The modelling requirements now become quite different, as do the availablility of specialized engineering staff and large scale computing facilities. The main uses of modelling are now likely to be for training and familiarization purposes and also for fault detection and maintenance. These requirements can probably be met with considerably less sophisticated models, which must retain the ability to operate over the whole flight envelope but need not go into the same depth to deal with secondary effects. The goal should be to implement a simulation based on component characteristics on a modern minicomputer which can be purchased relatively cheaply; using this philosophy it would be quite feasible to provide on-site simulation and training facilities at most major air bases.

The engine manufacturer, of course, would maintain an on-going program of improving the model in the light of service experience, and also to deal with uprated and modified versions of the orginal engine. Similar efforts would be followed by the control designer as new control technologies mature and the aircraft manufacturer would maintain an interest in applying uprated engines to the existing aircraft or entirely new aircraft.

Thus, the modelling process can be assumed to be an on-going activity which will continue as long as the engine remains in service.

CONCLUSIONS

The problem of modelling gas turbine engine dynamics has been discussed from the viewpoint of engine thermodynamics. It has been noted that the most appropriate form of model is one based on a description of the individual components. This approach provides a very flexible model which allows various levels of detail as required by the many model users.

It has been noted that, as propulsion systems become, more and more, an integral element of the aircraft flight controls, there is a need to consider the engine as part of a major system with design responsibility spread among the major participants in the development. Interest in a comprehensive model could, therefore, be expected from designers of airframe, flight controls, engines and fuel controls. Each would need to contribute his specialized expertise to successfully develop comprehensive models.

The pacing item in the development of large scale propulsion system models has been the growth of computer systems. Model development can be traced from early analog computers through to modern parallel digital processors. The ability to store, retrieve and reuse software has allowed a continued step-by-step evolution of increasingly sophisticated models which, in turn, allow the development of increasingly sophisticated and capable propulsion systems.

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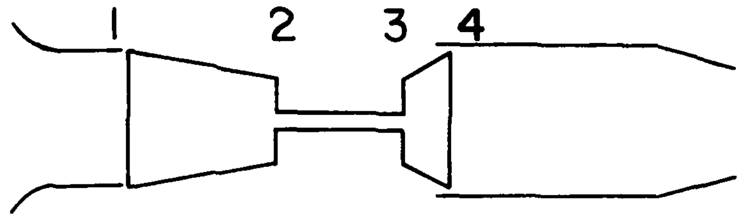

Figure 1 Station Numbering

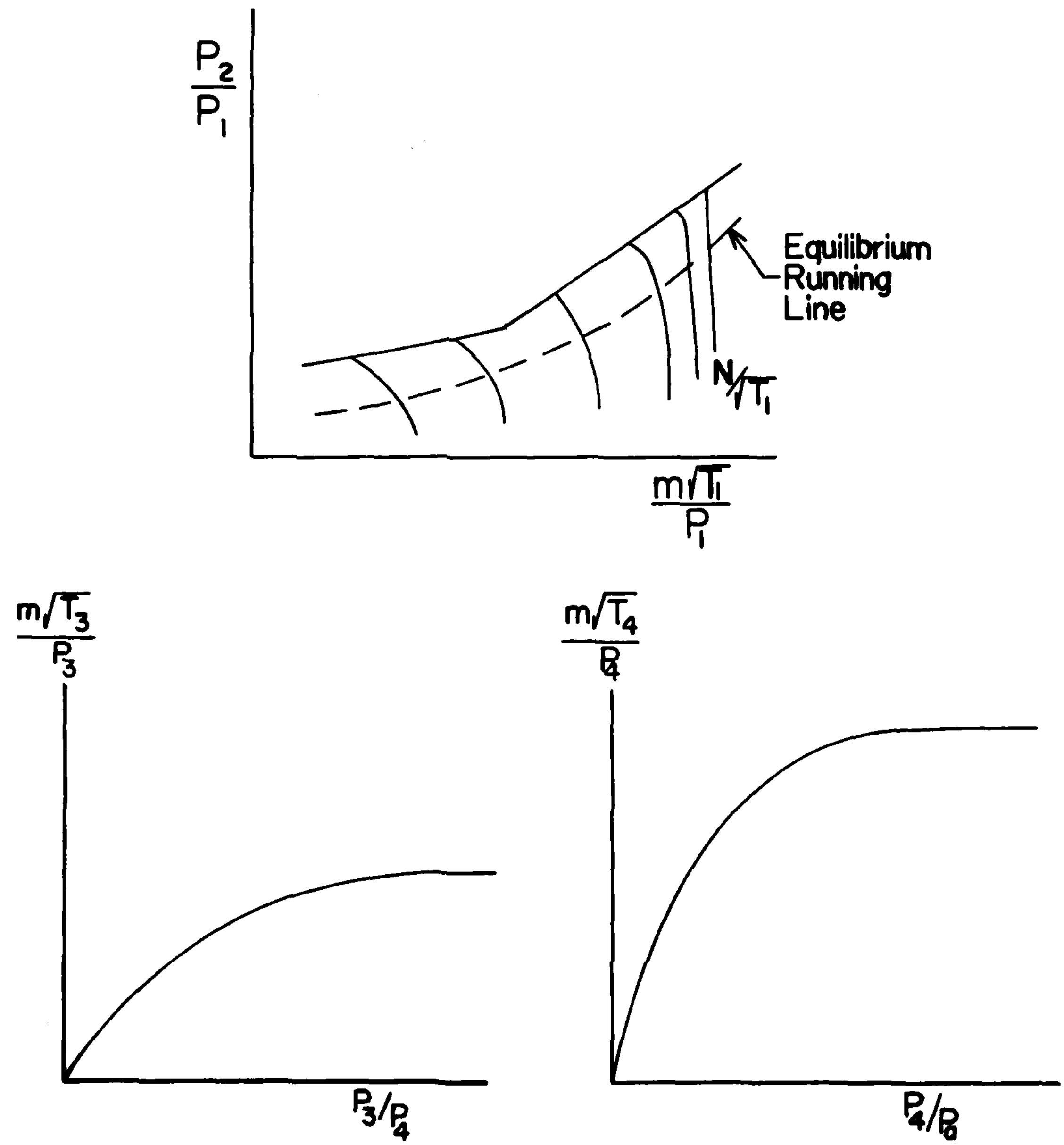

Figure 2 Component Characteristics

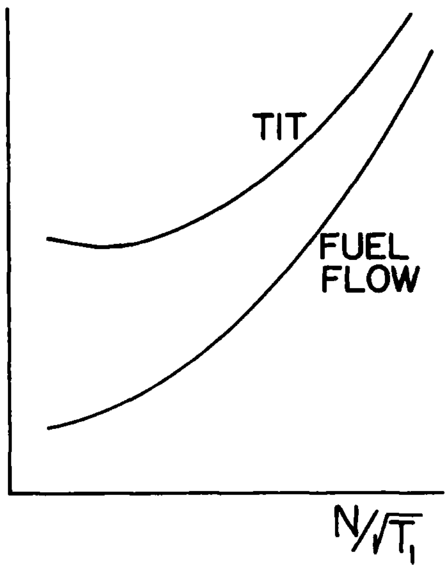

Figure 3 Parameter Variation With Speed

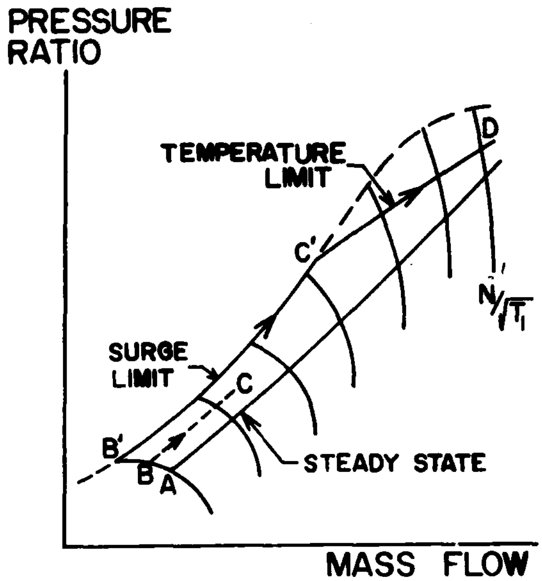

Figure 4 Acceleration Trajectories

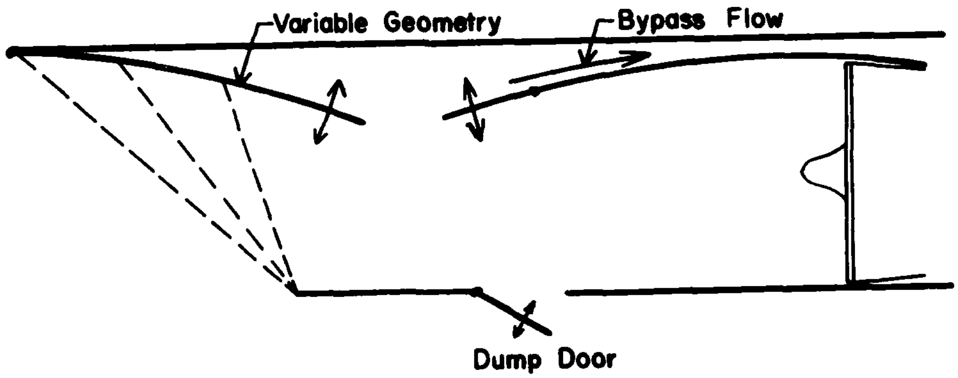

Figure 5 Modern Supersonic Intake System

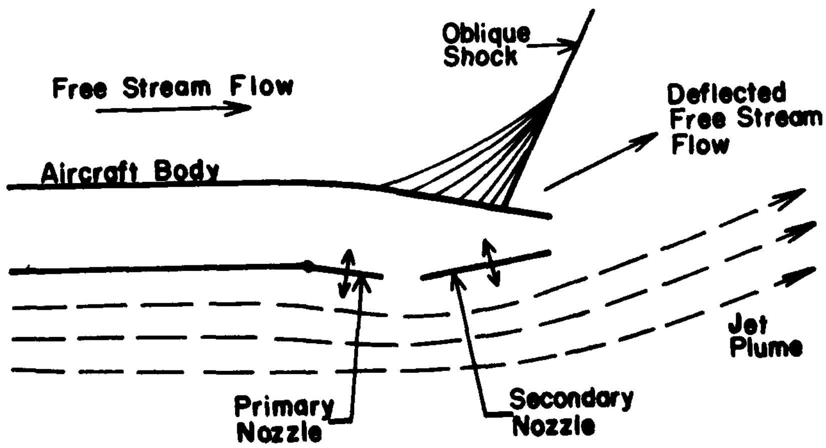

Figure 6 Exhaust Plume/Airframe Interaction

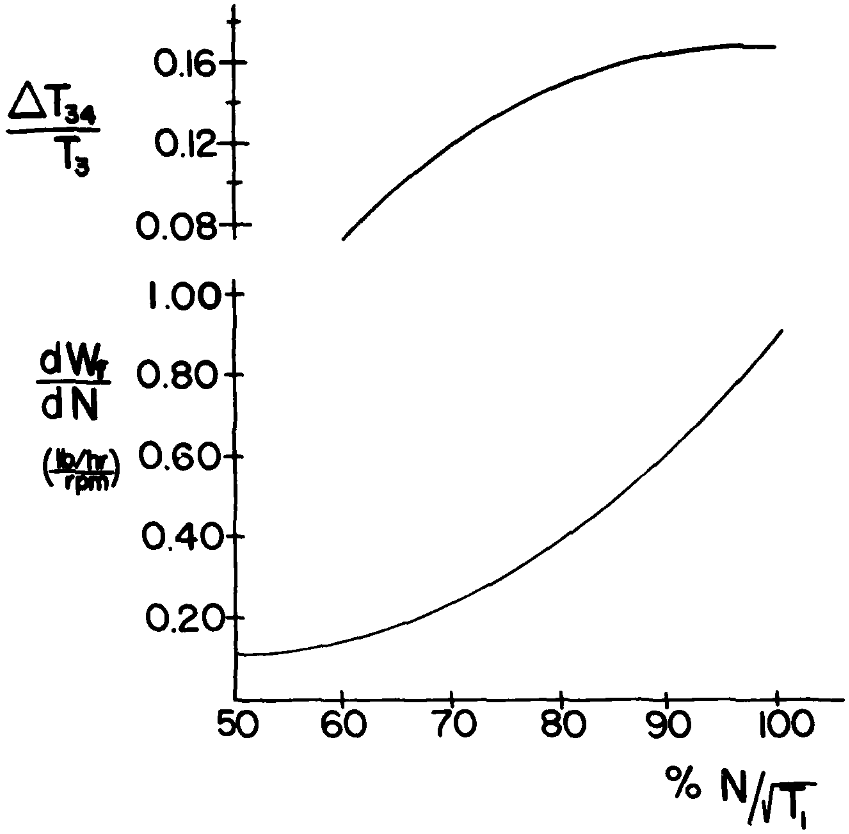

Figure 7 Parameters Determining Time Constant

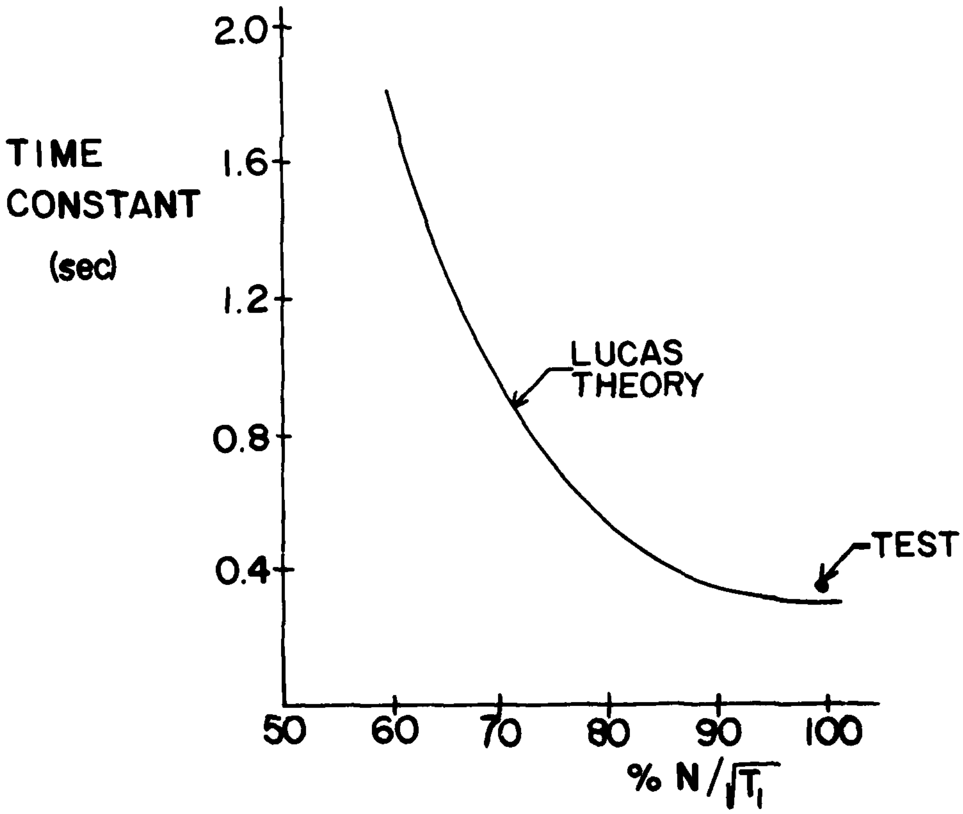

Figure 8 Variation of Time Constant with Speed

DISCUSSION

J. Dunham, UK:

Do you believe that simulation of engine start-up is needed and is possible?

Author's Reply:

I believe it is desirable, particularly for establishing light up fuel flows and starter power requirements. I am not convinced, however, that realistic calculations are possible; information on compressor characteristics below about 30% speed is seldom available, and the turbine characteristics are probably even more difficult to determine. Modelling may be useful for establishing re-light characteristic at altitude, where the rotational speeds have not dropped below idle.

L. Giorgieri, It:

When you talk about computers do you mean special purpose computers linked to the engine or a main computer in which are integrated all the functions of the aircraft including navigation and attack? What is your opinion of the direction this technology will take in the future?

Author's Reply:

When one speaks about computers being imbedded in an aircraft system, there are essentially two schools of thought. The fully integrated systems are sought after and promoted by the overall systems designers and sometimes by the user (airforces). They offer the potential of lower life cycle costs since there is commonality of design throughout the system. However, this is very difficult to achieve contractually. Computer control functions are implemented in software and every specialist company wants to protect its technology. Thus, most suppliers are interested in the distributed system which dedicates a processor for each major function. The other compelling argument for distributed systems is the vulnerability of an aircraft which places all controls in a central processor. It is the author's opinion that the dedicated processor will emerge as the trend of the future with a central computer providing a supervisory role.

Col Rougevin-Baville, Fr:

Quelles applications proposez-vous aux forces armées pour une modelisation mathematique du moteur: meilleurs connaissances théorique, instruction des personnels, diminution des coûts de maintenance, analyse des incidents...?

Author's Reply:

I believe mathematical models can effectively be used in all three areas suggested. There is no doubt that a suitable model can give a greatly increased understanding of engine behaviour, without hazard to the engine or the pilot; this is especially important for new engine types where there is very little operational experience. We are currently involved with providing models for training operating personnel on a gas turbine powered warship, with a particular emphasis on fault detection; the detailed models can be used to simulate faults such as turbine damage, compressor fouling and internal leakage effects. Similar work is being done for pipeline gas turbines to provide a diagnostic and maintenance tool, with a view to reducing maintenance costs and diagnosing trouble at an early stage.

It must be emphasized that such models are valuable as a tool to train operating personnel to *recognize* changes to engine performance and hopefully to understand why the change has occured. It is, therefore, not a substitute for qualified maintenance engineers.

GENERALIZED DIGITAL SIMULATION TECHNIQUE WITH VARIABLE ENGINE PARAMETER INPUT FOR STEADY STATE AND TRANSIENT BEHAVIOUR OF AERO GAS TURBINES

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SUMMARY

Computerized simulation of aircraft gas turbine engines is a good tool to investigate powerplant and aircraft systems taking into account a lot of system parameters. This paper presents a generalized digital simulation technique with variable engine parameter input for steady-state and transient behaviour of typical turbine engines as turbojet, turboprop and especially helicopter engines with and without heat exchanger and variable turbine. Examples are shown as helicopter flights close to the ground at low speeds in complex flow environment resulting from different interaction effects.

1. INTRODUCTION

Most of the airbreathing powerplants of today are based on gas turbine engines. As missions and aircraft-engines-systems have grown more complex in the last years the computer aided studies of many engine configurations have grown along with them. Accordingly the importance of large-scale and high speed computers increased. At the present, however, the valuation and effectiveness of large computer programs are partially considered suspect. The confidence in using large and complex programs depends on some typical factors like the transparency and difficulty in checking the logic program-flow or the investment needed to understand, implement and operate the programs.

The purpose of this paper is to describe some main features, uses of mathematical models and experiences in developing, generalized digital simulation techniques for aero gas turbines. Investments and costs of aircrafts and powerplants in development have grown up within the last ten years. Therefore greater emphasis must be placed on exploring and testing methodes to evaluate the suitability of aircraft-engine-systems. The need to predict the steady-state and transient behavior of aero-engines is becoming more important. Mathematical models on digital computers enable the aircraft or engine engineer to examine the performance of powerplant systems throughout the total flight regime.

Some of the main areas of developing improved computer codes for simulating any gas turbine engines are e.g.:

- general optimization methods for the internal engine components as well as the total aircraft / engine system including mission aspects
- studies of new concepts of multirole powerplant systems including Variable Cycle Engines (VCE) for supersonic and subsonic flight regimes
- calculation and prediction of engine weight and engine volume (length and diameter)
- reliable simulations of steady state and especially transient effects of aero gas turbines
- development of effective control systems to achieve an optimum matching of all powerplant components
- Life Cycle Cost analysis
- studies of engine failure monitoring and diagnostic systems combined with gas path analysis methods
- prediction methods of overall external and internal installation effects including variable inlet and exhaust nozzles

- improved analysis of engine operating behavior in relation with ambient effects like pressure and temperature distortion.

Nowadays and in future typical aero engines can be classified generally as follows:

(1) Low bypass-ratio engines for supersonic and subsonic flight missions such as one-, two- or three-spool, one-, two- or three-stream turbofan engines with or without mixed flow, heating in main stream and / or bypass-duct including components for Variable Cycle Engine

(2) High-bypass-ratio turbo-fan engines for subsonic flight missions in multi-spool and / or multi-stream configurations including components for VCE-capabilities

(3) Turboprop and turboshaft engines for subsonic airplanes and helicopters with two or three rotors, provisions for variable components and heat exchanger devices.

Fig. 1 shows some schemes of the abovementioned engine types.

The intent of this paper is to present some computerized simulation methods in use atthe Institute of Aeronautics and Astronautics of the Technical University of Munich for the Calculation of steady-state and dynamic gas turbine performance including optimization procedures.

Ten years before the development was started for digital computerized calculations of gasdynamic cycles for any conceivable gas turbine engine. The component matching programs based on published reports e.g. some NASA-Reports like /2, 3/.

The object was to perform computerized methods matching numerous components of different powerplant systems without using complex large-scale, program codes. In contact with engineering teams of different aircraft companies, postgraduate students and governmental institutions it came out that the acceptance, understanding and handling of complex computer based engine synthesis programs often caused problems. This called for the development and test of computer codes that should be: more readable by all users, more easily checked in logic algorithm and design etc.

The development of digital simulation codes concerns three types of modular program structure:

(1) Modular codes for each individual engine type e.g. two-spool two-stream turbo fan engine without mixed flow.

(2) General modular codes with the capability of simulating any conceivable engine cycle configuration including a simple and wide variety of variable engine parameter input and automated optimization techniques for different aircraft missions.

(3) Combined versions of modular program codes for special types of aero gas turbines as shown in Fig. 1.

First some main features of the basic program systems will be presented especially about point (2) and (3). Second, some versions of the abovementioned engine types of aero engines are discussed exemplary the main type of turboshaft engine for helicopter devices in steady state and dynamic performance.

2. MODULAR COMPUTERIZED SIMULATION TECHNIQUE

Fig. 1 shows some typical engine configurations and the function f(x) describing generally the engine system with the independent variables x_i. These variables are typical engine parameters at any position of the engine being calculated e.g. pressures, temperatures, mass flow rates etc.

2.1 Standard input procedure for steady-state engine simulation

As previously mentioned the turboshaft engine will be presented as major example. On Fig. 2 are reviewed some of typical features for steady state calculations. This figure shows a two-spool engine with schemes of its component maps. The determination of any off-design point B_j in a component characteristic, is based on the gasdynamic laws of Conservation of Mass and Energie and furthermore on physical conditions as mechanical performance balance on the rotors or the static pressure balance in components mixing different fluid flows. The calculation of steady-state operating points B_j requires a special solution technique for a system of nonlinear equations in relation with various engine matching conditions such as rotor speeds, mass flows in ducts etc. For a steady-state solution the system of nonlinear equations can be satisfied by special iteration techniques for example the multi-dimension Newton-method (sometimes called Newton-Raphson-Iteration method). The minimum number of the engine parameters (resp. independent variables x_i) to be varied and the engine matching conditions (resp. dependent variables ($f_i(x)$)) to be satisfied, are shown in Fig. 2.

Depending on the number of component maps (compressor, high pressure turbine, low pressure turbine) and the special conditions of the recuperative heatexchanger the Newton-Iteration technique needs a minimum-number of six variables. In Fig. 3 the definitions of the 6 independent and dependent variables are listed. The iteration procedure starts with the known values of the design point A or those of guessed operating points B_i. The identification of an operating point in a component map is carried out by the definition of a general map parameter Z as following scaling equation (Fig. 4).

$$Z_c(j) = \frac{\sum_{k=2}^{j} \sqrt{\left(\frac{\dot{m}_{red,k} - \dot{m}_{red,k-1}}{\dot{m}_{red,1}}\right)^2 + \left(\frac{R_k - R_{k-1}}{R_1}\right)^2}}{\sum_{k=2}^{n} \sqrt{\left(\frac{\dot{m}_{red,k} - \dot{m}_{red,k-1}}{\dot{m}_{red,1}}\right)^2 + \left(\frac{R_k - R_{k-1}}{R_1}\right)^2}}$$

j = 2, 3, 4, ... n identifies the discrete points on a guiding line in the compressor map, in that case the speed lines. The length of the guiding line is divided into equal segments between the minimum and maximum value. The mentioned map-parameter Z helps a lot to handle easily any component maps as a matrix. Moreover the Z-parameter has a good suitability as an independent variable in the iteration procedure. If the measured component maps of an engine are not available scaled similar maps should be used. Partially an analytical determination of component maps is possible /4, 5, 6/. Fig. 5 shows the performance of a numerically generated map for a cooled high-pressure turbine /5/. A scheme of the cooling airflow system is shown in Fig. 6.

The performance characteristic of turbomachines with variable geometry can be applied using some different maps for each turbine position. The interpolation between the different settings is possible. One of the key factors in handling large computer programs is the ability to pick up any desired value or engine parameter without changing the structure of the program. The way we choosed is shown in Fig. 7.

All engine datas resp. program variables are compiled in a matrix-form. The first and major block includes gasdynamic, physical and geometric datas for each defined engine position. The gasdynamic- or major-matrix is followed by the blocks of the component maps, the cooling and air systems, the ducts and pipes etc., last not least the block for results like thrust or shaft performance and fuel consumption etc.

This matrix-structure for all engine datas allows simple variable parameter handling especially for parameter input and output, for optimization devices and time depending values of powerplant control. The overall computer program consists of modular parts as shown in Fig. 8.

Fig. 9 illustrates a typical engine component representing the module "compressor" with the upstream station A (entry), down stream station B (outlet) and furthermore several possibilities defining steady state and time dependent input and output positions and engine parameters. Similar modules are used for all major components.

2.2 Standard Input Procedure for Transient Simulation

Simulating the transient behavior of gas turbine engines some typical transient effects have to be taken into account. Following the illustration in Fig. 9 the program codes for calculating steady state performance must be modified. Some steady state equations - shortly described in section 2.1 and indicated in Fig. 9 - have to be added by further differential time depending terms. Typical kinds of equations which have to be modified including transient terms are: Conservation of Mass, power balance and Conservation of Energy. Exemplary the energy balance for the module "compressor" can be written

$$\dot{E}_A(t) + P_i(t) = \dot{E}_B(t) + \dot{E}_i(t) + \dot{Q}_i(t) + \dot{E}_{Stor.}(t)$$

$\dot{E}_A(t)$, $\dot{E}_B(t)$, $\dot{E}_i(t)$	enthalpie rates	i = 1, 2, ...
$P_i(t)$	rotor power	
$\dot{Q}_i(t)$	heat flow rates	
$\dot{E}_{stor.,i}$	rotor-energy $\dot{E}_r(t) = (\frac{\pi}{30})^2 \, \theta_{pol} \, n \, \frac{dn}{dt}$ heat storage in material $\dot{E}_{m,i}$	
$\dot{m}_i(t)$	mass flow rate	

Calculating the dynamic behavior of the engine on a digital computer the differential terms must be replaced by difference equations. The iterative solution of the difference equation system can be done in the same way which is used in the steady-state iteration-technique shown previously. In order to avoid convergence problems time steps about 0,10 sec are recommended.

Much attention is paid to particular aspects of simulating specific transient effects by mathematical modelling. This investigations concerns basic physical or transient effects as e.g.

- heat exchange between the fluid flow and the material of blades, cases etc.
- mass storage in components and ducts
- time delay in the combustion processes
- transient changes of seales or clearances causing efficiency changes in the components.

These transient phenomena should be included in the engine simulation programs. It should be pointed out that the influence on the accuracy of the calculation results could be very different. Thus the effect of heat exchange is more important than the effect of the mass storage in a finite component volume. For some simulations the time depending mass storage effect can be neglected.

Simulating the heat exchange process and for simplifiing the modular computation technique all parts of the engine can be replaced by equivalent plates with a typical mean temperature level. The heat transfer coefficients can be derived from appropriate transfer coefficients of basic thermodynamic examples. A typical plate module is shown in Fig. 16 representing the dynamic simulation of a recuperative heatexchanger by a difference-equation procedure. The heatexchange simulation based on mathematical models can be classified in specific plate modules e.g.

- compressor blades ⟶ uncooled plates
casings
disks
...
- turbine blades ⟶ cooled plates
casings
...
- heatexchanger ⟶ modified cooled plates
...

2.3 Expanded Program Version with Variable Engine Parameter Input

Contrary to the standard input procedure discussed in section 2.1 the minimum number of independent resp. dependent iteration variables ($x_{i,min}$ resp. $f_{i,min}(x)$) is not sufficent for the solution technique of engine matching condition in many cases. The difficulties can arrise if complex optimization or control problems have to be solved for example if calculated program output values must equal a given input function. These calculated values can be time or mission depending curves for engine performance, fuel flow rates etc. Based on the major matrix (shown in Fig. 7) the number of the matching iteration variables x_i and $f_i(x)$ can be increased without changing any program structure. Accordingly it can be written for the numerical iteration technique:

independent variable x_i $\quad i > i_{min}$

dependent variable $f_i(x)$ $\quad i > i_{min}$

with engine parameter

x_i: $T_{amb.}, p_{amb.}$, distortion-parameter,
control-parameter, optimization parameter, ...

see sketch in Fig. 13 reviewing typical engine parameters.

Thisability incorporated in the engine synthesis program is a very comfortable feature to calculate performances and engine simulations most flexible. The examples presented in the following chapters will demonstrate the benefits of the program handling using variable engine parameter input.

2.4 Concept of a Modular Universal Synthesis Program

Taking into account the wide variety of air breathing propulsion systems and the broad areas of application as described in chapter 1 it became desirable to develop an overall computerized simulation system which should be able to assemble any conceivable engine configuration for any possible operating condition. Fig. 10 shows schematically the main flow diagram of the advanced modular Universal Synthesis Program (entitled MUSYN). Compiling any configuration of a powerplant system one can start the synthesis program using a special code for the input data concerning the labelled positions of the selected engine. The schematic model of a typical shaft power engine with recuperator and variable low pressure turbine is shown in Fig. 11. It is apparent that the flexibility of the above-mentioned general program MUSYN enables arbitrary off design studies especially of aero gas turbine engines with variable geometries resp. thermodynamic cycles.

As shown in Fig. 10 the major engine component modules which can be assembled are:

- Compressor
- Turbine
- Combustion chamber
- Nozzle
- Duct
- Inlet
- Heat exchanger, recuperator
regenerator
- Mixer
- Splitter
- Burner, afterburner

Besides the subroutines for the individual components further subroutines are used for special devices:

- Air system
- Ambient conditions
- Shafts
- Thrusts, power

and for aero-thermodynamic resp. material properties.

3. STEADY-STATE PERFORMANCE OF TURBOSHAFT ENGINE WITH RECUPERATIVE HEAT EXCHANGE

Future developments of advanced helicopter powerplant systems include different aspects improving fuel consumption, specific weight, part load range, contingency ratings, dynamic reponse characteristics etc. These features can be achieved for example by

- higher turbine inlet temperature and higher pressure ratios
- improved aerodynamics, combustion and clearance control
- variable geometry
- heat exchange (recuperator)
- variable air and cooling system
- digital engine control system.

Demonstrating the potential of the above presented steady-state simulation technique Fig. 12 illustrates typical results of partload studies for two engine types. Both shaft engines have a design of P_A = 1100 kW and a variable low pressure turbine. The design point values are optimized and listed in Fig. 12. The engine with the total pressure ratio of R = 8 has a recuperator as shown schematically in the block diagram of Fig. 2 and 11. This diagram shows the specific fuel consumption as a function of relative shaft power. The advantage of the recuperative cycle engine relative to the conventional engine is apparent. Besides the improved fuel consumption the variable turbine enables the control system to run the engine in the mean power regime near a constant turbine temperature range. One of the main disadvantage of the recuperator-engine is higher overall weight of the total powerplant.

4. TRANSIENT BEHAVIOR OF AERO GAS TURBINES

Besides the steady-state performance of aero gas turbines - briefly discussed in the previous chapter - there are a lot of further requirements for unsteady operation of aircraft powerplants. Among other points some of the desirable features for modern helicopter engines are:

- High reliability during low altitude unsteady flight missions
- Excellent handling characteristics without surge problems during slam acceleration or deceleration
- Stable automatic engine control systems
- Insensitivity to intake pressure and temperature distortion

Some examples are given to illustrate the capability of the digital computer techniques for dynamic simulation devices.

Fig 13 shows schematically the signal flow diagram of a typical conventional powerplant.

system of a twin-engine helicopter. A corresponding control has been incorporated in the helicopter engine system calculating the following examples.

4.1 Acceleration and Deceleration Simulation of a Medium Power Turboshaft Engine

As mentioned above a high heat flow between the hot air resp. gas and the material can occur during slam acceleration or deceleration. Fig. 14 shows the computed curves of the heat flow into some components. The total acceleration time was about 2 seconds.

Fig. 15 shows a typical reacceleration procedure of a hot engine after a short-time deceleration. In comparison with the cold acceleration which started from part load conditions and had been run for a long time, the hot acceleration tries to pass over the TET-temperature limit more than the cold acceleration. This is caused by the energy storage in the heated material.

4.2 Transient Behavior of a Turboshaft Engine with a Heat Exchanger and a Variable Low Pressure Turbine

The computerized transient simulation of a turboshaft engine with heatexchanger includes the basic transient effects of the heatexchanger. Fig. 16 shows the plate model of a counterflow recuperator and a scheme of the time dependent transient behavior.

Fig. 17 presents the results of some temperature responses to a step-function change ΔT-hotside-inlet solving numerically the heatexchange process by a simplified difference equation system. The simulation results are in sufficent agreement with the solution of basic thermodynamic reports /8/. The dotted curve in Fig. 18 represents the power rise of a recuperator engine during a slam acceleration.

The advantage of a variable power turbine improves the steady state as well as the dynamic performance as shown in Fig. 18 (upper curve). In comparision to the engine with fixed geometry the variable one shortens the acceleration time for more then 30 % by opening the variable stator nozzles for about 1 second.

4.3 Engine Failure Simulation of a Twin Engined Helicopter

The latest feature being illustrated exemplary concerns a real measured engine failure simulation during flight tests compared with a computed failure simulation of a light weight twin engined helicopter. The flight test was done without any pilot inputs through the power lever and the collective pitch as shown in Fig. 19. The measured and the computed results of the automatic acceleration of one engine and the downrunning of the immediately stopped engine are in good agreement.

5. UNSTEADY HELICOPTER FLIGHT NEAR THE GROUND

Recent years more attention was paid to helicopter operations near the ground (NOE nap-of-the-earth). The flight missions near the ground require manoeuvres with normal cruise, low level flights and contour following flights above and through covers. The so called "Dolphin"-flight is a typical NOE-flight for testing the capability of the helicopter-powerplant system. Fig. 20 shows schematically such a dolphin-flight-manoeuvre. The results of the computerized simulation of a typical test flight near the ground according to the manoeuvre (30 sec) of Fig. 20 is shown in Fig. 21. A light weight helicopter was used for this flight (MBB-BO 105, gross weight 2400 kg, two Allison 250-C engines). The range of mission was about 600 m and two jumps of 10 m height at a mean speed of about 60 kt were included. Curves of the change of collective pitch, torque and main rotor speed are given by flight test measurements. Distortion effects were taken into account as described e.g. in / 1 /. Fig. 22 shows the working line in the compressor map.

6. CONCLUSIONS

The computer program for digital calculation of steady-state and unsteady performance presented in this paper enables a realistic computerized simulation of powerplant behavior even for complicated engine types. In order to minimize costs and to limit the computer capacity the program was structured and written with the intention to work on medium sized digital computers (less than 100 000 words capacity). Furthermore only 0.1 to 0.5 sec of CPU-time is necessary for the calculation of one operating point.
For engine types which can be classified in groupes, the standard modular program (chapter2.1 2.2) is better accepted by aircraft- and powerplant engineers than the modular universal synthesis program (chapter 2.4). It is prefered because of its higher credibility, greater transparency, better handling, wider flexibility etc.

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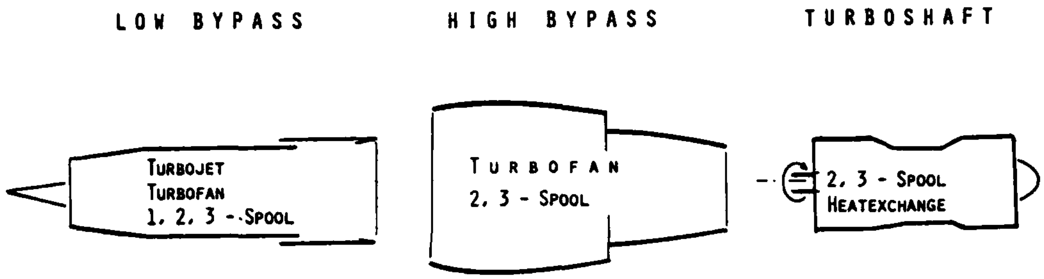
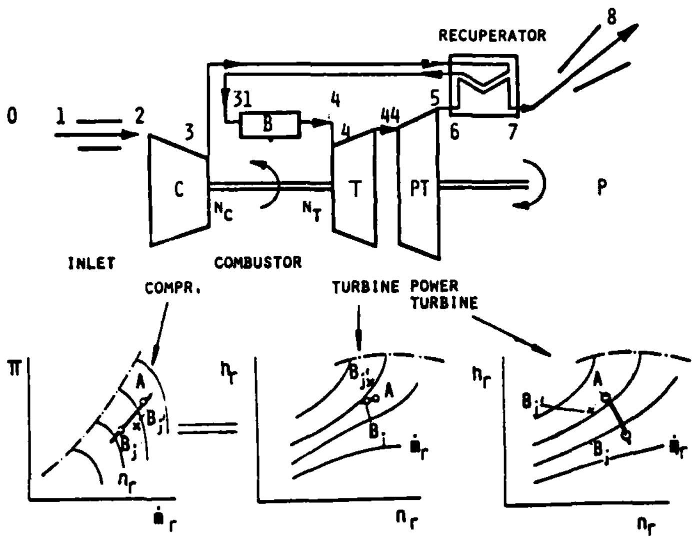


VCE	VARIABLE CYCLE ENGINE	VGE	VARIABLE GEOMETRIE ENGINE
VSCE	VARIABLE STREAM CONTROL ENGINE	VHE	VARIABLE HEATEXCHANGE ENGINE
D.B VCE	DOUBLE BYPASS VCE	VC	VARIABLE COOLING ENGINE

OVERALL FUNCTION POWERPLANT
ENGINE PARAMETER

$$f(x) = \begin{bmatrix} f_1(x_1, x_2, x_3, \dots, x_n) \\ f_2(x_1, x_2, x_3, \dots, x_n) \\ \\ f_n(x_1, x_2, x_3, \dots, x_n) \end{bmatrix} = 0$$

Fig. 1 Aero gas turbine configurations. General presentation.



$$\begin{bmatrix} \frac{\delta f_1}{\delta x_1}, \dots \dots \frac{\delta f_1}{\delta x_n} \\ \\ \frac{\delta f_n}{\delta x_1}, \dots \dots \frac{\delta f_n}{\delta x_n} \end{bmatrix} * \begin{bmatrix} \Delta x_1 \\ \\ \Delta x_n \end{bmatrix} = \begin{bmatrix} -f_1(x) \\ \\ -f_n(x) \end{bmatrix}$$

Independent Variables $x_i = x_6 = x_{i,min}$
Dependent Variables $f_i(x) = f_6(x) = f_{i,min}(x)$

Fig. 2 Turboshaft engine with recuperator, component maps and multi-dimension Newton-Iteration technique (schematically).

INDEPENDENT ITERATION - VARIABLES $x_i = x_{i,min}$

$x_1 = z_C$ Z-Par. Comp. Map (Table No....)

$x_2 = z_T$ Z-Par. High-press Turb. Map (Table No...)

$x_3 = z_{PT}$ Z-Par. Power Turb. (Table No....)

$x_4 = T_{t,6}$ Temp. Recup. Entry (Table No....)

$x_5 = T_{t,4}$ TET (Table No....)

$x_6 = T_{t,7}$ Temp. Recup. Out.(Table No....)

CEPENDENT ITERATION - VARIABLES $f_i(x) = f_{i,min}(x)$

$f_1(x)$ Power-Balance High-press Turb. (Table No....)

$f_2(x)$ Mass Flow Control High-press Turb.(Table No...)

$f_3(x)$ Mass Flow Exhaust (Table No....)

$f_4(x)$ Mass Flow Power Turb. (Table No....)

$f_5(x)$ Temperatur-Compar. $T_{t,31}$ (Table No....)

$f_6(x)$ Temperatur-Compar. $T_{t,7}$ (Table No....)

Fig. 3 Newton-Iteration technique for standard input procedure for the turboshaft engine shown in Fig. 2 (see Fig.7).

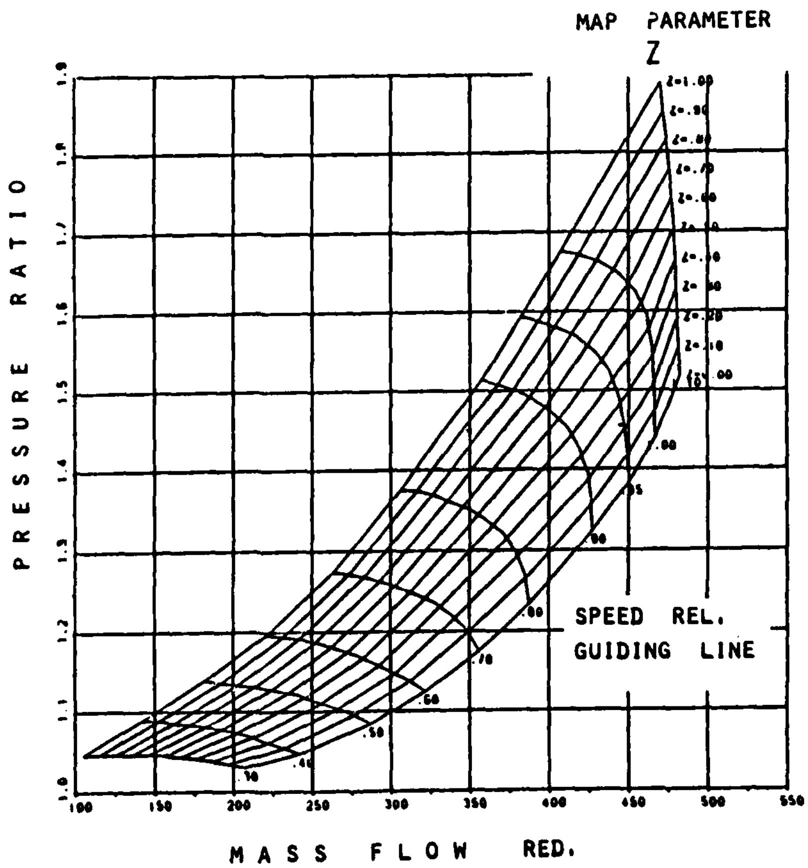


Fig. 4 Typical component graph with the map parameter Z (example: compressor map)

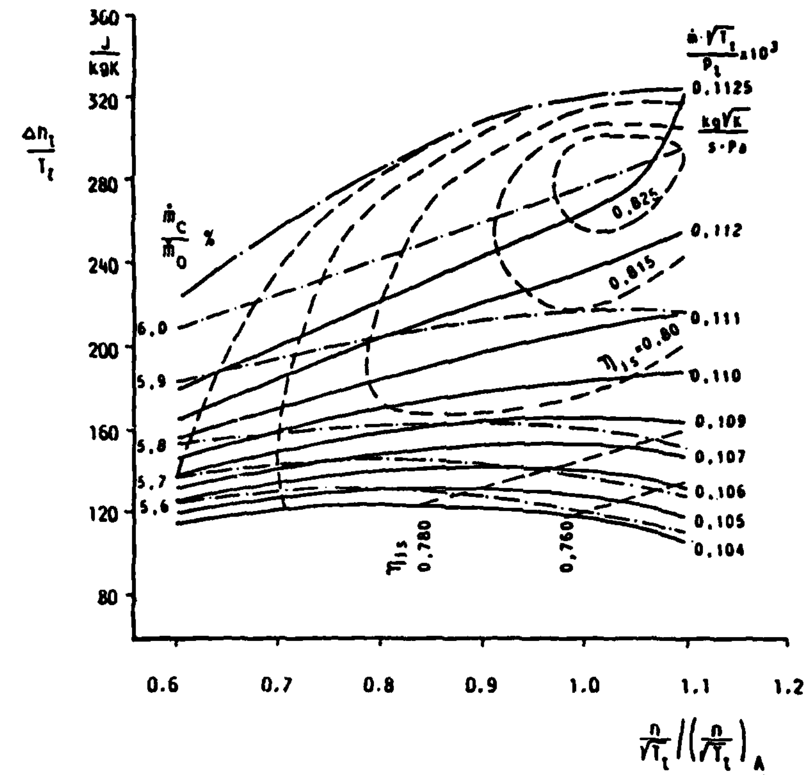


Fig. 5 Performance map of a cooled high pressure turbine (scheme fig. 6)

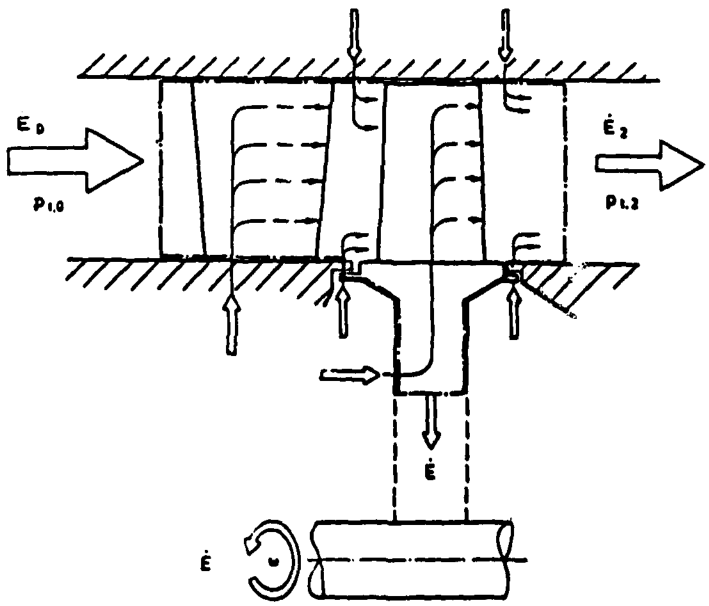


Fig. 6 Flow diagram of the cooled turbine (schematically).

POWERPLANT PARAMETER / VALUES

GASDYNAMIC MAJOR-MATRIX

Gasdyn. Value	Engine Area E0 1	E1 2	E2 3	E3 4	E31				E9 16
$\dot{m}_{tot}$	1	16	31	46	-	-	-	-	226
$\dot{m}_{air}$	2	17	⋮	⋮	⋮	⋮	⋮	⋮	⋮
$\dot{m}_{fuel}$	3	18	⋮	⋮	⋮	⋮	⋮		⋮
⋮	⋮	⋮	⋮	⋮	⋮				⋮
T_t	⋮	⋮	⋮	⋮					⋮
P_t	⋮	⋮	⋮						⋮
Mach-No.	⋮	⋮							⋮
Enthalpie	15	30							240

Component Maps
....

Air Syst.
Cool.Syst.
....

Nozzle / Duct
....

Heat Exchanger
Heat Transfer
....

Ambient Cond.
Distortion
Failure
Control
....

Unsteady,Dynamic Proc.
Heat Exchange
Rotor Inertia
....

Thrust, Power
Spec. Values
....

Fig.7 Gasdynamic-major-matrix.
Example: Turboshaft engine

MODULAR MAJOR-PROGRAM

A Powerplant - Configuration

Main Program
- Input
- Output
- Limiter
- Plot
- ....

B Thermodynamic Basic Modules
- Spec. Heat = f (T)
- Enthalpie, Entropie
- ....

C Gasdynamic Modules, Engine Components
- Ducts, Nozzles, Mixer
- Compressor, Turbines
- Combustors, Duct-Heating
- Heat Exchange Plate Model
- Heat Exchanger

D Steady-state / Dynamic Modules,Control
- Control - Values
- Rotor - Inertias
- Derivatives
- ....

E Mathe. - Modules
- Solution of Equation-systems
- Iteration-Techn.
- ....

Fig. 8 Modular major program (scheme)

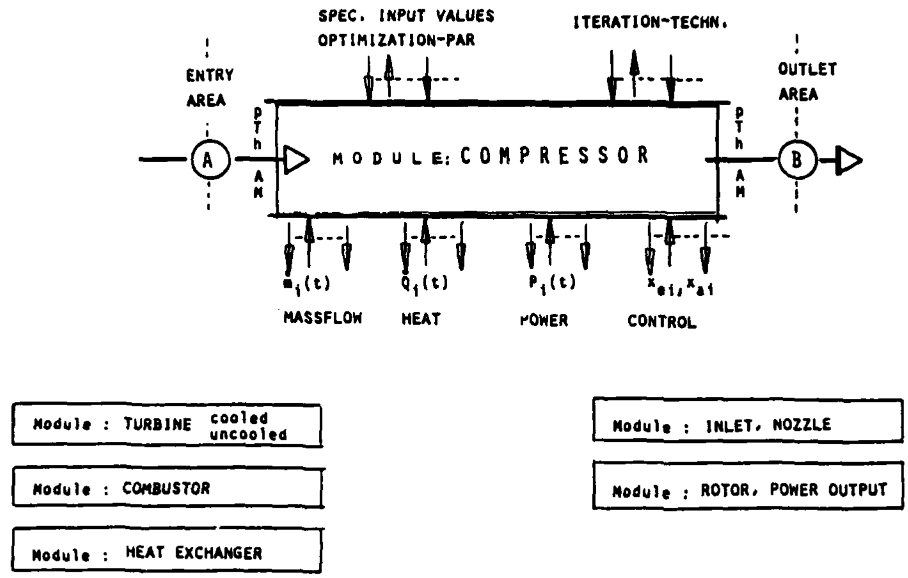


Fig. 9 Typical engine component: module compressor

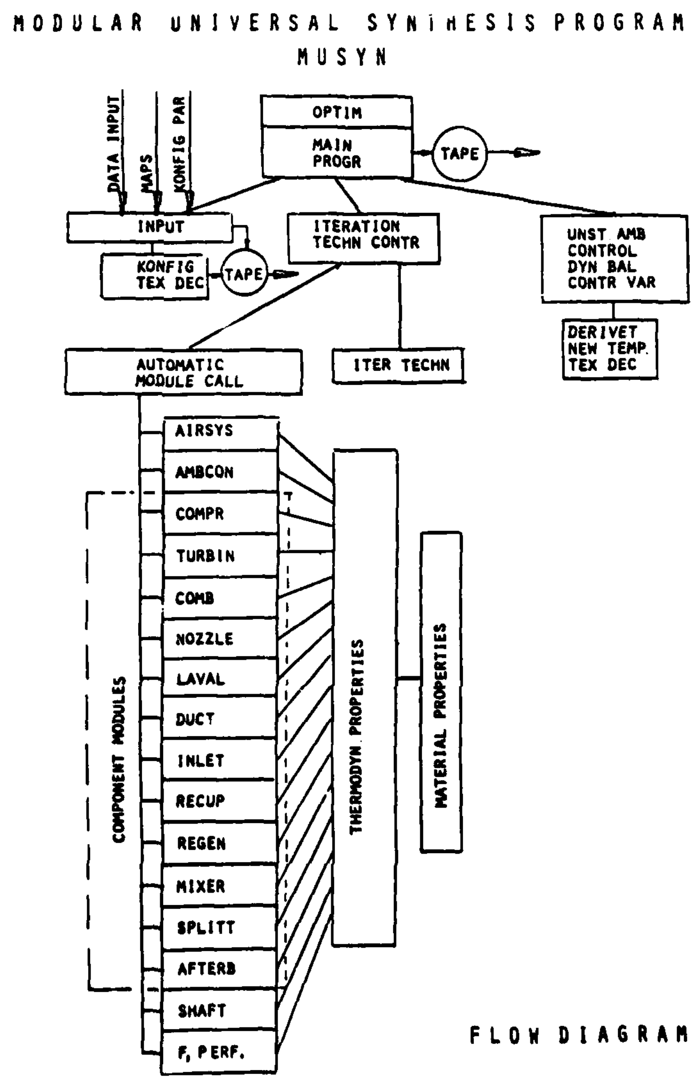


Fig. 10 Flow diagram of the modular universal synthesis program

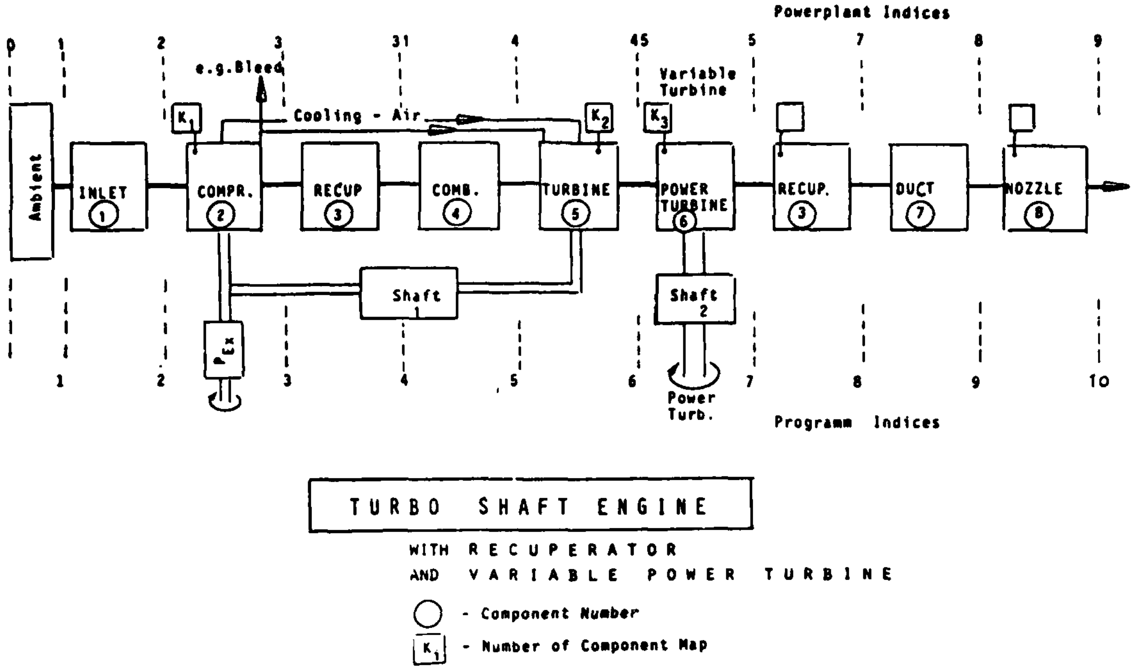


Fig.11 Turboshaft engine with recuperator and variable power turbine

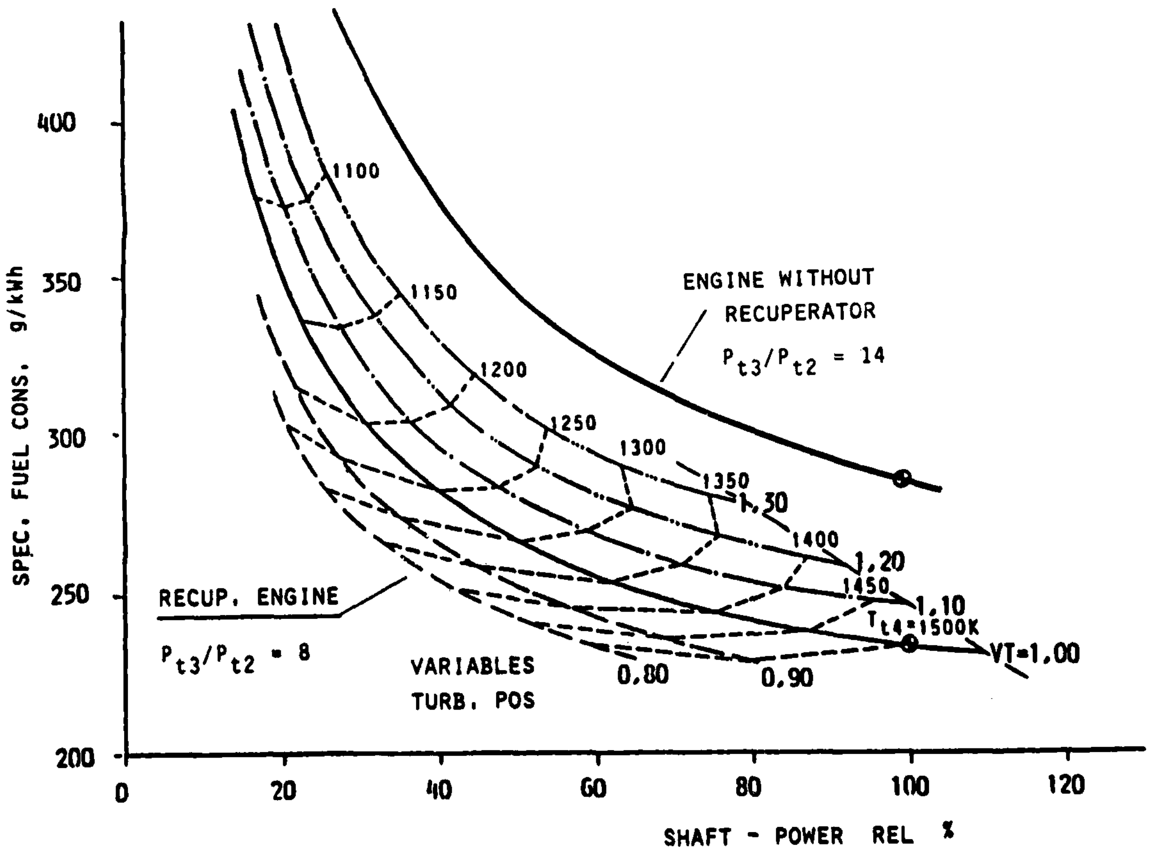


Fig. 12 Influence of variable turbine on SFC at part power.
Turboshaft engine without and with recuperator, P_A=1100 kW
(optimized design point).

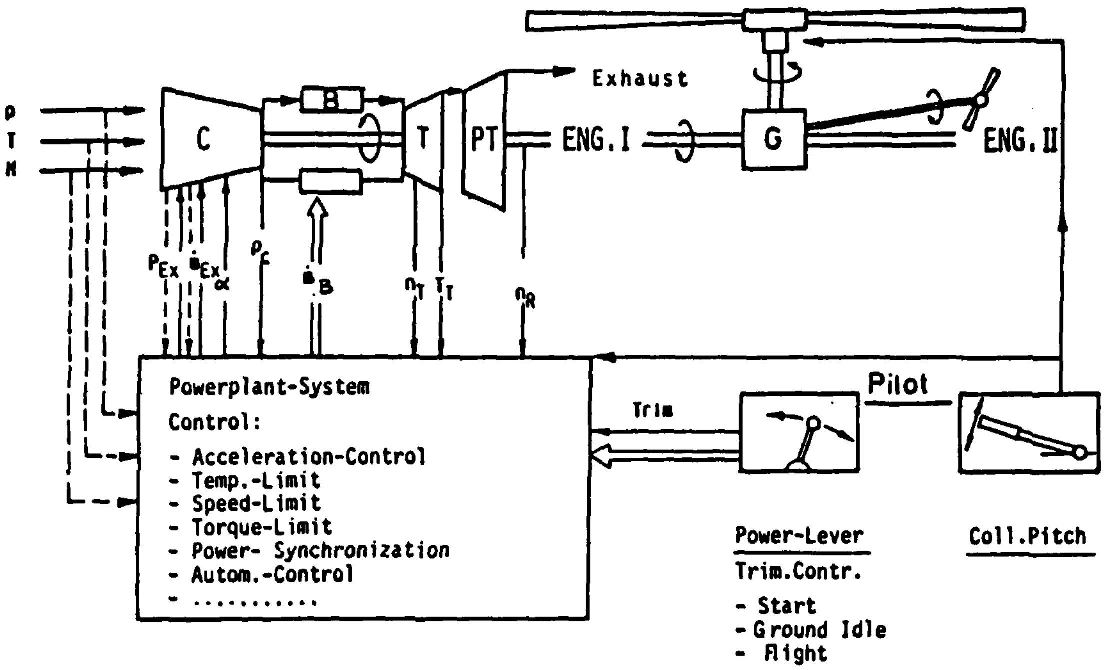


Fig.13 Signal flow diagram of a twin-engined helicopter

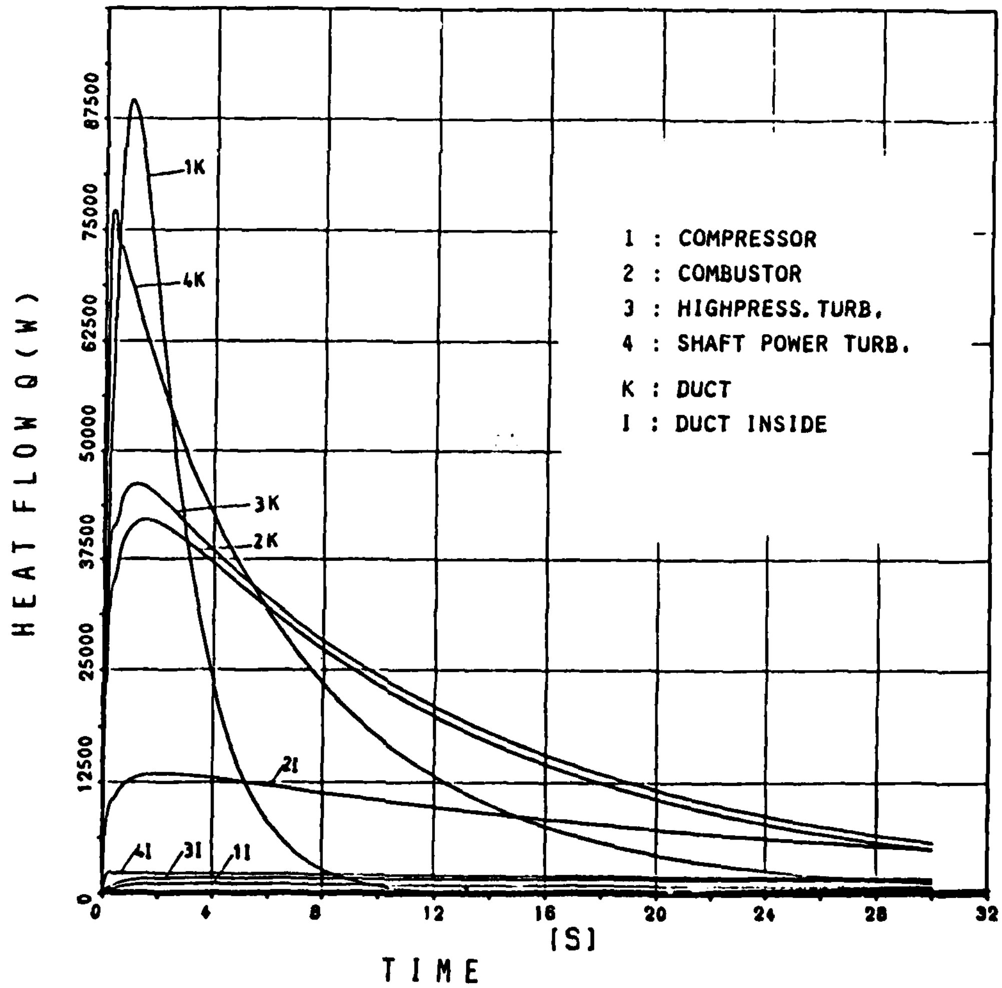


Fig.14 Heat flow into some components of a turboshaft engine (P_{max}=1100 kW). Slam acceleration.

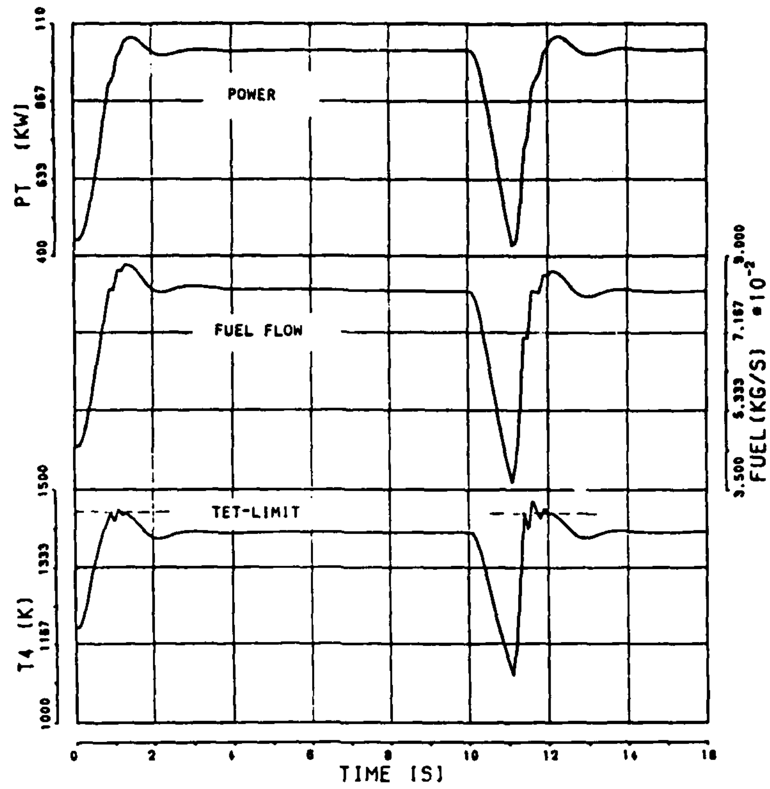


Fig.15 Typical deceleration-reacceleration procedure.
Turboshaft engine (P_{max}=1100 kW)

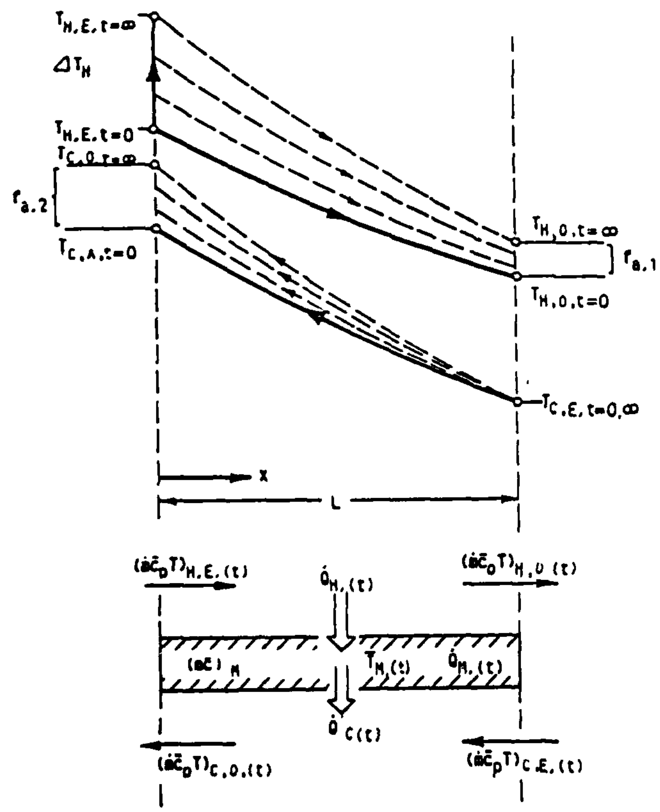


Fig.16 Plate model of a recuperator and the transient behavior (scheme)

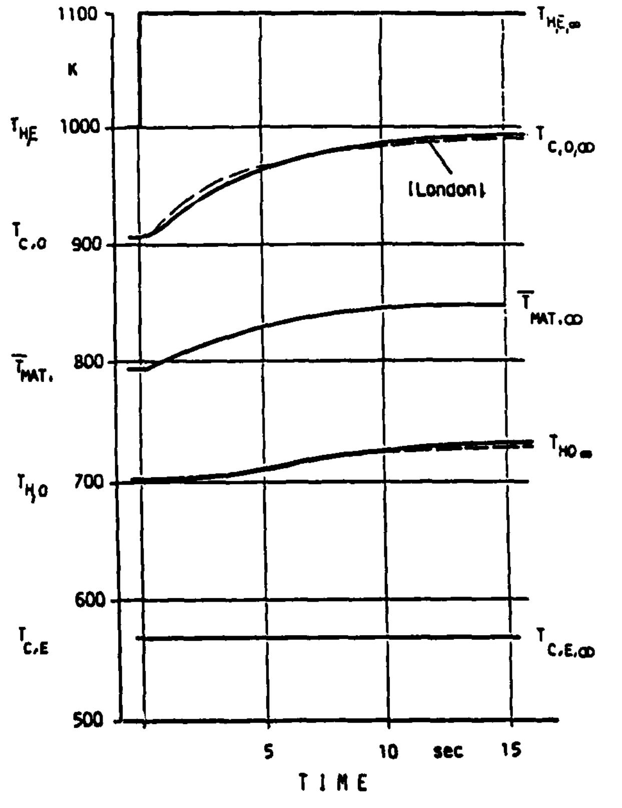


Fig.17 Temperature response to a step-function change
ΔT-hotside-inlet (turboshaft engine, P_{max}= 1100 kW

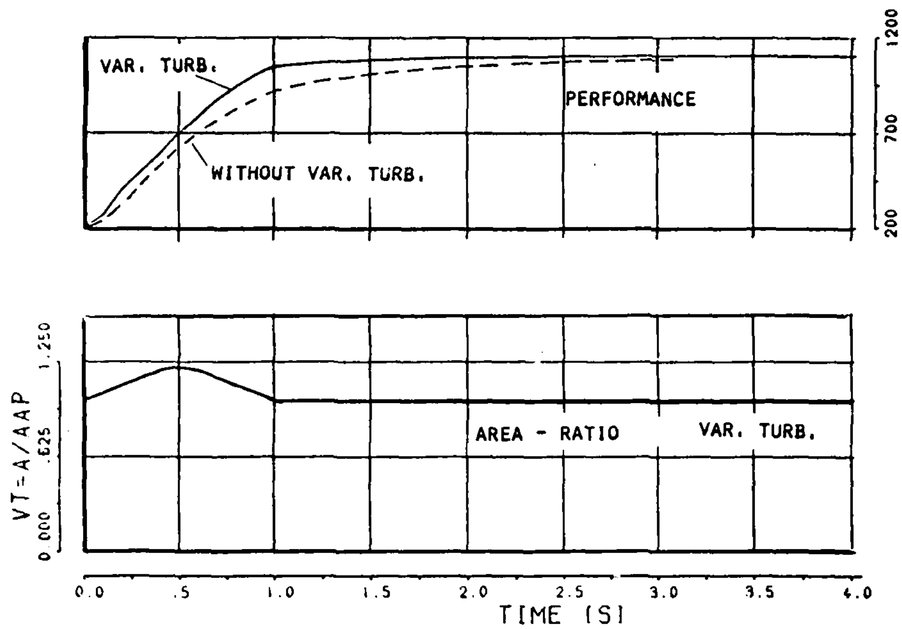


Fig.18 Slam acceleration of a recuperator engine with and without variable turbine (P_{max}= 1100 kW)

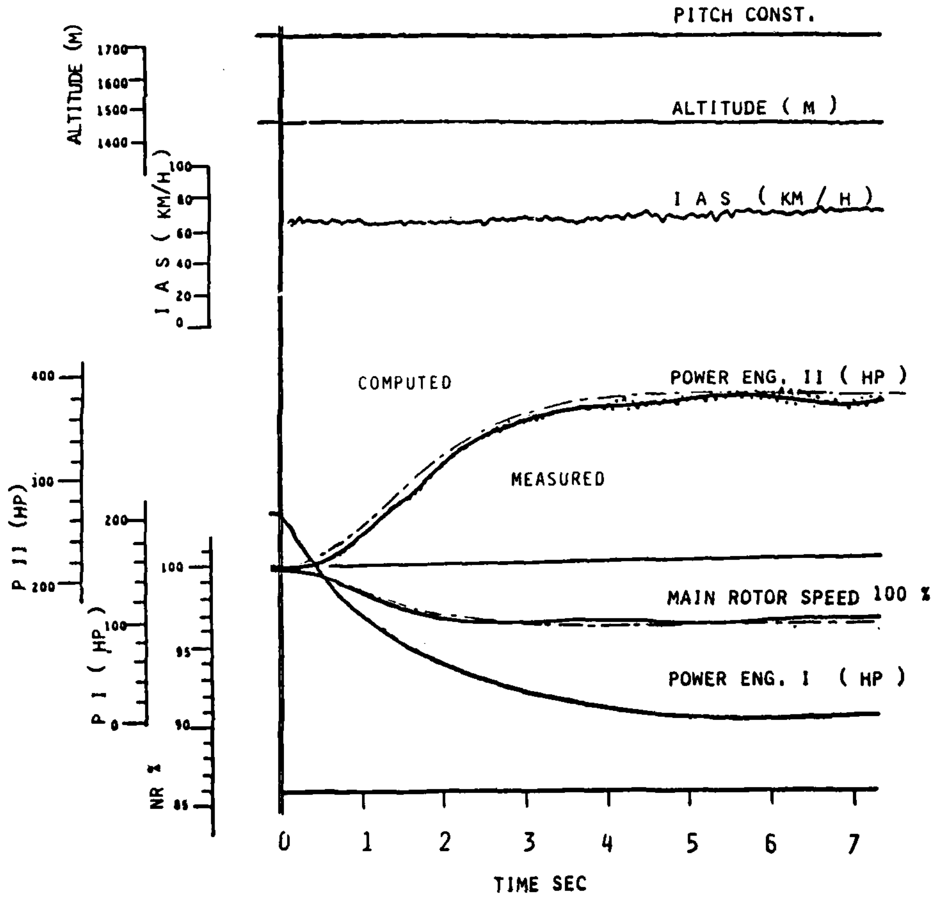


Fig.19 Simulated engine failure (engine no.II) on a twin-engine helicopter (MBB BO-105). Flight test results vs computed result

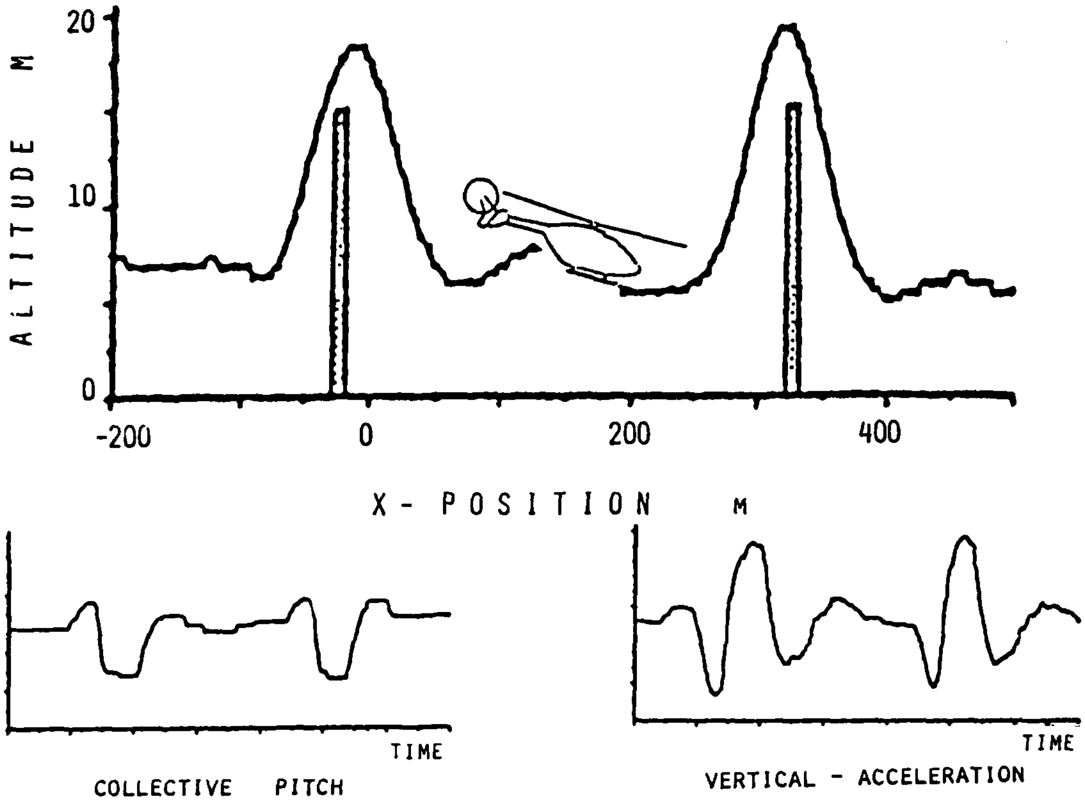


Fig.20 Helicopter testflight near the ground (MBB BO-105).
Dolphin - flight /14/.

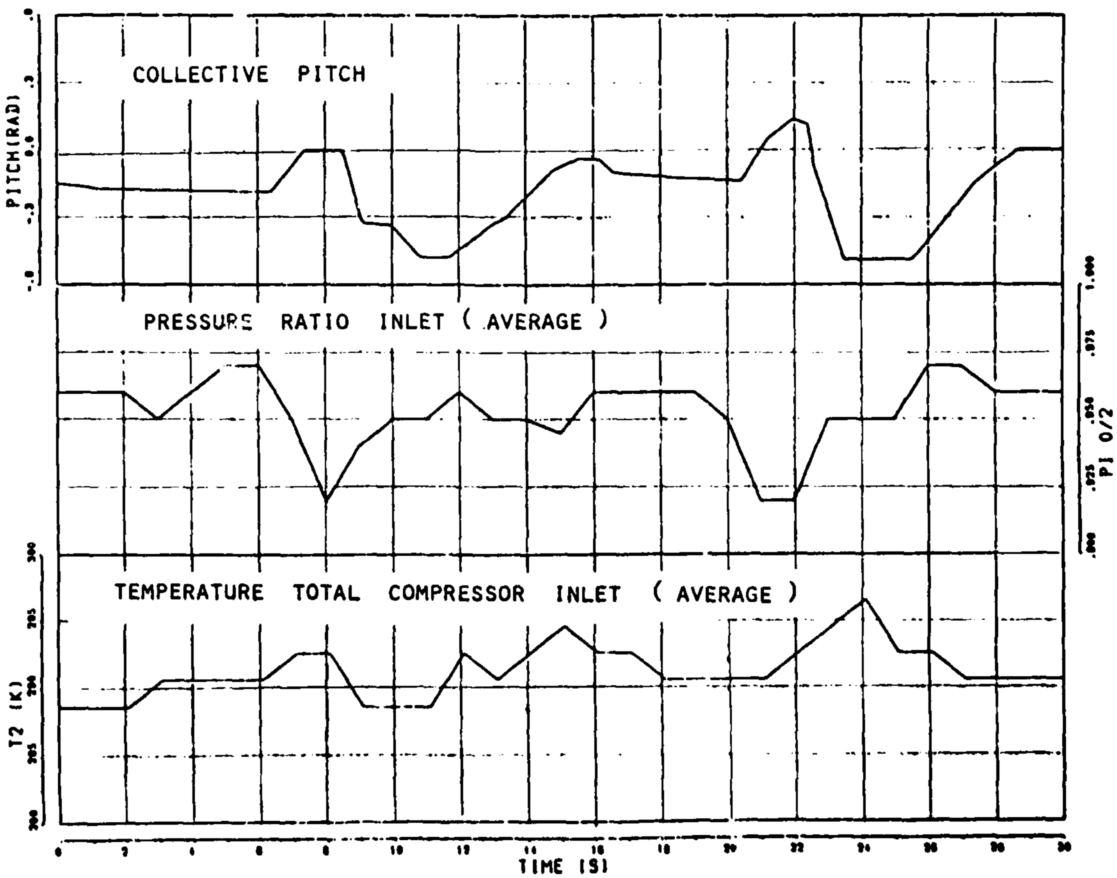


Fig.21 a

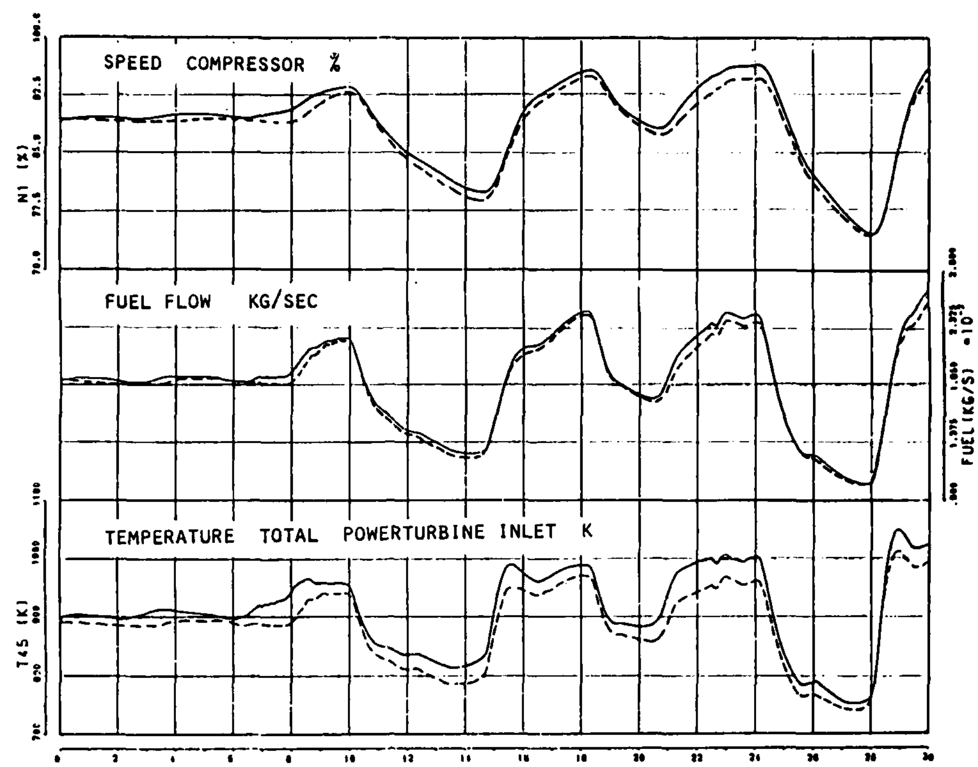


Fig.21 b

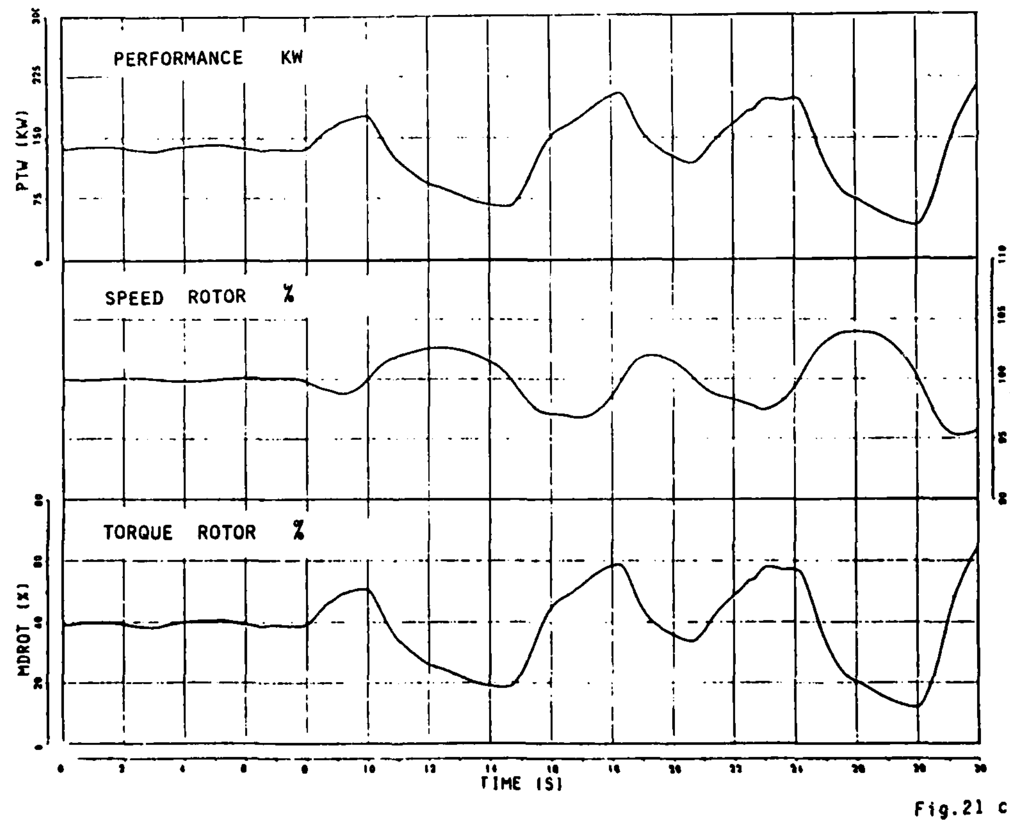


Fig.21 c

Fig,21 Dolphin-testflight (MBB BO-105). Computed resp.measured results including inlet distortion effects (dotted curves: without distortion effects)

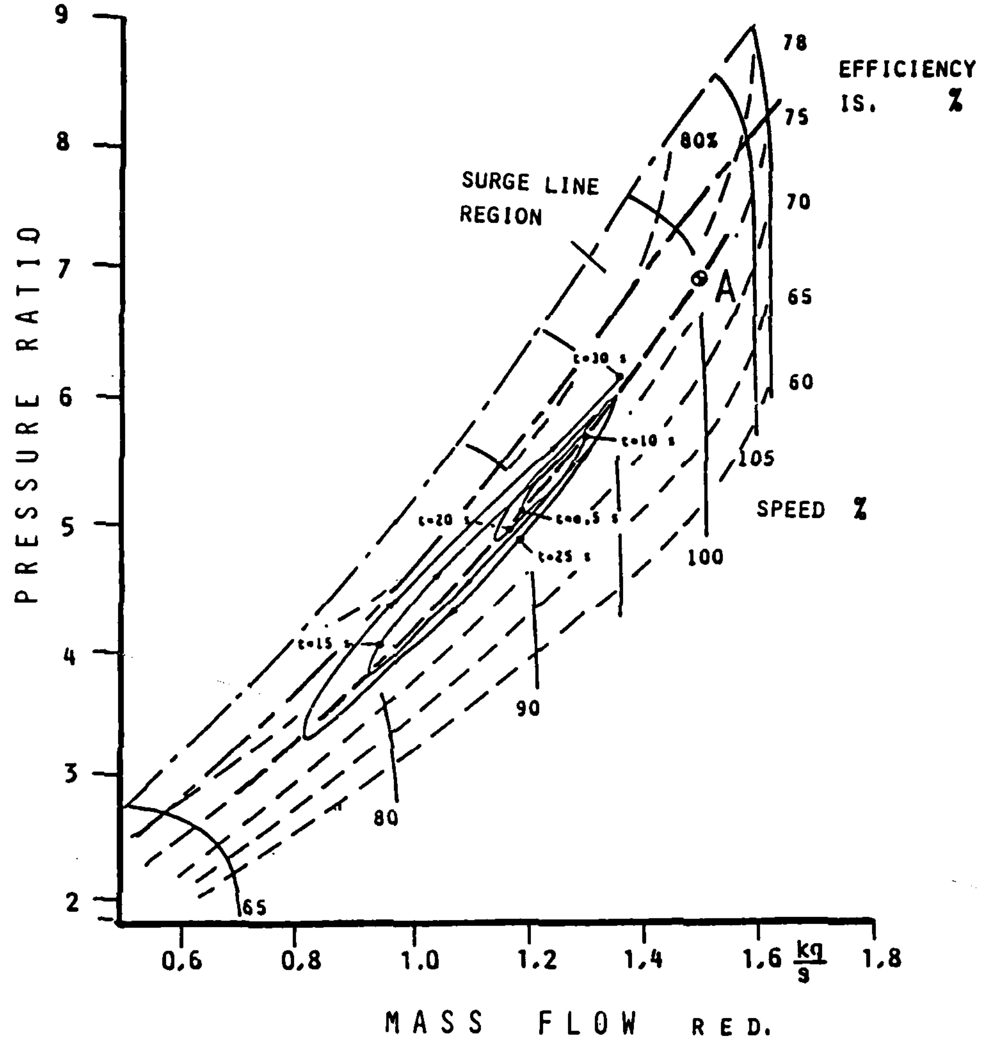


Fig.22 Compressor map with the operating line of *the Dolphin-testflight (fig.20, 21)*

DISCUSSION

M. Berthier, Fr:

Nous avons vu au long de la semaine que les jeux radiaux, axiaux, et au niveau des labyrinthes sous les redresseurs évoluaient beaucoup dans le temps et même en fonction du profil de vol qui précède la manoeuvre. Avez-vous pris en compte dans votre modèle les effets des variations des jeux?

Author's Reply:

Transient tip clearance effects resp. seal and leakage effects on turbomachine performance are important simulating the transient behaviour of aero gas turbines. These effects are difficult to quantify. Provisions and tests are made to take into account the mentioned effects but the object of this paper is to present procedures and experiences with computerized simulation techniques with limited computer capacity used e.g. in connection with simulation studies of aircraft-engine-systems.

IMPROVEMENTS IN THE DYNAMIC SIMULATION OF GAS TURBINES

by
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SUMMARY

For several years thermodynamic simulations have been used to predict both steady state and dynamic engine performance. This is particularly important for engine and control system designers in order that predictions can be made without recourse to expensive engine running. This Paper reviews the current status of these models at NGTE and the extent to which they have been capable of reproducing engine performance.

Simulations based on steady state component characteristics, known inertias and air volumes have not necessarily been capable of predicting the transient response of real engines under all circumstances. While it is possible to alter the assumed inertias of the rotors to tailor the response of the simulation to that of the real engine, this is not physically justified and the paper outlines alternative approaches aimed at more accurate prediction of the fastest transients.

In addition, work at NGTE investigating control problems associated with variable geometry showed that the critical period of an acceleration was during the first fraction of a second. Current models could give misleading results and a stage-by-stage representation of the compressor should give a more accurate representation. The concepts and techniques of the model together with a comparison of accelerations are presented.

SYMBOLS

A	area
C_p	specific heat at constant pressure
G	power
I	moment of inertia
K_1	$\left(1 + \frac{\gamma - 1}{2} M^2\right)^{1/\gamma-1}$
K_2	$\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\gamma/\gamma-1} \Big/ \left(1 + \frac{\gamma(\gamma - 1)}{2} M^2\right)$
K_3	$\left(1 + \gamma M^2\right) \Big/ \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\gamma/\gamma-1}$
L	length
LCV	calorific value of fuel
M	Mach number
N	shaft speed
P	pressure
PR	pressure ratio
Q	non-dimensional mass flow ($W\sqrt{T}/P$)
R	gas constant
T	temperature
V	volume
W	mass flow
α	blade setting indicator
γ	ratio of specific heats
η_{cc}	combustion efficiency
τ	torque

<u>Subscripts</u>

1	LP compressor inlet
2	HP compressor inlet
3	HP compressor outlet
B	bypass
C	compressor
F	engine fuel
H	hot stream
i	i th stage
S	static
ss	steady state
T	turbine

1. INTRODUCTION

Modern aircraft powerplants have very complex control systems and it is generally accepted that computer simulation of the powerplant behaviour has a considerable part to play in the development of these control systems. Suitable simulations allow a more cost effective and wider insight into the problems, including exploration of the extreme limits of the engine operating range without risk.

This paper describes the normal modelling approach used at NGTE to simulate engine behaviour and hence facilitate the development of control systems. However, anomalies associated with shaft responses exist in these models when compared with actual engine runs and a physically plausible solution has been developed and is presented.

In addition the advent of digital electronic control systems opens up the opportunity to control variables such as compressor stator rows or bleed flows to optimise engine performance. In order to investigate these possibilities an alternative model is described which enables the surge behaviour of compressors to be predicted.

2. THE ENGINE MODEL

The model described in the first part of the paper is based on the method developed by Saravanamutoo and Fawke[1]. The gas turbine is broken down into a series of discrete components and inter-component volumes and a thermodynamic analysis applied. Details of a full thermodynamic model as used by NGTE have been described by Cottington[2] and therefore only a brief outline is given in this paper.

The major assumption is that the performance of each component can be described by the steady state characteristics and that dynamic equations can be introduced to describe the transient behaviour between the components. Derivation of the dynamic equations and the assumptions made have been reported by Foss and Beard[3]. For brevity only the form of equations used is presented here.

For a volume where no heat is released the equations of continuity and state may be used to derive the pressure P.

$$\frac{d}{dt}(P) = K_1 \frac{\gamma RT}{V}(W_{IN} - W_{OUT}) \qquad \text{....(1)}$$

For volumes where heat release does occur, such as the combustion chamber or reheated jet pipe, the energy equation must also be included and the resulting dynamic equations take the form:

Continuity:

$$\frac{d}{dt}\left(\frac{P}{T}\right) = \frac{RK_1}{V}(W_{IN} + W_F - W_{OUT}) \qquad \text{....(2)}$$

which gives:

$$\frac{d}{dt}(T) = \frac{T}{P}\left(\frac{d}{dt}(P) - \frac{K_1 RT}{V}(W_{IN} + W_F - W_{OUT})\right) \qquad \text{....(3)}$$

Energy:

$$\frac{d}{dt}(P) = \gamma RK_2 \left(W_{IN}\, T_{IN}\, C_P + W_F\, LCV\, \eta_{cc} - W_{OUT}\, T_{OUT}\, C_P\right) \qquad \text{....(4)}$$

For shafts the speed N is determined by the mismatch of power between the compressors and turbines. The rate of change of shaft speed is therefore given by

$$\frac{d}{dt}(\tfrac{1}{2} I N^2) = G_T - G_C \qquad \text{....(5)}$$

The power difference can be expanded as:

$$G_T - G_C = W_T\,(T_{IN} - T_{OUT})_T\, C_{PT} - W_C\,(T_{OUT} - T_{IN})_C\, C_{PC} \qquad \text{....(6)}$$

Finally for some simulations an equation is required to take into account the mixing of bypass and core flow. Cottington[4] has pointed out the difficulties associated with this control volume and he suggested the approach currently used at NGTE. A mixing plane is introduced at the entrance to the jet pipe after which the two gas streams are fully mixed. In the steady state it is then assumed that the static pressures of the hot, cold and mixed streams are equal at this plane. The rate of change of flow across the mixing plane will transiently be determined by the inbalance of the static pressures across the plane and the momentum equation may be invoked.

For the cold stream:

$$\frac{d}{dt}(W_B) = \frac{A_B}{L_B}(P_{SBIN} - P_{SOUT}) \qquad \text{....(7a)}$$

and for the hot stream:

$$\frac{d}{dt}(W_H) = \frac{A_H}{L_H}(P_{SHIN} - P_{SOUT}) \qquad \text{....(7b)}$$

The air flow emerging from the mixing plane is the sum of the two flows W_B and W_H. If the mixing losses are known these can be incorporated into the jet pipe losses as an overall jet pipe pressure loss.

Making use of the above equations allows a complete thermodynamic model of an engine to be developed. A typical information flow diagram of a 2-shaft bypass engine is given in Figure 1.

3. ACCURACY OF SIMULATION

3.1 Steady state accuracy

The accuracy achieved from this form of model is open to some debate. The initial emphasis is to obtain a good steady state description and this is inevitably dependent on the engine data available. The combination of individual component characteristics as measured on rig tests is not always a good representation of the complete engine. It is therefore desirable to have a comparison of results from the computer simulation and a real engine, because then a certain amount of optimisation is possible. At NGTE Spey Mk 502 engine test data have been compared with a computer simulation. Initially there were differences between predicted and measured steady state values. In order to overcome this a computer program was written to calculate the unknown steady state values of the engine from those that were measured. The program itself used the same basic thermodynamic equations as those in the model. This enabled a steady state working line to be superimposed on the characteristics which were then modified, using an iterative procedure, to provide a consistent steady state match.

When the adjusted component characteristics were introduced good agreement was obtained between predicted and measured results. Hence if the data are available a good match can be obtained.

3.2 Perturbation tests

Although there has been some mismatch under steady state conditions, the major problem has occurred for transient manoeuvres. In order to obtain some direct comparison between the engine and simulation a frequency response analysis was first carried out using Pseudo Random Binary Noise (PRBN) perturbation, applied separately to the fuel flow and nozzle. This was a comprehensive analysis and only sample results are included here in Figure 2 to give an indication of the correlation between engine and simulation. The frequency responses were obtained with the perturbations applied at shaft speeds of 92% N_H and 74% N_H. The output parameters measured were low pressure compressor delivery pressure, high pressure compressure delivery pressure, low pressure turbine exit pressure and the two shaft speeds. It can be seen from Figure 2 that some mismatching occurs between the engine and simulation results. However, the essential shapes of the curves are very similar and the degree of mismatch is satisfactory as a basis for small perturbation transfer function determination. Nevertheless various approaches were attempted to improve the correlation. These attempts included varying the control volumes, shaft inertias and applying fuel lags and delays. Although these variations had some minor influence, none were found to improve the standard model sufficiently to justify their use.

3.3 Large scale transients

The purpose of these tests was to determine the response of the engine well away from its steady state running line and was achieved by applying a fast change in fuel flow. The applied fuel schedule consisted of a step change followed by a ramp, the magnitudes of which were varied. Figure 3 compares the slam accelerations of the engine and simulation. It can be seen that the engine has accelerated much more slowly than the simulation and in addition both engine and simulation have settled to what are apparently different steady state conditions. In reality the simulation had indeed reached a steady state condition but the engine had not and after a further five to ten minutes it gradually achieved the same conditions predicted by the simulation. This behaviour is attributed to long term heat soakage effects changing clearances and was not represented in the simulation.

Quite clearly, although the simulation has a relatively good steady state match, comparison of large scale transients indicates that the simulation has not represented all aspects of the engine behaviour under these conditions. The results quoted are for a Spey because that engine had the most detailed information available. However, the general conclusion also applies to thermodynamic simulations of other engines to a lesser or greater extent.

4. MODEL IMPROVEMENTS

Several approaches were tried in an attempt to improve the simulation during slam accelerations. These included reducing HP turbine efficiency, changing control volumes, imposing lags on HP compressor mass flow and air bleed, changing the shaft moments of inertia and varying combustion chamber efficiency. Of these approaches only the latter two had any great effect.

Changing the shaft moments of inertia is the simplest and most convenient method of overcoming the problem. However, the factor varies from simulation to simulation and indeed from shaft to shaft, which means that trial and error has to be used to determine the correct combinations. A more complicated approach can also be adopted in which shaft inertia is scheduled against some parameter such as heat soakage, but this again requires considerable attention to detail and increases the computation. Results achieved by using a fixed combination of inertia factors are shown in Figure 3. A cursory glance suggests this has given the desired result but a more detailed analysis shows that changing the inertias can also affect the small scale transient response, sometimes in an adverse way. Care therefore has to be taken in adopting this solution.

It can be argued that given the work involved, this simple solution is satisfactory as it only involves changes to one or two numbers in the model equations. However, some of our models now would require variations of inertia for different operating conditions and this therefore increases the complexity of the technique. In addition to these arguments, the biggest criticism of this approach must be that there is absolutely no physical justification for increasing the shaft inertias.

The second approach of varying the combustion chamber efficiency[4] was more fundamentally appealing. It effectively reduces the rate of energy release which has some physical justification and, if the detail can be kept to a minimum, it would be as convenient as the inertia technique. The argument for this approach is that the combustion efficiency falls off in relation to over-fuelling. This can be interpreted as either a reduced degree of atomisation or a temporary distortion of the stoichiometry of the combustion zone.

The approach adopted by Cottington[4] was to assume

$$\eta_{cc} = \lambda \cdot \eta_{ccss} \qquad \text{....(8)}$$

where

$$\lambda = 1 - K\left(\frac{\Delta W_F}{W_{Fss}}\right) \qquad \text{....(9)}$$

and

$$\eta_{cc} = \text{combustion chamber efficiency}$$
$$\eta_{cces} = \text{equilibrium combustion efficiency}$$
$$\Delta W_F = \text{over-fuelling}$$
$$K = \text{constant}$$

The value of K was adjusted to fit the measured engine data and Figure 3 presents a comparison between engine and simulation. Cle rly the correlation between them is good, however this work was carried out for an engine under sea level static conditions. When further work was attempted at altitude conditions, using a simulation of a three shaft bypass engine, there was no unique curve for W_{Fss} covering the range of ambient conditions, even when using the normal non-dimensional groups. A new approach was therefore adopted using a correction factor based on excess torque and took the form shown below.

$$\eta_{cc} = f(x)\,\eta_{cces} \qquad \text{....(10)}$$

where

$$x = \left(\frac{\tau_T - \tau_C}{\tau_C}\right)\frac{1}{\delta\sqrt{\theta}} \qquad \text{....(11)}$$

and

$$f(x) = \frac{1}{1 + x} \qquad \text{....(12)}$$

δ = relative air pressure (P_1/1.013)
θ = relative air temperature (T_1/288)

In order that the combustion efficiency factor could be used at altitude conditions it was necessary to include the non-dimensional term ($\delta\sqrt{\theta}$). Figure 4 shows the results of a slam acceleration under sea level static conditions for the three shaft bypass engine, compared with the factored and unfactored simulation results. As can be seen a considerable improvement has been achieved. In addition, to enable a comparison to be made at altitude, some flight results were obtained covering a slam acceleration at 10.76 km and Mach 0.8 and these results are compared in Figure 5 with results obtained from the simulation. Again good agreement has been achieved.

In order to obtain a direct comparison between the inertia factor and combustion efficiency factor they were first calibrated by using the simulation under a controlled acceleration. The shaft inertia factor was then increased by a fixed value of 72% which resulted in the two techniques giving almost identical results. Having calibrated in this way a direct comparison was carried out using a fuel spike, ie a fast application followed by a fast removal of fuel, the total transient lasting approximately 1.25 seconds. The shape of the actual fuel spike is shown in Figure 6 together with a stepped function as achieved by the computer implementation, updating every 25 m sec. The results achieved are given in Figure 7 which shows a comparison of the responses of the shaft speeds against time for the two factored simulations and actual engine results. However, it should be pointed out that the simulation was not matched closely in the steady state to the engine, because of the dearth of experimental data.

Clearly, the main discrepancies occur for the IP and LP shaft responses. Comparison of the IP shaft results shows that the response of the shaft for both forms of factor is very similar and no advantage is indicated for either. However, this is not the case for the LP shaft, where there is a much greater divergence between the two correction factors. The engine results can only give a qualitative comparison but they indicate that the combustion efficiency factor is more appropriate.

The combustion efficiency factor has only been tried on one simulation at altitude and two at sea level static conditions. Additional engine simulations need to be compared and much more experimental data are required from actual engines. However, indications are that the simulation with the combustion efficiency factor is more accurate, physically plausible and is relatively easy to implement. Furthermore, because the combustion efficiency factor is associated with large transients this modification does not affect the results of frequency response analysis whereas the inertia factoring can have an adverse effect.

5. ALTERNATIVE MODELLING TECHNIQUES

Compressor surge is a major consideration in the design of gas turbines and their control systems. The compressor designer must provide adequate margin between engine working line and surge line, and the control system must ensure that sufficient of that margin is retained during transients, in particular during a slam acceleration. The prediction of surge and working lines during both steady running and fast transients is therefore of major importance.

To provide sufficient surge margin at off design conditions, variable geometry or bleed valves may be provided. It is probable that greater advantage may be achieved from these variables if their control, either closed loop or scheduled, was integrated within the overall engine control scheme and could constitute a major contribution to surge avoidance.

The control strategy required to achieve this is not yet understood, and no satisfactory basis exists for optimising advanced control systems for surge avoidance. This is not a task that can be undertaken by experiments on some long-suffering engine due to the high costs involved and the possibility of damage; recourse must be made to simulation as the only practical alternative.

Corbett and Elder[5] have demonstrated that compressor simulations, constructed by assuming quasi-steady measured stage characteristics linked by stage volumes described by unsteady flow equations, can exhibit surge-like instabilities in the appropriate operating region. NGTE have adopted this approach in modelling a complete gas turbine. The simulation was based on a two shaft turbofan having a bypass pressure ratio of 4 to 1 with no mixing, and includes an eight stage version of an experimental research compressor containing three rows of variable geometry.

Three models have been developed:

i. Basic simulation using current NGTE techniques, ie lumped continuity model.
ii. Application of modified Corbett-Elder equations using overall compressor characteristics, ie lumped momentum continuity model.
iii. Detailed simulation incorporating modified equations and full stage-stacked compressor model, ie stage-stacked momentum continuity model.

The philosophy behind the three model approach is that it enables the two major differences between the established techniques (ie the model equations and the stage-stacking) to be considered separately.

5.1 The Lumped Continuity Model

NGTE models as outlined in the first part of the paper are described as 'lumped continuity models' because of the manner in which compressors and turbines and their intercomponent volumes are represented. The underlying assumptions are:

i. Momentum effects are associated with much shorter time scales than those exhibited by the other equations. Assuming them to be instantaneous, they have little overall effect on the engine dynamics and therefore can be ignored.
ii. Little or no energy is added to the gas stream during its passage through the intercomponent volumes. Conservation of energy can therefore be approximated by assuming adiabatic conditions in these volumes with temperature ratios across components given by their steady state characteristics.
The equations described earlier apply to this model.

5.2 The Lumped Momentum Continuity Model

Corbett and Elder[5] have shown that the method described above, but with the momentum equation included, is capable of predicting surge on an isolated compressor modelled stage by stage. In order to separate the effects of the momentum equation from the stage by stage approach a 'lumped momentum continuity model' was developed. The additional momentum equation takes the form

$$\frac{d}{dt}(W) = \frac{K_3 A}{L}\left(P_{IN} - P_{OUT}\right) + \frac{F_{net}}{L} \qquad \ldots.(13)$$

where F_{net} would, in a gas turbine, represent pressure forces acting on the walls and drag forces.

In steady state

$$F_{net} = K_3 A \left(P_{OUT} - P_{IN}\right) \qquad \ldots.(14)$$

now the pressure ratio may be represented by

$$PR = \frac{P_{OUT}}{P_{IN}} = f\left(\frac{W\sqrt{T_{IN}}}{P_{OUT}}, \frac{N}{\sqrt{T_{IN}}}\right) \qquad \ldots.(15)$$

Hence

$$F_{net} = K_3 A\, P_{OUT}\left(1 - \frac{1}{PR}\right) \qquad \ldots.(16)$$

In a similar fashion to Corbett and Elder[5] it is assumed that the above steady state expression for F_{net} is also satisfied transiently, ie the net force is equal to that experienced in steady state for similar inlet conditions. The resultant expression for the equation of momentum becomes

$$\frac{d}{dt}(W) = K_3 \frac{A}{L}\left(P_{IN} - \frac{P_{OUT}}{PR}\right) \qquad \ldots.(17)$$

The information flow diagrams for the two approaches are given in Figure 8.

5.3 The Stage-Stacked Momentum Continuity Model

The object of this model is to study the performance of the HP compressor and therefore the remainder of the simulation has been kept as simple as possible. This led to the following simplifications:

i. No internal airbleed or power offtakes.
ii. Components, other than the HP compressor, have fixed efficiencies.
iii. Specific heats are constant for each component.
iv. Pressure losses in the combustion chamber, bypass duct and jet pipe are constant percentages of the inlet pressure.
v. Combustion efficiency is constant.
vi. Nozzle discharge coefficients are constant.
vii. Isentropic flow with $\gamma = 1.4$ occurs in both final nozzles.

The first two models have simulated the compressor as a single entity using overall steady state characteristics. The stage-stacked model uses the momentum continuity approach but simulates each stage of the HP compressor separately using the individual steady state stage characteristics. Thus the model becomes complex in terms of computation as the compressor is now represented by eight sets of equations as opposed to one for the previous models. Details of the derivation of the equations for all three models are given in references 3 and 6.

This model has two main advantages. Firstly it allows the compressor stage to be transiently mismatched, which leads to the possibility of simulating the onset of surge, and predicting the transient movement of the surge line. Secondly the simulation should be capable of being used to determine the optimum settings for the variable geometry both under steady state conditions and transiently.

6. APPLICATION TO A HP COMPRESSOR WITH VARIABLE GEOMETRY

The HP compressor modelled has eight stages with the inlet guide vane and the first two stators having variable blade angles. Each stage consists of a rotor followed by a stator with the exception of the first and last stages which in addition include the inlet and outlet guide vanes respectively. Thus the first stage contains two variable surfaces and the second stage the remaining one.

Stage data were obtained from a computer prediction using a program developed by Calvert[7]. Starting from a geometric description of the compressor, the program predicts the overall characteristics, the individual stage characteristics and the surge and choke points. The program was run for five different blade settings covering the likely range of operation. The values are given in Table 1.

It was assumed that the performance of each stage would depend only on the geometry of the stator immediately preceding that stage. In order to validate this assumption, the program was run using two sets of mismatched geometry (Table 2). Comparisons of the characteristics for a particular blade setting, when the neighbouring blades were matched and unmatched, showed this assumption to be valid provided the shaft speed was below 90% of design.

Hence, for the stage stacked model, the first three stage characteristics depend only on the setting of the inlet guide vane and the first and second stators respectively

$$\left.\begin{aligned} PR_i &= f_i\left((W\sqrt{T}/P)_i,\ (N/\sqrt{T})_i,\ \alpha\right) \\ TR_i &= g_i\left((W\sqrt{T}/P)_i,\ (N/\sqrt{T})_i,\ \alpha\right) \end{aligned}\right\}\quad i = 1,2,3 \qquad \ldots\ldots(18)$$

Furthermore the performances of the last five stages are independent of the variable geometry

$$\left.\begin{aligned} PR_i &= f_i\left((W\sqrt{T}/P)_i,\ (N/\sqrt{T})_i\right) \\ TR_i &= g_i\left((W\sqrt{T}/P)_i,\ (N/\sqrt{T})_i\right) \end{aligned}\right\}\quad i = 4 - 8 \qquad \ldots\ldots(19)$$

6.1 Transient comparisons

The stage-stacked and lumped models were compared by looking at their open loop responses to a change in fuel flow. A fuel step of 0.3 to 0.37 kg/s passing through a 1 ms lag was used, with the variable geometry fixed at $\alpha = 0.5$. Figures 9 and 10 compare the responses between the models. In Figure 9 the response of compressor delivery pressure P_3 and compressor inlet mass flow W_2 are shown. Initial steady state discrepancies are due to interpolation errors. Transiently, the two lumped models show very little difference while the stage stacked model, although qualitatively similar, exhibits a larger and longer disturbance. The lumped models show an initial drop in W_2 of 1 kg/s with recovery within 15 ms while the stage stacked model shows a drop of over 2 kg/s and recovery after 25 ms. The differences in the models on the compressor map are shown in Figure 10 where overall pressure ratio PR_{32} is plotted against non-dimensional mass flow Q_2. All three loci move towards surge, actually passing the steady state surge line predicted by Calvert[7], the excursion being significantly greater for the stage stacked model. The pressures and flows for the individual stages showed no obvious differences and the mass flows collapsed onto a single line suggesting little flow mismatch. Nevertheless all the results indicated that it was the first 20 ms which is the most crucial for surge. In practice the rapid rise of fuel used for this comparison is unrealistic as typical fuel systems will limit its rate of change. Figures 11 and 12 compare the responses when the fuel lag is increased to 10 ms. Now all three models give very similar responses with similar excursions towards surge.

6.2 Frequency Response Analysis

In order to investigate the dynamic differences between the models a frequency response analysis was carried out. However, the Dynamic Systems Analysis Suite (DYSAS)[8] used at NGTE could not cope with a complete engine simulation with the stage stacked model, because there were too many differential equations. Thus the analysis was carried out on all three models allowing only the HP compressor and combustion chamber to vary in order to obtain a direct comparison. The first step was to linearise the models and to calculate their eigenvalues for blade settings of $\alpha = 0.5$ and shaft speeds of 95% at sea level static conditions. The results are presented in Table 3.

The two eigenvalues for the lumped continuity model are due to the combustion chamber dynamics and are also found in the other models. The presence of the momentum equation in the lumped momentum continuity model adds another eigenvalue representing the overall compressor response. In the stage stacked model, there are a number of complex eigenvalue pairs showing the presence of high frequency lightly damped pressure/flow oscillations[9]. In addition all eigenvalues have negative real parts; a positive real part would indicate an unstable condition. Figure 13 compares the frequency responses of the three models for the HP compressor inlet flow W_2 versus fuel W_F. As only the compressor and combustion chamber are modelled, these responses assume that the HP shaft speed, the compressor inlet pressure and temperature are fixed and the HP turbine is choked. All show a similar response up to 50 Hz, after which the three models begin to diverge. Thus fuel systems would need a bandwidth greater than 50 Hz for the effects of stage mismatch to be noticeable.

6.3 Surge prediction

One of the objectives for the stage-stacked model was to predict the onset of surge. This was not practical with the hybrid model due to handling difficulties, such as freezing the model condition when the instability occurred, in addition to sampling problems and other effects. However, as described in the previous section, stability can be inferred from the location of the eigenvalues of the linearised system and this method was applied to derive a surge criterion.

The model used was the digital version of the stage stacked compressor with combustion chamber. In accordance with expectation, the further one went to the left on the compressor constant speed lines (ie lower Q_2), the more positive the eigenvalues became and hence the more unstable the system. Using this approach a steady state surge line was determined by iterative calculation of the eigenvalues for ten constant speed lines with blade angle settings of $\alpha = 0.5$. The model conditions being:

Inlet pressure P_2 = 1.013 bar
Inlet temperature T_2 = 288 K

Figure 14 shows the results of the analysis and allows a comparison with the predicted surge line of Calvert[7] using an empirical approach. The methods give very close agreement over mid range speeds (80-100%), but differ at the two extremes. At low speeds, the Howell[7] method predicts surge to be at the condition of greatest pressure ratio on the constant speed lines, whereas the eigenvalue prediction is to the right of this. The limited experimental data available from a nine stage version of the compressor tends to confirm the eigenvalue predictions. At high speeds however, the eigenvalue method predicted no surge line, even though the analysis was carried out well beyond the Calvert surge line. The reasons for this are not understood but two possibilities exist.

i. Insufficient data, the data points were very close together at these speeds with poor definition of curvature.
ii. A different mechanism is precipitating the onset of surge as predicted by Calvert.

The stage stacked model assumes one dimensional flow and is therefore not capable of modelling rotational disturbances which may initiate surge. Nevertheless, it does aim to determine whether any disturbances, whatever their cause, are longitudinally unstable. Therefore the prediction of a zero or positive eigenvalue is expected to be sufficient to indicate surge but is not a necessary condition for a flow instability. Examination of Figure 14 suggests that the instability is surge at low and mid-speeds, but another instability is excited first at high speeds.

Finally comparison of Figures 10 and 14 gives some insight to the advantages of the stage stacked simulation. The Calvert and eigenvalue predictions of the surge line match very closely, and from Figure 10 it can be seen that the transient locus passed the predicted steady state surge lines without any instability. Hence it can be concluded that transiently the surge line, for the stage stacked simulation, must have moved to the left. No further results are available but these do suggest that the stage stacked model could be a powerful form of simulation for predicting surge during transients.

7. DISCUSSION AND CONCLUDING REMARKS

This paper has reviewed the modelling techniques currently used at NGTE for transient simulation of gas turbines in support of control law development work. It has outlined some of the drawbacks and in bringing these to the forefront it has attempted to provide relatively simple solutions to the more obvious faults. By so doing it has been shown that the simulations can be made to match the steady state performance relatively easily and factors introduced to improve the transient performance have been included with little cost in computing time. In addition it has emphasised the lack of data available for direct comparison of simulations and real engines and has suggested that for fast transients, experimental data achieved by using fuel spikes is a useful alternative which can be adopted. In general the simulations can run in real time or tenth real time and are ideally suited for control engineering purposes.

In addition the paper has described the modelling of a gas turbine using a stage-stacked compressor representation. Initial investigations into the capabilities of this model and differences between the earlier thermodynamic models have been presented.

The model predicts a compressor flow instability which agrees well with the expected surge line over low and mid speed ranges and calculation of the eigenvalues was shown to be a reliable and convenient way of obtaining the data. The technique in addition to predicting steady state surge lines can also be used to determine the distance from instability during a transient.

It has been shown that transient stage mismatch does alter the behaviour of the compressor but the effect lasts for less than 25 ms. After such time the compressor may be regarded as matched. This effect may only be seen in response to a very rapid rise in fuel, above the capability of present day fuel actuators. In addition, during this transient the steady state surge line was passed with no instability thus indicating that the surge/instability line varies transiently.

With modern day engines becoming more complex it may be necessary for the control engineer to make use of this form of model in order that he can take full advantage of the variable geometry. Such a model will also make it possible to develop techniques for running closer to the surge line thus obtaining greater optimisation of the powerplant as a whole.

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Reports quoted are not necessarily available to members of the public or to commercial organisations.

9. ACKNOWLEDGEMENTS

The authors would like to acknowledge the assistance given by their colleagues at NGTE without whom the work could not have been done. We are also grateful to Rolls-Royce for supplying engine data for developing and testing the simulations.

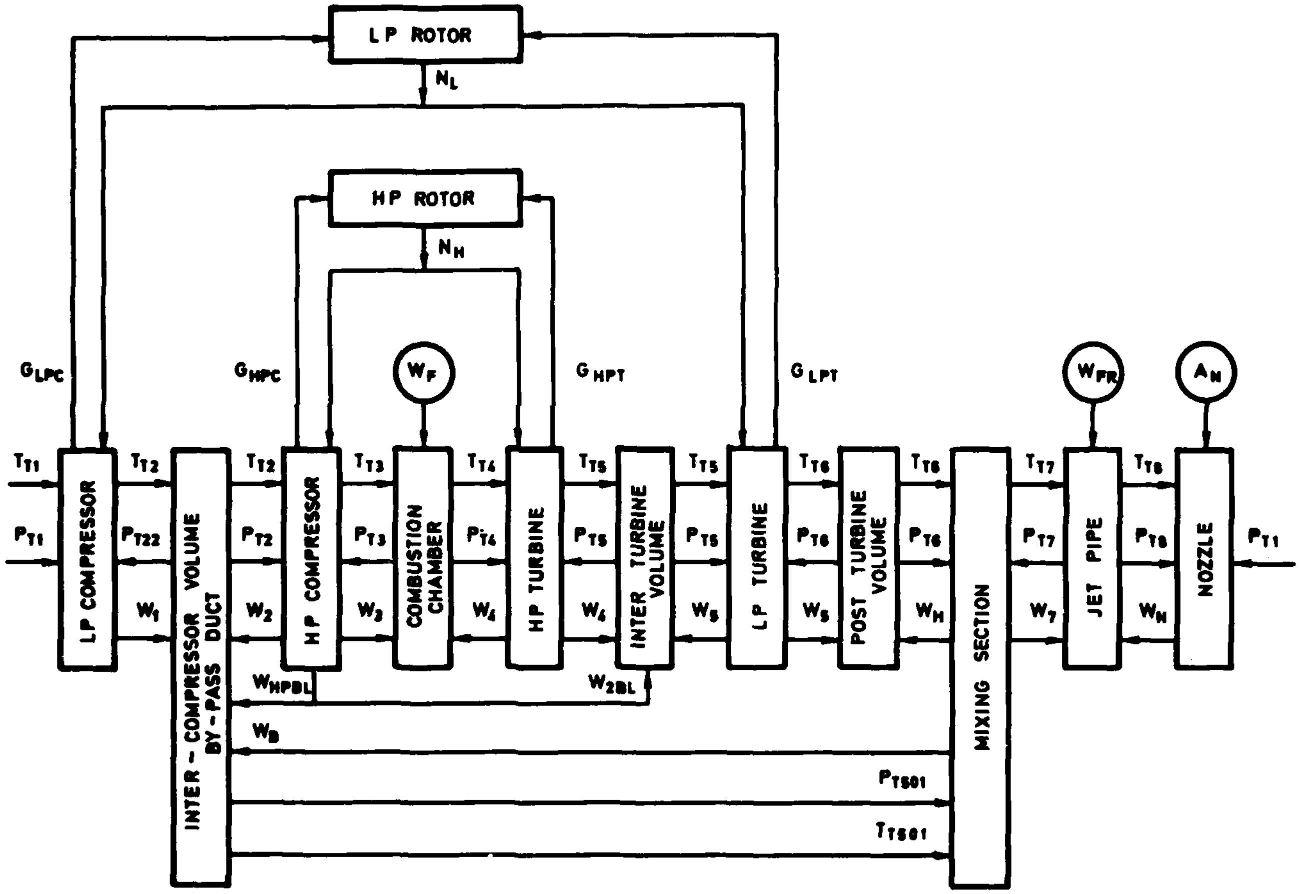

FIG.1 MODEL INFORMATION FLOW

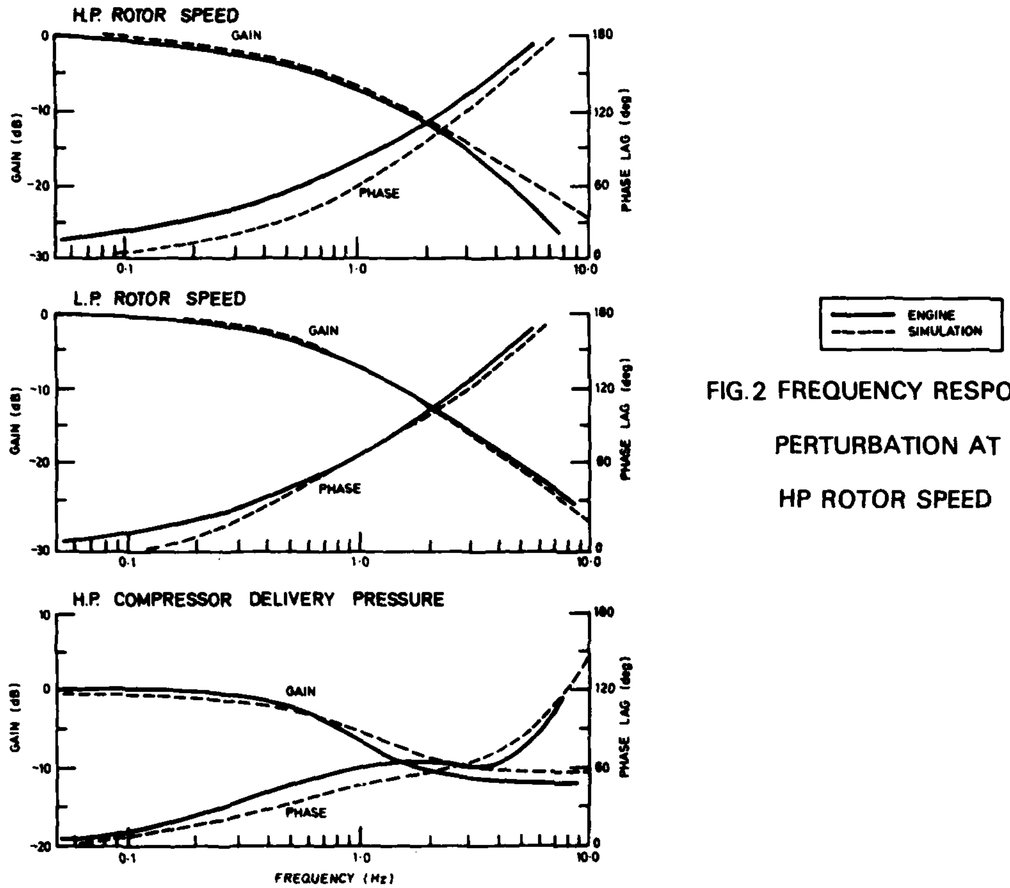

FIG.2 FREQUENCY RESPONSE TO FUEL PERTURBATION AT 92% HP ROTOR SPEED

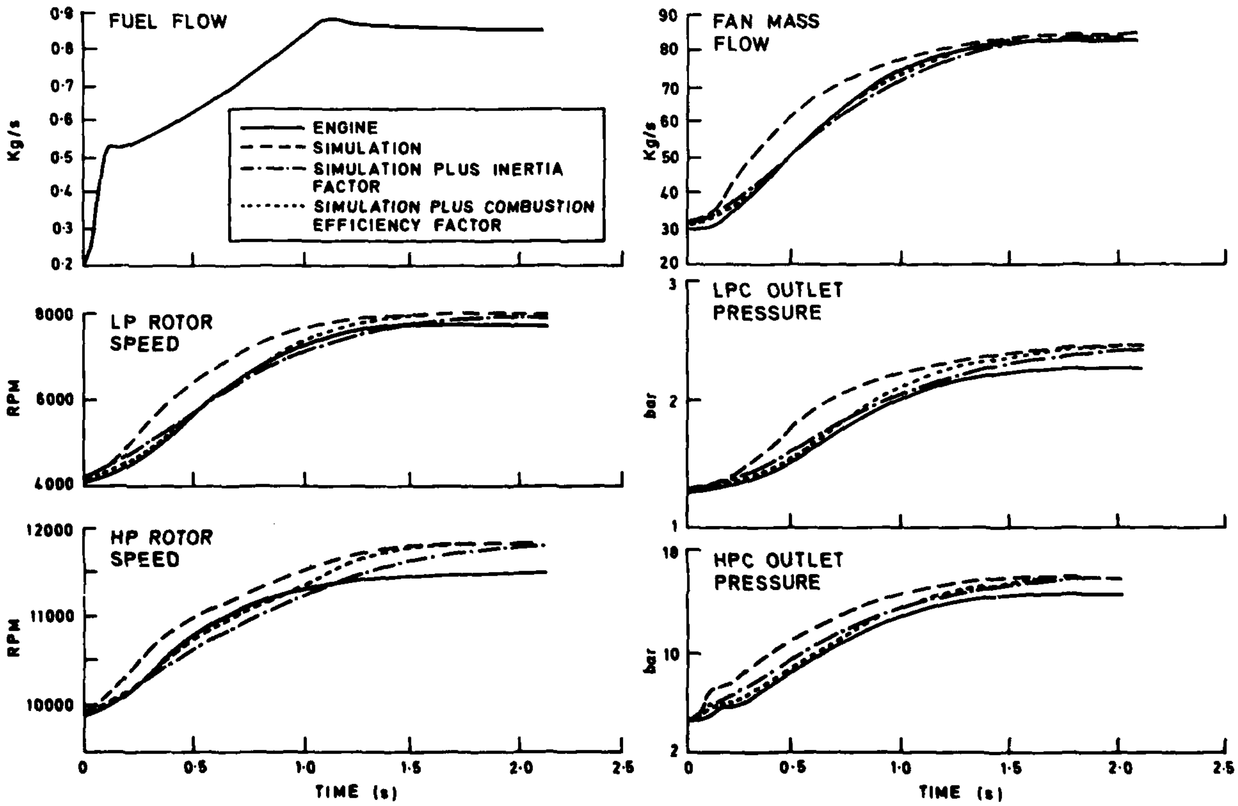

FIG.3 LARGE TRANSIENT-ACCELERATION

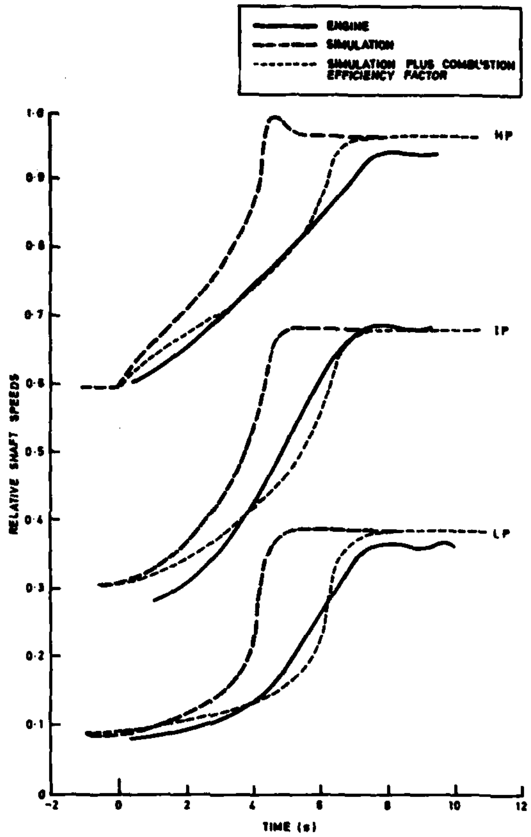

FIG.4 SLAM ACCELERATION AT SLS

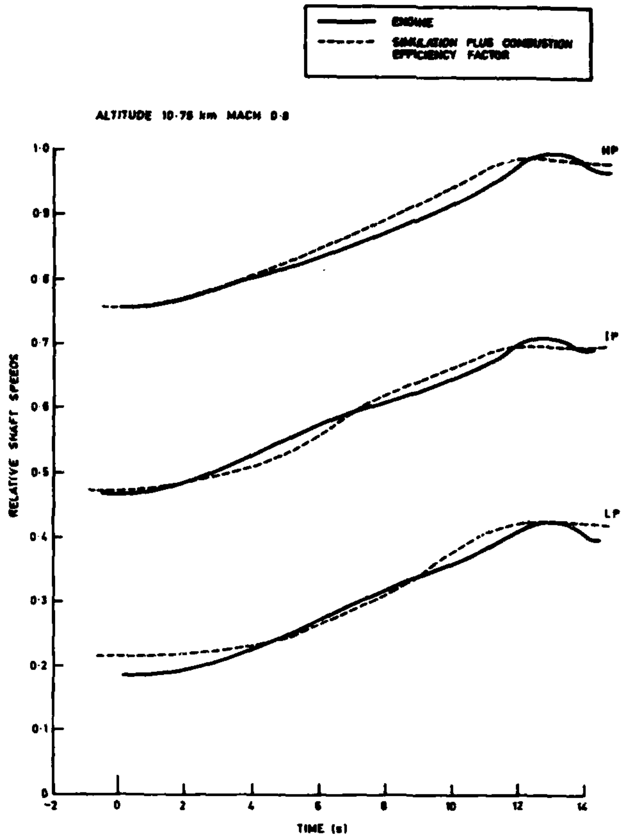

FIG.5 SLAM ACCELERATION AT ALTITUDE

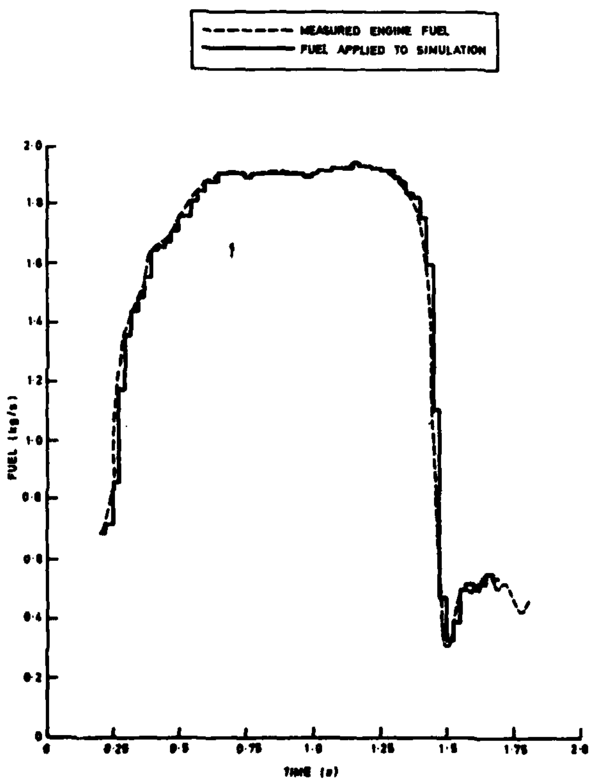

FIG.6 FUEL SPIKE

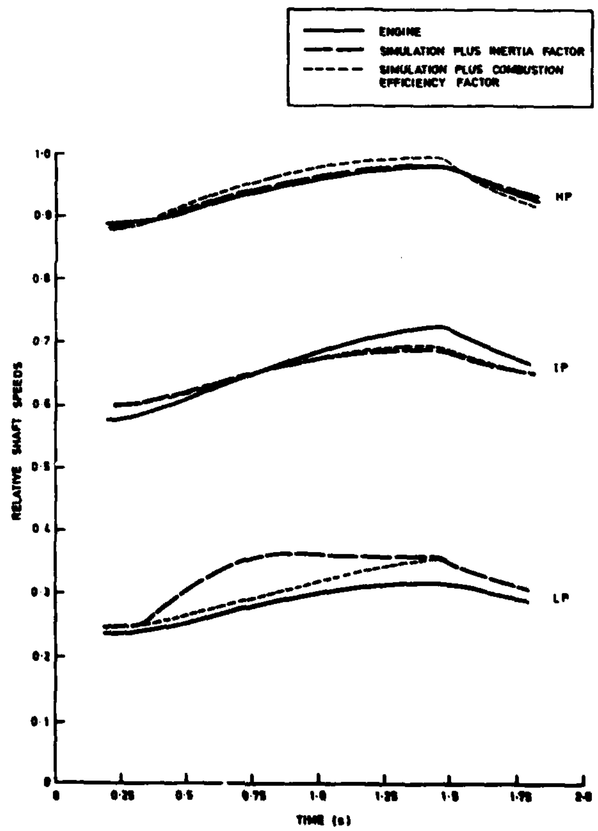

FIG.7 RESPONSE OF SHAFTS TO FUEL SPIKE

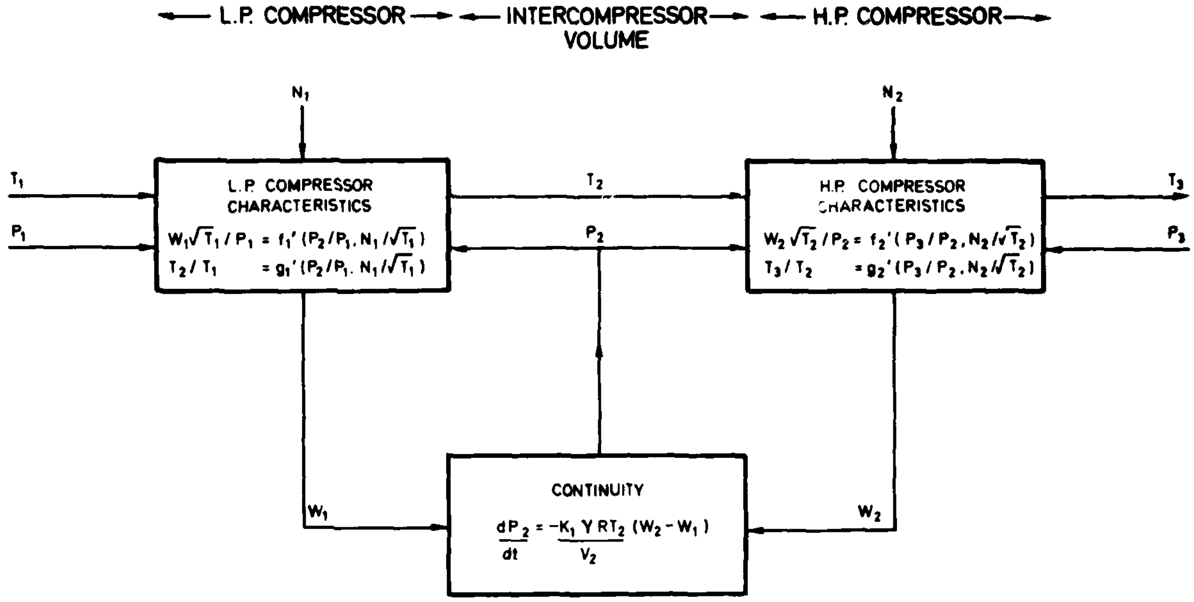

FIG.8a CONTINUITY MODEL

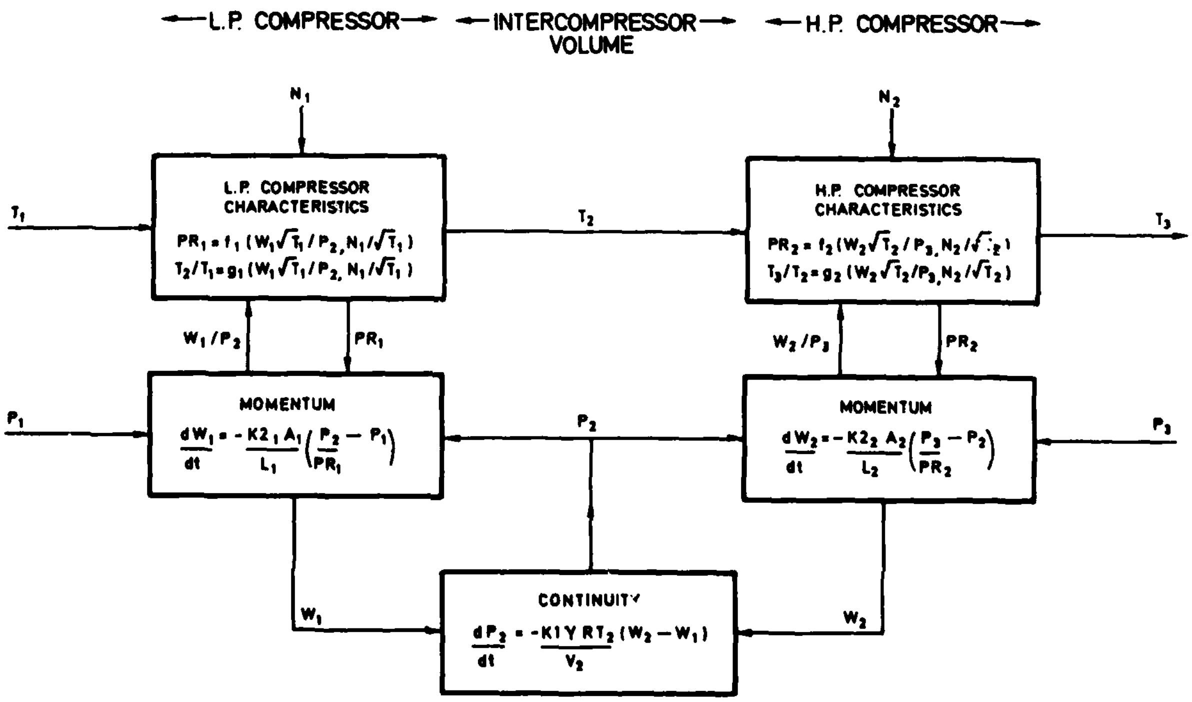

FIG.8b MOMENTUM CONTINUITY MODEL

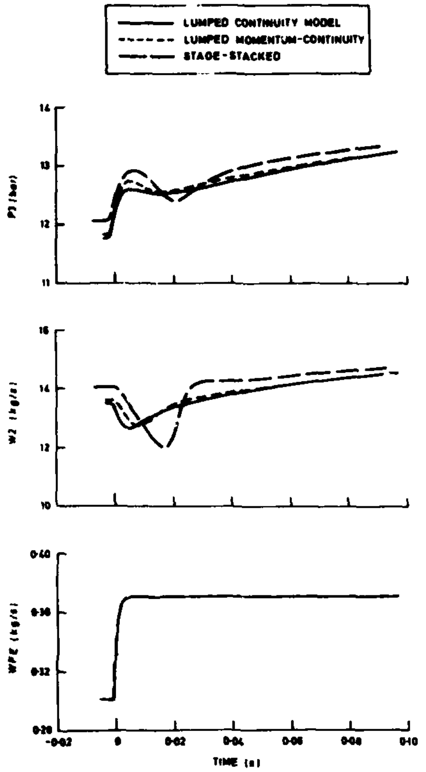

FIG.9 RESPONSE TO 1ms FUEL LAG

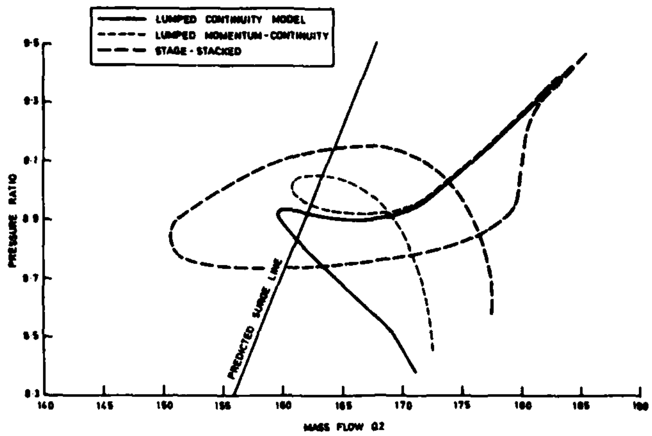

FIG.10 RESPONSE OF HPC TO 1ms FUEL LAG

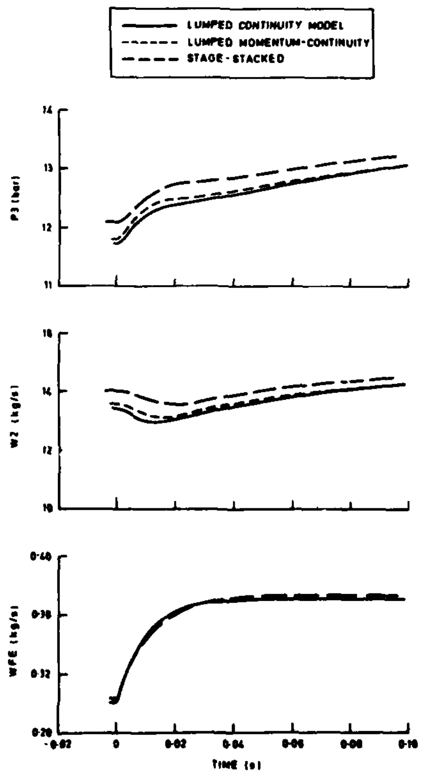

FIG.11 RESPONSE TO 10ms FUEL LAG

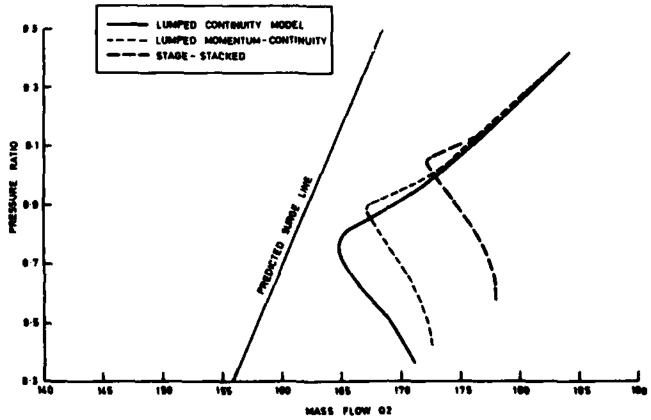

FIG.12 RESPONSE OF HPC TO 10ms FUEL LAG

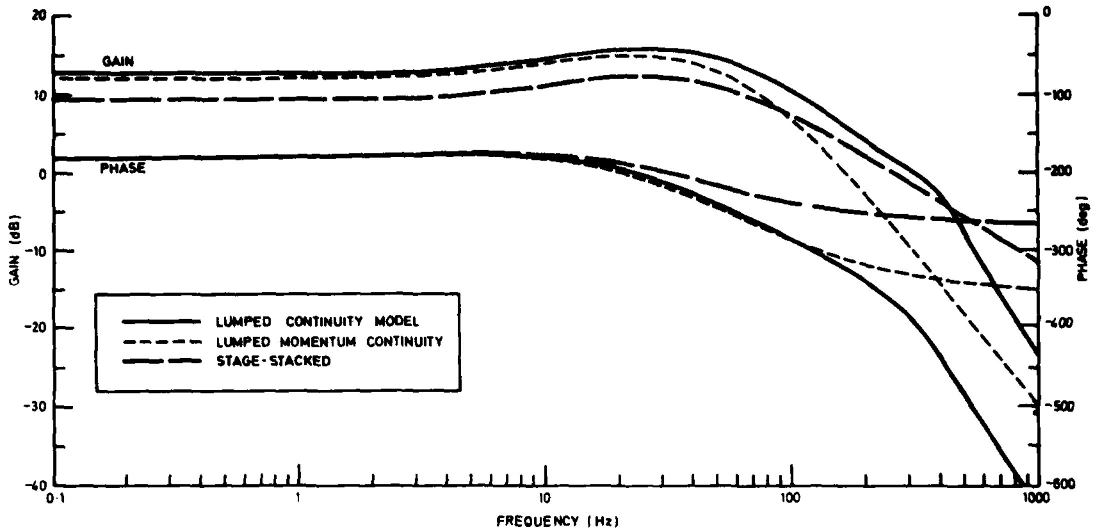

FIG.13 FREQUENCY RESPONSE W2 V WF

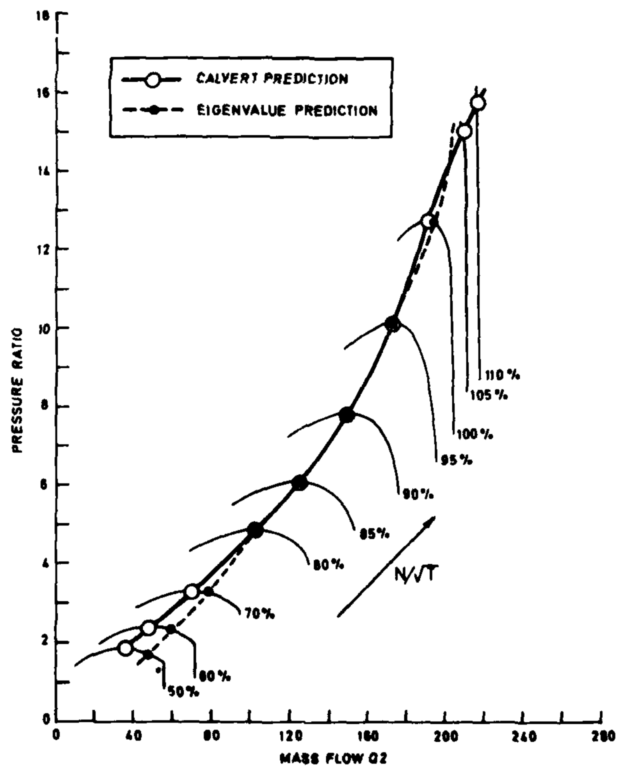

FIG.14 PREDICTED COMPRESSOR SURGE LINES

BLADE SETTING INDICATOR α	BLADE ANGLE RELATIVE TO NOMINAL SETTING - DEGREES		
	IGV	STATOR No. 1	STATOR No. 2
0	-10	-5	$-2\frac{1}{2}$
0·25	0	0	0
0·50	10	5	$2\frac{1}{2}$
0·75	20	10	5
1·00	30	15	$7\frac{1}{2}$

TABLE 1 BLADE SETTINGS

TEST	BLADE ANGLE RELATIVE TO NOMINAL SETTING - DEGREES		
	IGV	STATOR No. 1	STATOR No. 2
1	20	0	5
2	0	10	0

TABLE 2 MISMATCHED GEOMETRY

LUMPED-CONTINUITY MODEL
-326.1
-93.39
LUMPED MOMENTUM-CONTINUITY MODEL
-346.7
-96.7
-497.4
STAGE-STACKED MODEL
-346.7
-107.3
-348.7
$-3097 \pm 28060j$
$-2506 \pm 23260j$
$-2440 \pm 18500j$
$-2262 \pm 14080j$
$-1813 \pm 9749j$
$-1299 \pm 5731j$
$-953 \pm 2433j$

TABLE 3 EIGENVALUE PREDICTION $\alpha = 0.5$ $N_H = 95\%$

DISCUSSION

M.J. Fabri, Fr:

Did you try to measure the combustion temperature by means of an optical pyrometer?

Author's Reply:

No, we did not attempt to measure the combustion temperature with an optical pyrometer; at the time this work was carried out the Spey engine had no suitable instrumentation. Furthermore, I am not sure an optical pyrometer would be adequate. The combustion zone would, under the initial phase of the transient, be in a very dynamic state. The pyrometer would measure the effective surface temperature of the combustion zone envelope, not the mean averaged temperature of the envelope volume. At best it would only give a qualitative answer.

The Spey engine has recently had an optical pyrometer fitted to measure the temperature of the second row of the HP-turbine blades. It may therefore be possible in our future programme to measure the rate of change of blade temperature and relate it to combustion performance on a qualitative basis.

K. Bauerfeind, Ge:

I must admit that I am slightly puzzled about the first part of your presentation where you try various "fiddles" in order to make your simulation agree with test results. MTU had hit exactly the same problems back in 1961/62 when dealing with VTOL-engines where a realistic transient simulation was also most essential. Our research work in this area has highlighted the fact that the reason for this discrepancy is certainly not a transient drop of combustion efficiency (as also suggested at the time because of the observed smoke in the exhaust - you can produce a lot of smoke with a fraction of a percent of combustion efficiency). It was clearly established that the always-neglected heat transfer between gas and material is the actual cause of the observed discrepancy. With a Spey-type engine up to 1/3 of the actual overfuelling can thus be absorbed during an accel. Our work has been documented in detail and reported on during an AGARD-meeting in 1968 in Toulouse.

Author's Reply:

We carried out a series of analyses to check whether various factors had an effect on the response of the engine simulation. Only two factors had sufficient effect to warrant detailed discussion in the paper. Nevertheless, we have looked at heat transfer using a method developed by Rolls Royce. We entirely agree with your view that the long time-scale discrepancy is due to heat soakage effects but our calculations do not support that hypothesis for the initial phase of a very fast transient, where the effect was very small. Consideration was also given for the need to incorporate heat transfer into the combustion system; the results from this had even less effect.

Comparison between our results and your own (AGARD 68) is difficult because it was not clear how you obtained your values of enthalpy for Figures 1 and 2. There is also a big difference between the values of heat flow given in these figures and the transient heat flow into the HP compressor blades and casing estimated from heat transfer rates and given in Figure 8. After one second, for example, the HP compressor is absorbing 90 kW whereas 1000 kW are needed to explain the discrepancy. Of course there are other components also absorbing heat, but do they absorb enough? This is the form of calculation we tried on the Spey and it had little effect. Furthermore, you make the assumption efficiency remains constant; should it vary, this would alter your curves in Figures 1 and 2.

We are surprised to hear you do not regard a transient drop in combustor efficiency as a correct explanation in view of the most cogent arguments you produced in 1968 to support that explanation. Indeed, the shape of the efficiency curve you suggest is very similar to the one we obtained from our own analysis.

Final comment by Dr Bauerfeind:

There is no discrepancy between Figures 1, 2 and 8 at all since at the start of an acceleration, say from idle, the vast majority of heat transfer naturally takes place in the hot section of the engine (due to the over-fuelling) and not in the compressors. With respect to combustion performance, the mechanisms I had described in the above paper are in the millisecond-regime and do therefore not apply to the problem you address; a combustion process is so fast that a slam idle-max dry cannot have any significant effect on combustor performance other than that resulting from the increased FAR's during the transient.

TECHNIQUES FOR DETERMINING ENGINE STALL RECOVERY CHARACTERISTICS*

by
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Sverdrup Technology, Inc., AEDC Group
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Arnold Air Force Station, Tennessee 37389 USA

SUMMARY

Stall of a compression system component in a gas turbine engine results in engine surge followed by rapid and rather violent post-stall behavior. The post-stall behavior may be characterized by the response of one component (e.g., rotating stall) or the interaction between engine components. Experimental investigations of engine stall propagation and recovery (or nonrecovery) require techniques which will characterize component transient and dynamic performance and component interactions during the stall and post-stall event. Test, instrumentation, data processing, and analysis techniques have been employed at the Arnold Engineering Development Center in the systematic evaluation of engine stall characteristics. The application of these techniques to the evaluation of stall characteristics of mixed flow augmented turbofan engines is discussed. This includes (1) an overview of stall-inducing test techniques necessary for producing engine stall (recoverable or nonrecoverable) in a simulated test environment; (2) instrumentation necessary in both response and extent for determining dynamics of the engine flow path and engine control response; and (3) digital and analog data acquisition, data processing in digital form, and analysis which primarily utilizes digital techniques (FFT, filtering, time series, cross plotting, etc.) and graphics display.

1.0 INTRODUCTION

Stability of aircraft engines over their operating envelopes is a primary design consideration. The degree of stability varies due to a number of factors such as engine operating condition, inlet flow conditions, age, and control tolerances. The degree of stability is generally a trade-off with performance; that is, for a given engine, increases in performance are attained at the expense of reduced stability margin. As a result, gas turbine engines have experienced stalls or surges throughout their history as aircraft powerplants. Most stalls result in momentary disturbances to engine operation and are self-recovering. Some stalls, however, do not recover, and engine shutdown and restart are required to re-establish normal engine operation. These stalls are generally characterized by low engine speed and high turbine temperatures with no effective thrust. Such stalls are referred to as nonrecoverable stalls, and also, stagnation or hung stalls. The nonrecoverable state (a state in which the engine is not responsive to scheduled control inputs) may be characterized by rotating stall and/or sustained surge or burner oscillations. Prolonged operation in the nonrecoverable state can result in turbine damage due to overtemperature. In recent years, the incidence rate of nonrecoverable stall has increased significantly. This has resulted in increased research into the nonrecoverable stall phenomenon and has required tests of compression systems and engine systems to define, characterize, and understand the nonrecoverable stall problem.

The increased incidence of nonrecoverable stall is directly related to the increased use of the mixed-flow, augmented, turbofan engines in tactical aircraft. Although the engine has advantages of compact size, wide performance range, and flexibility, it is also a highly interactive system in which interaction between components can directly affect the stall/surge characteristics through complicated feedback paths. The point of instability (surge line) for these engines is well defined. Testing and evaluation techniques have been developed[1,2] to determine the effects on engine stability of rapid acceleration and deceleration, inlet distortion (influence on both core and fan compressor), augmentor instabilities, variable geometry scheduling, etc. However, until recently, testing was confined to determining the engine surge line with little attention given to the nature of the subsequent instability. In a like manner, engine design and development were concerned primarily with performance and stability margin as opposed to post-stall behavior and recoverability. Thus, neither analytical tools nor data were available to support the understanding of nonrecoverable stall or the development of nonrecoverable stall avoidance design criteria for use in future engine system development.

Recently, testing has been performed to characterize engine recovery influences and provide the basis for math modeling of engine post-stall phenomena. This paper includes a discussion of the methods/techniques which should be used in the testing and analysis of turbine engine nonrecoverable stall characteristics based on experience at the Arnold Engineering Development Center (AEDC). The paper is based primarily on observations made from two engine tests in a four-engine test program sponsored by the AEDC and the Air

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*The test effort reported herein was performed by the Arnold Engineering Development Center, Air Force Systems Command. Work and analysis for the effort were done by personnel of Sverdrup Technology, Inc., AEDC Group, operating contractor for aeropropulsion testing at the AEDC. Further reproduction is authorized to satisfy needs of the U. S. Government.

Force Wright Aeronautical Laboratories (AFWAL). The engines, engine instrumentation sets, and control interface electronics were supplied by turbofan engine manufacturers.

2.0 TEST VEHICLE

The discussion will be centered around the techniques used in the testing and analysis of the recovery characteristic of mixed-flow augmented turbofan engines. As mentioned previously, this type of engine has a high degree of coupling between components which requires careful consideration of component performance as affected by a rather complex set of interfaces. It is assumed that testing and analysis techniques used to determine recovery characteristics of such an engine could be applied in a less extensive fashion to nonaugmented turbofans and turbojets. In addition, the developing technology concerning stall and rotating stall has shown that observations and simulations of simpler systems, when used to predict characteristics of complex systems, have failed to adequately account for the effect of altered component interactions. A schematic of a mixed-flow augmented turbofan is shown in Fig. 1.

3.0 TEST TECHNIQUES

The general test procedure for determining recovery characteristics is to configure the engine for design or off-design operation, acquire a steady-state data point to document engine conditions prior to stall, and with transient and dynamic data acquisition systems operating, induce an engine stall. The engine is then allowed to continue in the stalled state until a recoverable or nonrecoverable state is established. Post-stall actions are implemented to evaluate their effects on recovery. If a nonrecoverable state is reached and recovery action is ineffective, the engine fuel flow is reduced to zero for engine shutdown based on engine turbine temperature limits or prolonged operation in the nonrecoverable state. Stall inducement, configuration altering, and recovery implementation techniques will be discussed.

The techniques for stall initiation are those which force one of the compression components to its stability limit. The resultant stall of this component then causes total compression system stall and engine surge. Such techniques are nozzle closure and augmentor fuel pulse for fan stall initiation, and mainburner fuel pulse, compressor exit inbleed, and power lever snaps from maximum dry power-to idle-to maximum dry power (Bodie stall) for compressor stall initiation. The nozzle closure technique produces a rather gradual increase in fan backpressure (Fig. 2) even if nozzle closure command is a step command. This is due to the rate at which the nozzle can move to closure in the presence of the operational pressure loads. The minimum nozzle area available with nominal geometry may be insufficient to cause stall. In this case, a nozzle plug or -- in the case of a convergent nozzle -- extension tabs may be used to further decrease exhaust area. A typical installation using a nozzle plug is shown in Fig. 3. Post-stall nozzle area scheduling may affect recovery in producing repeated stalls, particularly if the initial closure is due to a schedule change which is not cleared to the nominal schedule at initial stall.

The augmentor fuel pulse required to stall the fan results in a very rapid rise in fan backpressure (typically less than 0.01 sec, whereas rise due to nozzle closure occurs in approximately 0.1 sec). The pulse is produced by either simultaneously supplying fuel to the fuel nozzle manifolds with the ignition source activated, or allowing the augmentor nominal light-off sequence to proceed with fuel supplied sequentially to the augmentor ring or segment manifolds with ignition activation at a specified point in the sequence. The second method will produce augmentor ignition in the presence of a highly fuel-rich flow field and will result in a higher rise rate in fan backpressure as shown in Fig. 4.

The compressor-initiated stall techniques cause a rise in compressor backpressure. The mainburner fuel pulse and Bodie stall techniques produce stalls due to a rapid rise in burner fuel flow. The mainburner fuel pulse is accomplished by setting a more rapid (or fuel-enriched) acceleration schedule in the control. When engine acceleration is requested, the fuel-enriched schedule will cause a rapid rise in burner pressure and will backpressure the compressor, causing stall. The procedure for obtaining mainburner pulse stalls is somewhat fuel control configuration dependent and may be induced by any procedure which will rapidly introduce fuel into the burner. The length of the pulse (or step) required to allow speed to decrease to a nonrecoverable stall susceptible state may be as long as one second. The characteristics of a fuel step required to induce a nonrecoverable stall are shown in Fig. 5. The fuel step rises to a level three times the normal high power fuel flow rate. However, the effects of the continued high fuel flow rate become more severe as airflow and speed decrease and fuel-air ratio increases. Note that the fuel increase in this case is such that the compressor backpressure is increased with a rather gradual slope to stall.

The Bodie procedure induces stall due to the inability of the engine to accelerate and increase airflow consistent with the rate of fuel flow increase on throttle snap back to high power. Most engines have some form of Bodie stall protection which prevents the rapid fuel flow excursions in moving between decel and accel schedules. This protection must be removed to utilize the Bodie procedure as a stall-inducing mechanism. The Bodie stall is particularly useful for determining the compressor surge/rotating stall boundary.

The compressor exit inbleed procedure to initiate stall has been used in a number of investigations of compressor stability limit. The inbleed airflow (typically 20% of compressor airflow) causes an increase in burner pressure due to the inability of the turbine nozzles to pass the increased airflow. The method is not recommended for tests to determine recovery characteristics. The engine response and transition characteristics following stall are of prime importance. The inbleed flow would affect this transition by inbleeding during the surge cycles (the flow could not be shut down fast enough to prevent this effect). In addition, the energy added to the burner and available to the turbine would reduce rotor speed rolldown rate. Due to the added complexity, alteration of normal engine response, and expense of installing such a system, the inbleed method of stall is not recommended for stall recovery tests.

In conjunction with the stall-inducing technique, other procedures are used to evaluate fan/compressor/fan duct interactions and the effects of altered characteristics of the fan and compressor. This evaluation is accomplished by setting the fan inlet guide vanes and the compressor inlet guide vanes and front stage stators off-schedule. This modifies the compressor and fan characteristics (i.e., head rise and flow coefficient) and changes the interaction between the two in terms of fan backpressure, fan pumping characteristic, compressor and fan airflow, compressor/fan duct flow split or bypass ratio and match between the rotor speeds of the two spools. This also affects the stall sequence between fan and compressor. In addition to altering fan and compressor variable vane schedules, the exhaust nozzle area schedule can be altered to load or unload the fan and change rotor speed match and flow split. Distortion also affects fan and compressor stall margin and the stall sequence. Circumferential and radial patterns may be used. Distortion may promote or degrade recovery depending on the way it alters a critical component's susceptibility to a nonrecoverable stall phenomenon (e.g., rotating stall). At the AEDC, distortion has been generated by both distortion screens and an air-jet distortion generator. The air jet generator with pattern generation under computer control offers the greater test flexibility, although installation and operational considerations may dictate the use of screens.

The general transient characteristics of the engine may be important to recovery. This includes the rate at which rotor speeds decrease and the average energy extraction rate (or power extraction) of the turbines. Since compressors are subject to rotating stall at low speeds,[3] fast rotor speed decay moves the engine rapidly into a nonrecoverable stall susceptible state. The rate of speed decay can be altered by rotor power extraction through a generator/dynamometer load set. In addition, the rate of speed decay can be altered, and surge cycle repressurization level can be affected through a change in the fuel roll-off rate with respect to speed following the initial surge. Slow fuel flow roll-off will increase energy input to the turbine and will also affect backpressure of the compressor, promoting high repressurization. These effects may be important to recovery and should be evaluated.

Other actions which alter compressor post-stall operation are the compressor bleed flow, downstream volume and throttle area, and high power compressor inlet or exit pressure perturbation. The bleed flows (other than for turbine cooling) are generally extracted at mid-compressor and at compressor exit. The mid-compressor bleed changes the pressure profile through the compressor and the flow coefficient in the rear stages. The exit bleed reduces compressor back pressure and effectively increases downstream throttle area. The effects of downstream volume on compressor surge/rotating stall boundary were studied by Grietzer.[4] Increased volume moves the boundary toward surge and away from rotating stall. If this characteristic is evaluated, it is clear that the increased volume must freely communicate with the burner volume. Connecting line area, length, and restriction can combine to produce a compressor-burner volume-increased volume interaction which produces a higher tendency to rotating stall or nonrecovery. The compressor disturbances which may be evaluated are augmentor rumble and burner perturbations which can be generated by use of a high burner ignition rate. The high ignition rate may also affect recovery through sustaining burner operation during surge.

If the engine transitions to a nonrecoverable state (which is desirable in recovery investigation), then recovery action may be taken. Each action is assumed to change the engine operation to a configuration favorable to recovery. Such actions include variable geometry (fan and compressor variable vanes and exhaust nozzle) reset, bleed valve opening, and abrupt fuel flow reduction followed by fuel flow increase to recovery. The compressor exit bleed valve opening and fuel flow reduction provide an effective increase in downstream throttle, tending to clear rotating stall. Opening the mid-compressor bleed valve alters compressor operation as discussed previously. Variable geometry reset changes compressor or fan airflow and stage loading.

The implementation of the stall and recovery techniques and modifications to engine control schedules require modification to the engine control system to accept these input commands. If the control is basically hydromechanical, then hardware modifications may be required to provide the off-schedule or false control inputs. The recovery actions and configuration changes are then input through auxillary electrical and hydromechanical hardware. A schematic of a typical engine condition modification and auxillary activation system is shown in Fig. 6. The stall and post-stall actions are implemented upon receipt of a stall signal. This signal is activated on the basis of the rate of decrease of compressor exit pressure. The action is then taken at some time following stall through a set of relay and delay circuits. Engine configuration setup prior to stall is through adjustment of trimmer motors for variable vane schedule offsets and a similar technique for exhaust nozzle area schedule. Fuel flow reduction may be accomplished through a manually set automatic throttle chop mechanism. These techniques may be implemented through the use of analog or digital control. Digital control generally has more flexibility. Implementation of a system of this type is necessary when the test vehicle control authority is through hydromechanical circuitry. Since engines of current manufacture have hydromechanical controls, these systems are appropriate for investigation of recovery characteristics of such engines.

Control of engine operation by digital processors is in the development stage and will likely be the control mode of future variable cycle engines. Investigation of recovery characteristics of an engine which is configured for full authority electronic control through computer processors greatly increases the flexibility of test technique implementation. A schematic of control functions is shown in Fig. 7. The digital electronic control is replaced for test purposes with a minicomputer. Schedules for fuel flow, variable vanes, and exhaust nozzle area are implemented as digital algorithms. The computer monitors the stall signal (rate of change of compressor exit pressure) and a nonrecoverable stall signal (compressor pressure ratio or a compressor speed-turbine temperature relationship) and issues actuation commands for recovery action which may be delayed in time. The nominal engine control schedules are maintained in the computer and may be restored at any time. The full authority electronic control offers high flexibility in testing for recovery characteristics and is the preferred technique of fundamental investigation of nonrecoverable stall.

4.0 INSTRUMENTATION

The measurement of stall and post-stall operational characteristics requires that special attention be given to engine instrumentation. The determination of engine component interactions requires extensive instrumentation of components and component interfaces. Instrumentation must be considered for steady-state, transient, and dynamic measurement in determining oscillating pressure and thermal disturbances, distortion, combustor characteristics, fuel control parameters, mass flow, facility interaction, rotor speeds, component and engine performance, and events marking stall detection and recovery action initiation. The measurement of transient phenomena is accomplished with fairly standard instrumentation for temperature, speed, fuel flow, variable geometry position/rate and pressure. The measurement of dynamic pressures and temperatures requires use of techniques which are peculiar to stall and post-stall testing.

The aerodynamic phenomena to be characterized during post-stall operation of turbofan engines consist of planar axial perturbations of low fundamental frequency (2-20 Hz) such as surge and surge-like large scale combustor oscillations, and high fundamental frequency phenomena (50 to 100 Hz) such as rotating stall, small scale combustor oscillations and augmentor-induced oscillations. Although these fundamental frequencies are not exceptionally high for conventional pressure measurement, the other components of the phenomena are. For instance, to characterize surge dynamics, instrumentation must be capable of measuring flow reversal phenomena which may require a frequency response of 1000 Hz or greater. Also, if the absolute dynamic level is required in addition to the frequency characteristics, the effects of transducer coupling on amplitude and phase characteristics must be evaluated. The preferred practice for temperature is the use of very fine wire thermocouples for dynamic temperature measurement using a number of wires per measurement point to insure survivability. The preferred practice for pressure measurement is flush mount or very close coupling of transducers (typically less than 5 cm from the measurement point). The length of tubing coupling the transducer to the measurement point must be small enough[5,6] so that the frequencies of interest are below the frequency at which the amplification due to tube resonance becomes unacceptable (typically the frequency at which amplification is 10%). The coupling length to the transducer may be increased, if the transducer is mounted flush with the coupling tube wall and the tube is extended beyond the transducer to a sufficient length for damping of the propagating wave and prevention of reflection and subsequent tube resonance. Effectiveness of this so-called "infinite tube" technique has been demonstrated to be effective in damping tube resonance[7,8] for pressure disturbances which are quasi-acoustic (small perturbation about a mean pressure). However, inaccuracies will occur if this technique is used to characterize wave forms which have nonlinearities and pressure deviations which are of the same order of magnitude as the mean pressure. This may occur in pressure response to a compressor stall and subsequent blowdown with high flow reversal rates. The inaccuracies are due to the fill and empty characteristics of the "infinite tube."

The high response pressure instrumentation has a high sensitivity to thermal drift. To provide a means for correction of the zero drift, a co-located measurement of high accuracy but low response is made along with each high response measurement. The high response instrumentation is typically directly coupled to the acquisition system since level as well as frequency content is desired. Some installations employ differential measuring high response instrumentation. One side is connected to the measurement point in the engine. The other side is connected to a manifold of constant pressure. The number of manifolds used is dependent on the magnitude of the pressure variation in the engine. Typically, three manifolds are used: one for the engine inlet and fan interstage measurements, one for the fan exit/compressor inlet, fan duct, augmentor duct, and front stage compressor measurements, and one for the remainder of the compressor, compressor exit, and burner measurements. This reduces the instrumentation dynamic range requirements since the output is referenced to a quasi-mean pressure.

Experience has shown that transducer sensitivities change with engine condition (unless the instrumentation is thermally stabilized, e.g., water cooled) and installation. Therefore, some method of in-place calibration must be used. If the manifold system is used, the sensitivity is determined by changing the manifold pressure during steady-state engine operation and recording the transducer output change due to this known change in pressure. If absolute transducers are used, the test program must be designed to provide steady-state points at various engine operating conditions to determine high response transducer sensitivity by comparison with co-located accurate measurements.

The extent of instrumentation is governed by considerations of suspected nonrecoverable state phenomena and characterization of component performance and interaction during stall. Circumferential and radial pressure, and temperature instrumentation is required at the fan inlet, fan exit-compressor inlet-fan duct inlet, and compressor exit (typically three circumferential, two radial). Disturbance propagation in the fan-augmentor duct requires axial and circumferential instrumentation (typically at nozzle entrance, fan duct-augmentor duct interface, and two positions along the fan duct). If single or multi-stage compressor performance is required, then circumferential and radial instrumentation must be considered at the single or multi-stage interface points. This is probably excessive instrumentation from an analysis viewpoint, but may be required if the data are to be used for math model development. Characterization of rotating stall requires stage-by-stage, circumferentially displaced (typically two) static pressure measurements. Inlet distortion pattern measurement requires more extensive instrumentation of the fan inlet. Dynamic mass flow measurement is made using either total and static pressure or hot film anemometers. Forward- and aft-facing probes have been successful in a dynamic mass flow measurement. Hot film techniques have had limited success due to component survivability and due to accuracy at low flow rates. Hot film measurement also requires co-located forward and aft pressure measurements to determine flow reversal. Other special instrumentation to be considered includes flame detectors for determining augmentor and combustor operation and extent of flame propagation during stall, inlet duct pressure instrumentation for assessing facility interaction, instrumentation of critical pressures in the fuel control for analysis of its coupling with stall phenomena, and instrumentation for events such as stall detection (discussed previously), nonrecoverable stall detection, and recovery actions taken such as valve openings and variable geometry reset.

5.0 DATA ACQUISITION AND PROCESSING

The extensive instrumentation for tests to determine engine recovery characteristics and the complex nature of component performance and interaction during stall and post-stall requires careful design of data acquisition, processing, and analysis techniques. The analysis of the dynamic data requires acquisition and processing techniques which preserve the dynamic response of the basic measurement and maintain the integrity of the phase relationship between measurements. The dynamic data may be stored in and accessed from analog (FM tapes) or digital (tapes and disk files) medium. The amount of data generated in tests of turbofan engine recovery characteristics requires consideration of means for efficiently accessing the data, parameter calculation from measured data, and analysis display flexibility. This has dictated the use of digital processing and analysis. Digital acquisition may be desirable if the capability exists. However, the capbility for digital acquisition of data at rates suitable for dynamic data is not generally available in an on-line mode, and off-line analog-to-digital conversion is required. The digital data format also allows flexibility of usage by multiple organizations and access from remote terminals. The data from tests at the AEDC have been duplicated and made available for access by computer systems at the AEDC, Wright-Patterson AFB, Ohio, General Electric at Evandale, Ohio, and Pratt & Whitney at West Palm Beach, Florida. Remote access of the data has been established for analysis performed by Systems Control, Inc., in Palo Alto, California. The AEDC experience with acquisition and processing will be discussed on the basis of dynamic data acquisition in analog form, analog-to-digital conversion and digital processing, and transient and steady-state data acquired and processed using digital techniques. A schematic of a typical data acquisition/processing system is shown in Fig. 8.

5.1 ACQUISITION SYSTEMS AND TECHNIQUES

Dynamic flow path data are the data of primary importance to stall recovery testing. For the purposes of this paper, dynamic data are data from measurements which have acquisition and processing designed for analysis of data with frequencies greater than 25 Hz. Transient data are limited to frequency identification up to 25 Hz. The dynamic data after conditioning are recorded on FM multiplex (MUX) tape drives. A typical 14-track tape has 12 tracks for data recording, one track for the tape servo signal, and one track for recording the time code. Each track can accommodate a number of channels, each with a different modulation frequency band. At AEDC, six channels are used for data recording, one for timing synchronization verification, and one for recording tape flutter. The timing synchronization signal is recorded on all tracks and is a combined 2-Hz and 200-Hz signal. The signal is used to check for errors in timing alignment from track to track following A/D conversion. The tape flutter is recorded with zero input and thus records only tape drive induced dynamics. The tape drive dynamic response is typically 2000 Hz or greater. Two MUX drives recording 72 channels each are generally required. High response measurements from 100 to 120 in number are recorded along with speeds, variable geometry position, selected low response pressure measurements, fuel flow, selected control parameters, and selected parameters indicating actions of external inputs (recovery actions through a relay-delay system) or control outputs of a digital control computer. The data for recording are selected to provide a self-consistent data set for analysis.

The dynamic data are continually monitored on-line with oscilloscopes (amplitude versus time). The time code channels on each MUX are also monitored. The time code carrier signal is used for the digitization clock in the A/D conversion process, and the modulated code is read to specify and trigger start and stop of the digitization process.

Thus, the time code integrity is of utmost importance. Automated monitoring of the time code is being implemented for future tests.

Transient data are recorded on a 100 sample per second digital data acquisition system. The system typically records 200 channels of 100 sps data. The data are acquired by sequential sampling -- the sequential rate being approximately 20,000 sps. Front end anti-aliasing filters are used on selected channels. The system uses an internal digital clock. Steady-state data are acquired to document engine performance prior to stall. These data are also acquired digitally. The data are obtained from low pass filtered, electronically coupled individual transducers and from transducers which are coupled to the measurement point by means of mechanical multiplexing. Other systems important to test control and on-line analysis are time history plots of selected engine data using low response multichannel recorders and dynamic data time history display of selected channels on an oscillograph recorder.

5.2 DATA PROCESSING

Data processing includes all operations necessary to cast the data in a form useful for analysis. This includes digitization of data acquired in analog form, conversion of raw count digital data to engineering units data (EUD), and any calculations necessary to produce engine performance parameters or parameters useful in characterizing post-stall phenomena. Steady-state data are converted on-line (DEC 10 computer) from raw count to EUD form using transducer sensitivities obtained from pretest calibrations. The steady-state data are then computer processed on-line to produce engine performance parameters. All or selected channels of the transient data are converted to EUD on-line at selectable data rates. Calculated data channels are created and are available for on-line analysis through use of interactive displays. However, it has been found that the most desirable method of displaying the transient is through quick turn-around hard copy plotting of selected channels (30 to 50) for each data point. These plots are produced from data on disk files and are available within 24 hrs of the completion of the test run. The full processing of the 100 sps data is performed off-line and recorded on tape.

The dynamic data recorded on the MUX tapes are processed to produce data convenient for applying digital analysis techniques. Following the test (a test consists of four to five 10-hr test periods), 40 to 60 data points are selected for digitization. The first process involved is analog to digital conversion. The data are played back through two low-pass anti-aliasing analog filters prior to digitization. The filters are six-pole linear phase filters with a 400-Hz cutoff frequency. The data are digitized at a rate consistent with post-stall phenomena. This has been assumed to be no more than 200 to 250 Hz. Based on a four sample per cycle criterion, the digitization rate used in processing data at the AEDC has been 1000 sps. The data are digitized by simultaneous sampling using the time code carrier (10,000 Hz) as the digitization clock and the time code for triggering digitization start and stop. Using this technique, the digital clock is recorded along with all the data. This has been found to be the best method of assuring phase integrity from track to track and underscores the need for assuring a recorded time code signal of low noise and high signal integrity. The data are digitized in MUX track units producing 10 channel tapes: six data channels, a timing synchronization channel, a tape flutter channel, and two unused channels.

Data from the raw count tapes are read into the facility DEC-10 computer for verification. This includes review of calibration data and start and stop times. The data are displayed on a graphic screen and reviewed for signal integrity (digital dropouts, excessive noise). Using Fourier transform analysis techniques, the phase relation between data channels is verified by examining the tape to tape phase variation of the 200 Hz timing synchronization channel. Tapes failing verification are redigitized.

Following raw count certification, the tapes are transferred to the base IBM 370 computer for EUD conversion using pretest or in-place calibrations. The data are then merged to a single tape (one data point per tape) with a specified order for the time-oriented data arrays. During this step, adjustments are made to the data for zero shift and calibration changes (from in-place calibration data), and parameter calculations are made and added to the array. These tapes are duplicated and transmitted to the engine manufacturer and the AFWAL for analysis by their personnel. Smaller arrays of selected data are formed and placed on disk file for access by analysis personnel. The smaller arrays consist of parameters selected for a particular phase of analysis.

The detailed consideration given to the digital processing of the dynamic data is indicative of the difficulty of performing the processing and handling of such a large volume of data. The digital data from a typical test of 50 digitized data points would produce a data volume of 150 x 10^6 words of information. Assuring the validity of such information with respect to absolute level, dynamic level, frequency, and phase is a task which requires special planning, careful attention to the details of procedures, and the maximum possible use of automated verification techniques.

One additional processing step which is unique to these tests is the processing of data to obtain dynamic mass flow. The technique which has proved successful utilizes forward and aft pressure measurement co-located in a single probe. The pressures correlate well with local Mach number, and the probes are sometimes referred to as Mach probes. The calculation assumes that one pressure is total and the other is stream static -- depending on the direction of airflow. Assuming that the flow is reversible to the total and static measurement points, the relationship between forward pressure, P_F, and aft pressure, P_A, is

$$\frac{P_A}{P_F} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{-\gamma}{\gamma - 1}}$$

If the specific heat, γ, may be assumed constant, the function is then nonlinear in Mach number, M. However, if M is small (and M^2 is even smaller), then the function may be expanded in a Maclaurin series as

$$\frac{P_A}{P_F} = 1 - \frac{\gamma}{2} M^2 + \frac{\gamma(2\gamma - 1)}{8} M^4 \; ....$$

Retaining the first two terms and rearranging gives

$$M \simeq \sqrt{\frac{2}{\gamma}} \sqrt{\frac{P_F - P_A}{P_F}}$$

The Mach number of the flow past the probe is then linear with $\sqrt{\Delta P/P}$ with accuracy within 10% for $0 < M < 0.5$, the range of values typical of flow inside turbine engines. To check the validity of the linear assumption, the Mach number at the compressor entrance was calculated over a range of engine operating conditions utilizing mass flow determined from a turbine flow parameter. The results are plotted in Fig. 9. Also plotted is the pretest calibration. Although there is scatter in the data, the agreement with pretest calibrations is generally good and justifies the linear observation noted above. The scatter may be caused by a temperature effect on the value of γ or the flow angle sensitivity of the probe. Pretest calibration indicated no angular error between $\pm 10^o$ of yaw.

If the compressor mass flow is calculated from compressor inlet Mach number and corrected to compressor inlet conditions, then corrected flow per unit area at the compressor inlet is given by

$$W_c = \frac{W\sqrt{T/T_r}}{P/P_r} = M\left(\frac{\gamma}{R}\right)^{1/2}\left(1 + \frac{\gamma - 1}{M^2}\right)^{-\frac{1+\gamma}{2(\gamma-1)}}$$

where P is inlet total pressure, T is inlet total temperature, P_r and T_r are reference pressure and temperature, respectively, and R is the gas constant. The calculation of dynamic mass flow greatly aids the analysis of stall recovery or nonrecovery. It also allows analysis of compressor performance on traditional maps of pressure ratio versus corrected flow or head rise versus flow coefficient.

6.0 ANALYSIS METHODOLOGY

Analysis is performed utilizing data from several systems depending on the depth and scope of the analysis. Overall engine operation can be determined from on-line time history plots. This includes data on low response recorders for assessing engine transient response (speeds, variable geometry position, critical control parameters, temperatures, and pressures) and dynamic data recorded on an oscillograph. From these data, along with steady-state data, an assessment can be made of overall engine performance, general surge characteristics, rotating stall initiation and termination, and other sequencing events in the engine. More detailed analysis of transient response can be made utilizing the digital transient data. This includes a broader view of engine response and permits analysis in terms of various calculated parameters characterizing the stall phenomena. The detailed characteristics of the stall event must be determined using the dynamic data.

6.1 ANALYSIS TOOLS

The analysis tools are dependent on the form in which the data are retained. If the data are retained as analog data on tapes, then the tools are analog tape drives, signal-conditioning equipment, spectral processors, x-y plotters, and perhaps digital FFT processors and minicomputers for further processing. This is generally limited to single or dual channel analysis with limited flexibility in processing. If the data are retained in digital form, the analysis tools are the digital computer, peripheral output devices (CRT and graphic displays, printers, hard copy units), and software. The AEDC experience has demonstrated the futility of turbofan engine stall recovery analysis utilizing data stored in analog medium. Hence, the analysis methods discussed are those utilized with digital analysis techniques.

Typical analysis software consists of parameter processing and plotting programs, time series analysis software utilizing the fast Fourier transform, and software for displaying spectral data. The processing and plotting program should be highly flexible with respect to parameter selection, parameter calculation, and parameter display (versus time, versus another parameter, multiple parameter display). The program is enhanced if digital filtering is used. Analysis at the AEDC has used a Butterworth low-pass or band-pass filter with selectable poles and cutoff or bandpass frequencies. The filter is an infinite impulse response (IIR) filter designed using a bilinear transformation with frequency prewarping. The filter was developed by Systems Control, Inc., Palo Alto, California. The phase induced by filtering must be accounted for in analysis. Zero phase filtering may be obtained by filtering the data twice in opposite directions in time. Figure 10 shows filtered (120-Hz cutoff) and unfiltered plots of an engine pressure during stall. Filtering the data eliminates spurious high frequency noise and sharpens the details in the frequencies of interest. Figure 11 shows plots of corrected compressor mass flow versus pressure ratio during stall with filters set at 60, 120, and 240 Hz and zero filtering. For this data, the 120-Hz cutoff is the best compromise between eliminating spurious noise and retaining the general characteristics of the surge trajectories. The use of a flexible plotting program with digital filtering is very useful in this analysis.

The fast Fourier transform (FFT) may be used to determine spectral characteristics of the data. The spectral analysis package used at the AEDC is based on the FFT computational technique described by Cooley, et al.[9] The package is a modification of a program developed by Systems Control, Inc. The program calculates statistical functions of two time series data sets including probability-density function, correlation function, power spectral density, cross spectrum magnitude and phase, transfer function magnitude and phase, and coherence function. A typical plot of cross spectrum magnitude and phase is shown in Fig. 12.

Displaying the spectral characteristics of a parameter versus time is important to analysis of stall-sustaining phenomena. Additional programs display power spectral density, cross spectral density, and cross spectral phase as a function of magnitude and time. Figure 13 shows a cross spectral map or "waterfall" plot. Figure 14 shows cross spectral magnitude (plotted as the length of a vertical line) as a function of frequency and time. Figure 15 shows cross spectral phase (indicated by the direction of an arrow from zero reference) as a function of frequency and time.

6.2 ANALYSIS METHODS

The analysis methods are divided into two broad areas depending on the results of previous investigations and theoretical development concerning recovery characteristics. For example, surge, stall and rotating stall of axial compression systems have been the focus of much study and theoretical development (i.e., Refs. 3, 4, 10, 11, 12, 13, 14, and 15). One area of analysis is then directed toward determining the degree with which previous observations apply to multistage compression systems in turbofan engines. On the other hand, the interactions affecting recovery in a turbofan engine have not been previously studied in depth due either to a lack of stall regime testing or to tests insufficiently instrumented to characterize phenomena. Analysis based on previous work employs directed analysis with correlative parameters for characterizing phenomena, while the analysis of system interactions relies somewhat on postulate formulation and verification or rejection in an iterative process.

The typical analysis directed by previous studies involves surge cycle analysis in terms of surge trajectory characteristics (blowdown and recovery) and transition characteristics to rotating stall. These characteristics can be studied using traditional compressor maps (Fig. 11). Rotating stall has also been studied extensively in low speed, one- to three-stage compressor rig tests. The characteristics of rotating stall described by frequency, number of cells, radial (full or part span), circumferential, and axial extent and distribution characteristics during initiation, when fully developed and when clearing, are important in analyzing in-stall phenomena. The rotating stall characteristics are identified by use of spectral techniques (Figs. 10, 11, 12, and 13) and expanded time history plots. In addition, the surge/rotating stall boundary may be correlated against previous theory, and the effect of compressor downstream mass flow throttling characteristics as affected by recovery actions may be analyzed.

The areas of analysis not guided by previous studies include the broad area of component interactions. Typical is augmentor influence on surge due to either oscillations at the surge frequency or the effect of higher frequency disturbances (rumble). The fan-compressor interaction may be analyzed for the effects on engine recovery of stall sequence, recovery sequence, and non-planar stall or recovery. Compressor/burner interactions may be analyzed for coupled compressor/burner oscillation, burner repressurization characteristics during surge, and burner operation during surge (on or off and high frequency disturbances). These analyses utilize spectral analysis for oscillation identification as to source frequency, amplitude, and extent, and rely on expanded time history plots in determining stall and recovery sequence, compressor repressurization, augmentor-fan-fan duct-nozzle interaction, etc. In any of the above analyses, the effects of control scheduling of variable geometry and fuel flow must be assessed.

7.0 CONCLUDING REMARKS

The complex nature of stall-related phenomena in turbofan engines requires special test techniques, extensive instrumentation of components and component interfaces, data processing which preserves and in some cases improves measurement accuracy and produces final data in digital form, and analysis of the data using digital techniques. Stalls are obtained in conventional manner by nozzle closure, mainburner and afterburner fuel pulse, and throttle transients. The investigation of recovery techniques and the effects of engine configuration depend on integration of test peculiar equipment with the basic engine control. Electronic engine control provides for the implementation of the most efficient and flexible test techniques.

Instrumentation considerations dictate that in addition to conventional steady-state and transient measurements, dynamic instrumentation for temperature, pressure, burner flame detection, stall detection, and mass flow measurement are required.

Close-coupled thermally stabilized pressure transducers are most desirable. Pneumatically referencing the pressure measurement using differential transducers can improve dynamic accuracy. A method of in-place calibration of dynamic pressure is essential to obtaining data of accuracy sufficient for analysis. Mass flow is best determined using forward- and aft-facing pressure measurements.

Data processing must produce data in a form convenient for analysis. The quantity of data and the required flexibility in analysis requires the data be produced in digital form. Digital acquisition is desirable. If digital data are produced from data recorded in analog form, data processing must be carefully planned and executed to maintain signal dynamic integrity in level and phase. Dynamic data processing must be designed to accommodate the production of data using in-place calibrations.

The characterization of dynamic phenomena during stall requires the use of digital time series analysis techniques built around the fast Fourier transform. Digital analysis provides great flexibility in parameter calculation, including mass flow, head rise, flow coefficient, pressure ratio, etc. Software programs for digital filtering and flexible display options are also important. The analysis method is based on correlation of data with existing theory and identification of causative factors primarily in the analysis of component interation.

The techniques recommended herein have demonstrated applicability in an extensive program directed at the identification of recovery characteristics and design criteria in an effort to produce future engines which are free of nonrecoverable stall.

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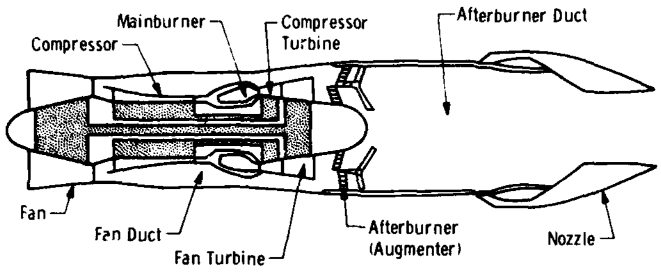


Figure 1. Augmented Turbofan Engine Schematic

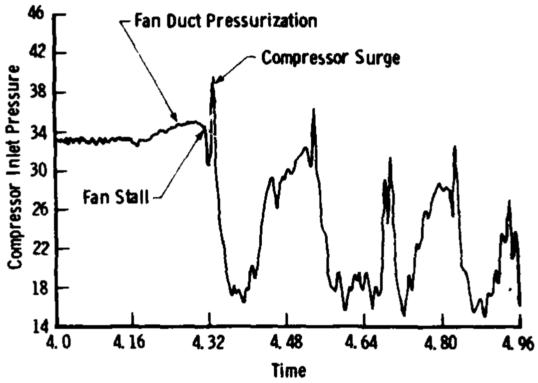


Figure 2. Typical Nozzle Closure Stall

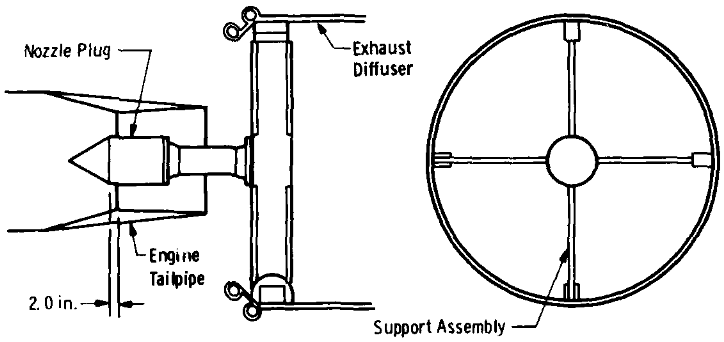
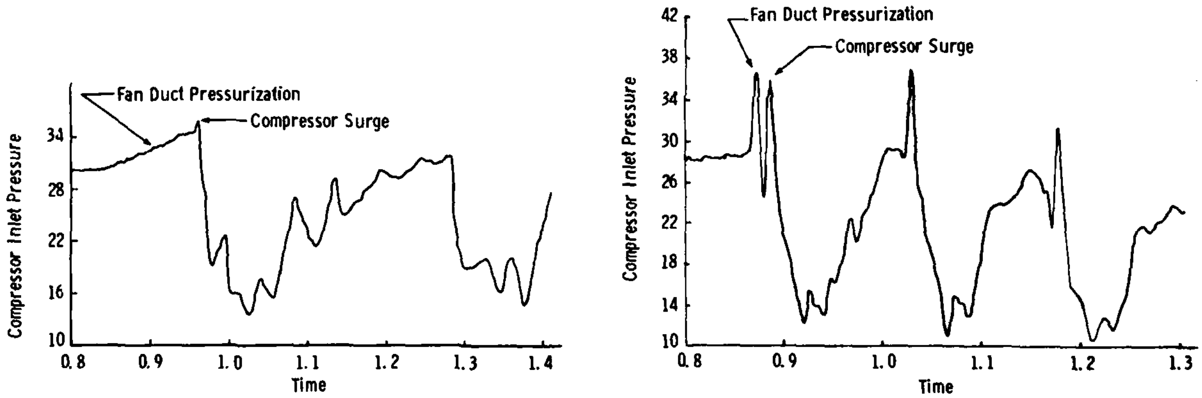


Figure 3. Plug Assembly for Nozzle Closure Stalls



a. Ignition at Start of Manifold Fill b. Ignition During Manifold Fill Sequence

Figure 4. Typical A/B Fuel Pulse

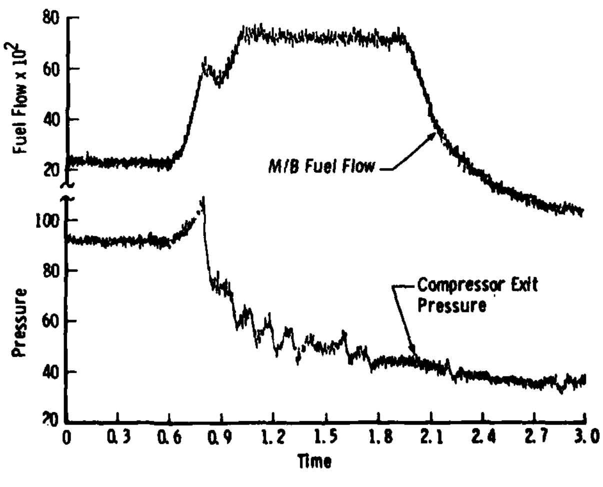


Figure 5. Typical Mainburner Fuel Step

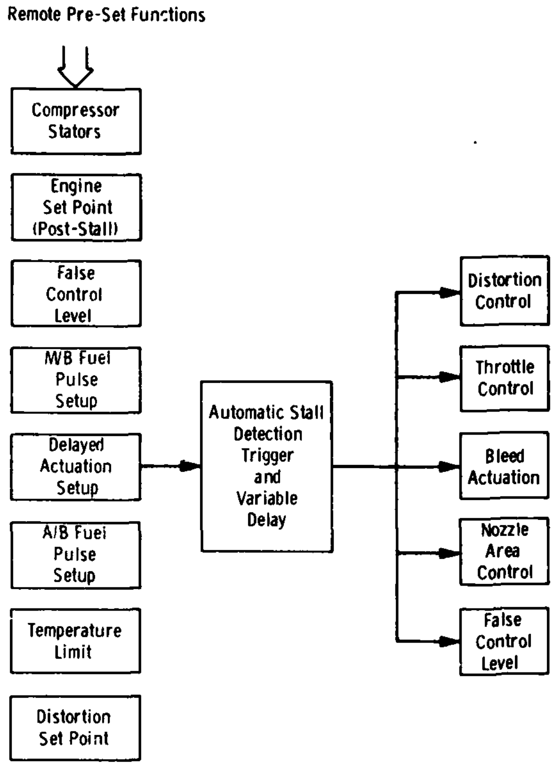


Figure 6. Preset Functions and Delayed Actuation System for Hydro-mechanical Control

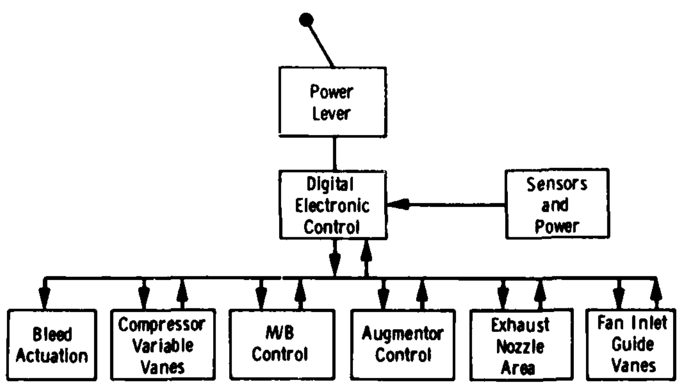
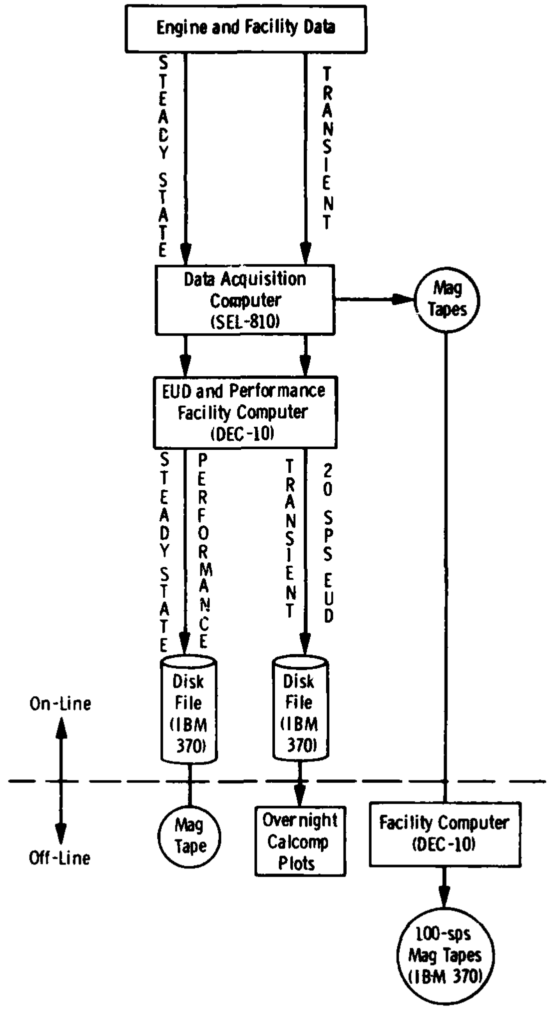
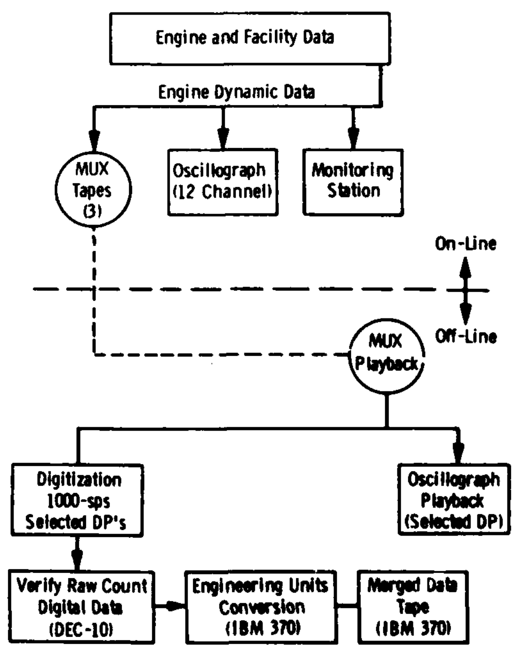


Figure 7. Full Authority Electronic Control



a. Steady-State and Transient Data



b. Dynamic Data

Figure 8. Data Acquisition/Reduction Schematic

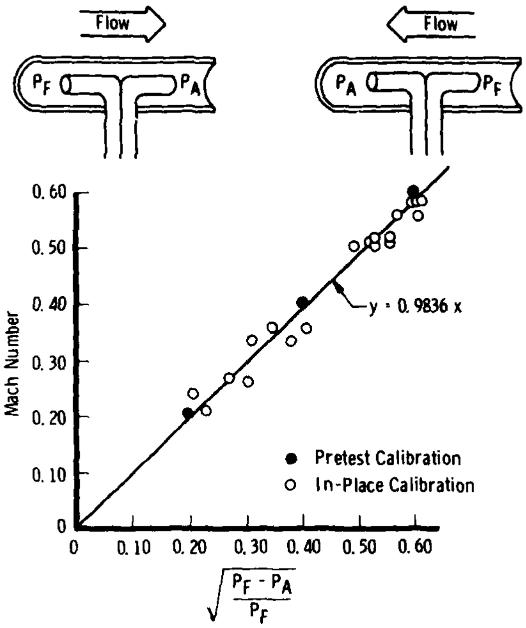


Figure 9. Mach Probe Calibration

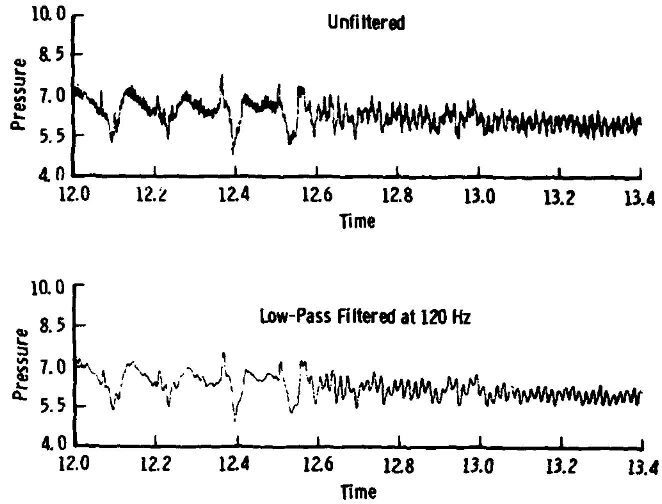


Figure 10. Filtered and Unfiltered Pressure/Time History

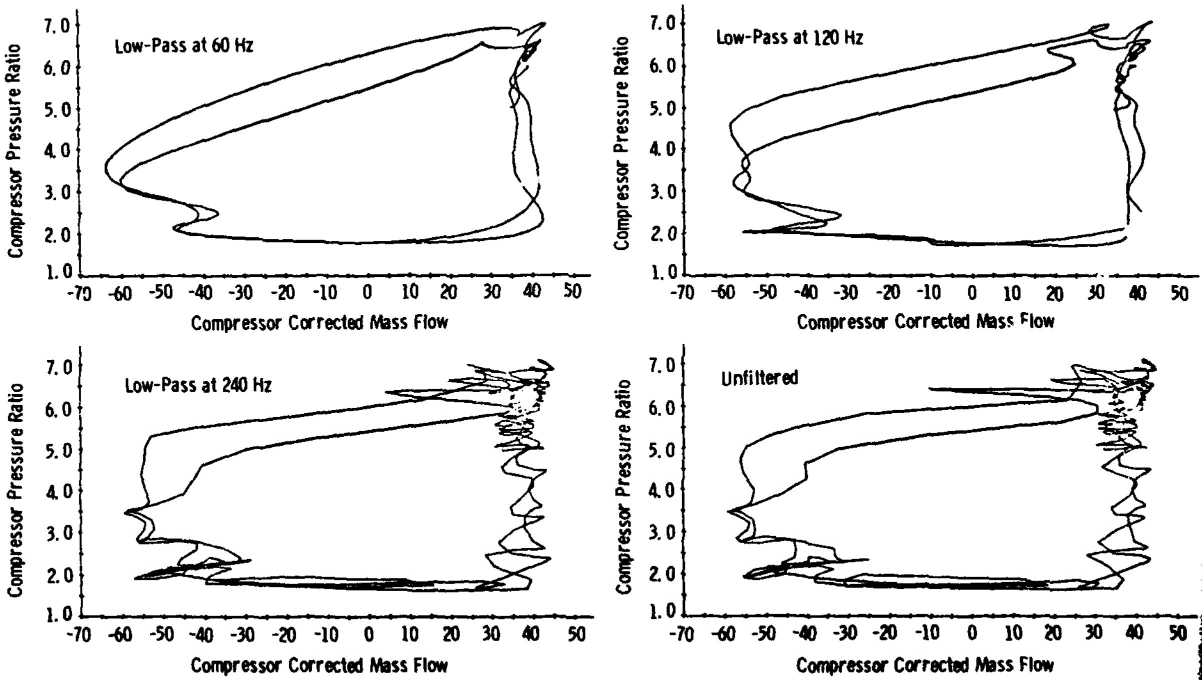


Figure 11. Filter Cutoff Effects on Surge Trajectory Plotting

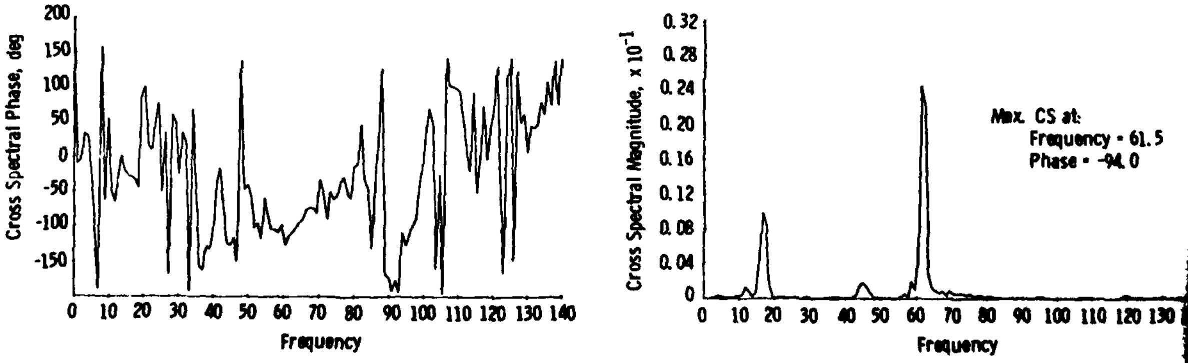


Figure 12. Cross Spectral Magnitude and Phase Plot

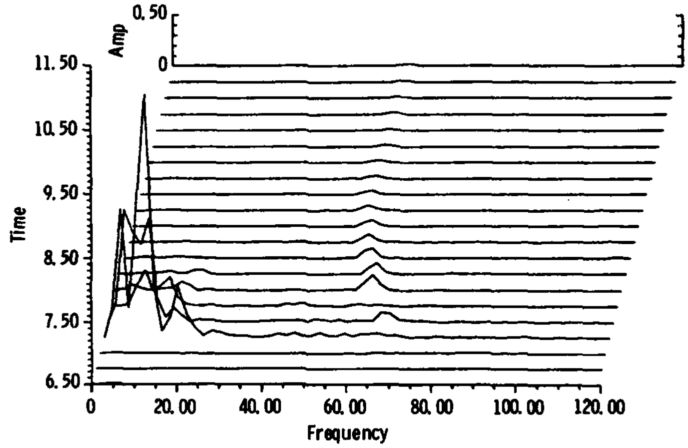


Figure 13. Typical Spectral Map

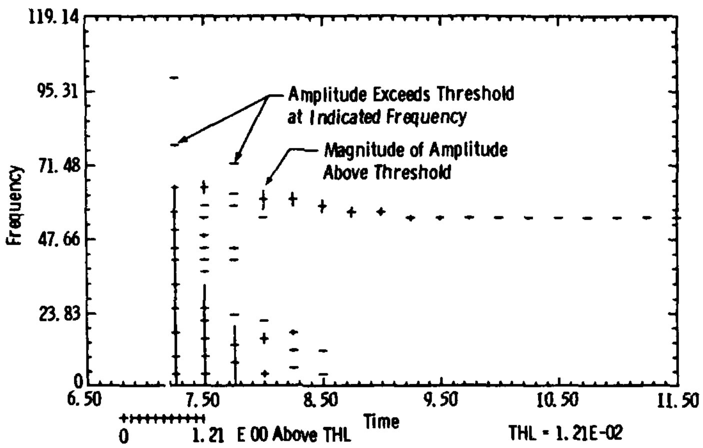


Figure 14. Typical Cross Spectral Phase Diagram

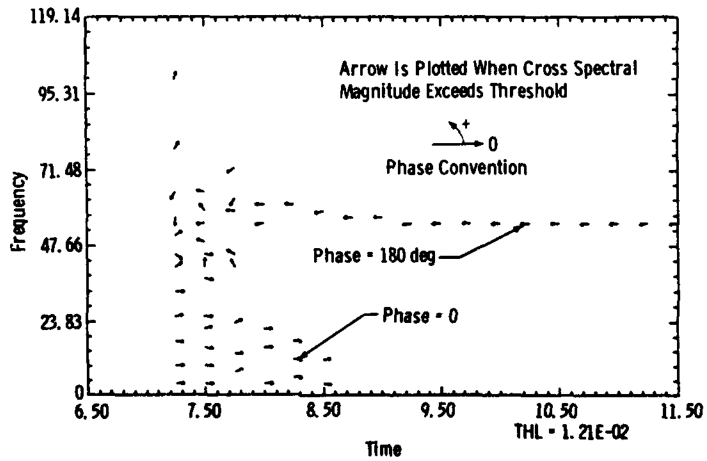


Figure 15. Typical Cross Spectral Magnitude Diagram

DISCUSSION

C. Cocheteux, Fr:

– Stall initiation:
Did you consider injecting air in the chamber in order to avoid engine overheating at high RPM?

Author's Reply:

Yes. But we rejected this method for several reasons. The costs of installing an exit compressor inbleed system are significant. Also, if the inbleed system cannot be shut down very quickly, this will affect post-stall engine response and spool rolldown rate due to increased mass flow through the turbines.

R.G. Hercock, UK:

Was there any difference in recovery characteristics, depending on the method of initiation?

Author's Reply:

Yes. The initiation technique may tend to sustain surging of the engine. For instance, if during an augmenter induced stall, the augmenter fuel flow remains high this may cause continued surging of the engine.

The same is true for nozzle closure. The initiation technique did not, however, affect the tendency of the engine to transition to a nonrecoverable state.

DESIGNING FOR STABILITY IN ADVANCED TURBINE ENGINES

by

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ABSTRACT

One of the most critical functional problems that a high technology turbine engine encounters is nonrecoverable stall. Presently, the only effective means of clearing the nonrecoverable stall is engine shutdown and subsequent airstart, potentially impacting the effectiveness of the weapon system. This paper addresses the design improvements that are required to make the system more tolerant to the operational environment. This paper also deals with establishing design criteria to be applied in the preliminary engine design phase to ensure resistance/avoidance of nonrecoverable stalls while ensuring adequate engine operability in the form of airstart capability and engine throttle response. This paper will identify the mechanisms of rotating stall, the design improvements to resist/avoid rotating stall, their projected effectiveness in reducing operational problems, and engine test results of some of these design improvements.

PROBLEM

Turbofan engines have several attractive characteristics when used for tactical fighter propulsion systems. The most important of these is the inherently high thrust to weight ratio at full afterburning combined with excellent fuel consumption when operating dry. These engines have contributed significantly to the high thrust loading and, therefore, high maneuverability and excellent fuel efficiency of modern tactical aircraft. These very important characteristics of turbofan engines have contributed significantly to the outstanding family of tactical aircraft of recent times. This same high thrust loading demands a stability and transient capability of the engines beyond that required in past generations.

An evaluation of the typical usage of tactical weapon systems over the past several generations shows some interesting trends as shown in Figure 1. As the airplane installed thrust to weight ratio increases the amount of time the engine spends at maximum turbine temperature decreases significantly; but at the same time, the number of major throttle cycles increases. This increase in the transient operation of the engines places a major challenge to the engine designer from a stability standpoint. The increase in weapon system thrust weight ratio improves the maneuverability of the airplane, frequently resulting in an increased distortion of the inlet flow conditions as well as adding a more demanding combustor stability requirement. In addition, the turbofan engine must withstand a significantly increased range of augmentor operation. The augmentor in the turbofan engine communicates directly with the compression system (even to very low bypass ratios, i.e., < 0.3) as opposed to being isolated as in past turbojet engines by a choked turbine as shown in Figure 2. This provides an additional destabilizing influence which must be designed for.

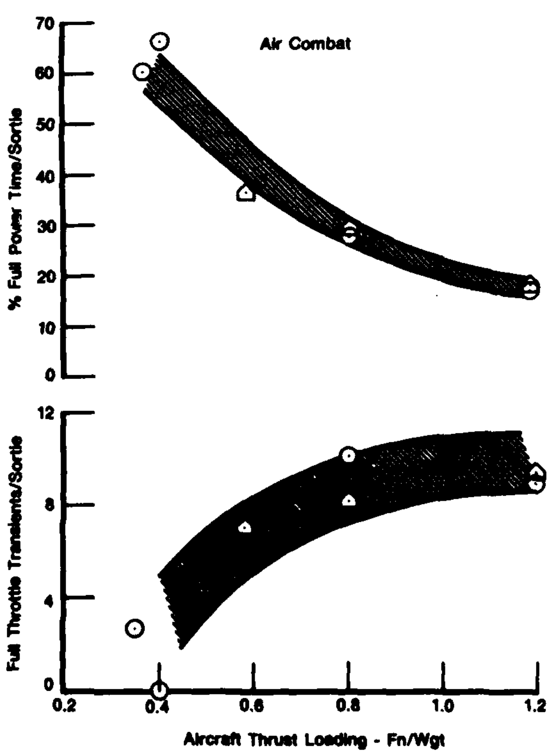

Figure 1. Trend of aircraft thrust loading on full throttle transients and time at maximum turbine temperature

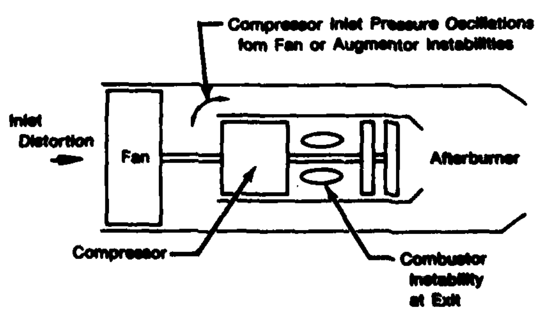

Figure 2. Compressor stability affected by augmentor perturbations

Historically, the compressors of past engines have experienced stability problems driven by combustor malfunctions or inlet disturbances caused by the extremes of aircraft maneuverability and transient response requirements of the engine itself. Past lessons learned have enabled the designer to provide for increased distortion tolerance in the compression system to minimize the loss of surge margin; but equally important, to attenuate the flow maldistribution in the compression process to minimize combustor and hot section temperature pattern factor problems due to flow maldistribution. Combustion system stability is enhanced by providing improved combustor primary zone stability and a continuous ignition source. The nature of augmentors is such that, although they may be extremely well behaved and stable at sea level, as pressures and temperatures decrease, such as in the upper left hand corner of the flight envelope, combustion stability margin is significantly decreased. In addition, because of the low fuel flowrates, fuel scheduling accuracy decreases with increased altitude resulting in increased probability of augmentor induced fan or compressor stalls in the upper left hand corner of the envelope. The instability can result in large changes in augmentor combustion temperatures, with no compensating nozzle area changes, which backpressure the fan forcing it into stall. Figure 3 below summarizes the experience of a modern tactical turbofan engine.

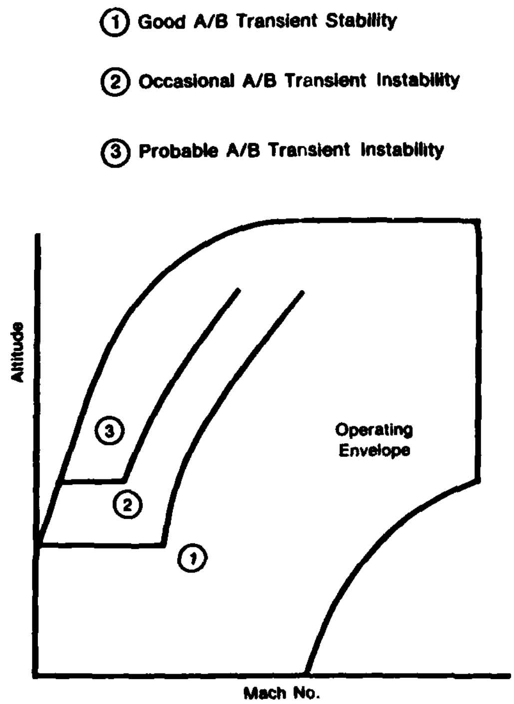

Figure 3. Flight envelope impact on augmentor transient success

The operating envelope for a typical turbine engine can usually be divided into these three areas based on the probability of augmentor transient success. Generally speaking, region one in Figure 3 is an area for unrestricted augmentor operation. Mislights, blowouts, or stalls should not occur since there is more than enough adequate margin to allow for normal transient operation. In region two, however, while most augmentor transients can be expected to be successful, an occasional mislight or instability will occur during a throttle transient but should not occur during steady state operation. Region three is the most difficult area where, because of the reduced margin, there is a high probability of an augmentor anomaly during transients although steady state operation is normally successful.

Although the vast majority of augmentor anomalies that occur will result in a self-clearing stall on the engine, a small percentage of these stalls within the turbine engine will, under some conditions, result in a nonrecoverable stall situation. This is one of the most critical functional problems that an engine may encounter. Presently, the only effective means of clearing the nonrecoverable stall is engine shutdown and restart, potentially impacting the effectiveness of the weapons system. Should a nonrecoverable stall occur, the pilot will observe a rapidly increasing engine exit temperature while the engine rotor speed is decreasing. Without some compensating action the temperature may increase to the point where turbine damage is possible. The rate at which the temperature increases and rotor speed decreases does vary with altitude and airspeed. The increase in temperature is a result of an interaction of the fuel control and the compression, burner and turbine system. Generally speaking, the control is unable to compensate for the unusual characteristic generated by the rotating stall mode. More will be said about this characteristic in a later section of this paper, but a compressor in a rotating stall mode has a significantly reduced air pumping capability for the same rotor speed (as well as a significantly reduced efficiency) as shown in Figure 4. This results in a very high fuel-to-air ratio in the combustor and, therefore, a high temperature on the turbine. At the same time, the turbine power available to drive the compressor reduces more than compressor requirements due to the poor compression efficiency in this operating condition. This will cause the rotor speed to fall rapidly as shown in Figure 5. The danger resulting from a nonrecoverable stall in a gas turbine engine is that the combustor gas temperature will exceed allowable limits for the turbine and/or that the rotor speed will fall below the self-sustaining level. Generally, it will then be necessary to shutdown the engine and allow it to cool somewhat before re-starting.

In the normal surge process (as opposed to nonrecoverable stall), the compressor will recover. Frequently, however, the original source of the stall remains and the compressor will re-stall following the initial recovery. In some cases, a series of these surge cycles may take place. In those events, the combustor gas temperature will tend to fluctuate at levels higher than the original value before the stall. The fluctuations in rotor speed will be less obvious because of the inertia of the rotor, but there will be a gradual trend to a lower level because the time-averaged value of compressor and turbine torque mismatch is negative.

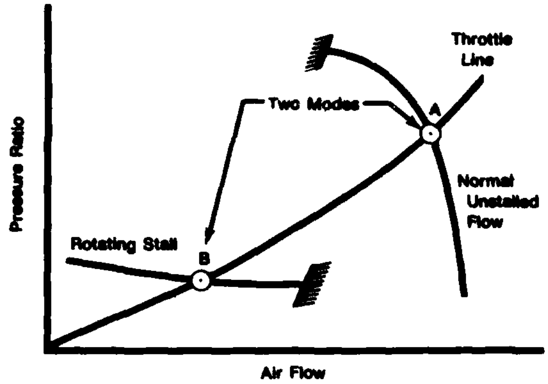

Figure 4. Two distinct compressor operating modes possible as a result of rotating stall

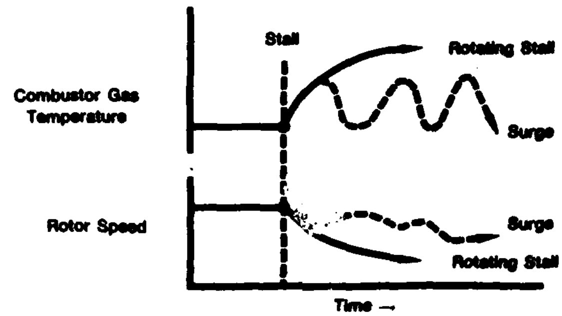

Figure 5. Engine characteristic during surge and rotating stall

Summarizing, we can say there are two possible results to an engine stall. The first of these is the normal surge cycle in which the engine experiences a series of rapid flow reversals and recovery. This will continue until the stalling cause is removed and normal operation is restored or until the engine degenerates into the second mode or more serious situation of operating the compressor in the nonrecoverable stall mode. The compressor in this mode is normally operating in a rotating stall situation and is characteristically very inefficient. Recovery from this situation usually requires a complete shutdown and re-start cycle.

ROOT CAUSE

A fundamental understanding of several of the compression system characteristics is essential if we are to define the root cause of nonrecoverable stall as well as some of the potential systems solutions. The following section will attempt to cover these concepts.

The compressor itself is the heart of the total engine system, particularly in terms of stable or unstable operation. The compressor performance is normally defined in terms of its pressure ratio and normalized flow operating characteristics as shown in Figure 6. The conventional way of plotting these data is to plot the variation in pressure ratio and efficiency along a constant, normalized speed curve as the flow is gradually decreased from wide open exit throttle until surge or some other form of instability is approached with decreased throttle flow area. The independent variables are corrected speed, which is N(RPM) over the square root of normalized inlet gas temperature, and normalized flow rate. The dependent variables are usually pressure ratio, which is exit pressure divided by inlet pressure, and adiabatic efficiency.

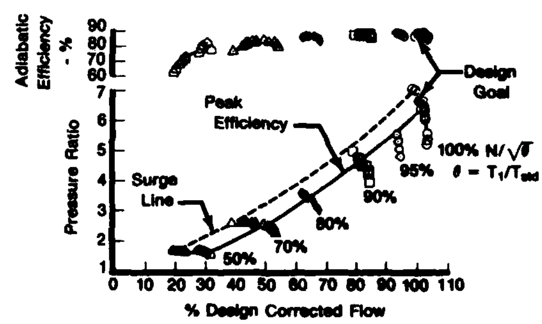

Figure 6. Compressor performance characteristics

In any system containing a compressor, there is usually some kind of downstream throttle as shown in Figure 7. The flow rate through the throttle is the function of the pressure ratio across the throttle or the pressure ratio across the compressor as the sketch in the lower right hand corner shows. The flow rate vs. pressure ratio characteristic can be represented by a line for a given combination of area and temperature. The figure also shows the compressor operating characteristic for a given speed and inlet temperature. The intersection of any one throttle characteristic and a compressor characteristic represents an equilibrium point for the system.

The stability of this point is a major design concern. For a stable equilibrium point, a small disturbance or perturbation will tend to decay with time and the flow rate and pressure ratio values will converge back towards the equilibrium point as shown in Figure 8. Conversely, an unstable system, depending upon the relative slopes of the two characteristics and other systems variables, would be divergent or unstable. In this case, a small disturbance will tend to grow with time as shown in Figure 9. The flow rate and pressure ratios will diverge from the equilibrium point. This is typical of what happens to a system as the operating point or equilibrium point approaches the stability limit of that particular compressor.

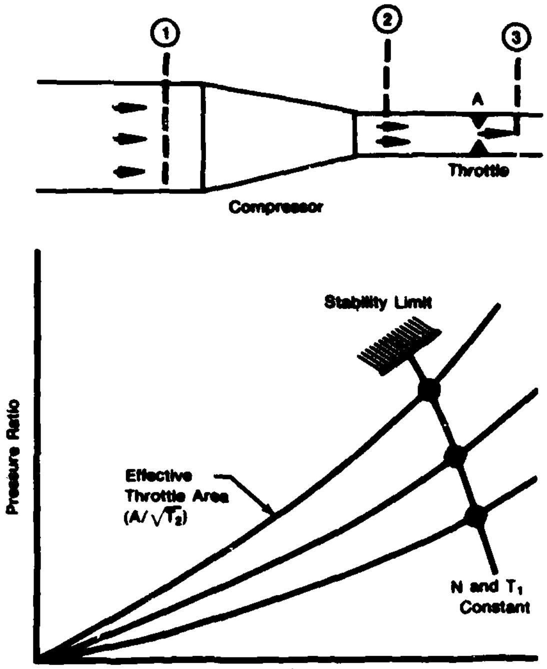

Figure 7. Compressor throttling characteristic

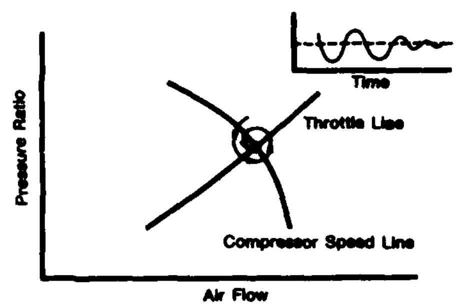

Figure 8. Stable system

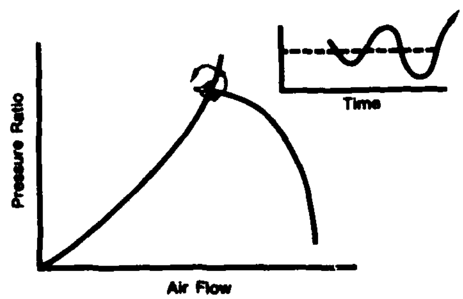

Figure 9. Unstable system

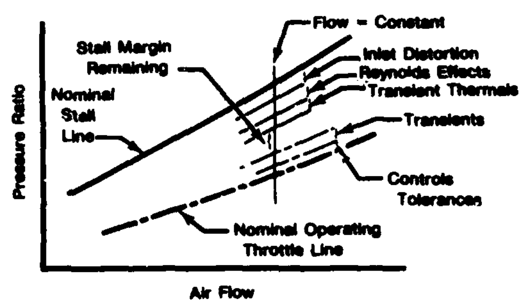

Figure 10. Compression system stability audit

A stability margin audit is a necessary process in design of any compression system to ensure stable operation in a "worst case" scenario. It is the intent of the audit to estimate the remaining stall margin after variations in both the surge line and operating line values have been accounted for as shown in Figure 10. In the design of a gas turbine engine, the designer must recognize the stability limits and insure that the system will operate with sufficient stall margin to allow for all known disturbances and transient operating conditions. This figure shows a typical stability audit accounting for conditions that reduce the stall line such as inlet distortion, Reynolds effects, and transient thermals in the compressor. It also shows variables which can raise the operating line on the compressor such as transient acceleration and control tolerances. The difference between the lowest stall line and the highest operating line is the amount of stall margin remaining under worst conditions. One job of a gas turbine engine designer is to assure that there is always sufficient stall margin remaining under worst conditions.

One of the many kinds of disturbances which must be designed for is a non-uniform distribution of compressor inlet pressure known as inlet pressure distortion. Figure 11 illustrates a variation in inlet pressure around the circumference of the compressor with high values in some places and low values in other places, producing some overall annulus average pressure.

A useful physical concept developed to understand the effects of non-uniform distribution is known as the parallel compressor model. As shown in Figure 12, we can view the compressor as though it were two compressors. One compressor operates in the region from low inlet pressure to a common discharge pressure; the other from the high pressure to the same common discharge pressure. If we plot these two points on our compressor map looking at a single speed characteristic, we can locate where each of these compressors is operating, as shown in Figure 13. In the region of high inlet pressure, the pressure ratio is relatively low. Conversely, in the region of low inlet pressure, the pressure ratio is high. The low inlet pressure ratio inlet case, therefore, establishes the limiting case for the overall compressor. The average or nominal pressure ratio is located between the two extremes. However, the stability limit is reached when the low inlet pressure reaches the capability of the overall compressor and causes it to stall as shown in Figure 14.

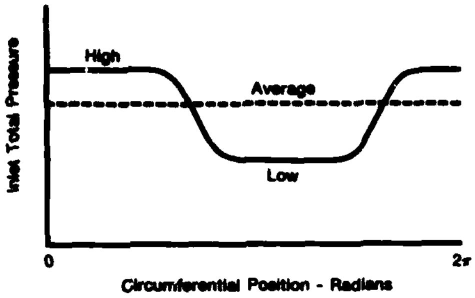

Figure 11. Inlet pressure distortion

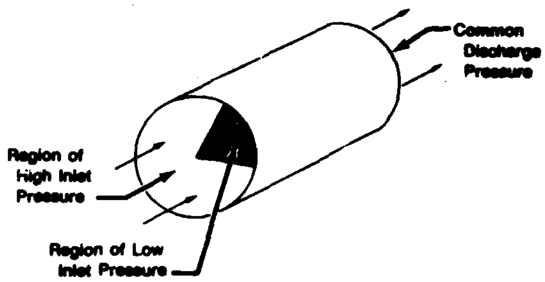

Figure 12. Parallel compressor concept

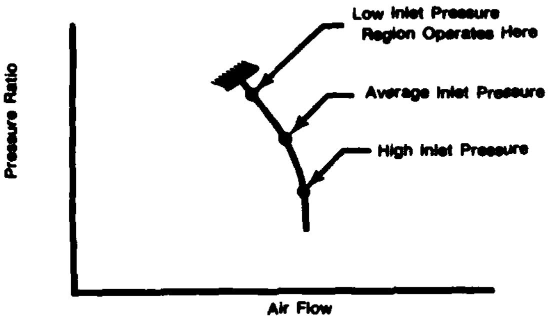

Figure 13. Pressure impact on compressor operation

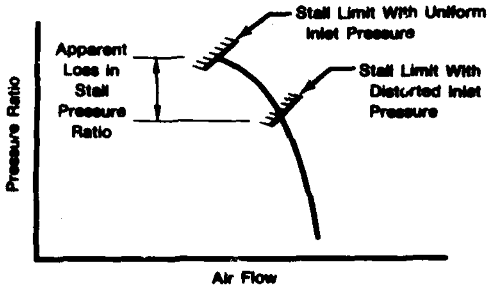

Figure 14. Inlet pressure distortion impact on stall line

The aircraft gas turbine engine designer, therefore, has to work closely with the aircraft inlet designer to insure adequate margin for the worst angle of attack and inlet turbulence level projected for the weapons system maneuverability requirements. The curve shown in Figure 15 indicates qualitatively the kind of information which must be allowed for in the design of the engine compression system. Until recently, the gas turbine engine industry had many different methods of measuring and correlating inlet distortion intensity. However, SAE Aerospace Recommended Practice 1420 by the SAE Committee S16 in March 1978 provided a common method shown in Figure 16 which can be used to predict the effects of both circumferential non-uniformity as well as radial variation. This system is in common use today.

Another type of inlet distortion that has to be considered by the engine designer is inlet air temperature distortion. Earlier we had said that the compressor characteristic was a function of the

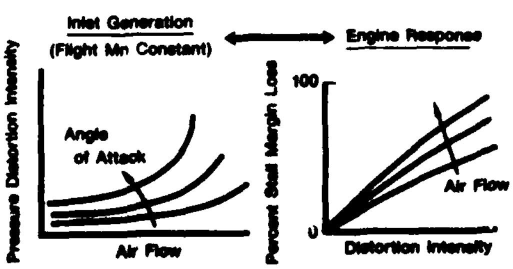

Figure 15. Inlet/engine compatibility

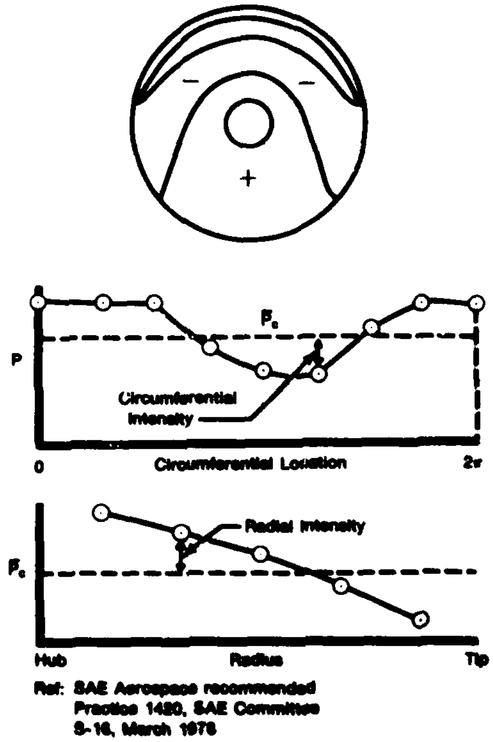

Figure 16. Distortion correlation

corrected speed and flow rate. The corrected speed, in turn, was a function of inlet gas temperature. Although the mechanical speed is constant, if there is a variation in inlet temperature, the corrected speed, therefore, varies. Again returning to the parallel compressor concept, we can visualize a portion of the compressor operating at low corrected speed while another portion operates at the high corrected speed as shown in Figure 17. These correspond to points of low corrected flow rate and high corrected flow rate. On the average, the corrected speed and flow rate for the compressor are somewhere between the two extremes. Note that the portion of the compressor at the low corrected speed (high inlet tmperatures) will determine the compressor stability limit shown in Figure 18.

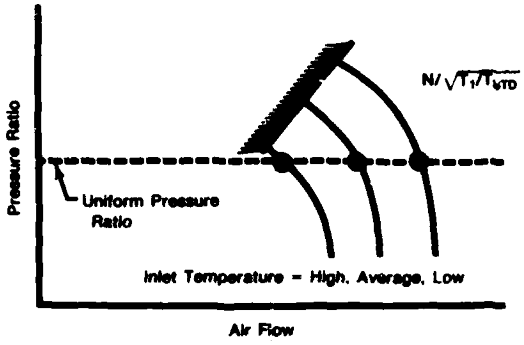

Figure 17. Temperature impact on compressor operation

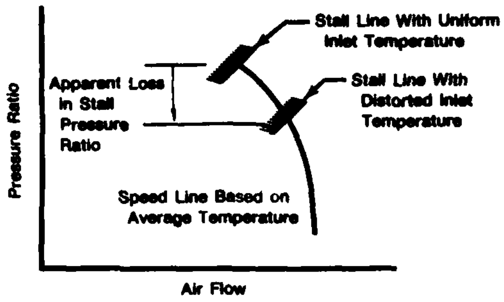

Figure 18. Inlet temperature distortion impact on stall line

As said previously, this type of distortion must also be allowed for in the design of the compression system. Temperature distortion is more difficult to precisely measure due to available instrumentation limitations. Consequently, the designers task is more difficult to allow for the temperature variations. Also, other effects such as gas property changes with steam or rocket exhaust or even engine re-ingested exhaust gases further complicate the problem.

In Figure 7, we discussed the stability of an equilibrium point defined at the intersection of a typical compressor characteristic and throttle characteristic. Figure 19 shows a similar equilibrium point defined as Point A. The compressor characteristic discussed was one of an unstalled normally operating compressor. There is however another possible equilibrium point that can exist in compressors defined here as Point B. This is the intersection of the throttle characteristic and the compressor characteristic that exists in rotating stall. In this condition, a portion of the compressor annulus is stalled and a portion is unstalled. Because the stalled cell tends to rotate around the annulus, this operating condition is normally referred to as a rotating stall situation. Thus, there are two possible equilibrium points, Point A with the compressor unstalled, and Point B with the compressor in rotating stall. Operation at Point B infers a section of the compressor operating at zero or slightly negative flow rate. This is an undesirable situation from an engine stability standpoint as described previously. Once a compressor has stabilized at Point B, however, it may be difficult to clear the condition and have the compressor return to the unstalled operation. For example, if the throttle line is moved to a higher flow rate by opening the throttle, the equilibrium point will move to the right on the compressor characteristic as shown in Figure 20, but can remain in stable equilibrium until a critical value is achieved. Unfortunately that increase in throttle area opening may require as much as 100% increase to achieve the critical value. In the gas turbine engine, therefore, a physical flow area increase of sufficient magnitude may not be either feasible or practical.

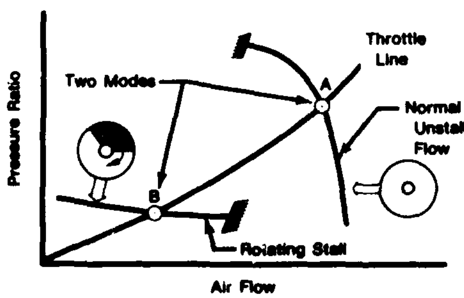

Figure 19. Compressor operating modes

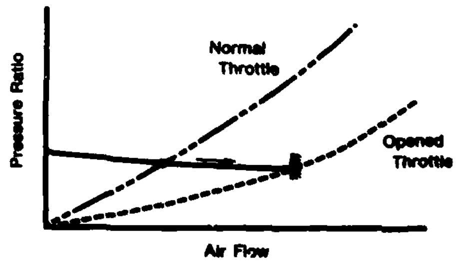

Figure 20. Rotating stall hysteresis

To understand the rotating stall mode we can return again to the parallel compressor concept as shown in Figure 21. The equilibrium point that was just discussed is really the annulus average condition. In the parallel compressor concept, we view one portion of the compressor, the unstalled portion, as operating in a normal unstalled compressor characteristic. The other portion, the stall portion, operates at a flow rate close to zero. When we average these two flow rates based on the relative amount of annulus area occupied by the two modes of operation, we obtain an average flow rate per unit area.

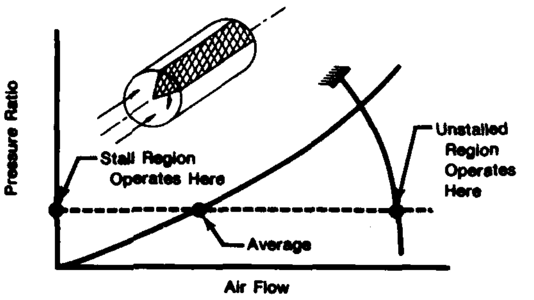

Figure 21. Rotating stall performance

A normal stall sequence is usually followed by a self clearing surge cycle as shown in Figure 22. The air flow then follows the re-acceleration path as shown. If the cause of the stall is not removed, the sequence will repeat itself. The surge cycle in this sequence if often called a blowdown and results from the higher pressure air downstream in the engine flowing backwards to area of low pressure. Since the compressor stalls and stops pumping air, turbine work is reduced more than compressor work and the engine rotor slows downs. A stall at high energy engine operation normally results in a surge and a self recovery. The more stall cycles taking place the slower the engine rotor system runs and the higher the probability of the compressor dropping into the rotating stall mode as shown in Figure 23. Repeated surge cycles occasionally result in incomplete recovery. This phenomena has often been referred to as a "hung stall". In addition, many engines did not have continuous ignition systems. Hence this phenomena was often masked by a combustor flameout and consequent engine dieout. Occasionally, when the compressor rotor speed drops below approximately 60% of design speed in flight, the stall becomes nonrecoverable and the engine must then be shutdown, cooled and re-started to clear the stall, thus summarizing the observed phenomena.

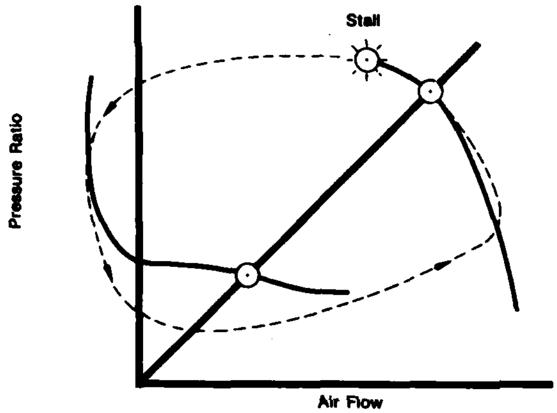

Figure 22. Recoverable compressor stall

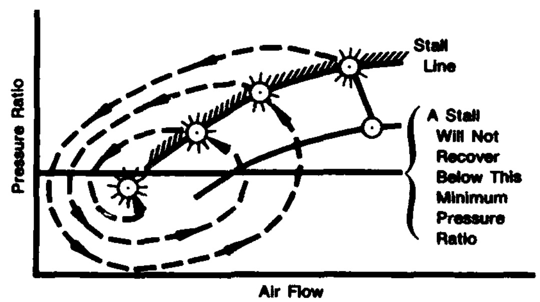

Figure 23. Nonrecoverable stall progression

SYSTEMS APPROACH

Significant progress has also been made in understanding the behavior of the engine system following compressor stall. Much of the initial work has been done by Greitzer, who reported initially on his work on an ASME paper in 1975. At that time, Greitzer defined a parameter "B" which he used to determine both analytically and empirically whether the compressor system would execute a surge recovery cycle or establish a rotating stall condition. When the value of "B" is large enough, a surge recovery cycle is established after stall. The downstream volume depressurizes reducing the pressure ratio until it's finally low enough for the flow to accelerate and fully restore the unstalled flow. When the "B" parameter is too low, it is possible for the transient operation to converge on the lower equilibrium point (B) point in rotating stall. Simplistically, this may be analytically modeled as schematically represented in Figure 24.

In Greitzer's analysis, the compressor's steady state pumping characteristic is represented by an actuator disk. The compressor's flow inertia is represented by a duct of length (Lc) and area (Ac) and the downstream storage volume capability is represented by a simple volume (V). Downstream of the volume is a throttle, shown in this figure as the B Parameter Formula derived by Greitzer. Note that the desirably large values of B are obtained at low system frequency, at low compressor length and high wheel speeds. The low system frequency is obtained with a large downstream volume, a small compressor annulus area or large compressor length. Post stall behavior of a compression system in a gas turbine engine is governed by conditions significantly more complicated than this simple parameter suggests, but the basic phenomena is as described here. Some of these additional considerations are discussed in Greitzer's paper. Other engine design considerations are being investigated currently in research programs at the university research centers and gas turbine engine companies. A first-generation evolution of Greitzer's concept has been accomplished at Pratt & Whitney Aircraft using current engine and rig experimental data. This results in a stability limit line (Stall Recovery Factor, SRF) as shown in Figure 25. This parameter attempts to predict the boundary region where a stall can result in the compressor dropping into the undesirable rotating stall mode of operation.

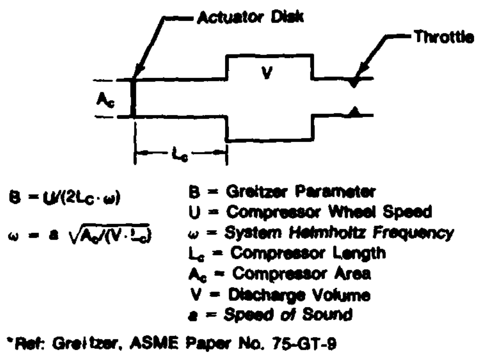

Figure 24. Greitzer B. Parameter*

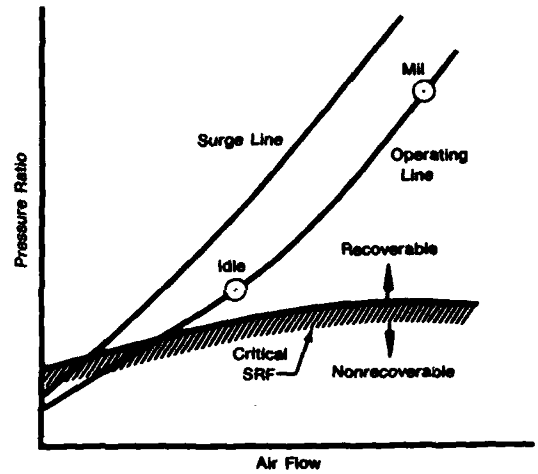

Figure 25. Engine recoverability determined by critical SRF boundary

The designer can use this model to configure new engines or modify existing engine systems to provide nonrecoverable stall margins for this low energy surge phenomena. In other words, the engine designer must configure the engine to satisfy the Stall Recovery Factor allowing margin not only for normal steady state engine operation but allowing for engine operation in the transient mode and off-design operation.

A pressure perturbation, such as from an augmentor pulse or a combustor pulse, may occur at a critical time in the transient causing the system to converge on the B equilibrium point instead of completing the surge loop as shown in Figure 26. Experimental investigation using full scale turbine engines have demonstrated this type of perturbation from several sources. The main engine combustor for instance, generally is unstable when the compressor is recovering from a stall and a sudden pressure change may result in a compressor pressure ratio perturbation at the critical moment. Similarly, there can be sources of pressure perturbation at the inlet. Fan exit pressure may drop as the compressor recovers causing a compressor inlet pressure perturbation. Also, if the engine has been operating with the augmentor lit just before the compressor stalled, the augmentor itself may cause that perturbation. All three of these phenomena have occurred during compressor stall recovery and have been observed during testing at Pratt & Whitney Aircraft. Each of these phenomena requires the engine system designer to provide the component dynamic characteristics to ensure problem avoidance and a completion of the full surge loop as opposed to dropping into the nonrecoverable rotating stall mode. Recognition of these phenomena coupled with the proper system design have shown dramatic decreases in the nonrecoverable stall rate. In production, turbine engine results of such a program are shown in Figure 27.

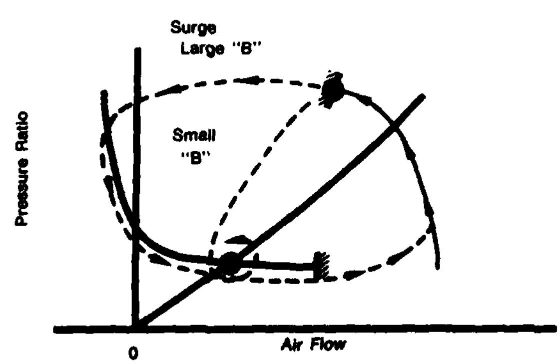

Figure 26. Post-stall transient

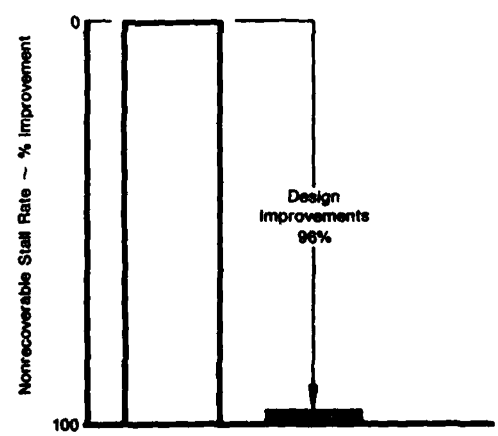

Figure 27. Nonrecoverable stall rate improvement

RESULTS AND CONCLUSIONS

As a result of the lessons learned to date, we conclude that the designer does have powerful tools with which to enhance the stability of advanced turbine engines. First and foremost is to know and to understand the technical challenge of the increasing stability demands of the modern engines. The general trend of increasing thrust/weight of the engines leads to increased thrust loading of the aircraft which in turns leads to increasing the transient requirements of these engines over their ancestors. The increased weapon system maneuverability requirements make it mandatory that the engine designer and airframe designer work together to create a system which can minimize the effects of the increasing demands of inlet distortion. This will allow the engine compression system to be designed with adequate surge margin distortion stability and attenuation tolerance as well as sufficient margin for the normal destabilizing events.

Secondly, and equally important for modern high technology turbine is that engine systems must be designed to eliminate the nonrecoverable stall phenomenon. One of the legacies of the lessons learned from past engines has been a basic understanding of the root cause of nonrecoverable stall. This has resulted in modern design analytical techniques which can be used to configure engine systems with sufficient forgiveness for nonpredictable stalling events, such as those experienced during a component failure. Simply said, we feel that the engine system dynamics can potentially be designed such that should a stall occur a full surge loop results over the full operating range thereby eliminating the nonrecoverable stall phenomenon.

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DISCUSSION

Ph. Ramette, Fr:

What do you think about the connection between the two points A and B which you show in some of your compressor map diagrams and Prof. Greitzer's theory for which the operating point in a compressor map is the intersection between

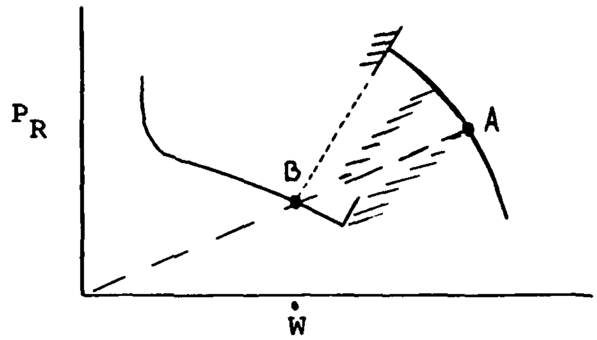
1) a compressor system line (i.e. which takes into account the compressor itself and the volume behind the compressor

and

2) a throttle line, since these two lines often cut in two points?

Author's Reply:

I am aware of no real compressor characteristics which, when combined with a fixed turbine flow parameter, would give a double solution such as you have described. These compressors show a transient non-steady "hysteresis loop" connecting the post stall steady state characteristics with the usual installed position.



I believe, however, that Dr Greitzer has used a mathematical model such as you described in some of his modeling activities.

W. Heilmann, Ge:

Referring to your figure 10: To what extent will the "Digital Engine Control" help to design high efficiency compressors with relatively modest surge margin?

Author's Reply:

Your question is a good one in that it points out that we should not simply substitute the full authority digital control for the hydromechanical one.

The first and most obvious benefit is that the additional accuracy reduces the control tolerances and therefore minimizes the running line variations. The more powerful benefit is to use the increase in computing power to compensate for Reynolds Number and distortion effects by causing the engine rematch only when needed. This will allow improved engine performance as well as a possible reduction in compressor stages.

J. Hourmouziadis, Ge:

Figure 25 shows a stall recoverability limit in the compressor map. Is it a characteristic of the compressor or of the complete engine system?

Author's Reply:

The stall recovery limit indicated in Figure 25 is an engine system limit line. The same compressor in a different measurement would react differently. Some of the major influences on the recovery limit behaviour are indicated in Figure 24. The value of the discharge volume, for instance, could be modified in a different engine and therefore change the system dynamics and resulting recoverability.

de Richmont, Fr:

D'après votre expérience, quel est le critère de distorsion de l'écoulement à l'entrée du moteur qui est le mieux corrélé à la perte de marge au pompage d'un turbofan?

De façon plus précise, pensez-vous que le KO maximum instationnaire suffit pour définir la perte de marge près du régime maximum? N'est-il-pas nécessaire de définir un nouveau critère qui tienne compte à la fois de la distorsion de pression totale et de la distorsion du "swirl"?

Author's Reply:

Pratt & Whitney's experience in treating the influence of inlet distortion on compression system performance has been limited to compression systems with inlet guide vanes. As such, inlet induced swirl has not a significant effect on the performance of that system.

I agree with your statement that the swirl effect must be considered for engines without guide vanes such as the RB199. This would imply that a new criterion should be derived which would consider all aspects of the environment.

CONCEPT MODERNE DE CONTROLE DE L'ACCÉLÉRATION : INTRODUCTION DE COMMANDE OPTIMALE DANS UNE STRUCTURE DE PILOTAGE PAR OBJECTIF

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RESUME

Les études théoriques effectuées sur les moteurs d'avions utilisent usuellement une représentation linéaire du système étudié. Cette approche est parfaitement justifiée quand on désire calculer les gains de régulation d'un point de fonctionnement stabilisé. Pour étudier les changements de régime, une représentation non-linéaire est plus précise mais les outils mathématiques d'investigation sont alors plus délicats à utiliser.

Nous présentons ici les résultats d'une étude soutenue par la DRET et effectuée en collaboration avec la SNECMA au cours de laquelle nous avons intégré les résultats d'une étude non-linéaire de transition et d'une étude linéaire de régulation dans le même régulateur.

NOTATIONS

N_B, N_H	: régimes des corps basse et haute pression
C	: débit carburant
S	: section de tuyère
$A_B(N_B,N_H,C,S)$ $A_H(N_B,N_H,C,S)$	: dynamique des corps basse et haute pression
F	: poussée du moteur
τ	: horizon d'optimisation
T_{5max}	: température T5 maximum admissible
(N_{BMAX},N_{Hmax})	: régimes maximum (N_B, N_H) admissibles
(N_{Bcible},N_{Hcible})	: raleurs désirées de (N_B, N_H)
Fcible	: valeur de la poussée stabilisée au point désiré
ψ_B, ψ_H	: composantes du système adjoint
$\alpha_H(N_b,N_H,C,S,\psi_N,\psi_B)$ $\alpha_B(N_B,H_H,C,S,\psi_N,\psi_B)$	: Equations d'évolution du système adjoint
C (ψ_H, ψ_B)	: Conditions finales que doivent vérifier les composantes du système adjoint

I - INTRODUCTION

Il faut distinguer deux types de fonctionnement d'un moteur pour lesquels les qualités requises du régulateur sont différentes :

- fonctionnement stabilisé : on demande au régulateur d'assurer une bonne stabilité du régime affiché, de compenser efficacement les variations des conditions de vol
- changement de régime : la transition doit être rapide mais les limites de fonctionnement (températures stabilité compresseurs) doivent être respectées.

Une représentation linéaire du moteur se prête à l'utilisation des techniques "LQR"(cf Skiva, De Hoff, Hall : "Design, Evaluation and Test of an Electronic, Multivariable Control for the F100 turbofan engine" AGARD - PEP 54 A, Cologne, Octobre 1979). Elles permettent d'obtenir aisément les gains de régulation d'un régime stabilisé. Il est théoriquement possible de les utiliser pour réguler des changement de régime. En pratique, cela conduit à des réalisations très lourdes et on évite cette approche. On préfère réguler un point de transition idéal délivré par un système de conduite de changement de régime (pilotage par objectif)

Nous avons exploré deux possibilités de génération de ces trajectoires idéales, l'une basée sur le Principe du Maximum, l'autre sur la Programmation Dynamique. Elles présentent l'avantage de ne pas imposer de modélisation linéaire du processus.

Le Principe du Maximum qui avait donné des résultats très simples et très intéressants pour des moteurs simples corps (cf Barrouil "Détermination de lois optimales de montée en régime d'un turboréacteur" AGARD PEP 54A, Cologne, Octobre 1979) conduit à des problèmes de précision de calculs très délicats et a dû être abandonné.

La programmation dynamique conduit sans difficulté à la tabulation des trajectoires idéales.

Notre étude s'est terminée par l'évaluation en simulation du système de régulation complet.

II - FORMULATION DU PROBLEME DE TRANSITIONS

II.a) - Modèle

Nous nous intéressons à un avant-projet de moteur double corps double flux en fonctionnement sans post combustion. Le modèle que nous utilisons est l'empilement des modèles des sous systèmes du moteur, basés sur l'expression thermodynamique des changements d'états du flux. Considéré comme une boite noire, on a un système :

- à deux commandes : débit carburant C et section de tuyère S
- à deux états : régimes des corps haute et basse pression : N_H et N_B

Le calcul d'un point de fonctionnement nécessite la résolution d'un système d'équations implicites pour laquelle une méthode itérative est utilisée.

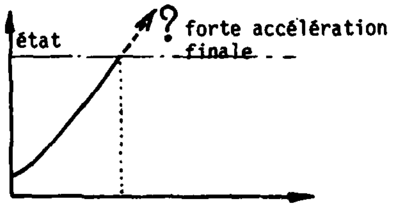
La réalité d'un moteur est plus complexe qu'un système du second ordre non linéaire et il existe notamment des modes pneumatiques très rapides au niveau des compresseurs et du mélange de flux dans la tuyère. On pourrait améliorer la qualité de la modélisation en augmentant l'ordre du système différentiel, en introduisant par exemple la pression P_9 de tuyère en variable d'état. La résolution des équations implicites à laquelle nous conduit le choix d'un modèle simple doit être interprétée comme le calcul à chaque instant du régime permanent des modes rapides du système.

Parmi les différentes grandeurs fournies par le modèle nous distinguons celles sur lesquelles portent les contraintes de fonctionnement du moteur :

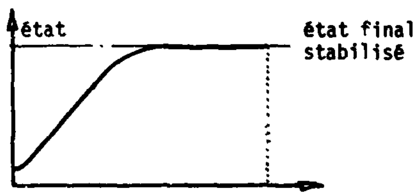
- T_5 : température devant turbine haute pression
- M_H : index de stabilité de fonctionnement du compresseur haute pression
- M_B : index de stabilité de fonctionnement du compresseur basse pression
- F : poussée du moteur.

II.b) - Critère de performance

Nous recherchons des transitions performantes. Pour utiliser des méthodes mathématiques de résolution il faut préciser le concept de "performance". Les critères de type temps minimum sont mal adaptés à notre problème car ils laissent le moteur dans un état très destabilisé en fin de transition. Il vaut mieux optimiser un critère de performance sur un horizon fixé, assez grand pour que le moteur soit stabilité en fin de transition.



critére de type temps minimum



optimisation sur horizon fixé

Pour optimiser les montées en régime, nous avons choisi le critère :

$$\mathcal{J} = \int_0^T F(N_B, N_H, C, S)\, dt$$

Il est clair que la maximisation de ce critère va dans le sens de la performance. Des essais ont été faits avec d'autres expressions tendant aussi à la recherche de la performance : les résultats sont tous très proches.

II.c) - Expression mathématique du problème

Formellement, notre problème s'énonce ainsi :

- Etant donné un système décrit par :

$$\begin{aligned}
dN_H/dt &= A_H(N_B, N_H, C, S) \\
dN_B/dt &= A_B(N_B, N_H, C, S) \\
T_5(N_B, N_H, C, S) &\geq T_5max \\
M_B(N_B, N_H, C, S) &\geq M_Bmin \\
M_H(N_B, N_H, C, S) &\geq M_Nmin \\
N_B \leq N_Bcible &\leq N_Bmax \\
N_H \leq N_Hcible &\leq N_Bmax
\end{aligned}$$

- trouver la loi de commande maximisant le critère de performance :

$$\mathcal{J} = \int_0^T F(N_B, N_H, C, S)\, dt$$

On notera que le point de fonctionnement désiré est introduit par ($N_B$cible, $N_H$cible) dans l'expression des contraintes.

III - CALCUL DES TRANSITIONS IDEALES

Deux méthodes de résolution ont été testées : principe du maximum et programmation dynamique. La première consiste à calculer une à une chaque transition optimale, la seconde à faire progresser de front la recherche de toutes les transitions optimales (cf planche 1).

La nature des calcul est très différente d'une méthode à l'autre.

III.a) - Utilisation du principe du maximum

Les résultats données par la théorie se résument à :

. *Les commandes optimales* sont celles qui maximisent compte tenu des contraintes à chaque instant le hamiltonien :

$$\mathcal{H} = -F(N_B,N_H,C,S) + \psi_B\, A_B(N_B,N_H,C,S) + \psi_H\, A_H(N_B, N_H,C,S)$$

où ψ_B et ψ_H sont les composantes du "système adjoint".

Le maximum de ce hamiltonien est constant dans le temps. Comme les accélérations sont nulles en fin d'évolution (le régime permanent est atteint) on a :

$$\mathcal{H} = \text{constante} = -\text{Fcible} = \text{poussée stabilisée au point de consigne}$$

. *L'évolution du système adjoint* (ψ_H, ψ_B) est régie par des équations différentielles assez complexes dans lesquelles interviennent les dérivées partielles par rapport à N_H et N_B des grandeurs $A_B()$, $A_H()$, $T_5()$ $M_B()$, $M_H()$, $F()$. Celles-ci ne sont pas connues s s forme analytique. Il faut donc calculer les dérivées partielles par petites perturbations des paramètres. Par exemple :

$$\frac{\delta F}{\delta N_H} \simeq \{ F(N_B, N_H + \delta N_H, C, S) - F(N_B, N_H, C, S)\} \;/\delta N_H$$

La qualité de ces calculs est très liée à la précision du modèle que nous avons été amené à améliorer à plusieurs reprises, non pas au niveau du modèle lui même mais à celui des méthodes numériques :

- notre modèle comporte un certain nombre d'abaques utilisées par interpolations entre les points de codage. L'interpolation linéaire introduit des variables brutales de pente aux points de codage qui se propagent dans le calcul des dérivées. L'interpolation d'ordre supérieur améliore la précision.
- la méthode itérative de résolution des équations implicites comporte un test d'arrêt qui détermine la précision des calculs. Pour obtenir une précision convenable de la dérivée, il faut être très exigeant à ce niveau.

Les calculs d'intégration du système adjoint sont très pénalisants en temps de calcul.

. *Les valeurs finales du système adjoint* doivent vérifier un certain nombre de consitions $C(\psi_H, \psi_B)=0$ dont l'expression peut être également assez complexe. L'organigramme des calculs est présenté sur la planche 2 ainsi que l'allure des résultats obtenus.

La résolution de ce sytème (détermination de ψ_H et ψ_B à $t = 0$ de sorte que $C(\psi_H, \psi_B) = 0$ à $t = \tau$) est longue et il faut renouveler ce calcul pour chaque point (N_H, N_B) initial différent. Par ailleurs, la précision du résultat final est incertaine. Ces raisons nous ont incités à abandonner l'approche "Principe du Maximum".

III.b) - Utilisation de la programmation dynamique

. *Maillage de l'espace d'état*

Dans le plan (N_B, N_H) nous considérons que le domaine utile est une bande de 1500 t/mn de large sensiblement centrée sur la ligne des points de fonctionnement stabilisés. Cette bande est maillée avec un pas de 200 t/mn en N_B et N_H. Ce maillage détermine des lignes de points parallèles à la seconde diagonale numérotées à partir du point plein gaz (numéro 0).

On admet qu'une trajectoire de montée en régime passant par la ligne (i+1) est dirigée vers la ligne i.

. *Discrétisation de l'expression du critère de performances*

Notons P_i le point (N_{Bi}, N_{Hi}). Posons

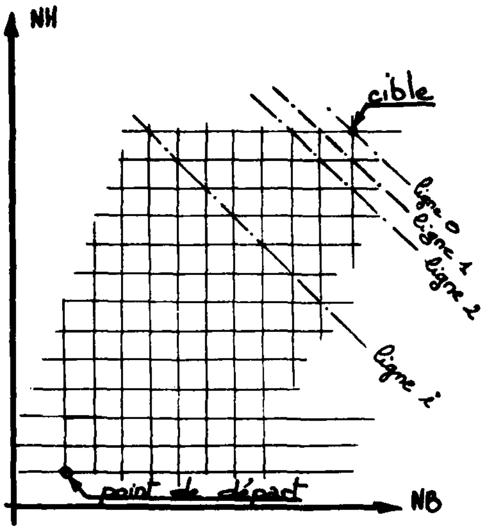
$$\hat{R}(P_i) \triangleq \underset{C(t),S(t)}{\text{maximum}} \int_{t=0}^{\tau} F(N_B, N_H, C, S)\, dt$$

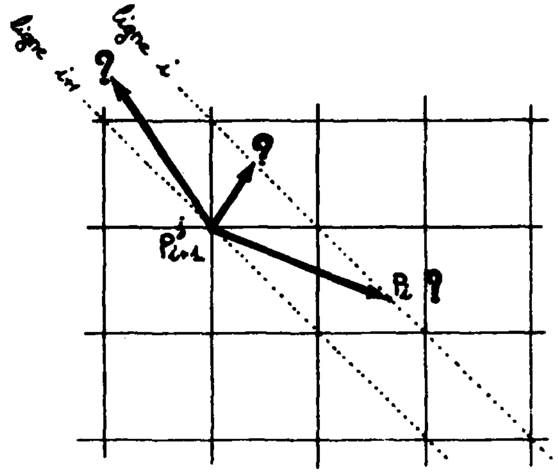
$\hat{R}(P_i)$ désigne donc la valeur optimale du critère de performance pour une transition partant du point P_i. On remarque que $\hat{R}(\text{cible}) = \text{Fcible} \cdot \tau$

On peut voir que si $\Delta\tau$ est assez petit et τ assez grand pour que le régime stabilisé soit atteint en fin d'évolution on a l'identité :

$$\hat{R}(P_{i+1}) = \max_{C,S} \left[\{F(N_{B_{i+1}}, N_{H_{i+1}}, C, S) - \text{Fcible}\}\, \Delta\tau + \hat{R}(P_i)\right]$$

P_i désignant le successeur de P_{i+1} dans le délai $\Delta\tau$



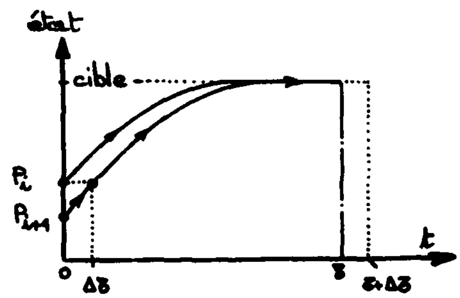


Cette écriture se prête tout naturellement à la résolution par programmation dynamique inverse.

. *Utilisation de l'équation récurrente en $\hat{R}$*

Supposons connues toutes les valeurs de $\hat{R}$ aux points de maillage de la ligne i. Considérons un point de maillage de l'espace d'état P^j_{i+1} de la ligne i+1.

Au point P^j_{i+1}, nous balayons avec un pas de recherche ΔC toutes les valeurs possibles du débit carburant C et pour chaque valeur de C, nous explorons avec le pas de recherche ΔS toutes les valeurs possibles de S.



On rejette les valeurs de C et S telles que les contraintes de fonctionnement ne sont pas respectées.

Pour les valeurs de C et S admissibles, on évalue $F(N_B, N_H, C, S)$, $A_B(N_B, N_H, C, S)$ et $A_N(N_B, N_H, C, S)$ On connait alors la direction du vecteur vitesse, le point P_i d'intersection avec la ligne i et la durée Δτ de la transition $P^j_{i+1} \rightarrow P_i$. Par interpolation entre les points de maillage de la ligne i, on calcule la quantité R au point P^{i+1}_i puis :

$$R(P_{i+1}, C, S) = (F(N_H, N_B, C, S) - Fcible)\,\Delta\tau + R\,(N_H(P_i), N_B(P_i))$$

A l'issue de l'exploration en C et S on sait évaluer :

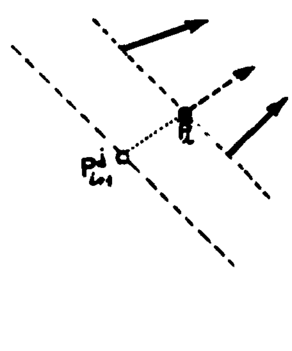
$$\hat{R}(P_{i+1}) = \underset{C,\ S}{\text{maximum}}\ R(P_{i+1}, C, S)$$

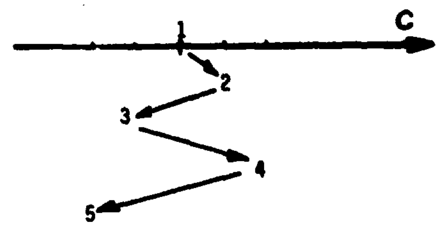
L'initialisation se fait sur la ligne 0 avec R(cible) = Fcible . τ

. *Amélioration de la méthode*

Un certain nombre d'aménagements sont nécessaires pour rendre cette méthode plus performante, sans altérer le principe même.

Le point le plus important est d'éviter d'explorer en chaque point l'ensemble des commandes C et S admissibles. On trouve une bonne initialisation de ces grandeurs au point successeur "le plus probable" P_i tel que le vecteur vitesse en ce point passe par P^j_{i+1}.





recherche en "zig-zag" de Cidéal

A partir de ces valeurs initiales de C et S, on effectue une recherche "en zig-zag" et on s'arrête dès qu'un optimum local est trouvé.

. *Résultats*

A l'issue de la résolution, on dispose en chaque point du domaine d'état admissible des valeurs idéales de débit carburant et de section de tuyère. Il s'agit donc d'une loi de commande en boucle fermée. On trouvera sur la planche 4 la représentation des vecteurs vitesse idéaux dans le plan d'état (N_B, N_H) pour le moteur que nous avons étudié.

IV - INTEGRATION DES TRANSITOIRES IDEAUX DANS UN SYSTEME DE REGULATION

IV.a) - Régulation en fonctionnement stabilisé

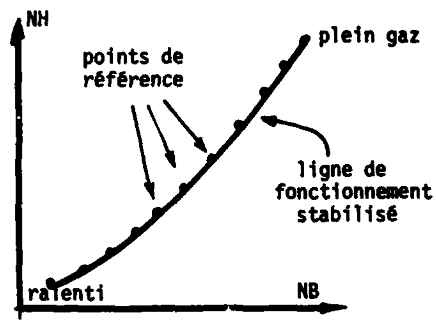
Les gains de régulation sont calculés par la méthode LQR déjà citée. La démarche consiste à :

- définir sur la ligne de fonctionnement stabilisée un ensemble de points de référence
- Calculer en chaque point de référence un modèle linéaire tangent au modèle thermodynamique non linéaire
- utiliser un package LQR pour calculer les gains de régulation G

Le schéma de principe de la régulation est présenté sur la planche III.a

L'adaptation consiste à afficher les gains obtenus par interpolation entre les points de référenches les plus proches.

On notera qu'en fonctionnement stabilisé, cette interpolation ne pose aucun problème de principe.



IV.b) - Régulation en transition vers le plein gaz

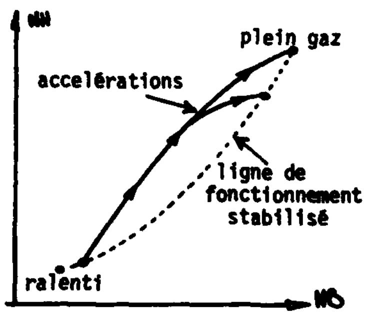
La mesure de (N_B, N_H) courants permet de lire dans les tables les valeurs idéales de C et S.

Le même schéma de base peut être utilisé mais il faut remarquer que la correction par les gains de régulation est fictive (cf planche III.b).

IV.c) - Régulation d'une transition vers un régime intermédiaire

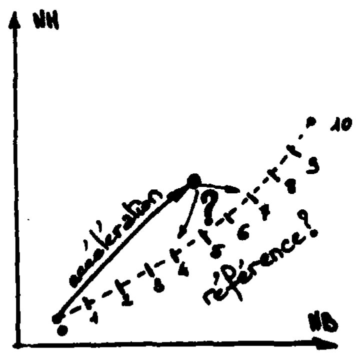
Le calcul des transitions idéales vers un régime intermédiaire montre que le début de telles accélérations sont assez proche des transitions vers le plein gaz.

On peut donc piloter toutes les montées en régime en donnant comme consigne de transition la montée au plein gaz quand on est loin de la cible et une trajectoire de raccordement quand on s'en approche. Ce principe peut être réalisé par un régulateur dont le schéma de principe figure sur la planche III.c dont nous avons vérifié la validité en simulation.



Il faut noter ici un problème au niveau de l'adaptation des gains de régulation et des valeurs de saturations des commandes : le point de transition courant ne se trouve pas sur la ligne de fonctionnement stabilisée et la méthode d'interpolation du IV.a doit être aménagée. On calcule les gains en faisant une moyenne de leurs valeurs aux points de référence "les plus proches".

Dans notre étude nous avons défini une proximité au sens des valeurs du régime du corps basse pression. La précision obtenue sur les limites des paramètres critiques T_5, NH et NB nous a paru satisfaisante.



IV - CONCLUSION

Les trajectoires "idéales" de transition que nous avons déterminées ne sont pas très sensiblement meilleurs que celles que l'on peut obtenir habituellement par une série d'essais et d'améliorations successives.

Les avantages que nous voyons dans la méthode proposée est :

- son aspect systématique. Un programme écrit en FORTRAN a été établi. Il calcule les trajectoires idéales de transition, les gains de régulation et les limitations de commande associées à un modèle de moteur donné.
- son aspect de référence. Les évolutions obtenues servent de point de repère pour juger de la qualité de régulateurs sous optimaux.
- sa rapidité de mise en oeuvre qui en fait un outil intéressant au niveau de l'avant projet pour tester rapidement les répercussions sur le système régulé d'une modification d'un étage de moteur.

PLANCHE 1 COMPARAISON DES METHODES: "PRINCIPE DU MAXIMUM" ET "PROGRAMMATION DYNAMIQUE"

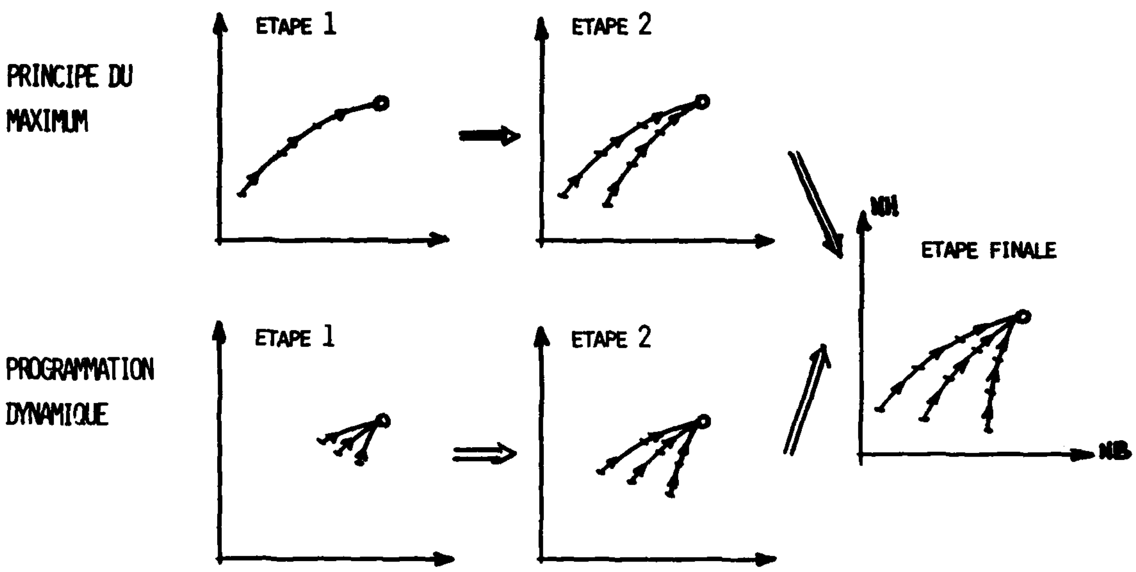
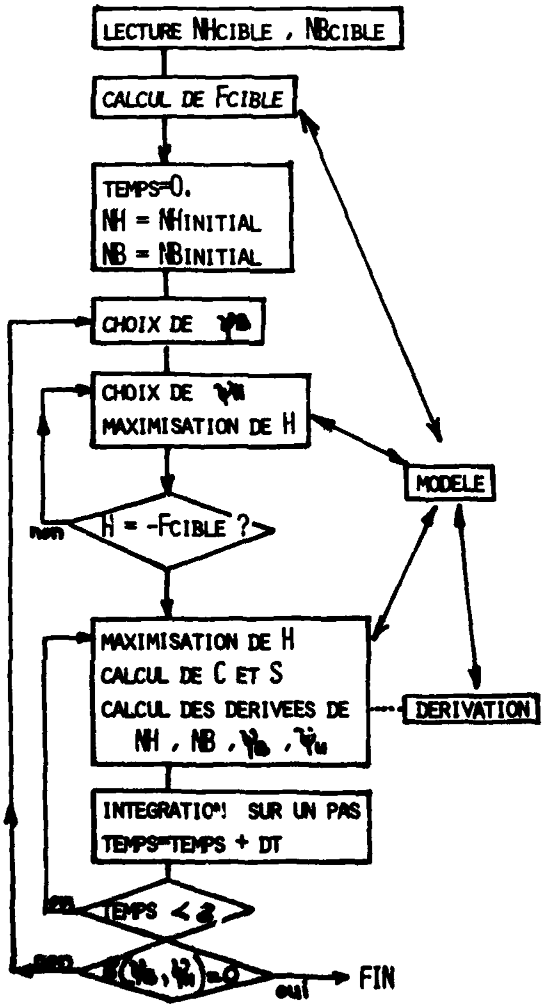


PLANCHE 2 : MISE EN OEUVRE DU PRINCIPE DU MAXIMUM

ORGANISATION DES CALCULS



EVOLUTION DU SYSTEME POUR DES INITIALISATIONS DIFFERENTES DE ψ_B ET ψ_H

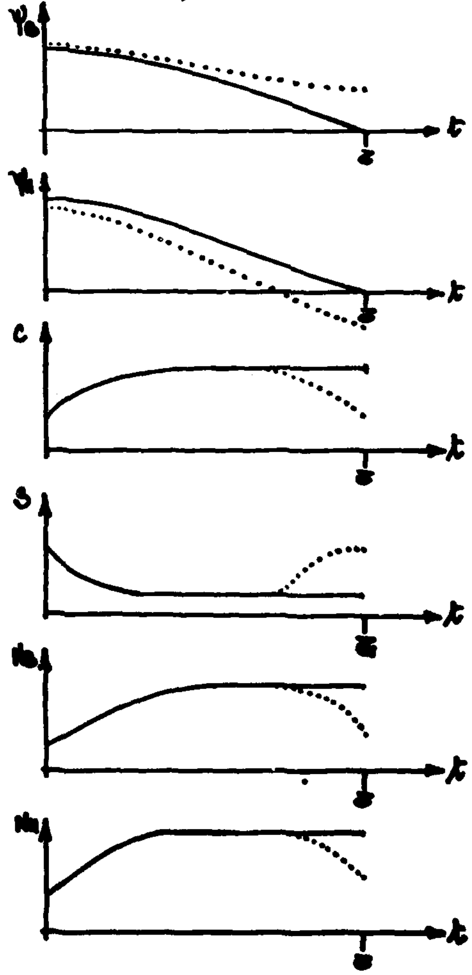


PLANCHE 3 SCHEMAS DE REGULATION

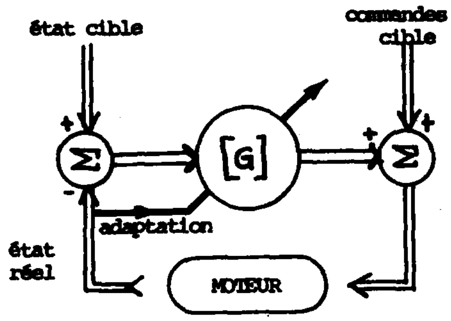


PLANCHE 3A

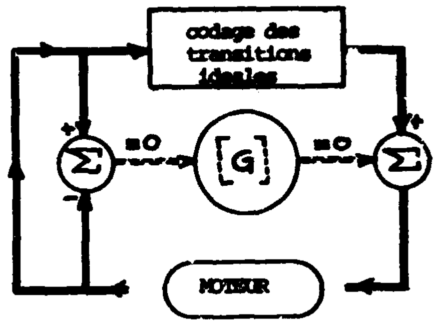


PLANCHE 3B

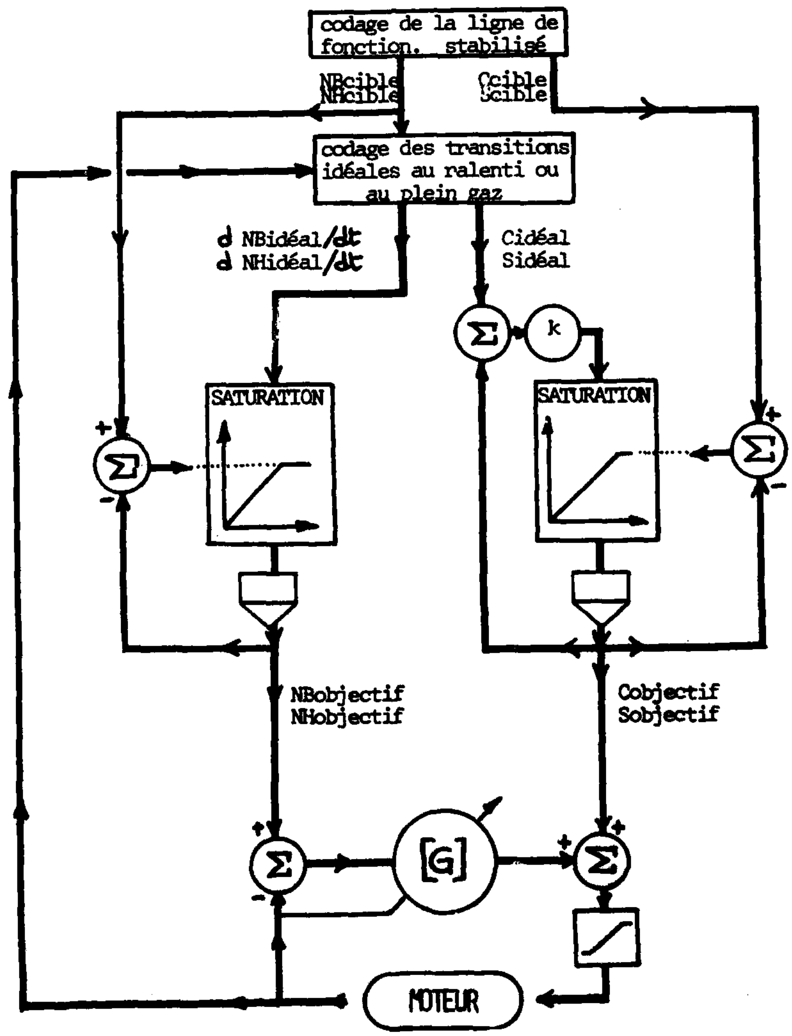
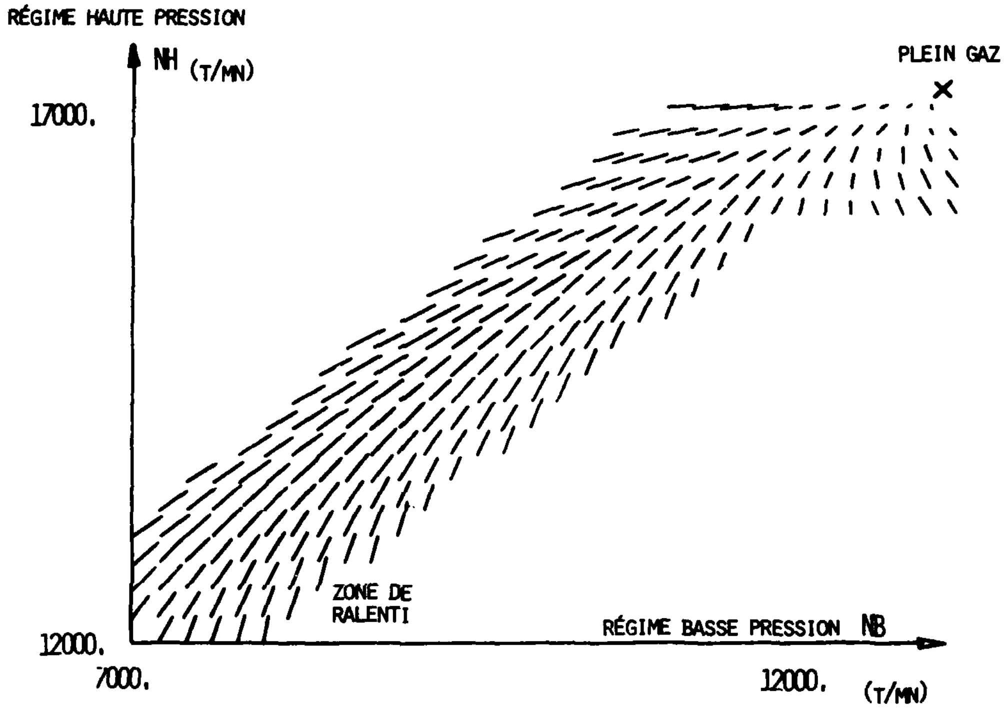


PLANCHE 3C

PLANCHE 4



LES SEGMENTS REPRESENTENT LE VECTEUR VITESSE IDÉAL

A NEW APPROACH TO REHEAT CONTROL

by

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SUMMARY

The task of the reheat (afterburner) control system is to provide as rapid a thrust response as possible without exceeding reheat stability limits or engine limits. As there is at least one reheat fuel flow to control (in some systems two or three fuel flows to different reheat manifolds can be controlled separately) and also a variable nozzle area to control, a variety of control strategies can be considered. The Paper describes a closed-loop controller, incorporating automatic buzz-avoidance, and gives the results of an engine test bed evaluation.

LIST OF SYMBOLS

A_{MAX}	Nozzle area at maximum reheat
A_{MIN}	Nozzle area at minimum reheat
A_N	Nozzle area
F_{RH}	Total reheat fuel flow
F_{MAX}	Total reheat fuel flow at maximum reheat
F_{MIN}	Total reheat fuel flow at minimum reheat
N_H	High pressure rotor speed
N_L	Low pressure rotor speed
P_{T1}	Intake pressure
T_{T2}	Low pressure compressor delivery pressure
P_{T3}	High pressure compressor delivery pressure
P_{T6}	Jet pipe pressure
T_{T1}	Intake temperature
θ_p	Pilots lever demand

1. INTRODUCTION

The task of the reheat control system is to provide as rapid a thrust response as possible without exceeding reheat stability or engine limits. On a modern bypass military engine (Figure 1) this is not straight forward because any changes in the jet pipe pressure (P_{T6}) are readily transferred, via the bypass ducting, to the low ressure compressor (fan). This will result in the fan being forced away from its optimum working line, and in the extreme a compressor surge may occur. Any change in reheat fuel flow (F_{RH}) must therefore be matched by a corresponding change in jet pipe nozzle area (A_N) to maintain the jet pipe pressure and hence retain the fan working line. The control system must also ensure that the reheat fuel is kept within the limits defined by the engine/flight conditions. At minimum fuel values the system should prevent rough, unstable, inefficient combustion and at the maximum values it should prevent over-rich combustion and the reheat instability known as buzz.

This paper discusses the problems associated with two possible ways of controlling reheat in the modulation range (ie reheat alight) and then describes in detail a combination of these schemes which overcomes their drawbacks. This latter system was implemented using a digital micro-processor to perform the control algorithms thus resulting in a control package which used the minimum of hydromechanical components. The control of reheat within the modulation range is of course only part of the overall system as depicted in Figure 2 which shows a flow diagram for full reheat control with prime, light-up and purging incorporated. These necessary stages in the sequencing of reheat are not discussed in any detail here because they are hardware dependant and also do not affect the main control theme of this paper. The paper also includes a brief description of a test bed facility[1] which utilises the flexibility of digital processing to quickly find the optimum running conditions of a reheat system. A brief outline of the hardware used for testing will be followed by typical results showing the features of the system.

2. POSSIBLE SOLUTIONS FOR TRANSIENT CONTROL OF REHEAT

With reference to Figure 1 it can be seen that the system to be controlled consists of two inputs (nozzle area and reheat fuel flow) which must be changed in the correct relationship to achieve minimum interaction with the engine. The solutions considered are discussed below.

2.1 Pure closed loop

It would be possible to have a closed loop control system which effectively monitors the fan's working point and adjusts one of the inputs, say the nozzle area, to keep the fan on its optimum running line, as the other input reheat fuel flow is varied. It is preferred to close the loop with the nozzle area as this removes any problems due to under/over fueling which can occur with closed loop control of the reheat fuel. This particularly applies to over fueling when in the extreme case, with a very rich mixture, an increase in fuel flow could cause a reduction in jet pipe pressure.

There are two problems with this approach:-

a. The determination of the fan's working point, which for test bed running can be conveniently measured by its pressure ratio and corrected rotational speed, could have serious errors in flight due to the effects of intake distortion on the pressure ratio measurement. The accepted way of overcoming this is to use the turbine pressure ratio, which is less affected by the distortion. The effective working line error can therefore be derived from the required turbine pressure ratio (found from a schedule against corrected fan speed) and the actual pressure ratio measured with two pressure transducers.

b. The second problem, of large transient working line errors, is not so readily overcome. The response of the nozzle is proportional to the working line error and when the fuel changes are rate limited (to prevent gross errors) this inevitably limits the speed of the transient.

Therefore with pure closed loop control, the fan's working line can be effectively controlled but only if the rate of change of reheat condition is relatively low. In consequence it was not selected for further development.

2.2 Open loop control

In this approach prior knowledge of the reheat steady state behaviour allows a schedule of nozzle area against reheat fuel flow to be defined. In one application of this method, the nozzle is positioned in response to the pilots lever demand, and the actual nozzle position is used to derive the reheat fuel flow using the pre-defined nozzle/fuel relationship. Unfortunately it is difficult to define this nozzle/fuel schedule over the flight envelope, particularly when engine limiters come into operation. Therefore a basically simple system ends up with a series of trimming schedules and dynamic compensation in an attempt to keep the working line error within acceptable limits in both steady state and transient conditions.

This therefore is a system which can be set up accurately for any one operating condition with no guarantee it will achieve the required performance over the whole flight envelope.

2.3 Open loop with closed loop trim

This combination of the above schemes, uses the defined nozzle/fuel relationship to approximately position the two actuators whilst applying closed loop control to keep the fan on the required working line. In this way, closing the loop around the open loop system obviates the need for complex trimming adjustments to the nozzle/fuel schedule and using the closed loop with a schedule, overcomes the problem of working line errors related to the speed of the transient. This type of system which is fully compensated for the complete flight envelope, and is capable of achieving fast transient reponses, has been selected at NGTE for research work into reheat control.

3. A NGTE REHEAT CONTROL SCHEME

The system outlined below, which was the first serious attempt at digital reheat control at NGTE, was designed to control the reheat in an experimental jet pipe having three fuel manifolds. Figure 3 shows an overall block diagram of the control system. The pilots lever demands a fuel flow between the limits defined by the engine/flight conditions. To keep the transient response of the system within the capabilities of the actuators, the fuel demand must be rate limited. The rate limited demand of reheat fuel is fed to the fuel actuators via a section which controls the split between the three manifolds. The split control is scheduled from the rate limited fuel demand, which is also used to derive the scheduled nozzle demand.

Thus the reheat control scheme consists of an open loop system with both sets of actuators being driven in a pre-defined relationship. Superimposed on this is the closed loop working line trim which is added to the scheduled nozzle value to produce the final nozzle demand.

It is appropriate to consider the individual blocks of the system in more detail.

3.1 Fuel control

The pilots lever (Figure 4) demands a percentage of total reheat fuel between the lower and upper limits as defined by the current engine/flight conditions. The maximum fuel flow for each of the manifolds is of the form

Primary	$f(N_H, P_{T3})$
Gutter	$f(N_H, P_{T3}, T_{T1})$
Colander	$f(N_H, P_{T3}, T_{T1})$

In each case the N_H (high pressure rotor speed) term is linear, the P_{T3} (high pressure compressor delivery pressure) term, is quadratic, and the T_{T1} (inlet total temperature) term is cubic. Thus the maximum total reheat fuel flow can be calculated. The ratio maximum/minimum reheat total flow is found from a schedule against P_{T3}.

The resultant pilots demand of total fuel flow is next rate limited to match the systems response to that of the slowest actuator. In this case a rate limit was chosen so that the nozzle (the slowest actuator) was within its maximum speed capabilities - thus allowing faster travel if the working line trim required it.

This rate limited pilots demand of total fuel flow must now be proportioned between the three manifolds to give the required optimum reheat performance. This is achieved by keeping the primary flow constant and varying the combined gutter and colander fuel flows to achieve the required total flow. The ratio between these two manifold flows was also varied by a schedule against the current total fuel flow. The top end point on this schedule has already been defined by the maximum allowable fuel flows, and the remaining points were established during the optimisation tests (see Section 4).

3.2 Scheduled nozzle position

The scheduled nozzle position is derived directly from the fuel/nozzle schedule utilising the rate limited total fuel flow demand. The actual fuel/nozzle relationship used for this is derived by scaling from a stored fixed schedule as shown in Figure 5. The pair of co-ordinates defining the minimum and maximum operating conditions again vary with engine/flight conditions. The fuel flow values are as previously described in the fuel flow section (3.1) and the related nozzle values are scheduled against P_{T3}. The shape of the curve was found during the optimisation tests (see Section 4) and is stored separately. The working nozzle/fuel schedule is produced by fitting this curve between the defined end points. The assumption that the shape of this single curve can apply over the flight envelope was verified at NGTE in 1970 with a similar type of reheat control.

This technique is easy to implement in a digital computer and it is thought to be an adequate definition of the fuel/nozzle relationship over the flight envelope bearing in mind that the system incorporates closed loop working line trim which will compensate for any discrepancies between the schedule and the actual performance.

3.3 Working line trim control

As discussed earlier the parameter used for working line control is the turbine pressure ratio P_{T3}/P_{T6} which being non-dimensional compensates for altitude effects. From the engine steady state performance a schedule of P_{T3}/P_{T6} against $N_L/\sqrt{T_{T1}}$ can be defined for the required running line. Thus by measuring N_L (low pressure compression speed) and T_{T1} (intake temperature) the demanded pressure ratio can be "looked up". The actual values of P_{T3} and P_{T6} (jet pipe pressure) are measured directly and hence the actual pressure ratio can be calculated. The difference between these demanded and actual pressure ratios (the pressure ratio error) is directly related to the required working line error.

This pressure ratio error is the input to the working line trim controller which is a modified two term controler (Figure 6) having proportional and integral components. A three term controller was tried but no great improvement was observed. This may be because of the limit on differential gain imposed by noise and the relatively slow nozzle response.

One problem associated with closed loop control systems is that of actuator wear due to the controller attempting to follow any noise in the system, because it is unable to distinguish between the noise and a genuine movement in the operating point. In this NGTE system it was found that the pressure ratio signal had as much as 0.05 of a pressure ratio of noise when minimum reheat was burning, reducing as reheat fuel flow increased. There are several ways of avoiding this noise problem, whilst keeping the capability of fast response, by using variable time constant low pass digital filters in the input to the controller. An alternative technique used successfully for this and an earlier project was to employ switched gains in the control loop such that when the error signal was small (0.1 of a pressure ratio), the proportional gain was reduced to zero and the integral gain reduced by a factor of 16, hence effectively filtering out any noise. Thus in steady state the system can follow any slow drifts in the operating point (eg due to heat soakage) without responding to the aerodynamic noise of combustion, but if the working line error signal exceeds the switch datum value the full gains are used giving fast response. Utilising switched gains in this way allows marginally higher gains to be used because of the effective dead band of the low gain region. It should be noted that when switching from high gain to low gain mode of operation, the integrator must be reset to the last trim output to avoid "bouncing" due to the loss of the proportional term.

3.4 Buzz control

The reheat instability of buzz, (violent self induced oscillations in jet pipe pressure) must be avoided because of the resultant structural damage. The reheat control system being discussed employs a technique of controlling the reheat fuel flow and manifold split ratio so as to keep the amplitude of buzz to acceptable limits.

Figure 7 shows a block diagram of the buzz controller including hardware and software. Full details of this technique were reported by Waters[2]. The buzz signal is obtained by feeding the P_{T6} signal into a broad band pass filter (centred on 150 Hz) to remove the DC and all unwanted higher frequency components. The resultant signal is full wave rectified and fed through a low pass filter to produce a DC voltage proportional to the AC buzz component of P_{T6}. In the software this signal is then normalised by P_{T6} to produce the buzz signal used by the controller.

This buzz signal is compared with a datum value and the resultant difference is integrated. The integrator output, which is set to zero if it goes negative, is subtracted from the demanded fuel flow to reduce the fuel and hence the level of buzz. The value in the buzz integrator is also used to modify the split in fuel flow between gutter and colander. During the optimisation tests (see Section 4) on the reheat system, it was found that the gutter fuel flow was the main contributing fuel to buzz amplitude. Therefore during buzz controlling it was arranged to bias the split in fuel flows towards the colander, thus requiring less reduction in total flow than if the split had been left as scheduled. The change in the fuel split ratio was chosen so that again optimum thrust was obtained at the buzz controlling point.

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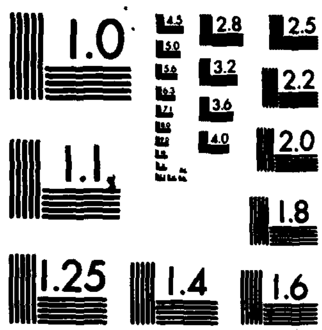

MICROCOPY RESOLUTION TEST CHART
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4. METHOD OF REHEAT OPTIMISATION

The general requirement was to find the optimum flow split between the three fuel manifolds over a range of effective dry nozzle area's (EDNA) in order to maximise the thrust from the engine-reheat combination. EDNA, of course, defines a particular working line on the low pressure compressor map.

In principle, utilising the power of a digital computer, it is possible to produce a scheme which will automatically "hill climb" to the optimum point. But as this would not produce any off-optimum data, the following approach was adopted, which is semi-automatic in operation and quickly produces the required data.

The major interest was in how the thrust varied with the ratio of the two main fuel flows (colander/gutter), as it was decided to use an automatic ramping of this fuel ratio whilst maintaining EDNA, primary and total fuel flows constant. During a ramp the control computer was used to automatically take several sets of data log points on limited parameters. At the same time, an indication of reheat efficiency (nozzle area/EDNA) was plotted against the fuel ratio on an oscilloscope for each of the data log points. Thus on completion of the ramp the operator can see where the optimum lies and then the ratio can be manually moved back to this point and a detailed set of engine data was then gathered. In this way, keeping primary flow and EDNA constant, a full set of optimisation data was obtained for various total reheat fuel flows.

The effects of primary fuel flow and EDNA can then be determined by either changing their values at the optimum points or by repeating the optimum data gathering, as above, with different primary fuel flows or EDNA values. The latter method was used to test the feasibility of the optimisation system.

To get meaningful results, care was particularly needed in two areas of operation:

Firstly the ramping of the fuel ratio, the total demanded fuel must be adjusted to compensate for errors in the fuel control valves and to keep the actual total flow constant. This actual flow was measured using a turbine flow meter. After corrections for non-linearity, the readings were averaged over one second periods (to minimise noise) and used to adjust the colander and gutter demands (keeping their ratio correct). Thus even though the primary flow and colander/gutter ratio were only held nominaly correct the total fuel flow was constant to better than 0.1%.

Secondly the transducer signals used for the data logging suffered from noise (mainly aerodynamic) and therefore a single sample of data would have produced inconsistent results. A short term repeatability of better than 0.5% (in some cases 0.1%) was achieved by letting the system settle for 5 seconds after ramping to the selected point and then the data were smoothed for a further 5 second period, with a digital low pass filter of time constant 10 seconds, to produce the logging data.

5. HARDWARE

The reheat control system was tested using a miscellaneous selection of experimental components. The components were not representative of flight standard units, but they were adequate to test the philosophy of the control and optimisation schemes.

The tests were conducted on a sea level static test bed using an early development Phantom Spey engine which was fitted with a non standard reheated jet pipe known as the DP20. This jet pipe was used during the early development of the latest "burn and mix" reheat systems and is therefore fairly modern in concept having three fuel manifolds. The primary, the gutter manifold in the hot stream and the colander which sprays fuel into the mixing shutes of the bypass air, are shown in Figure 8.

The main engine control was the standard Spey hydromechanical unit (CASC). To preserve engine life all testing was performed at the engines maximum continous rating of 93.5% N_H.

Figure 9 shows a diagrammatic representation of the reheat control hardware. The Spey normally uses a vapour core pump and hydromechanical reheat control system. This unit was extensively modified to leave only the vapour core pump in operation. The reheat fuel flow control valves used were some early experimental units produced for work in this field. Within their accuracy they are true flow control valves, ie within their operating range they will pass the demanded flow regardless of upstream pressure or downstream restrictions. The valves were fast enough in operation to allow them to be used to prime the manifolds prior to light up. Reheat ignition was achieved by a hotstreak injector system using an existing unit after modification to reduce its metered output.

The exhaust nozzle control system used most of the standard Spey hydraulic components, but it was changed to a position control servo by the addition of an input drive jack, electricaly operated spool valve, position transducer and some external electronic amplifiers.

The interface between the engine transducers/actuators and the digital control computer was of "in house" design and manufacture, and included all the necessary signal conditioning and drive circuits as well as the conventional multiplexer and ADC system.

The digital computer used was the Texas development system the 990/4, it was selected and purchased when the TMS 9900 micro-processor, which it uses, was the only 16 bit micro-processor with inbuilt multiply and divide instructions available. The computer is still in use at NGTE for control studies rather than the second generation of faster micro-processors because of the investment in the machine code software which has been made. With the extra software necessary to allow detailed monitoring of the performance of control, the cycle time was marginally slow, for reheat, at 20 milliseconds.

6. RESULTS

6.1 Modulation control

It serves no purpose to demonstrate the performance of this type of control system with a perfect match between the nozzle/fuel schedule and the reheats steady state operation, which would be so easy to achieve at sea-level-static. Therefore to demonstrate the operation of the working line trim, the schedule was purposely offset by a nozzle area of 10% at the top end producing the open loop response shown in Figure 10 for a transient from minimum to maximum reheat. This of course produces an unacceptably high pressure ratio error using open loop control. Bringing the working line trim control into operation and repeating the same transient shows the benefits of fast working line control. The pressure ratio error has been contained within acceptable limits, and the nozzle moves smoothly to its correct area. The transition of the working line trim term from a very slow ramp to a fast ramp and visa versa at the gain switching pressure ratio error is readily apparent. This rather "steppy" nature of the trim is smoothed out by the relatively slow response of the nozzle servo system.

Figure 11 shows the action of the control system when performing a transient into buzz. (The buzz datum was depressed to avoid any possible damage to the jet pipe or engine.) Here, even though at these conditions the buzz signal is very noisy, the effect is readily seen. When the buzz signal is greater than the datum, the nozzle area and hence fuel flow (not shown) are reduced, and conversly when the buzz signal is lower than the datum.

Having considered the operation of reheat during the modulation range, it is worthwhile briefly examining its behaviour during light up. In the system used, whilst the reheat manifolds are being primed, the nozzle is pre-opened under working line control by the selection of a lower fan running line. With reference to Figure 12 this takes the fan away from surge ready for the ignition of reheat which will of course move the fan nearer to the surge line. Thus during this stage of operation the pressure ratio quickly drops (A to B) to the new working line and then slowly increases as N_L increases due to the miss-match in the engine. The nozzle area stays sensibly constant during this period (B to C). When reheat light up occurs (C) and is detected (D) the normal running line is selected, but the engine is again miss-matched, with N_L too high, and the system tracks back to the starting condition (A) as shown. As well as this light up to minimum reheat, Figure 12 also shows a trace for a light up to maximum reheat with very similar results. Thus during the reheat light up the only major excursion in the fan's working point has been into the "safe" area away from surge.

6.2 Optimisation

Because the engine-reheat combination used was a "one off", the optimisation system was only tested (for feasibility of operation) at one condition - sea level static.

A typical result from the optimisation tests is shown in Figure 13. The minimum fuel ratio was limited at high fuel flows by the reheat fuel pump being unable to provide the required pressure. The maximum fuel ratio was stopped at 1 for the lower flows as it was of no interest beyond this point. The data for each constant total flow line were gathered in approximately two minutes.

The top four total flow curves show a definite optimum value even if marginal at the highest condition. The lower four curves show a trend towards a maximum at low fuel ratios (increasing gutter flow) - signifying that the gutter will not burn efficiently below a given fuel flow.

The dotted line is the chosen trajectory for the control of reheat in the modulation range. The selection of this trajectory was a compromise between the shape of the fuel/nozzle area relationship, the required fuel valve operating rates and the minimum colander fuel flow. This is discussed in detail in Reference 1 Section 4.

7. CONCLUSIONS

1. During reheat modulation control, the closing of a fast working line trim around an open loop system has been demonstrated to give tight control of the engine working line, thus simplifying the open loop scheduling requirements.

2. The ability to "buzz ride" should ease the constraints on the maximum fuel flow which can be utilised.

3. It has been shown that the use of a digital computer to process the control algorithms enables the optimum performance of the power plant to be quickly determined. The usefulness of such a technique during early development work has been demonstrated.

8. REFERENCES

1. Z. M. Jawor, M. J. Porter, Engine evaluation of a microprocessor based reheat control on a Spey/DP20 Engine, September 1981, NGTE 81024

2. J. H. Waters, A digital controller applied to the limitation of reheat combustion roughness, AGARD-CP-151, September 1974

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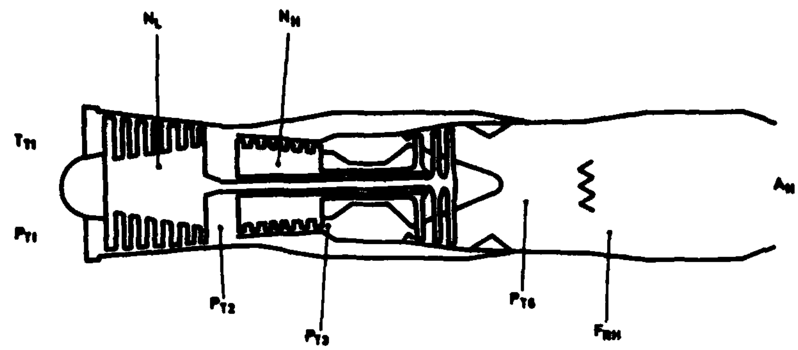

FIG.1 BYPASS MILITARY ENGINE

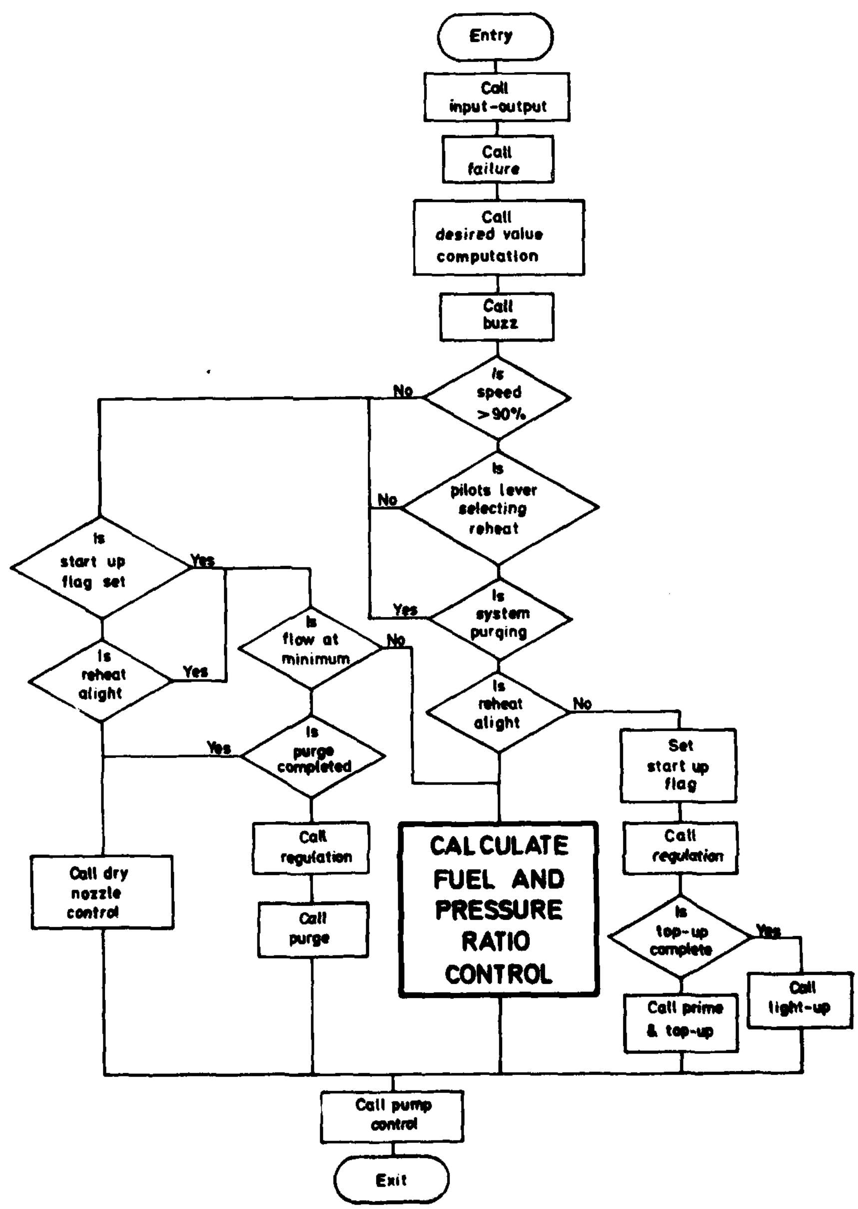

FIG.2 FLOW DIAGRAM OF COMPLETE REHEAT CONTROL

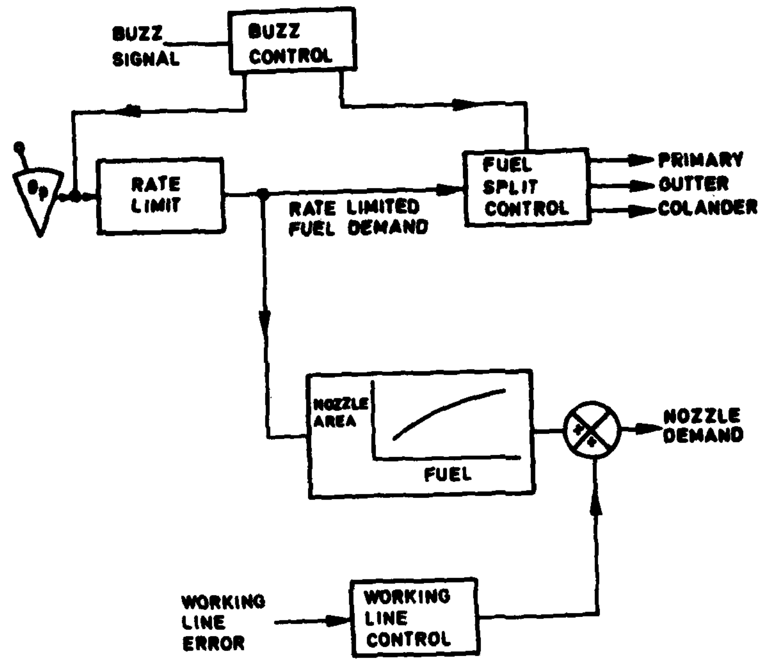

FIG.3 OVERALL BLOCK DIAGRAM OF REHEAT CONTROL SCHEME

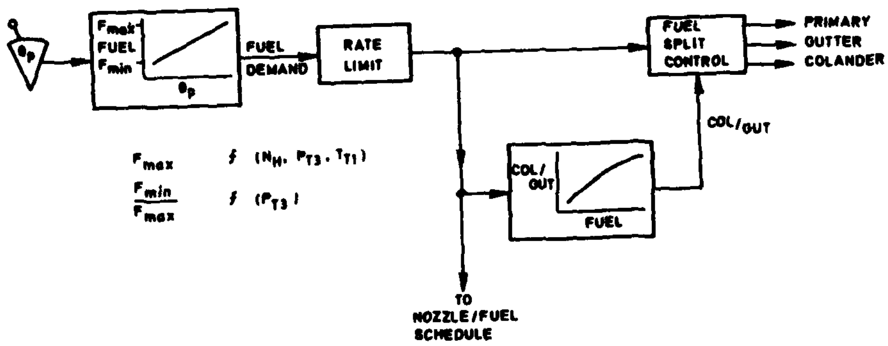

FIG.4 FUEL CONTROL

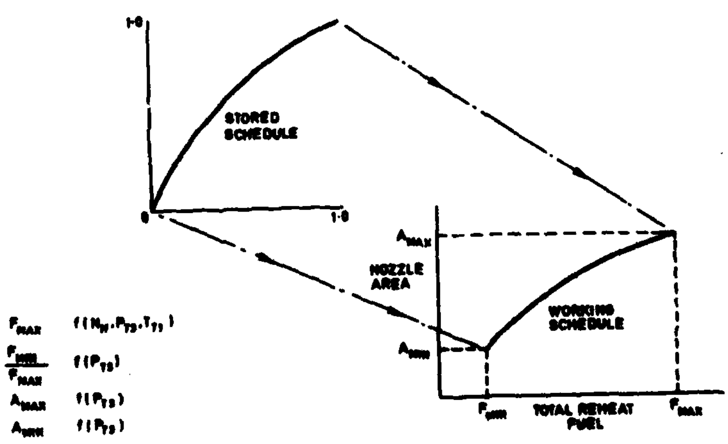

FIG.5 DERIVATION OF THE WORKING NOZZLE/FUEL SYSTEM

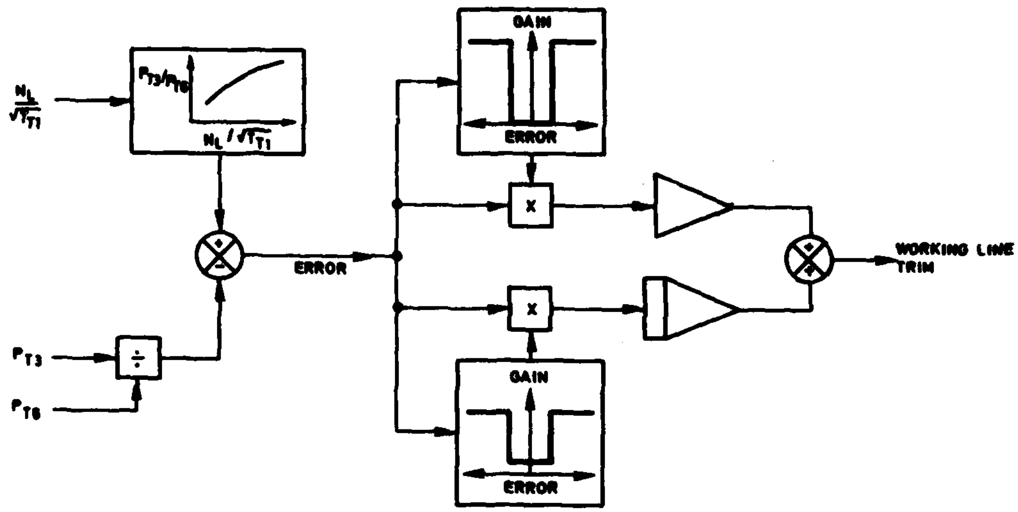

FIG.6 BLOCK DIAGRAM OF WORKING LINE TRIM CONTROL

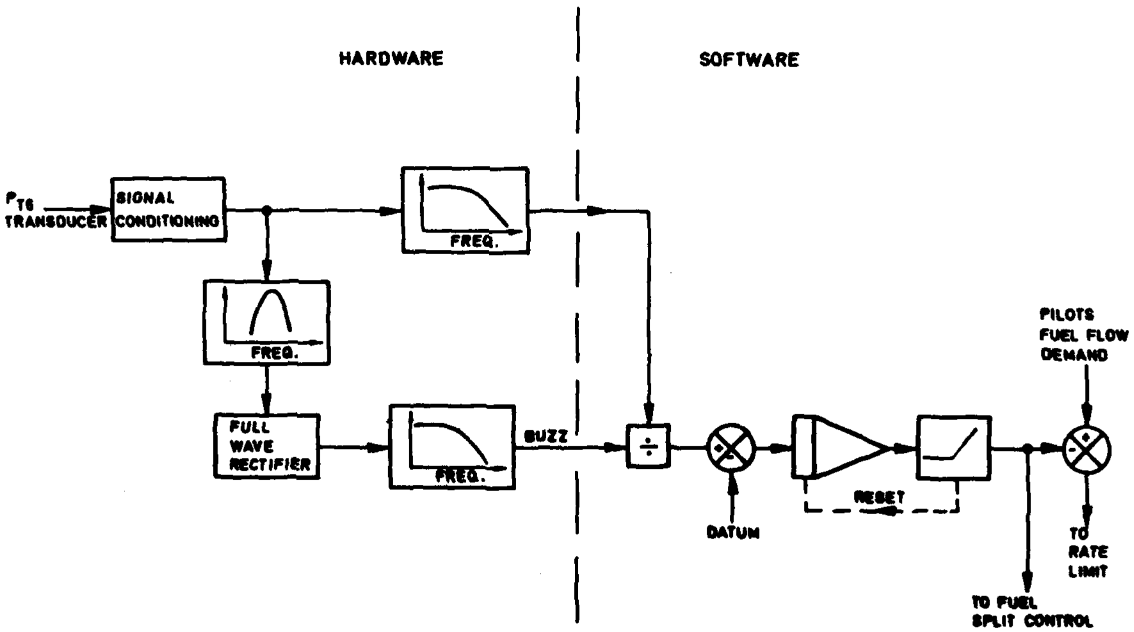

FIG.7 BLOCK DIAGRAM OF THE BUZZ CONTROLLER

FIG.8 SPEY/DP20 REHEAT MANIFOLDS

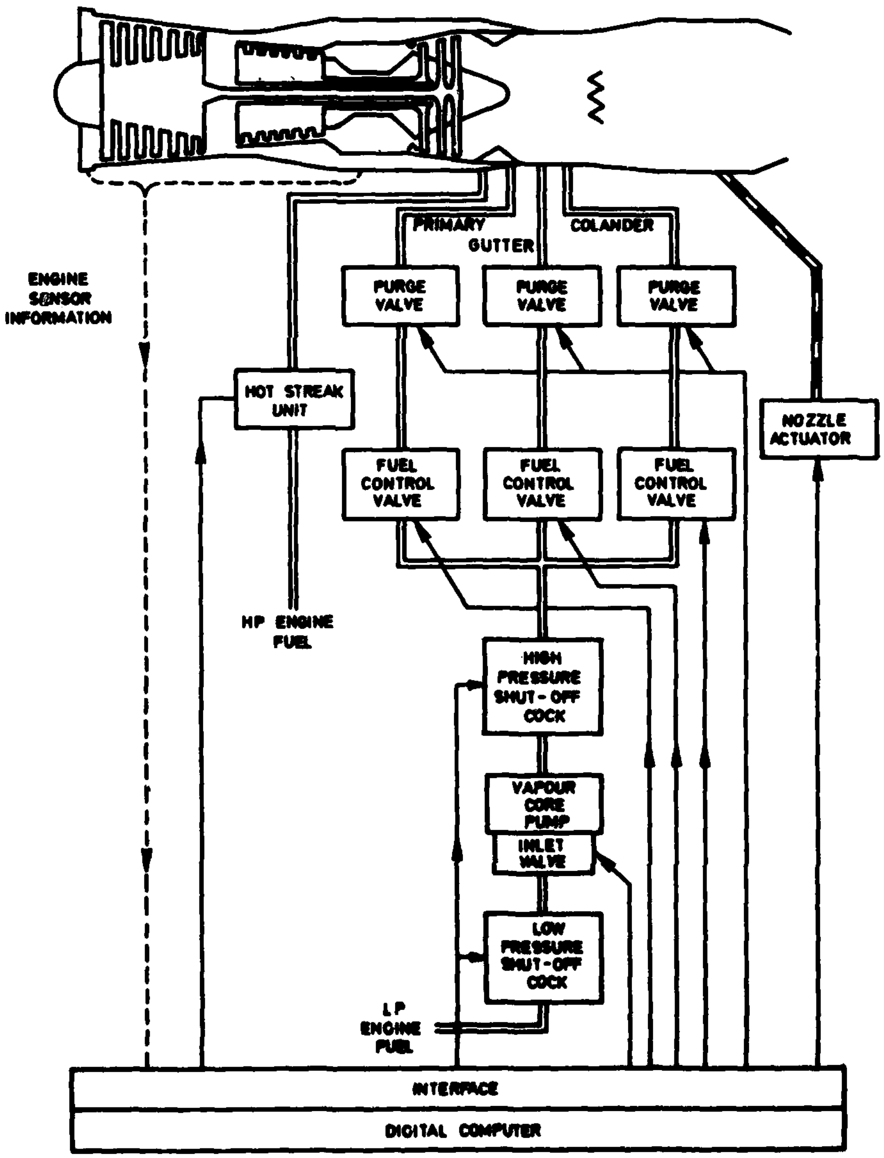

FIG.9 BLOCK DIAGRAM OF REHEAT CONTROL HARDWARE

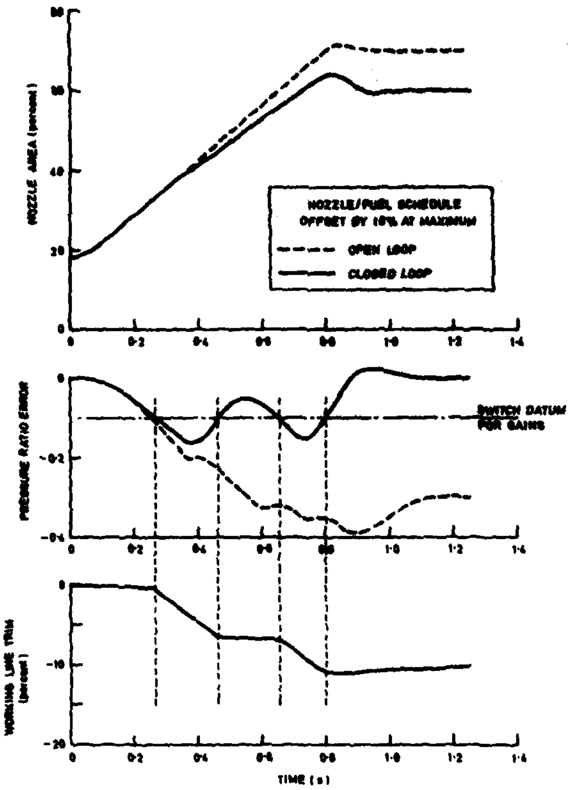

FIG.10 EFFECT OF WORKING LINE TRIM DURING A TRANSIENT

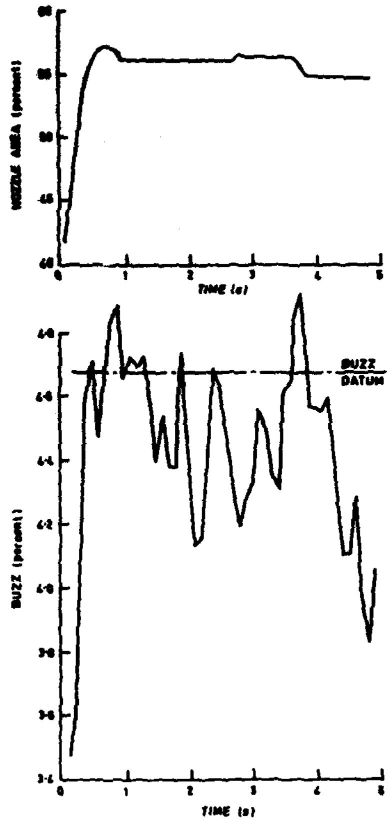

FIG.11 A TRANSIENT CONTROL USING A DEPRESSED BUZZ DATUM

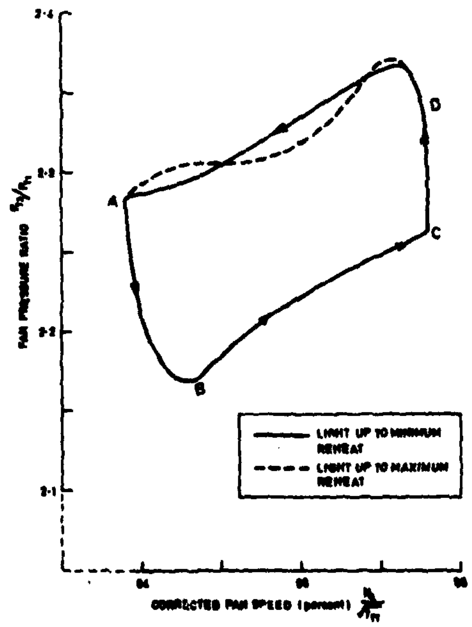

FIG.12 REHEAT LIGHT-UP SEQUENCE

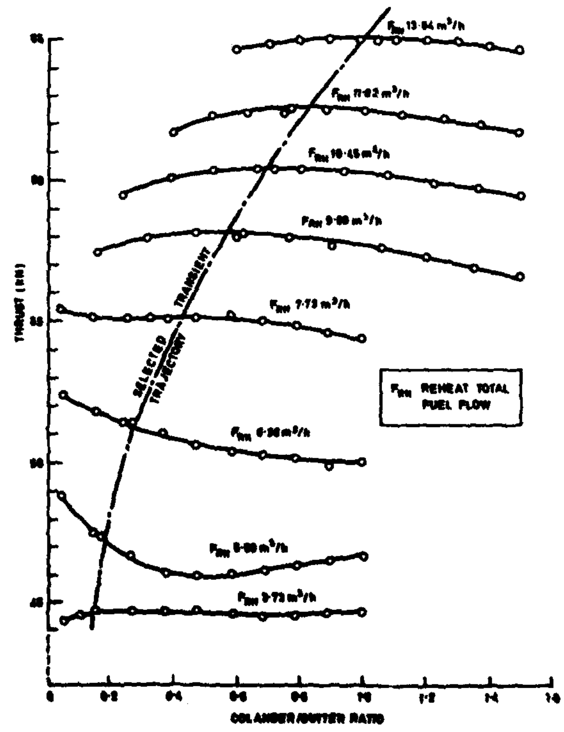

FIG.13 OPTMISATION RESULTS AT A FIXED EDNA

DISCUSSION

K. Bauerfeind, Ge:

With your proposal of a pure closed loop control during very fast reheat transients you are obviously attempting to exercise a tight control on the fan running point during transients. It is not surprising that both your simulation and the engine test on a Spey have produced positive results since you have only offset an originally perfect match between reheat fuel flow and nozzle area (at one particular flight point i.e. SLS) by a proportionally increasing amount with increasing degree of reheat. This is the easiest possible ride you can define for your working line trim. This trim will, however, fail to cope transiently with the more severe cases where it has to suddenly change from say a −10% to a +10% nozzle area correction within less than 0.1 s for example. These more severe cases are realistic because of:

- effect of different flight conditions relative to a perfectly matched SLS case
- tolerance changes in the fueling over modulation range
- reheat efficiency changes relative to datum over modulation range
- irregularly delayed reheat light-ups.

Author's Reply:

The effect of different flight conditions and reheat efficiency changes relative to datum must surely be similar to the example demonstrated in the paper, i.e. effective changes in the general slope of the nozzle/fuel schedule.

The more severe case of fast changes in the effective slope of the nozzle/fuel characteristic due to fuel valve tolerances would not occur with this type of control, because of the envisaged design of the hydromechanical fuel control system.

The problem of irregular delayed reheat light up (and also irregular burning during modulation), both occurring at altitude conditions, are more difficult to answer as I have had no personal experience at controlling reheat at these conditions. However I would like to mention two possible ways in which this system may improve matters over a current open loop system.

i The changing of the colander/gutter fuel split ratio over the modulation range provides much smaller colander flows at light up, which should improve the light up performance and incidentally provide a smaller increase in thrust when reheat lights.

ii Holding a correct working line, i.e. preventing jet pipe pressure from falling, will aid the combustion process thus helping to counteract spurious burning characteristics.

If these problems or similar ones still exist, the flexibility of digital control can be used so that the working line control is adapted so that it always responds to positive errors (preventing surge), but some negative errors (due to poor burning) can be ignored during certain phases of operation, i.e. after light up or at the beginning of some transients.

POTENTIAL BENEFITS OF A FULL AUTHORITY DIGITAL ELECTRONIC CONTROL (FADEC) INSTALLED ON A TF30 ENGINE IN AN F14 FLIGHT TEST DEMONSTRATOR AIRCRAFT

by
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SUMMARY

Grumman completed a systems analysis, preliminary installation design, and flight test definition for an integrated F-14/TF30/FADEC demonstrator aircraft flight test configuration under contract with the Naval Air Propulsion Center, (NAPC). Pratt & Whitney Aircraft (P&WA) was awarded a simultaneous contract to define the modifications, tests, and installation support to incorporate FADEC's on the TF30. An overall plan for the F14 design and modifications, procurement, fabrication, ground test and flight test program was developed. The proposed program includes a 22 flight development demonstration of F-14/TF30/FADEC fuel controls technology and inlet control on an F14 aircraft. The major P&WA tasks that support the program include design and development of the FADEC on the TF30, and the planning and performance of systems, sea level and altitude chamber tests. The study efforts identified many potential areas of F14 and TF30 generic or general aircraft and propulsion system benefits which could be accrued through the use of integrated aircraft/engine FADEC technology. These include improved air inlet-engine compatibility, improved propulsion system reliability and maintainability, increased loiter time, mission fuel savings, reduction in weight, and incorporation of some F14 propulsion system independent features into FADEC. This paper reviews these benefits in detail.

NOMENCLATURE

A_C	Inlet Capture Area
A/B	After/Burner
AFM	Advanced Fuel Management
AICS	Air Inlet Control System
α	Angle-of-attack
APC	Approach Power Compensator
β	Angle-of-Sideslip; Beta Angle
BIT	Built-in-Test
C_D	Coefficient of External Drag
CADC	Central Air Data Computer
CSDC	Computer Signal Data Converter
DLC	Direct Lift Control
EPR	Engine Pressure Ratio
FADEC	Full Authority Digital Electronic Control
FMEA	Failure Mode and Effect Analysis
H/W	Inlet Aspect Ratio
IMN	Indicated Mach Number
IRT	Intermediate Rated Thrust
M	Maintainability
Mo	Mach Number, Free Air Stream
MCB	Mid-Compression Bypass
MTBF	Mean Time Between Failures
Ps	Static Pressure Measurement
P_T	Total Impact Pressure Measurement
Ps_e/Ps_O	Exit Static Pressure Ratio
P_{T2}/P_{TO}	Inlet Total Pressure Recovery Ratio
P&WA	Pratt & Whitney Aircraft
PLA	Power Lever Angle
Q_C	Dynamic Pressure
R_s	Specific Range
SFC	Specific Fuel Consumption
W_C	Corrected Weight Flow
WRA	Weapon Replaceable Assembly

1. INTRODUCTION

The U. S. Navy is funding propulsion technology studies, ground tests, and flight tests directed toward development and refinement of highly reliable digital electronic engine control systems. The overall objectives of these efforts include the development of integrated engine, inlet, and aircraft flight controls for future military aircraft. A FADEC program was initiated in 1976 toward achieving these objectives, under the sponsorship of the Naval Air Systems Command (NAVAIR) and directed by the Naval Air Propulsion Center (NAPC). This resulted in the design, fabrication, and testing of engine-mounted, flight-type, FADEC's for advanced military aircraft gas turbine engines. They were designed to satisfy a most demanding set of engine control variables. Incorporation of these advanced digital electronic controls and the resultant reduction in the use of complex hydromechanical hardware are projected to result in more than a 43% reduction in acquisition cost, a 25% reduction in weight, and a 130% improvement in piece part reliability on an overall control system basis relative to a system configured with current production technology. Other FADEC payoffs include optimization of engine performance, improved propulsion system stability, and increased capability for aircraft control system integration.

A demonstrator engine of the F100 family was tested with a FADEC in complete control of all engine functions. The success of nearly 2000 hours of similar demonstration engine tests have proved that the FADEC concept is functionally suitable and that the electronics could be designed to safely accommodate system faults. Confidence in digital control equipment is also supported by over 800,000 hours of operational experience accumulated in F-15 and F-16 aircraft with a digital electronic supervisory control on F100 production engines. With this background of experience at P&WA, coupled with the experience of other engine manufacturers, the propulsion community is now in a good position to exploit the digital computer to benefit all new aircraft designs. See References 1 and 2.

The Naval Air Propulsion Center as a continuing part of this effort, identified the F-14 powered by two PWA TF30 turbofan engines as a potential flight vehicle to demonstrate aircraft integrated control technology using a dual fault tolerant FADEC configuration interfaced to an Advanced Fuel Management system (AFM). This led to the award of two study contracts by NAPC: one to Grumman in May 1981 and one to P&WA in September 1981.

The major tasks performed by Grumman as part of this study contract were:

- Preliminary evaluation of propulsion/flight control integration technology and other benefits of FADEC
- Baseline FADEC and aircraft system interface requirements definition
- Preliminary design of the F14 system to accommodate FADEC
- Preparation of a plan and cost estimate for final design, manufacture, test, installation and checkout of equipment in an F-14 test aircraft
- Preparation of flight test plans and cost estimates.

The results of these studies culminated in a proposed flight test program to be conducted on an F14 at the Grumman flight test facility at Calverton, New York. Key tasks milestones and the schedule leading to and including the proposed 22 flight program are shown in Fig. 1. The recommended F14/TF30/FADEC flight test configuration is shown schematically in Fig. 2. It retains the present F14 propulsion system on the right side

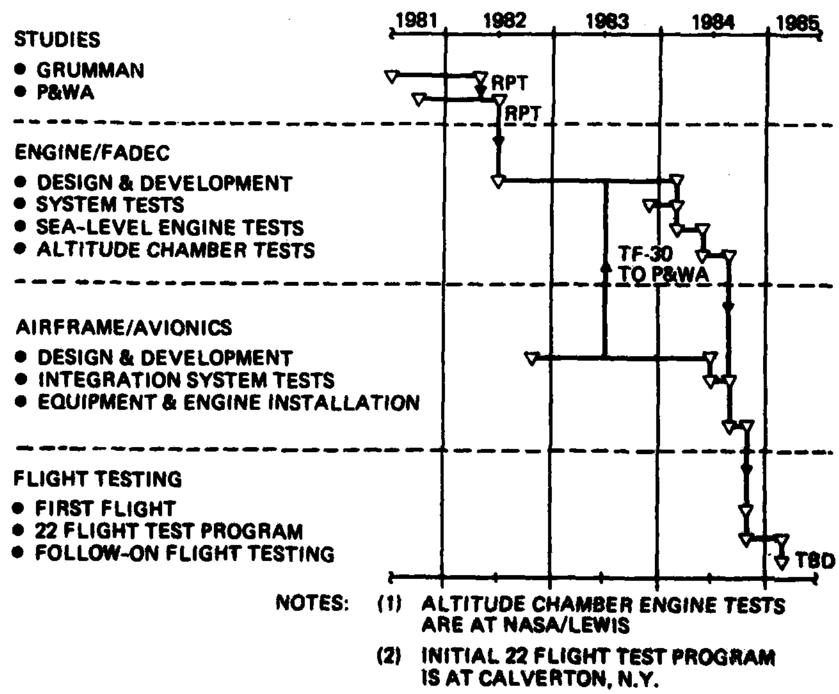

Fig. 1 F-14/TF-30/FADEC Proposed Development Program Schedule

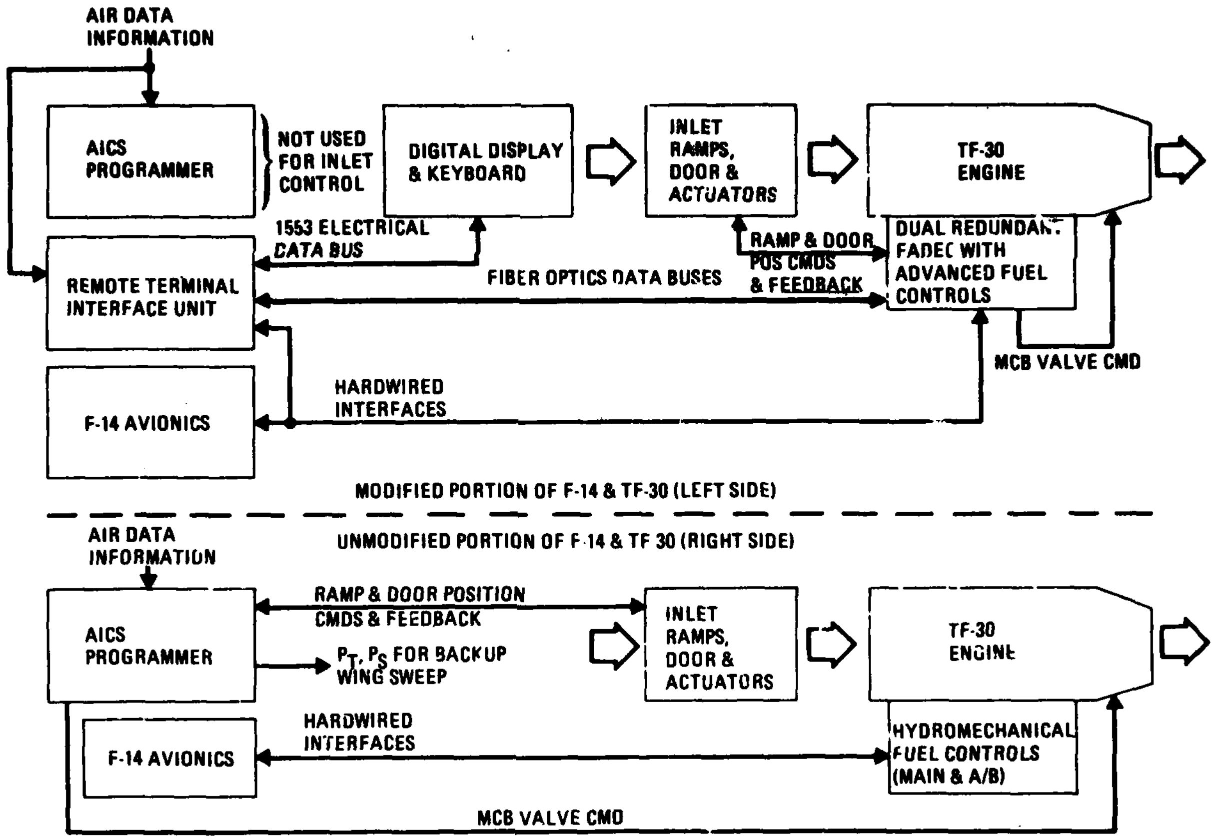

Fig. 2 Recommended F-14/TF-30/FADEC Flight Test Configuration – Overview

of the aircraft and incorporates the TF30 FADEC system and other interface and display electronics changes on the left side of the aircraft. This flight test configuration has the following features:

- Maintains left/right propulsion system independency
- Includes redundant FADEC processors
- Incorporates fiber optics data bus (1553 format)
- Incorporates electrical data bus (1553 type)
- Uses digital display I/O for engine control
- Integrates air inlet/engine controls (left side only)
- Incorporates MCB function into FADEC
- Incorporates APC into FADEC
- Incorporates Mach lever function into FADEC

This paper reviews in detail the benefits of a dual FADEC channel fault tolerant control configuration integrated with advanced fuel management of a TF 30 engine and discusses those benefits that can be demonstrated on an F14 flight test aircraft.

2. F-14/TF30/FADEC INTERACTIONS

A technical review was conducted to determine the beneficial and undesirable interactions that can exist between the TF-30 FADEC controlled propulsion system and the F-14 vehicle and its flight controls. The objective of the review was to identify those beneficial couplings or interactions that could be exploited to demonstrate improved aircraft/propulsion performance and compatibility and those undesirable effects that should be suppressed or negated.

Table 1 summarizes possible interactions noting their applicability to the F-14 FADEC control configuration with pertinent comments. The first four, A through D, were investigated further to determine if technology benefits could be obtained and demonstrated via flight testing. Effect E required no further study and is included primarily for completeness. Further investigation and discussions with F-14 aerodynamics personnel revealed that the present F-14 wing sweep schedule is "fuel optimum with Mach number" and

Table 1 F-14/TF-30/FADEC Interactions

COUPLING OR INTERACTIVE INFLUENCE	APPLICABLE TO F-14	FURTHER STUDY REQD	COMMENT
A WING SWEEP/GLOVE VANE AERODYNAMICS. L/D EFFECT COMBINED WITH ENGINE THRUST (T) PERFORMANCE TO YIELD OPTIMUM R_s	YES	NO	WING SWEEP ANGLE SCHEDULE IS FUEL OPTIMUM AS CONSTRAINED BY VEHICLE STRUCTURAL LIMITATIONS. PART-TIME CONSTRAINT RELAXATION FOR CRUISE MAY YIELD SAVINGS.
B AIRCRAFT AERODYNAMICS, L/D ENGINE EPR-CONTROLLED THRUST, NOZZLE AREA & ALLOWABLE AIRCRAFT/INLET ATTITUDE ENVELOPE EFFECTS TO YIELD OPTIMUM R_s	YES	YES	ENGINE FUEL CONTROL SCHEDULES ARE FUEL OPTIMUM WITHIN THE CONSTRAINTS OF STALL & DISTORTION MARGINS FOR THE FULL A/C ENVELOPE. PART-TIME CONSTRAINT RELAXATION FOR CRUISE CONDITIONS AS WELL AS NOZZLE AREA MODULATION WOULD YIELD FUEL SAVINGS.
C THRUST VECTOR GENERATED MOMENTS ABOUT THE A/C cg ($\underline{M} = \underline{F} \times \underline{\ell}\,cg$)	YES	YES	ASYMMETRICAL THRUST DUE TO LOSS OF ONE ENGINE'S THRUST CAUSES A SIGNIFICANT FLIGHT CONTROL SYSTEM DISTURBANCE, WHICH IS OF PARTICULAR CONCERN AT HIGHER ANGLES-OF-ATTACK (α).
D INLET/ENGINE COMPATABILITY, ENGINE AIR MASS FLOW, ENGINE STALL & DISTORTION MARGINS, & RELATIVE AIR MASS VELOCITY VECTOR ($\alpha, \dot{\alpha}, \beta, \dot{\beta}$) DYNAMICS	YES	YES	MAJOR THRUST OF THE FADEC CONTROLS INTEGRATION PROGRAM IS TO EXPLORE BENEFICIAL TECHNOLOGY GAINS IN ENGINE PERFORMANCE, AIRCRAFT OPERATING ENVELOPE & ENGINE STALL/DISTORTION MARGINS.
E WING SWEEP/GLOVE VANE AERODYNAMICS EFFECT ON ENGINE AIR MASS FLOW DISTORTION	YES	NO	NEGLIGIBLE EFFECT OF THE WING SWEEP SYSTEM ON THE F-14 AIR INLET AERODYNAMICS

effect A was dropped from further consideration. Effect B requires the use of multiple engine control schedules, available through FADEC, which could be selectable through a "CRUISE" or "MANEUVER" function selection. The "CRUISE" function selection picks out the necessary set of engine thrust schedules from a multiple set corresponding to reduced stall margin (EPR controlled) cruise operation. There is a possible fuel savings of 1% through this application of FADEC.

Effect C suggested the possibility of suppressing the adverse yaw effects resulting from the asymetrical thrust condition existing during the loss of engine thrust due to a stall or a complete engine failure. A FADEC output signal might be provided to the F-14 yaw flight control system to assist the pilot in defending against a possible aircraft departure and spin entry condition. This concept was partially explored during the study and requires further definition of the unreliability aspects of an engine stall logic signal interface with the aircraft flight controls system prior to any possible implementation.

Effect D is the baseline technology objective of the F-14 controls integration effort. The engine air mass flow distortion and the resultant inlet/engine system stall margins are directly effected by the inlet geometry and/or the relative air mass velocity vector, aircraft angle-of-attack, (α), aircraft angle of sideslip, (β), and engine airflow requirements. Modified inlet ramp and bleed door schedules, using all available engine and inlet data, should provide the ability through FADEC to significantly increase overall inlet/engine system stall margins and provide the ability to operate the ramps and bleed door to increase propulsion system performance. FADEC control of the engine also promises the possibility of reducing or eliminating the engine stall sensitivity to throttle power level angle (PL) transient inputs.

2.1 F-14 Inlet-Engine Compatibility

The requirements and feasibility of providing improved inlet performance and inlet-engine stability, throughout the F-14 flight envelope using FADEC were reviewed. To do this, a partial assessment of the F-14 inlet performance and inlet-engine stability that presently exists was made, and the appropriate levels quantified. The potential for improvement in performance and stability was then identified for the various speed ranges of the aircraft. Methods for providing this improvement by rescheduling variable inlet geometry and engine corrected airflow demand, as well as through the scheduling of new functions within the engine and inlet were studied. After review of these methods at Grumman and PWA, several specific approaches were identified for development, using the FADEC processors as the central integrators.

To provide adequate background, a summary of the inlet design is presented below.

2.1.1 Inlet Description

The F-14 inlet was designed to provide adequate amounts of sufficiently high quality air to ensure good performance and stall-free operation of the TF-30 engine. It was concurrently developed to efficiently accommodate advanced turbofan engines with higher airflow. (See Reference 3.)

The inlet has a two-dimensional, four-shock external compression system with essentially horizontally oriented ramps. It is rolled 10 deg out of the horizontal plane (Fig. 3) and has a 3 deg initial fixed ramp and three computer-controlled variable compression ramps, which are programmed to vary with Mach number only. Any airflow variation or angle-of-attack/angle-of-sideslip effects at any particular Mach number fall within the stable operating limits of the basic inlet ramp positions. The inlet aspect ratio, H/W, is 1.35 and the duct is four engine diameters from cowl lip to engine compressor face. The centerline is essentially straight. The compression ramp boundary layer is bled through a throat bleed slot and dumped overboard through the bleed/bypass door. This door was variable during the flight test program but is fixed in a compromized position for production aircraft. FADEC will permit the optimization of the door exit area.

The inlet was designed to meet the α/β shown in Fig. 4 based upon early wind tunnel tests. The α levels were, in 1968, as high as or higher than those obtainable by any then-current fighter aircraft. The supersonic high-α limits reflect control surface effectiveness at high altitudes and limiting g loads, whichever is most constraining at any Mach number. The supersonic low-α limits reflect a design goal of 1g above Mach 2.00 to preclude the requirement to vary inlet geometry with α. Extreme β angles during subsonic flight were anticipated to be quite large below M = 0.60. After early flight development the F14 was successfully flown at much higher α's, β's (Fig. 5).

2.1.2 Inlet Control System

The primary function of the air inlet control system (AICS) is to position the three variable ramps in each inlet to some preset schedule of angles, which varies the ramp position as a function of flight Mach number only. The resultant diffuser wall angle, throat area, and scoop height are shown in Fig. 6. Each inlet is independently controlled *by its own* AICS. The required positions are derived from an averaged calibration using both the left side and right side pitot-static probe data.

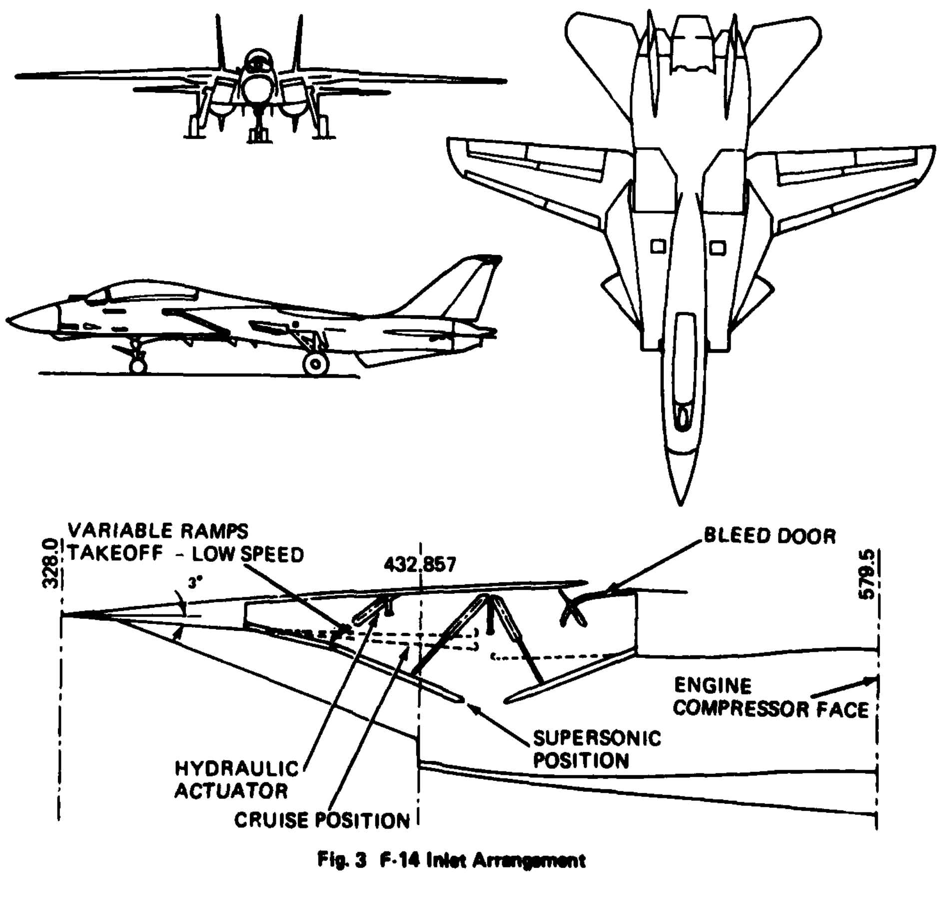

Fig. 3 F-14 Inlet Arrangement

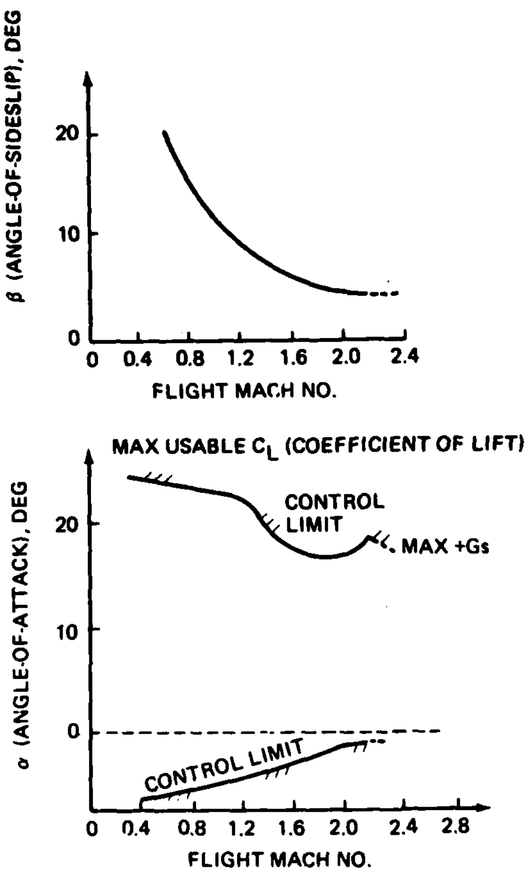

Fig. 4 F-14A Inlet Design Limits – Alpha & Beta

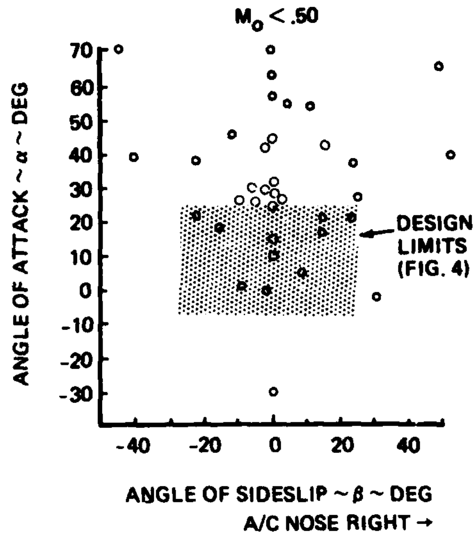

Fig. 5 Low Speed Maneuvering Flight Test Points

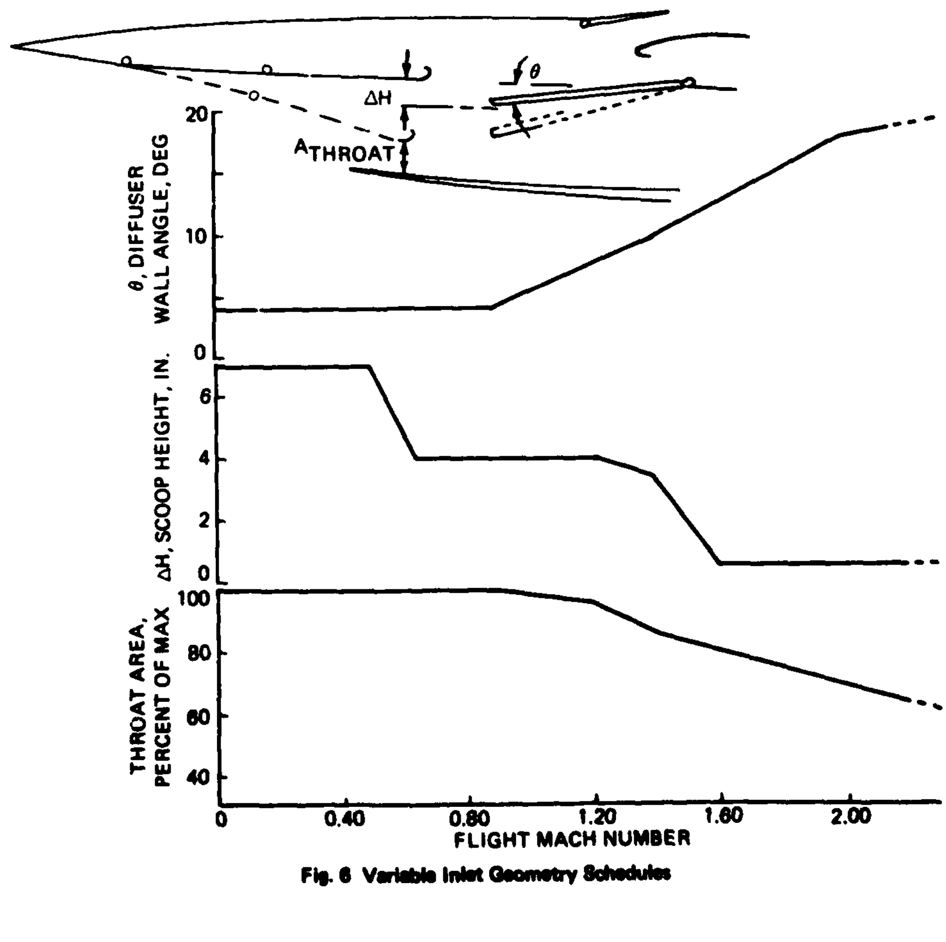

Fig. 6 Variable Inlet Geometry Schedules

2.1.3 FADEC/Inlet Benefits

The external drag of the F-14 can be reduced in the supersonic flight regime by high flowing the engine, perhaps for only short periods, during accelerations and combat. In addition to the greater thrust this would provide, the external inlet drag could be reduced by repositioning the ramps to provide a larger throat area. The various ramp positions shown in Fig. 7 for $M_0 = 2.0$ provide an almost constant level of total pressure recovery while allowing a continuous range of mass flow ratio and a unique variation of external drag. The minimum drag level, at a mass flow ratio of unity was measured with a normal shock version of the inlet model. The penalty in drag to provide the efficient pressure recovery levels of the multiramp configurations is denoted as inlet drag (C_D). Obviously, the higher the mass flow ratio the lower the drag, and pressure recovery need not suffer. The table in Fig. 7 shows the sensitivity of F-14 external drag per pound per second of engine corrected weight flow, at several Mach numbers. Subsonically, there is no significant benefit to be had, because the slope of external drag coefficient versus mass flow ratio is very low. High-flow inlet-engine matching was discussed with P&WA and it was determined that the potential for an increase of approximately 5% for constant corrected fan turbine inlet temperature (FTIT/θ) does exist. This approach will be evaluated in the F-14/FADEC Flight Test program.

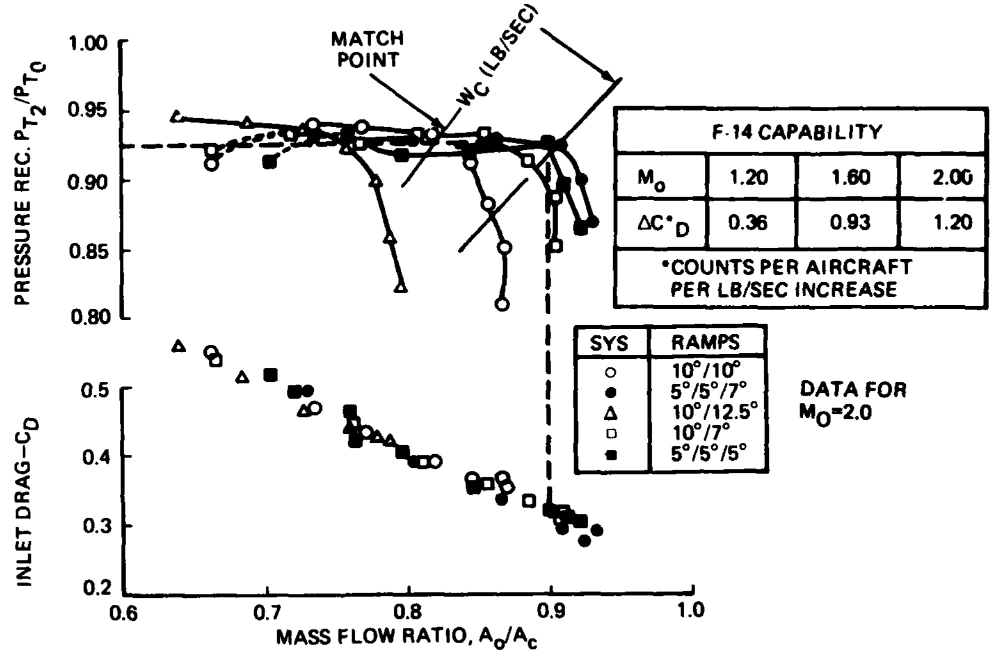

Fig. 7 Potential Drag Reduction Due to High-Flowing Engine

The F-14 inlet/engine compatibility was evaluated and documented during a flight test program conducted in 1978. Results of this program are shown in Figs. 8 and 9, in terms of the total distortion index including turbulence effects, K_{D_2}. The values at which engine stall would be expected for a "worst-case" engine are compared to the expected inlet capability for a "worst-case" inlet. The comparisions are for maximum or near-maximum engine corrected weight flow, with the MCB open.

The effects of engine afterburner transients, either lighting or retard from max A/B, significantly reduce high α operation at the higher altitudes, at subsonic conditions. At the lowest Mach number shown, $M_0 = 0.30$, the A/B lighting restriction precludes stable operation at α above 30 deg at 15,000 ft. and above 18 deg at 30,000 ft. At the higher Mach number, $M_0 = 0.60$, the inlet-A/B lighting compatibility is somewhat better. In addition to discussing with P&WA means for improving the engine distortion tolerance, there were several approaches identified that can be taken to reduce the inlet distortion generated at high α.

The MCB is open above 18 deg α, subsonically, to ensure that the engine has a relatively large stall margin for high α operation Tailoring the opening of MCB to meet the increased inlet distortion occurring at high α's would allow more thrust to be available for maneuvering flight. Approximately 6% could be available with the MCB closed. This item was discussed with P&WA and will be included in the F-14/TF30 FADEC system development.

Inlet instabilities (buzz) have been reported from time-to-time during fleet operations. The present inlet ramp schedule has been selected to eliminate any potential buzz situations. However, it is possible that if the ramp positioning and engine airflow control accuracy is relatively poor on a given day, then buzz may occur. Fig. 10 shows 1/7th scale model wind tunnel data that demonstrates the significant effect of ramp deployment on increasing the stable airflow range at $M_0 = 1.2$ & 1.6. Using FADEC the inlet control system will be restructured to provide a better variation of ramp and bleed exit door positions with engine corrected weight flow. This would allow a significant increase in

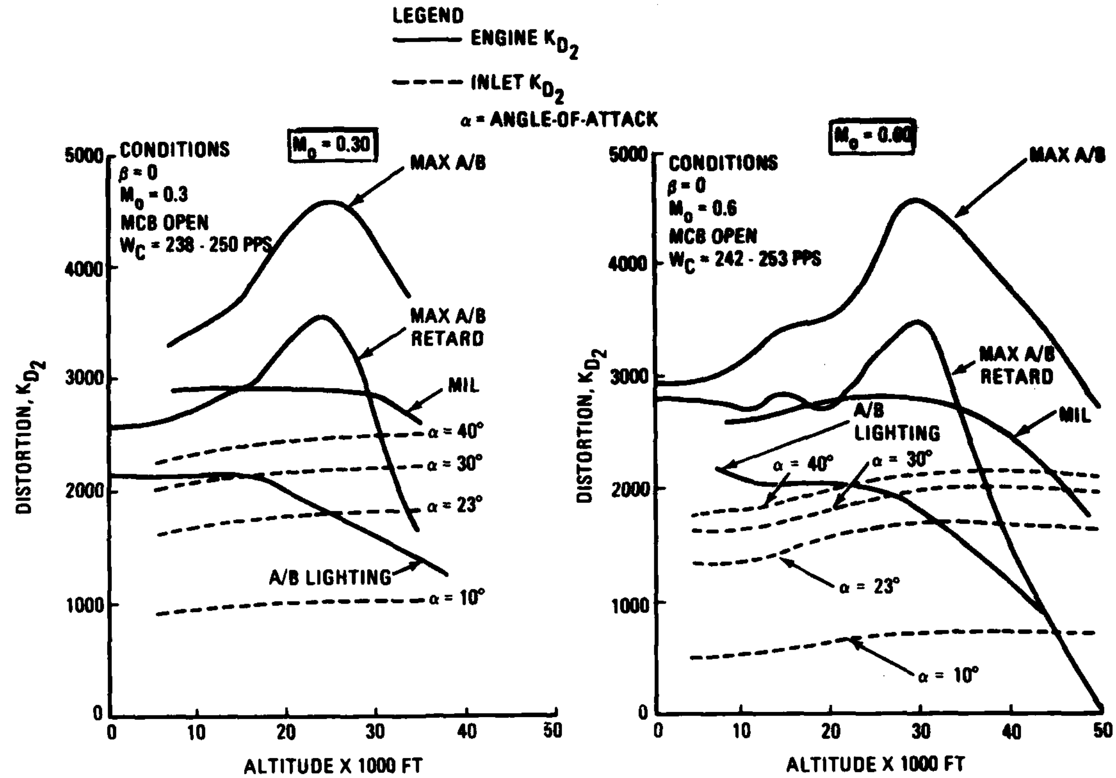

Fig. 8 F-14/TF-30 Inlet-Engine Compatibility – M_0 = 0.3 & 0.6

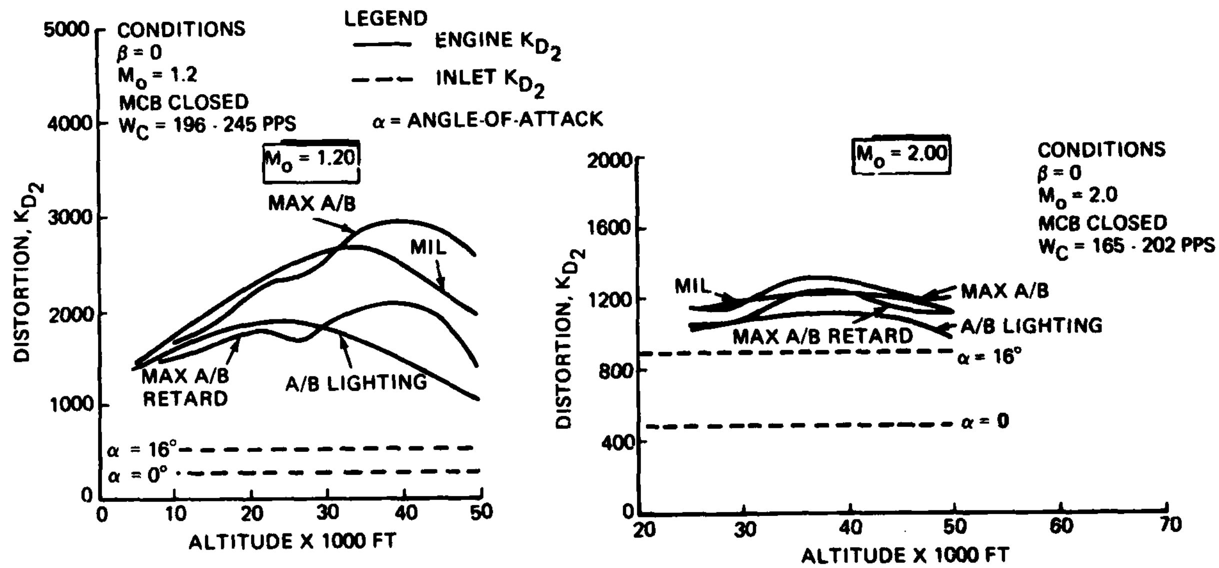

Fig. 9 F-14/TF-30 Inlet-Engine Compatibility – M_0 = 1.2 & 2.0

the stability corridor shown in Fig. 11, as well as almost eliminating any potential for inlet buzz. The FADEC system is an ideal means for implementing this approach and it will be demonstrated during the flight test program.

High speed flight above M_0 = 2.00 is characterized by relatively high turbulence, which is the result of excessively high levels of engine corrected weight flow. This is shown in Fig. 12 which compares the engine face turbulence measured during two accelerations by an F-14. This first acceleration resulted in an engine stall at a turbulence level of about 2.8%. The engine corrected weight flows during this acceleration were up to 8% greater than the inlet design values, and the inlet was probably operating supercritically. The drastic reduction in turbulence that accompanied the lower engine corrected weight flows on the second try, after adjusting the engine Mach lever control, allowed a stall-free and adequately rapid acceleration.

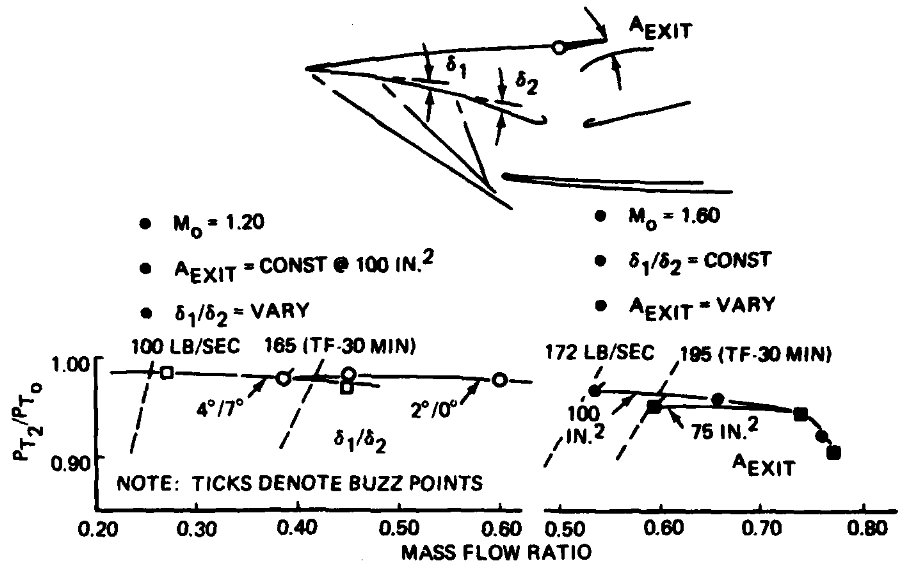

Fig. 10 Inlet Variable Geometry Effects On Buzz Margin at Level Flight Conditions

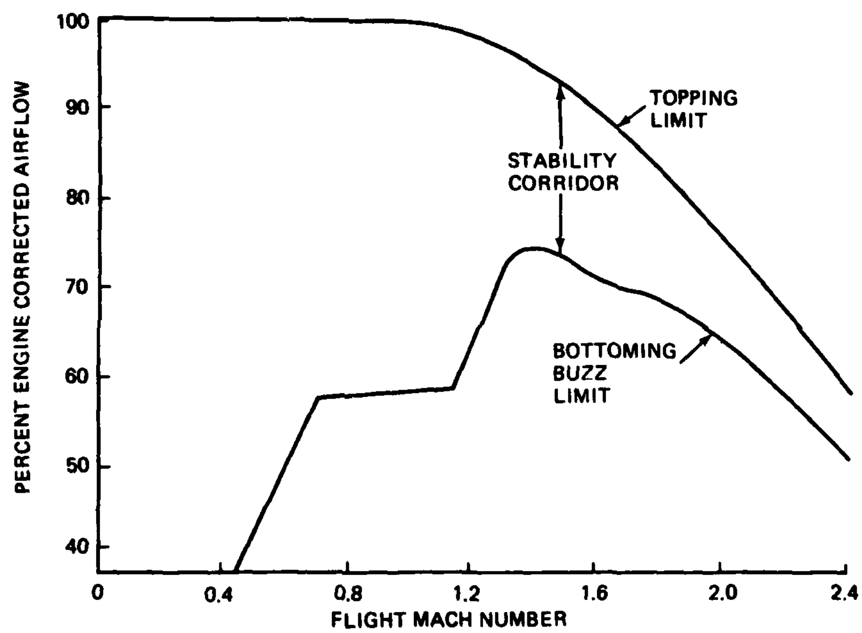

Fig. 11 F-14A Inlet Design Limits – Corrected Airflow

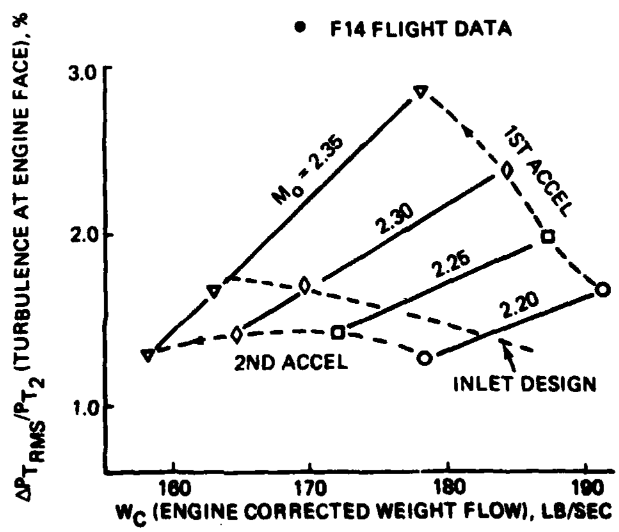

Fig. 12 Effect of Engine Corrected Weight Flow Scheduling on Supersonic Compatibility

Providing efficient inlet-engine matching for high supersonic flight as a means to ensure subcritical inlet operation using the FADEC system will be demonstrated during the proposed flight test program by including engine corrected airflow as one of the control parameters for determining the inlet ramp positions.

3. F14 FUNCTIONS FOR POTENTIAL INCORPORATION INTO FADEC

Table 2 lists the F14 functions that were considered for incorporation into the FADEC. The functions annotated with a "yes" were pursued further under this study. The approach power compensator (APC) with direct lift control (DLC) function was not incorporated into FADEC, but will operate in series. The results of the study concluded that the digitally integrated function of APC with DLC, Mach lever trim, and AICS as they presently exist would require approximately 2500 words of memory. The more sophisticated AICS functions envisioned for the F-14 FADEC program will require more memory but will not represent a problem in terms of FADEC capacity.

Table 2 Potential FADEC Functions

FUNCTION	RECMD FOR FLT TEST DEMO	PROS	CONS
APC WITH DLC	NO*	• ELIMINATES 1 WRA • MORE RELIABLE APC & DLC FUNCTIONS	• SLIGHT I/O, MEMORY & THRUPUT IMPACT • DEVELOPMENT & FLT TEST REQUIRED
FLIGHT CONTROLS	NO	• ELIMINATES MANY WRAs • MORE RELIABLE SYSTEM • MORE VERSATILE SYSTEM (DIGITAL)	• LARGE I/O, MEMORY & THRUPUT IMPACT • MAJOR DEVELOPMENT & FLT TEST PROGRAM REQUIRED
WING SWEEP/GLOVE VANE	NO	• INCREASED RELIABILITY OF THE WING SWEEP/GLOVE VANE FUNCTION	• DEVELOPMENT PROGRAM REQUIRED • NO A/C PERFORMANCE GAINS
N_1 OVERSPEED DETECTION	NO	• ELIMINATES 1 WRA • MORE RELIABLE OVER-SPEED INDICATION	• MINOR DEVELOPMENT PROGRAM REQUIRED
MACH LEVER TRIM	YES	• ELIMINATES 1 WRA • MORE RELIABLE MACH LEVER TRIM FUNCTION	• DEVELOPMENT & FLIGHT TEST PROGRAM REQUIRED • SLIGHT SOFTWARE & I/O IMPACT
THROTTLE CONTROL	YES	• FADEC, COST, WGT, RELIABIL-ITY & MAINTAINABILITY GAINS	• DEVELOPMENT & FLIGHT PROGRAMS REQUIRED
AICS	YES	• ELIMINATES 2 WRAs • MORE RELIABLE AICS • IMPROVED ENGINE STALL & DISTORTION MARGINS	• DEVELOPMENT & FLIGHT TEST PROGRAM REQUIRED • SOFTWARE & I/O IMPACT

NOTE: *FUNCTIONALLY, FADEC WILL BE IN SERIES WITH APC & DLC

Digitizing the F-14 flight controls function was estimated to require the following types of I/O functions for each of two redundant flight computers:

Input/Output Type	No. of I/O
Analog Inputs - AC	25
Analog Inputs - DC	16
Discrete Inputs	37
Analog Outputs	12
Discrete Outputs	46

The number of software memory words required to implement the F-14 flight control functions into quad redundant computers, similar to two FADECs per engine, is estimated at 12,000 to 15,000 words. Grumman and P&WA concluded that the flight controls requirements far exceeded the flight test FADEC processor capacity and it was agreed to drop incorporation of this feature or function from further consideration.

The wing sweep and glove vane control logic and algorithms are presently implemented in a digital computer (the CADC). FADEC could improve the reliability of the function. The wing sweep function was examined to determine if any engine interrelationships exist. The optimum wing sweep control is a function solely of Mach number and therefore no interactive benefit can be obtained through the use of FADEC. Therefore, inclusion of this function was dropped from further consideration.

The N_1 overspeed detection function is a straight forward task which does not have to be verified via flight test. The present independent overspeed caution capability was retained in the flight test configuration for crew safety considerations.

A review of the Mach lever trim set function and the associated FADEC interface trade issues was conducted and recommended FADEC interface configurations were derived. The following paragraphs discuss this function in detail.

3.1 Mach Lever Trim

The Mach lever trim set shown schematically (Fig. 13) provides a closed loop control of a Mach lever mechanical input to the engine main fuel control as a function of aircraft Mach number to control engine airflow within a topping limit, which prevents excessive distortion at maximum thrust, and a bottoming limit, which prevents engine buzz at idle settings. These limits are shown in Fig. 11. If the Mach lever fails in the 1.6 to 1.8 IMN range, a maximum reduction of 22% in the level of MIL thrust will result. To minimize the consequences of a failed Mach lever, the absence of a signal or zero voltage commands the Mach lever motor to the minimum Mach condition.

The FADEC flight test configuration will accomplish the Mach lever trim set functions listed in Table 3 for the left engine. The replacement of the Mach lever by FADEC eliminates parts, reduces costs and will increase reliability and reduce maintenance.

3.2 Throttle Control

A review of the F-14 throttle and APC set functions and the associated FADEC interface trade issues was conducted and a recommended FADEC interface configuration was derived. The following paragraphs discuss this function in detail.

The F-14 throttle control and APC provide three different modes of operation; AUTO, BOOST and MANUAL. The inputs, outputs and elements used for these modes are identified in Figs. 14 and 15.

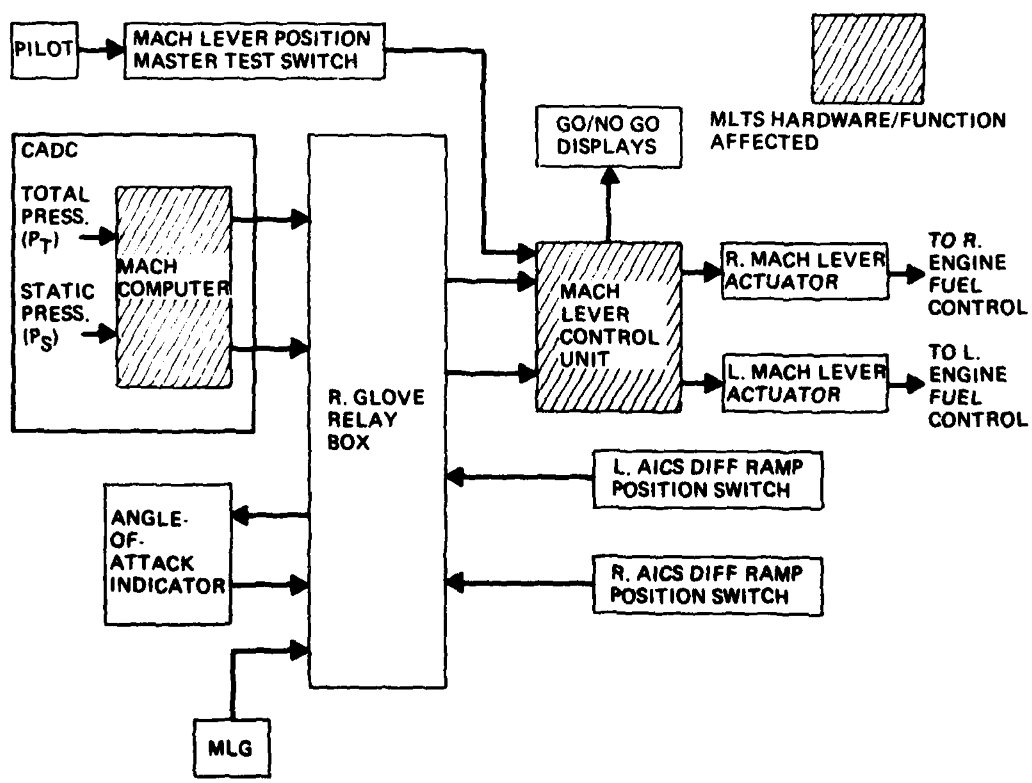

Fig. 13 Mach Lever Trim Set

Table 3 FADEC Mach Lever Trim Functions

FUNCTION	CONDITION	REQUIRED RESPONSE
1. CONTINUOUS SELF-TEST	PASSES	PROVIDE GO
2. COMMANDED BIT*	PASSES FAILS	CYCLE M.L. & PROVIDE GO PROVIDE NO GO
3. COMMAND MACH LEVER POSITION	MACH < 0.25 MLG DOWN MACH > 0.25 MACH < 0.9, MLG UP & AOA > 16 ± 1 COUNTS	POSITION M.L. AT 0° POSITION M.L. AT 0° POSITION M.L. PROPORTIONAL TO MACH POSITION M.L. AT 42° ± 2°
NOTES: ABOVE ASSUMES MACH LEVER ACTUATOR & THE MACH LEVER ARE BOTH REFERENCED TO 0° *INTERLOCKED WITH MAIN LANDING GEAR (MLG) FOR SAFETY		

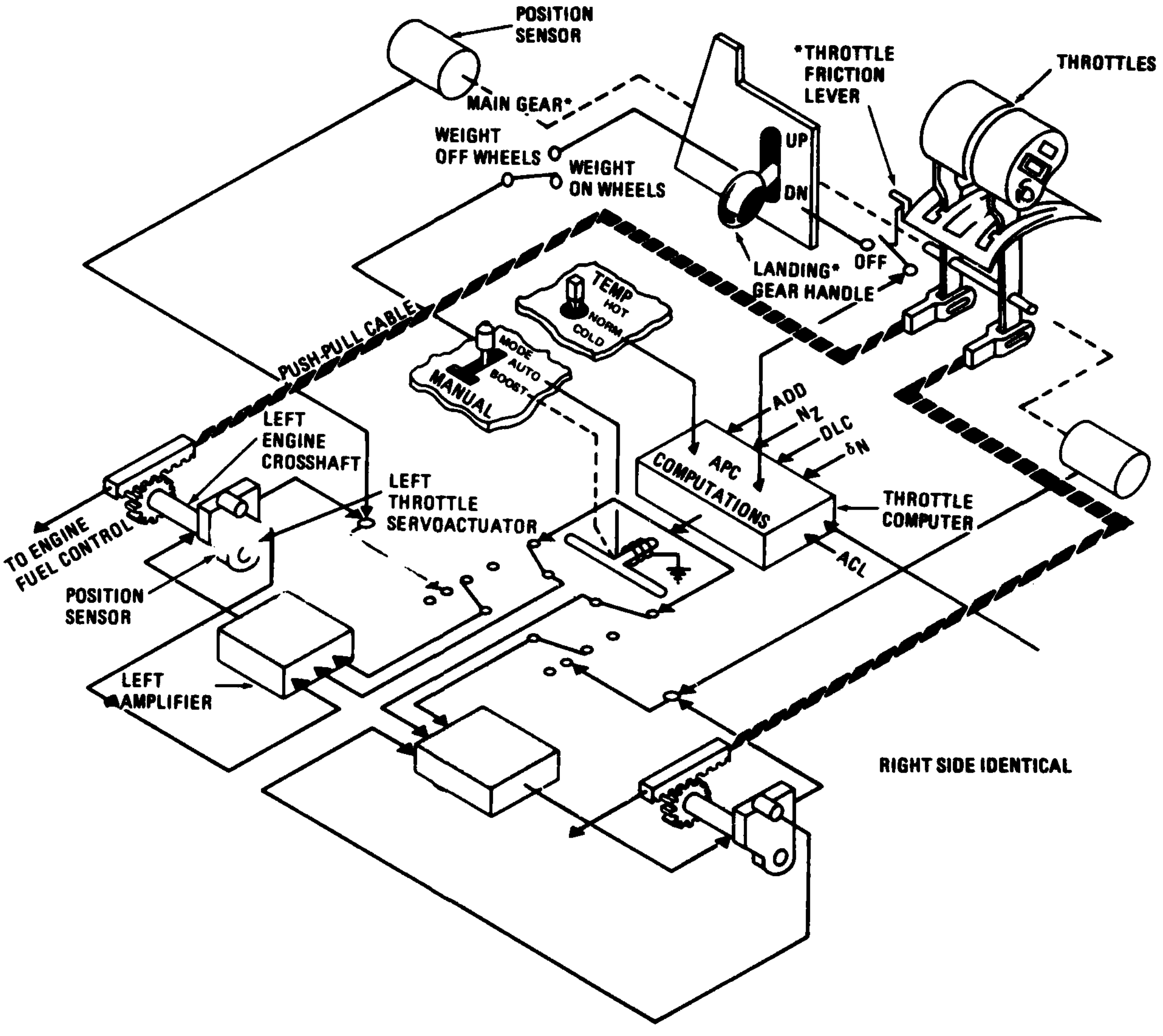

Fig. 14 Engine Throttle Controls

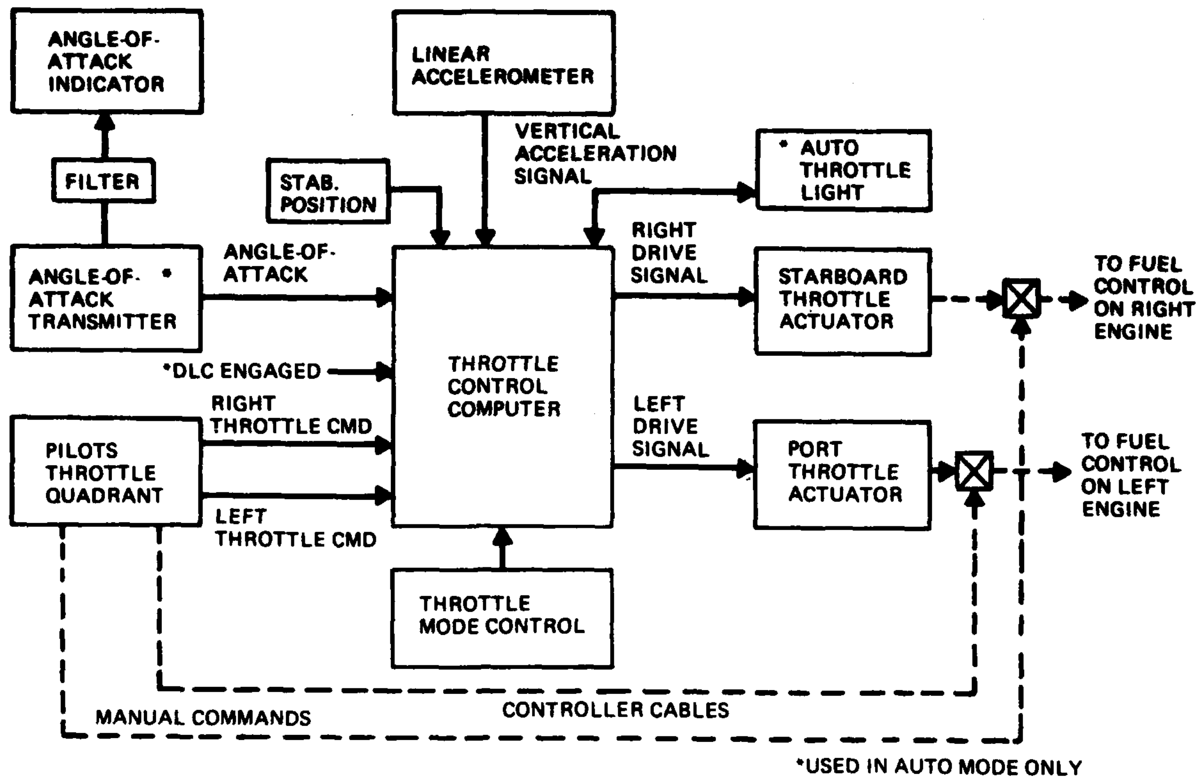

Fig. 15 Approach Power Compensator

The automatic mode (AUTO) of throttle control (APC) is a closed loop system that automatically regulates the basic engine thrust to maintain the aircraft at an optimum approach angle-of-attack for landing. In addition to pilot manual flight control, the F-14 auto throttle is compatible with the automatic carrier landing system (ACLS) and direct lift control (DLC) modes of flight path control.

Engine thrust is automatically regulated to maintain optimum angle-of-attack for landing by the throttle control computer portion of the APC set. The α signal from the α probe on the left side of the forward fuselage is the controlling parameter within the computer for providing responsive but stable system operation under normal and abrupt control inputs, for balanced and varied atmospheric temperature conditions.

Engagement of the auto throttle can not be accomplished with the throttles outside of the auto mode limits or until all controller interlocks are satisfied. Engagement of the auto throttle permits computer commands to be transmitted to the two control amplifiers (left and right engines) located in the throttle computer, where the error relative to the throttle actuator position feedback signals on the respective side is resolved. The resultant error in the amplifier comparator network produces a follow-up command to the throttle rotary actuator to null the error. Actuator authority is limited by internal switches and the rotation of the engine cross-shaft back-drives the throttle to auto throttle commands.

The throttle control computer controls both throttles during the AUTO mode of operation. In the AUTO mode inputs of α, data relative to direct lift control and horizontal stabilizer position, and vertical-axis acceleration rate are fed into the computer, which uses them to command the throttle fuel shaft position necessary to control α by thrust adjustment. In the boost mode, the computer receives throttle position synchro-resolver error signals from the throttle lever synchro-resolvers. These are referenced to the throttle actuator position transducers, and the error signals are used to drive the actuator to achieve the commanded setting.

If a rotary actuator malfunction imposes an excessive cross-shaft torque the mechanical clutch slips so that pilot control is overriding. The limit switches are set to close at 1 deg above IDLE and 1 deg below intermediate for the automatic mode of APC. If the throttles are not within the limit-switch range, the automatic mode cannot be implemented.

The push-pull cable assemblies are the mechanical links between the throttle control quadrant and engine. A three-section cable is used for each engine throttle control. The cable consists of three contoured stainless-steel tapes suspended between ball bearings and encased in flexible conduit. During MANUAL mode throttle control, translation of the tapes within the cable conduit transmits command loads to operate the throttle rack and sector gearbox on the engine cross-shaft. During the boost mode of operation, the cable is used as a follow-up to the throttle lever position. The cable is a backup control if both the AUTO and BOOST modes malfunction.

3.2.1 Interface Alternatives

Various alternatives were considered for interfacing the throttle quadrant power level angle (PLA) information with the *FADEC processors* to control the engine thrust level via fuel cross-shaft position. The study concluded that the configuration with a dual cross-shaft position transducer (synchro or resolver) feeding each FADEC processor is best. It retains the push-pull cables and the AUTO, BOOST and MANUAL modes and preserves the present APC BIT operation, avoids right/left engine throttle shaft tracking response difference problems, and allows demonstration of DLC, APC with FADEC in the F-14. The flight test configuration will incorporate dual or redundant shaft position transducers driven from the left engine cross-shaft. The throttle position or PLA command will then be transmitted to the FADEC processors without any mechanical or hydromechanical fuel control backups. Like the Mach lever, function significant cost and weight savings, improved maintenance and better reliability will be realized if this control can be done through FADEC.

4. ADDITIONAL FADEC TECHNOLOGY BENEFITS

Additional F-14 generic or general aircraft system benefits that might be realized through application of integrated aircraft/engine FADEC control technology are:

- No trim engine installation
- Improved engine installation
- Reduced aircraft weight
- Improved aircraft propulsion system reliability
- Engine performance monitoring and diagnostics
- Improved Engine/aircraft performance
- Reduced maintenance manhours
- Reduced engine and aircraft costs

4.1 No Trim Engine Installation

The use of FADEC with a TF-30 promises the ability to delete the periodic engine trims required due to aging/wear of the standard hydromechanical fuel control hardware. The following savings are projected for 302 F-14 aircraft with FADEC controlled TF-30 engines:

- Time and fuel used for a typical trim run with no adjustments

		Fuel Used	
Power Setting	Time, Min	lb	kg
Idle	5	50	(23)
IRT	3	400	(181)
ZONE IIIA/B	2	1100	(499)
Max A/B	1/2	438	(199)
Idle	5	50	(23)
		2038	(925)

- Trim run with adjustments required

  Fuel Used = 2 x 2038 = 4076 lb (1849 kg)

- Average trim run (assumes half of trim runs required adjustment)

  Fuel Used = 1/2 x 2038 x 1/2 x 4076 = 3057 lb (1386 kg)

- TF-30 FADEC trim elimination savings in fuel

  3057 x 6 trims/(A/C)/year x 302 = 5,539,284 lb/yr (2,512,147 kg/yr) = 19,395 bbl/yr saved

In addition, deletion of the trim requirement could save an estimated total of 21,7[illegible] maintenance man-hours per year for the 302 aircraft.

4.2 Improved Engine Installation

There are improvements when the TF-30 advanced fuel management is combined with FAD[illegible] Replacing the motove flow centrifugal impeller pump and gear box with a new vane-type pu[illegible] could result in horsepower extraction savings at several engine settings for the F-14 aircraft of:

Power Setting	Power Savings/Engine
Max	10 hp (7.35 kW)
Cruise	4 hp (2.94 kW)
Idle	0.5 hp (0.36 kW)

The fuel savings for the fleet air defense and fighter escort mission for these lower power extractions are estimated to be 1912 and 1816 barrels per year respectively, based on 302 aircraft.

4.3 Reduced Aircraft Weight

P&WA has estimated a 130 lb weight savings per engine for FADEC with Advanced Fuel Management. FADECs in an F-14 could also allow elimination of the AICS, Mach lever, APC with DLC and N_1 overspeed detection functions. The AICS pitot-static and α probes and the APC accelerometer would be retained for interfacing data with FADEC. Some torque motor electronics would have to be added in a unit to provide the P_t, P_s, and α data to the FADEC processors. The F-14 is assumed to have a redundant 1553B data bus capability in place and therefore only the dual redundant electro-optical conversion need be added to interface with the FADEC fiber optics data links. It is assumed that the FADECs would power these added electronics.

The resultant savings for an F-14/TF-30 type aircraft configuration utilizing FADEC and Advanced Fuel Management technology could be in the 325 to 335 lb (147.7 to 151.96 kg) category. This would result in savings of 13,000 bbl and 6400 bbl per year savings for the fleet air defense and fighter escort missions respectively, based on 302 aircraft. The benefits of replacing the AICS, which is a high maintenance item on the F-14 would be considerable.

4.4 Improved Aircraft Propulsion System Reliability

P&WA projects a 5.5x improvement in MTBF for the TF-30 advanced fuel control with FADEC processors over the existing hardware. Similarly, there is a preliminary design goal of one mission abort per million flight hours vs 1400 for the existing TF-30 engine fuel control system. Since the fuel control system is a significant, although not a major portion of the overall engine MTBF, there would also be an improvement in the overall TF-30 engine reliability.

The overall F-14 propulsion reliability would be improved by at least a factor of two due to gains for FADEC, Advanced Fuel Management, and deletion of the AICS, APC and Mach lever trim units.

4.5 Engine Condition Monitoring And Diagnostics

The availability of the FADEC processor digital computation and data storage capabilities make it an attractive candidate for inclusion of engine condition monitoring and failure diagnostic functions.

The engine condition monitoring function could encompass routines and algorithms to perform calculation and data storage for the following:

- Major/minor heat cycle counting
- Low cycle fatigue history counting
- Limit exceedance occurrences and exceedance time counting
- Engine and A/B on time and cycle counts
- Vibration limit exceedance occurrences
- Delta performance changes
- Sensor failure or out of limits occurrences
- Abnormal start detection and counts
- Stall detection and occurrence counting, etc.
- Failure accommodation history recording/counting

This history data could then be used to obtain health and projected life information for limited life and cycle fatigue critical engine elements. A maintenance program could then be developed for the TF-30 which would be keyed to actual engine wear/degradation usage and not just to fixed time schedules. Engines which are not being subjected to strenuous demands such as seen in air/combat maneuvering flights, might yield savings in maintenance (M) hours due to the resultant, relaxation in M interval times.

The failure accommodation capability which P&WA plans to incorporate into FADEC to provide a fail operational capability can be exploited to perform a diagnostic observation and recording function. This would assist maintenance personnel in allowing more effective post-flight troubleshooting.

4.6 Improved Engine/Aircraft Performance

P&WA also suggests that improvements in engine/F-14 aircraft performance are anticipated from FADEC in the following areas:

- Automatic TF-30 stall detection and recovery
- Reduced part power engine transient response times
- Elimination of full power thrust overshoots
- More precise engine scheduling.

An improvement in SFC, expansion of inlet/engine stall margins, and improved engine transient responses could extrapolate to F-14 aircraft performance improvements in:

- Increased loiter time
- Increased combat maneuver envelope
- Improved manual and automatic carrier landing control precision.

5. CONCLUSIONS

FADEC technology is capable of providing significant gains in F-14/TF-30 type system performance, reliability, weight savings, fuel savings, lower life cycle costs, and maintenance manhours saved. The air inlet/engine controls integration promises improvement in F-14/TF-30 inlet/engine compatibility and performance characteristics. An F-14/TF-30/FADEC controls integration flight demonstrator test program can be achieved for reasonable costs. The introduction of digital FADEC processors has many projected advantages and savings for a TF-30 type configured F-14 aircraft system. Pursuit of this technology is strongly recommended for future applications. In particular, the architecture advantages of combining future quad redundant fly-by-wire flight controls with two engine, duel FADECs should be exploited to the fullest extent possible.

6. REFERENCES

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2. Vizzini, R. W., Lennox, T. G., and Miller, R. L., "Full Authority Digital Electronic Control," SAE 801199, 13 October 1980.

3. Tindell, R. H., Hoelzer, C. A., and Alexander, D., "F14 Inlet Development Experience," ASME 82-GT-5, April 1982.

DISCUSSION

J.T. Bakker, Ne:

You spoke of the integration of two FADEC controlled engines with a fly-try-wise flight control computer. Do you also plan integration with the Fire Control Computer, especially with regard to throttle-automation?

Author's Reply:

Future designs will use fully integrated engine, propulsion system and aircraft flight controls and will where applicable integrate the fire control computer. As I mentioned, it is my opinion that integrated controls with aircraft controls are applicable to all types of further designs for everything from cruise control on subsonic patrol aircraft to the obvious most complicated VSTOL fighter. This will save aircraft design take-off gross weight, repair cost, decrease maintenance, increase survivability and durability.

D. Giovanni, It:

1) Was there any particular reason in selecting the left engine *for installation of the* FADEC? Is it the more critical one?

2) Do you plan to profit out of FADEC performance to cope with disturbances in left engine induced by gun firing?

Author's Reply:

1) The left engine was selected because all previous propulsion system performance tests (inlet rakes, bleed, etc.) were done on the left side. Therefore a correlation with previous data could be made. That was the only reason for selecting the left engine, there was no other difference.

2) Early in the development phase of the F14 gun gas ingestion was encountered. Several deflectors were tested. A successful one was tested and no gun gas ingestion problem has re-occurred. Therefore, we have no problem to evaluate with FADEC. So if we did, I am sure the FADEC could be used.

ROUND TABLE DISCUSSION

Chairman: Mr H.I.Bush, Director Turbine Engine Division
Air Force Wright Aeronautical Laboratories
Wright-Patterson, Ohio

Introduction:

This is the last official function of this particular symposium, the Round Table Discussion. We have until 1700 hours and I suspect it would be best, if we tried our best to close the session at 1700, so we can get our closing remarks from our chairman. So, we have about 1 h 10 min and I think that even if we had no questions at all, we should just sit here for an hour and 10 minutes, but I don't think we will have to, I think we will have plenty of response.

I want to thank everybody because we did get very good response on questions, in fact we have more questions than we could handle. However, I think that is good. Please do not feel left out if the questions you raised are not handled here on the floor. I hope that everybody feels free to participate and that the questions which do come up, at least in part will satisfy everyone as far as the questions they put in.

So let me re-introduce all the people, so you know who is up here:

Mr Fabri, who is the Head of the Turbomachinery Department of ONERA in Paris, sitting next to him is Mr Pollitt, who is Consultant to Rolls Royce for Military Affairs and as until very recently he was also the Chief Test Pilot for Rolls Royce. Next to him is our recent arrival, Mr Ben Koff, he is the Senior Vice President for Engineering for Pratt & Whitney Aircraft, West Palm Beach, Florida. Next to him is Professor Saravanamuttoo, who is in the Department of Mechanical and Aeronautical Engineering in the Carleton University, Canada. And last is Dr Bauerfeind, who is the Head of Performance and Control Department, MTU Munich.

So we discussed how to start this, I think we should let it move along. So that if we have issues, I may have to cut it off to go to a different subject, but again, bear with me, that it is my first time presiding over this type of an activity. We discussed how to start and thought we should open it up just with questions from the floor and see what that brings us.

So I think, with no more discussion I will ask for questions from the floor to any of the individuals or more of them on the Round Table.

Is there anyone who would like to start the issues? If not, let us proceed to the first question.

The subject of automated engine performance, the degree of integration of the engine into the overall combat weapon system and the degree of isolation that the engine should have from the pilot has been raised by Mr Bakker, Mr Stetson and Mr Robinson. So I am going to ask Mr Pollitt: Just how isolated should the engine operation be from the pilot and how can we approve integration of the engine into the overall total weapon system.

Mr Pollitt: I was pleased to see that right from the start Grumman have already taken my advice and that was only in two days, so there is hope for the industry. It's obvious that is the way that we are going to go, Tom and I have already discussed this. We can obviously get very fast response, for instance, simply by modulating just the after-burner part of the system. In other words, for all combat you just modulate the afterburners between mil and max A/B and with these modern engines with very high ratios of afterburning to dry thrust this is clearly one way to go in the short term. The difficulty with that, as mentioned by Tom, is that for the chasing combat pilot, the guy who has the advantage in any air combat situation, if you either put air brakes in and out to modulate air speed, that is the most marvellous visual cue to the guy behind that you are going to try to slow down. Similarly afterburning systems are like putting a big red light on every time to say that you are going to put on more power. And the more that you can do to hide your intentions the better. And therefore, I still believe that for subsequent generation aircraft we probably for all sorts of reasons including I R will have to go for non-afterburning engines, though the problem still remains. As far as integrating the pilot into the system, I would really say that you could fully automate it to optimise the total energy management to the whole flight control. And all I would really say, is that the pilot would have some kind of override, some instinctive cut out, whereby he is still, we hope, the best brain left on board in terms of deciding what to do next as far ahead to see what the enemy is visually doing. And therefore you could foresee situations where you would want more or less energy than the computer was demanding, and in that case the pilot would still select it.

Bush: I see, so as far as response goes, I think I heard you say, that from the pilot's point of view you need the combination of dry engine operation in order to be able to reduce the amount of flagging and very rapid response, even in the dry engine situation, so it is the response rate in the dry power regime that you are looking for.

Pollitt: What I was saying is that you could very quickly, with an engine existing today, achieve fantastic response rates of less than a second, simply by modulating the afterburning and never coming out of afterburning during the combat. The difficulty, as I say, is that by staying in afterburning all the time you are signalling your intentions.

Comment from the floor:

You also run out of fuel! **Pollitt**: Hence the other argument for having shielded hot gas exits from wherever they may be and practically not having yards of fire coming out of the back.

Bush: So do I take it then that the combination of dry power and rapid response would be best from your point of view if you could get them both?

Pollitt: In the ultimate, yes. I mean, no pilot wants a huge infrared plume sticking out of the back with somebody with an AIM-9L coming round the corner!

Bakker: What I would like to do is to go a step beyond that. And I mean that, take all your engine instruments out of the cockpit. I would like to get rid of the throttle; it is a bogey thing, it's always in the way. The guy behind me has said things like that already. He would like some gain from a fairly short movable throttle; I would like to do without it. I also think that you should give the pilot a performance gain in each flight phase. He could be in flight phases where he really wants to make small performance corrections, he might be in flight phases where he really needs quick response and only two throttle settings: max and idle. I also think that you could automate a lot of functions so that from the pilot's standpoint workload goes down. I also think from hearing all the people here talking about cycles, that you could decrease the number in all of the flight phases by a tremendous amount. I would like to give you just two examples. One of them involves a high speed terrain following aircraft, where the object of the pilot is to reach a certain point at a certain time. You can do that in two ways by automation. Either you can let him fly at constant ground speed using the throttle or you can say, okay, we will take care of that time over the turning point and we allow you a speed band of say 30 kts to manoeuvre in. If you do the last, then on every hill the pilot has to climb, he will lose speed, but the system says, nothing to worry about. The average speed which he has to make to reach his turning point on time might rise by 1 kt.

Subsequently when going down the hill, he would pick up speed again. So on average you just have to change the fuel flow by maybe 100 pounds per hour instead of going through all that cycling.

When I saw the picture that was flashed on the board this week, the throttle movements looked to me as hilly as the terrain.

The other subject I would like to raise, is that if you really integrate an engine to go for speed control, for instance, in the last phase of combat you might be tracking a target which is flying in front of you and the only thing you are interested in is to reach a certain distance from the target without any overshoot speed in the shortest time possible. The fire control computer has all the necessary information readily available, and if you could automate say, "go to 1000 ft behind him in the shortest time possible", the pilot work load would go down immensely. On the other hand it is the most difficult thing the pilot has to do and especially in that regime you find a lot of idle-max power-idle .... situations. Thank you.

Bush: Thank you, do any of you wish to respond to that? Let me say that I think what I hear is that there should be more explicit requirements or goals or objectives in the area of throttle response and in the area of more automated use or combinations of aircraft manoeuvering, including speed, target acquisition and engine response in order to be able to get that word down into the place where it counts, and that is where the engine is first being designed. I sometimes wonder if the types of things that you told us about here ever get through the red tape of bureaucracy to get down to where we can use them as design targets. And that may be a situation that we might be able to help.

One problem that I see to say, going to the equivalent of the inflight control system at the moment, is that every time in the United States so far and to a large degree in Europe, that we go to a full authority either analogue or digital control, there has always been an insistence on having a mechanical back-up of some kind of emergency fuel system; that, of course, ruins your whole idea of having a short throttle displacement, pressure signalling, and so on. So we still have to educate the people who write the requirements into the fact that you can have a triplex or duplex system and that your standby control is a simple DC Electric Manual.

Covert: Part of my remark has just been remarked already, but I think that in all of these great ideas about full automation there is much merit in reducing the pilot's workload, particularly in combat. I guess I have some minor concerns that when people get around to doing this they remember that combat is a very difficult situation and you don't want to be in a position that if you lose a data computer somewhere along the line, that your engine goes out. So there are a variety of redundancies that I think have to be built in. And the other thing I think that you have to worry about, at least in the future; if it should ever happen – and we all hope it doesn't, – that we have to go back to war, then nothing seems to work as well under those circumstances as it does in peacetime, and so the whole maintenance procedures will have to be completely reviewed. I think that what we are talking about is not an evolutionary change but a revolutionary change and while I think it is a push in the right direction, let's just bear in mind that it has a long shadow which is not over solid.

Koff: I just want to make two comments, one is that a lot of the advanced technology people in the airframers are thinking very positively about vectoring nozzles. Most of the companies are doing some work in this area.

We have recently tested a vectored nozzle in dry, afterburning and full reversing, for shorter take-off distances, shorter landings, that can be done in 1/5 the distance of an icy runway. But these nozzles can also be used in flight for manoeuvre. Now, when we go to that and there is no question about it, the flight control system is going to have to be linked up with the engine and the pilot in moving the throttle is actually going to be controlling the engine, otherwise it would not make much sense.

That is going to give another dimension perhaps to the question that we have brought up. Then the other part of the question is rather interesting, most of the companies are working for governments that are at peace and the peace-time requirements for cyclic operation are quite a bit different from war-time. There are several countries that have used the equipment that we use today in war. And their view is a little bit different when you talk to their pilots. Their pilots say they do not have any durability problems because when you fight a war you do not keep the engine on wing for seven or eight years which is the equivalent of some of the things that we are designing for. And so I think we have got to have a look at what we are designing for and make sure that we do not put all our efforts into a lot of peace-time long range activities when in fact these systems do have to work in war. It will be interesting to see what results come out of the recent combat experience where our equipment was used against the other guys.

Bakker: I would like to make some comments about the concept of such a system, and I do not think you understand me quite, because some people are really afraid of that. We are already flying without any mechanical back-up, fully electrical, so that it won't make a hell of a difference, if the throttle was automated in the same way, say quadruplex, on the electric side. The other side is, I have not heard here any comments about the improvements especially in engine life in peace-time, because if you go to automation you would produce much less cycles. So you either could go with the same engine for war performance or you could go to longer engine life.

Koff: Normally there is design for one requirement. If it has so many cycles to meet, then that is what we design for and very rarely do you then end up trading. I do not say it could not be done but I have not seen many of these trades pulled off.

Bush: Well let me shut the subject.

Major Goulios of the Hellenic Airforce would like to hear some discussion regarding improvements in the ability of the aircraft operation user to use all of the life of each engine that is available. That is a paraphrase; let me explain it a little bit more, let me ask Mr Koff and Dr Bauerfeind to expand on what I just said. Essentially will you discuss the methods of tracking low cycle fatigue and time and temperature engine usage that is in the F100 and the Tornado engines and offer some comments on how either data recorded or use of this data could be improved and please look at it from the point of view of the operator.

Koff: I saw a paper that had a little box on it, which showed on the F100 that you track the hours, you track the cycles at two levels of turbine temperature and of course you monitor the other critical parameters to make sure you have not exceeded rotor speed and things like that. I think that in the future we have opportunities to have much more data when we introduce digital electronic control.

Whether it will be useful, I cannot say, but we would store a lot more data than we could actually read out. So when the plane comes back, the maintenance sergeant can go up and read out a lot of things and they will tell him much more than the current black box does. I do not really know whether or not we will need much more, but I think it will tell us about more faults that we can find in the engine and so, from the maintenance point of view, I think it is going to be useful.

Bauerfeind: Well, as far as the Tornado is concerned, we have not got a life cycle recorder on the aircraft. What we did right at the beginning of the programme, was to put a recorder into a few number of the aircraft. These aircraft then flew what was considered typical missions and the recordings we got of the cycles used during these mission gave us a basis for working out the overall life of the various components in the engine. Now the problem is a rather complex one, since the RB199 engine is an engine of typical modular design and we are all convinced that it would pay off to have a good recording system where we do sophisticated recording, not just RPM against time but also including temperatures in the components to really be able to compute the actual LCF life used up. And it is very likely that the three Nations participating in the Tornado programme will have their own and separate approaches to this problem finally. On the German side we are all convinced that it would be cost effective to finally include LCF counter on this engine but it is very true that we have to solve the logistic problem on the ground, because just having the data available would not do the job, you have to make sure that you process the data in the correct manner, because if you introduce this system, then it would be very dangerous, if somebody made a mistake or something goes wrong somewhere and you have a disc in the engine installed for longer than acceptable, because then you might produce a disc failure and that is just about the worst thing that can happen.

Heilmann: Please allow me to make a comment. When we calculate the temperature we have to distinguish between the cold and the hot parts of the engine. By the hot part of the engine I mean the last compressor stages and the first turbine stages. For these stages the proposed system will calculate the temperature and its evolution with time. In other words, the computer which is integrated in the system calculates the real stresses in each disc at any time. From these data the life used up can be computed and the remaining life can be read directly with the appropriate ground equipment. The computer box is integrated in the propulsion system and is delivered with the engine. In our opinion, there is no sense in having a simple cycle counter in an engine with a pressure ratio of about 30 and a temperature of about 1700 K. You need a temperature measuring device and a system to calculate the disc temperature and the actual thermal stresses. This way we have an opportunity of using the discs for much longer than you could if you only have a cycle counter which counts the number of cycles and speeds. Thank you.

Ramette: I would like to comment on what you have just said. For the modelling, I have the feeling that all the models that are used in the engines, especially in the hot part of the engines are actually much too simple and in particular, if you take for example the temperature of the air flow at the entrance of the high pressure turbine, the model usually calculates an average temperature and in fact there is a very strong distortion of temperature at the inlet to the high pressure turbine. The model usually does not calculate the distortion of temperature, but it can lead to very large changes, especially when you make a calculation of the stresses of the blades. So I have the feeling that for the modelling it is actually too simple, and a lot needs to be done to make the model more precise. Thank you.

Wing Commander Rolands, RAF: In my opinion we have just heard the very reason why on Tornado there is no engine life recorder. The designers and the engineers, such as yourselves say that the models are not good enough with the result that the Tornado is flying with no engine life recorder on board. In my opinion we ought to have one now, perhaps even a simple one that just gives the time temperature counts as it is done on the Harrier aircraft. I would like to see time, temperature and RPM somehow put together in a fairly simple algorithm that would then produce a number in counts, and it is then my job as a maintenance engineer to match the counts number with the condition of the piece parts which are seen during modular maintenance. If the counts number is not very accurate, because the model is not good, I do not care as long as it is fairly close to the condition of my turbines when I see them on strip when they are taken apart during overhaul. We already have an engine life recorder which could go into Tornado today and I would like to see the personnel at MBB apply the same sort of pressure as I am attempting to put on from the user's point of view.

Bauerfeind: I would just like to make a very short comment on this. I think we all agree with what you said with the exception of one statement. I think it will be very difficult to see by inspection the LCF life taken out of a component, it is virtually impossible because you can only do this by a destructive test. In the normal inspection you do not see how much LCF life has been taken out of a disc and this is a typical case where you have to build up experience and then refine your model. I think it is also very true what you said, the correct model might finally be a complex one, but I think it is still better to have a rather crude model to start with rather than having no counting at all and then to continue refining the model because Tornado will be in service for many many years to come.

Koff: I think we might add one thing that helps this case. That is, you know in commercial service they have been gaining a lot of experience on how long you can run a disc before the initial life is used up and all of the companies have life management groups which are working very hard to perfect their internal modelling of the disc, to say at least when it runs at a certain reasonable cycle, it should produce this much life. Those models are quite sophisticated today provided you stay within certain temperature ranges and certain speed cycle ranges. I believe that their predictions for the initiation of cracks are becoming very accurate compared to 10 years ago. I also mentioned that the materials that we are going to go for in the future are going to be a lot more tolerant on the residual life. Today we only talk about crack initiation, that's initial life, but depending on where the crack is and depending on the design and depending on the material you can have a substantial crack residual life which may greatly exceed the initiation life and this will really help the maintenance people.

Covert: First I think that the idea that the Wing Commander mentioned is terribly appealing and I would like to see someone try it. I would also suggest that you record pressure because it is not temperature alone but also the heat transfer rate and so you would need this additional parameter. But there is one thing I have a mild concern about, and Mr Koff alluded to this, and that is that in the civil usage there is some reason to believe that the damage and the deterioration is uniquely related to the processes that the engine has gone through. In a fighter I don't think we have enough knowledge to know whether or not such a unique relationship exists. I remind you, years ago, I think it was the early models of the F86, I am not sure which one it was, had a marvellous new material 71T6 in the main spar and that worked out very well under normal circumstances but when you flew across the hills of Korea at a Mach-Number in excess of about 0.6, those wings were good for about nine flights before there were 40 in cracks in the webs. So we have to learn more about this. I agree that some work is needed here to make sure that our models are even unique in terms of damage and its rate of accumulation.

P.Neal: I would like to take a certain amount of issue with the previous figures. To calculate cycle life in discs, in commercial and military designs, involves fairly sophisticated finite element programs, temperatures, stresses etc. and particularly transient stresses. When you design a modern engine you also have to do a deflection analysis, in other words you are controlling tip clearances and movements. You cannot use these sophisticated programs to that extent because you have to analyse a complete mission; but we already have programs which are very much simpler but which largely give the information required.

We already have programs which we use which can follow missions and calculate tip clearance changes. And in commercial usage a large number of airlines have AIDS which give us basically the information to do this. I think when you are going to FADEC digital systems of electronics all the information is there, it is simply a matter of collecting it and running it on a second level program to calculate the fatigue damage for any given mission.

Ramette: I have a question for all the people here. During the whole meeting I have never heard about detection of cracks. I think it could be very helpful if someone could say something about this. Because if the pilot could know when a crack appears that would be very, very nice for him!

Remark from the audience: "I don't think so!"

Koff: There is considerable work going on all over to try to detect small cracks in parts, and I think that the industry has progressed tremendously, even in the last 10 years. And in the last 5 years in particular, I can remember when we were talking about trying to detect a .030" crack. Now we actually have laboratory equipment that can detect .005" cracks. The difficulty of course in introducing some of this equipment is, when parts fret and wear you then have to prepare this surface in order to inspect it, and of course that means taking the rotor totally apart. And this shows you the advantage of having a low number of parts, minimising the number of joints. But there is a lot of work going on. As a matter of fact, I think that most of the effort that is going on in structures is in crack detection, the avoidance of cracks, and in how you incorporate that into design. I just do not think it is a subject that should be discussed at this kind of a meeting, but there are major efforts going on for crack detection and designing materials that can last longer with a given crack. But not detection in flight!

Bush: I think there is one more aspect which may seem mundane but I don't think is and that is from the point of view of the operator, there is more to keeping track of low cycle fatigue than time and temperature aspects of the engine. With the modular designs, there is the whole bookkeeping problem of keeping track of time and life and cycles of modules that may shift engines throughout the life of these parts. For us to be able to do that with any degree of accuracy and any degree of economy of manpower I think is going to take some kind of automation. The problem we have is that quite often the engineer, in trying to simplify the system or to put equipment on board the aircraft or in the engine that will keep track on this, fails to understand from the maintenance point of view, what data is required or how the data should be handled, so as not to make the job even harder in the field in terms of keeping track of this information. I think the way to get around this problem is that some of the maintenance squadrons and logistics people, who have to overhaul the engines and keep track of parts, need to be involved in the generation of this type of equipment and software right from the start. I know from personal experience that this can be done. It means getting into the field at the squadron level at the start of a programme and letting the maintenance people and the logistics people be involved in the generation of that system right from the start. If you don't do it that way you end up with beautiful techniques that nobody uses and I think that one of the lessons we have all learnt in generating these kinds of automated bookkeeping procedures, if that is what you want to call them, is that you will have to keep the operator in the cycle right from the start of the program and then there can be some success to it.

Bush: Any other comments to this before we proceed to a different subject? Dr Dunham raised this issue, which is a different slant on the first subject of this afternoon. Let me read the question: "Mr Pollitt spoke about the need for rapid thrust changes in combat, which may be achieved at the expense of extra weight. How can we quantify the desirability of rapid thrust changes so to be able to make proper trade-off studies?"

Pollitt: The only way that we are going to be able to evaluate any changes in thrust response is by using one or preferably all of the European and the United States air combat simulators. And these are now developed to a point where the results are very pretty and entirely believable in terms of success and failure rates. I mean, to the extent that I understand it, particularly at McDonnel I saw people coming back from real missions, who were very sceptical about it, who actually had to be forced off the machine at midnight, because it was all so real. Now, in order to do that somebody has got to raise some money to do the evaluations; they are pretty cheap to do and we could find out very quickly. It may well be that my theory is totally wrong and I am quite sure that any mathematician here could prove that I am wrong.

In the total energy management that we are talking about the difference in all this maneouvering between a one second acceleration and say three or four second acceleration in the total energy is very small indeed. But to go back in history, we had a problem when we re-engined the F4 with a Spey for the Royal Navy. The total time for both the J79 and the Spey from light up to max A.B. was almost identical. But the characteristic of the thrust increase was quite different. The J79 was almost a straight line but the early Spey system produced almost no increase in thrust for about three seconds and then you got it all in the last one, and we had to change that, even though it met the specification, as we received such abuse from the British F4 pilots about the system.

It increased the workload as they were not sure when the power was coming on or what was happening. So there will be a certain psychological element in any test you could do.

Koff: In order to get rapid throttle response you claim you need to add weight and I guess I am having a little struggle there. I do not recognise that we ever add weight to get throttle response. The design of the turbomachinery and a matching of the components and the ability to put fuel in without exceeding the limits of operability of the compressor or the limits of operability of the combustor are normally the limits for how fast you can accelerate. And as a matter of fact, if you are willing to take some temperature overshoots, or if you design the turbine to take an additional overshoot in temperature, or if you design the turbine blades with more cooling margin you can actually get additional throttle response in many of the current engines without any expense in using up part's life. But you do have to pay attention to how you are cooling a lot of these parts when you get that momentary temperature overshoot.

Saravanamuttoo: You should also be careful in looking at the difference you are going to get on engine life. If you say, well, we can do a four second acceleration, but would rather do a two second acceleration, you would certainly pay a very severe penalty in terms of the overtemperature and the thermal shock for doing that. I would like to remind you that there is a definite upper limit to the amount of energy you can put in in a fixed geometry engine, which defines the minimum possible acceleration time you can get between two speeds. If you really want to keep the acceleration time down you have to keep the rotor speed up. You may be able to do it by variable geometry means, such as thrust vectoring, but I think people have to clearly understand that super fast acceleration times will have to be paid for in life.

P.Keeling: You have just been talking about the handling of the engine, and cooling of the engine.

The problem that we usually find is that another factor comes into the equation and that is the thrust. Because we always want more thrust and ask for more handling but the engine manufacturer wants more life, i.e. wants more cooling. So there is a three way argument between thrust, the handling, and the cooling.

Bush: Let's proceed to the next question:
Dr Heilmann wants to know to what extent the digital engine control will help to design high efficiency compressors with relatively modest surge margins.

Saravanamuttoo: My basic response to that is, it doesn't. The only thing that a digital control system is going to do for you, is give you a better control of the acceleration fuel schedule; there could be slightly less tolerance on these fuel schedules than you have with a hydro-mechanical control, but the surge margin, the surge line of the compressor and the operating line of the compressor primarily are fixed by the aerodynamics and thermodynamics of the cycle.

Bauerfeind: As I tried to explain in my paper, we are working on a concept called "Mode Control". And mode control tries to feed more intelligent information into the control systems in order to achieve what you are obviously after, that is operating with less surge margins at the normal operating conditions. When I say normal operating conditions, these are the operating conditions apertaining to about 99% of a mission, where you have a climb, some manoeuvres, descend and landing. But with fighter aircraft, and if I take the case of the Tornado, there are the extreme situations where a pilot flies in the upper left hand corner of the envelope, pulls high angle of attack, at the same time swings his wings, perhaps has an emergency with one engine failing and the other engine has to provide all the power to swing these wings and at the same moment accelerate the engine. This is just about the worst combination you can think about and if you have to cater for this, and with normal control systems you do because you don't want to risk a surge in such a situation, then you need a reasonably large surge margin in your compressors. Now the concept of mode control tries to prevent this handicap that you have to carry around additional surge margin, just to cover one extreme case. What we try to do is to include bleed valves for instance in the engine which bleed air off the compressors and lower the running lines so that we can cope with the situation. But in order to operate these bleed valves and do other tricks and trims in the controls we need the information, and this is where the digital control comes in. If we get a lot of detailed information from the aircraft computer, about flight condition, angle of attack, so that I can compute Reynolds Number effects, the effect on the surge line and the power offtake requirements, then I can adjust my controls and provide the surge margin to cope with this critical case.

Under normal conditions, and I think this has also been mentioned before, I think it was the paper on the TF 30, it pays to reduce the surge margin in order to reduce the SFC for cruise for long range performance. So I think, digital control does come into the business here, and it can cope with extreme situations which otherwise I would have to provide the surge margin for.

Bush: Another question is very close to what we had talked about just shortly ago, the ability to decrease the response time. Is it possible to fulfill the requirements for short handling time and quick response with dry fighter engines? In other words, engines which don't afterburn. Would you like to handle that also, Dr Saravanamuttoo?

Saravanamuttoo: Yes, let me just show a viewgraph for civil engines. If you consider the thrust response for civil engines, it is laid down by the FAA in the US and the CAB in the UK. They basically call for an acceleration between idle, which is defined somewhat differently in the two systems and maximum in not less than 5 seconds. That lower flow curve shows in fact what is an acceptable response, it is an undesirable response, but it is quite acceptable and it meets the requirements. The dotted curve up above shows the fastest possible at the same time which represents the best thrust response you could get in that time. Now if you really want to get this very rapid thrust response on a dry engine you have to maintain a high core speed under all conditions.

If you let the core speed fall off to get a lower thrust it would be difficult to get a rapid response, even with variable geometry.

Bush: Would anyone else like to comment on a dry engine in terms of being used for a fighter combat type engine?

Cliff Hosser: (Grumman) Most of the designs that we are working on are taking advantage of the composite materials and the advanced state of the art in terms of thrust to weight, higher thrust to weight. We have come up with a few configurations of airplanes that are so called supersonic cruisers where in fact you can do a whole mission with dry engines. So I think there may well be a future for this possibility.

Bush: I take it from what you said, you intimate that if dry engines for supersonic aircraft are a fact in the future, we have got to learn how to make their response rates acceptable. Does anyone else have any comments?

Dunham: The key to faster response must be variable geometry and the intelligent control of it by digital computer. It was pointed out that the acceleration rates are limited by rather fundamental things like thermodynamics. But there must be scope for using variable geometry in such a way as to optimize a path near the surge line. We don't yet understand how to do that. The other thing we do not understand how to do, is how to detect when you are nearly at surge but not quite there. At NGTE, 12, 15 years ago we found it possible to have a closed loop control which would detect surge as you approached it and avoid it, but only on certain engines and only at certain speed ranges and only at sea level. We still do not know, and I don't think anybody else knows how to do that in general. If we could find out then we could certainly minimize the acceleration time, but we don't know yet. But I do not think there is a law of nature which says, we shall never find out.

Bush: Yes I agree that the idea of looking for that type of device is good, but I would hope we would not wait for it to come before we took action. I think that is a long range problem.

Heilmann: I would agree with Dr Saravanamuttoo, but the other point is, if we design dry engines, the dry engine has a higher moment of inertia. That is another fact which contributes to the longer response times because the engine is bigger if it is a dry engine.

Saravanamuttoo: I am not quite sure in fact that because an engine is bigger it has a slower response. I made a bit of a study from first principles and you have to remember that as an engine gets smaller, although the inertia goes down, so does the energy release. Actually it has been shown that the response rate is proportional to the linear dimension. I think it is in fact not quite so, because when you look at the small engines the discs change shape, and you get fatter discs so that the moment of inertia does not in fact go down as fast. In the bigger engine the shape stays about the same. So although you would certainly go a bit slower on a bigger engine it would not necessarily be in linear proportion to the small engine.

Pollitt: I would also like to state that there are no afterburning helicopter engines in the world, that I know of, and they all accelerate ever so quickly because they have to. You have just got to match the rotor speed otherwise the rotor claps hands over your head and nobody likes that. Secondly, the graph I put up showed clearly, I thought, that regardless of the size of the engine, Rolls Royce manages somehow to produce the response which is needed by the aircraft. And in the case of the Harrier engine we have to produce fast acceleration times because it is the only way you can hover the thing, and that is relatively quite a big engine with 22 000 pounds of dry thrust.

Saravanamuttoo: Are you also sure that the Pegasus engine has the same thrust response as the Viper?

Pollitt: Yes.

Saravanamuttoo: I also think that the reason that your helicopter engines show the faster response is they are starting from a considerably higher speed, where the time lost is much less.

Pollitt: Yes.

Hercock: Perhaps from Mr Pollitt's point of view, how quickly can you lose your forward thrust on the Pegasus just vectoring the nozzle?

Pollitt: Under 1 second.

Bauerfeind: I would like to come back to this argument of dry engine versus reheated or afterburning engine for supersonic fighters. I think from the discussion it has been established that there will be a disadvantage with the dry engine with respect to dry response times. But there will be another disadvantage and that is, when you modulate thrust in the supersonic regime with the afterburner engine you just take the afterburner from between min and max and the thrust alters by more than 70 – 80%, perhaps as much as 90% in the high supersonic regime. While you are modulating the afterburner you keep the mass flow of the engine constant and this helps the intake control. With a dry engine modulating the thrust between idle, that is minimum thrust and maximum thrust, you take the mass flow right from min to max and this would add to the problems.

Hosser: I thoroughly agree with the whole point that was brought up, I think I have already said it, but inflight vectoring is one of the ways that you can do that at high speed within the two degrees of deflection you need. You run the engine at max power but change thrust by dumping the nozzle in and out.

Hercock: If you design the intake properly, as on Concorde you can modulate the thrust in the dry mode without any problems, any buzz problems at all. It is possible by proper intake design.

Ramette: I think what we want, if we consider transients in a compressor map is to go as fast as possible from one point in the map to another. So if I take an acceleration of the engine, what we want is a fast acceleration without going too near the stall line. I do not think it is by knowing when we are stalling is the best way. The best way should be to have some model to know that we are at a certain distance from the stall line and to be able to optimize the curve from point

A to point B by trying to keep at this distance from the stall line. I am sure that the optimization could be different from what we actually do now. For example it could be possible that to optimize the time we would have to make a slight deceleration at the end of the acceleration.

Bush: I would like to make one other comment and that is the fact that what we are trying to do is to modulate in the tail pipe some combination of pressure, temperature and mass flow, to get fast changes in some manner.

As was mentioned up here the mass flow has problems supersonically with the inlet. If you are operating dry you don't have the temperature range operated over quickly by controlling the fuel flow and nozzle area as we do with an afterburner. The only thing we have left to modulate is the turbine inlet temperature and pressure which is the operating point on the compressor map. So we had better quarrel with the energy management within the engine both within the turbine and within the compressor, and depending on how hard or how badly we want the fast response the answer, I think, lies in reducing the amount of RPM modulation and expanding the amount of pressure and temperature modulation, which means we need to be able to manage the energy being extracted out of the turbine and the energy being swallowed by the compressor, and the cost to do that requires control of effective flow areas both in front of and behind the turbine.

Fabri: I think there is a further difficulty, as you are thinking as if you would have a succession of steady states but you actually have transients and we showed this in one of the papers that you cannot stall a compressor even on the test bench just by suddenly giving too much fuel in the main combustion chamber or the reheat chamber or by some other means.

So, when you think of the stall limit in the steady state operation this may have to be modified so that you have to think of other things that happen during the acceleration. Transients, we think, will have more and more importance in the future.

Koff: I think that most tests that are run on compressors today consider both the steady state line where you creep up very gradually and also the degradation when you go up more suddenly. And there is a distinct difference and I think that everyone is now monitoring that difference.

Bakker: I am sure there is some difficulty in making a totally dry engine. However, I have seen a tendency over the years that the afterburner part of the thrust gets bigger and bigger and I think it would be very advisable if we could come up with an engine, which, say, for 95% could do the job in dry and only when you really need it should you add a small portion of thrust increase by afterburning. This means that I could make my comments against air-traffic pilots in the dry engine and only if I am really up against threat would I have to use afterburner. I think that would give me a far better controllable engine.

Bauerfeind: I think it depends very much on your mission. We have come across this problem with the RB199, and as I said in my paper, we chose a bypass ratio in the order of 1.2, because we had a requirement for long range good dry SFC, particularly in the throttled back case. And when you make an engine with a bypass ratio of 1; 1.2 or .9 like the TF30, then you automatically get the potential of high boost with the afterburner, because you have got a lot of cold air, which you can heat up to 2 000 K plus, and this gives a reheat boost at sea-level static in excess of 70%. Now after you have got this capability, it is up to the control system engineer to put a stop somewhere, and we are actually thinking in terms of what you have just said that is introducing an economic rating, where the pilot is normally only allowed to use 80%, 90%, whatever you want to define, of this boost available and only in an emergency situation overrides this stop and then gets the full power. So we are also thinking along your lines.

Bakker: But with 80% of the afterburner boost you still have a big flame sticking out of the back, which is not only inadvisable for the pure close combat type of situation but also for the missiles which are a threat against you. So what I would rather see, is 80% of total performance with the dry engine (or 95%) with just 20 – 25% afterburner power increase.

Bauerfeind: If you want to live with only 20% boost then you have to increase your dry engine, you have to make the core and the fan and everything bigger, and then when you cruise in the interdiction mission for instance then you have to throttle back much more than with the engine which gives 80% or 75% boost. And there is quite a penalty on your SFC. This is a problem, and that is why I said, it depends on the mission for which you design an engine.

Bush: I think this is very indicative of what we need to do more of and that is from the operators point of view what he has to have and from the engineers point of view the fact that it is hard to do and what else has to be given up in order to get it. I think we need more of that kind of dialogue between the people who have to build these things and the people who have to use them.

Our time is up; I think I just took my prerogative as the leader and made the last statement unless someone else would like to have a wrap up statement. It is very close to 5 o'clock. Would anybody on the board like to say something as a conclusion? So I thank you very much, I think this has been very successful, and I am very glad that we had the participation we did, I think it was very good, thank you very much.

I would like to turn the meeting over to Professor Covert.

Covert: First I would like to apologise to Mr Ramette for mispronouncing his name worse than I usually mispronounce French names. Second of all I think that in all of this discussion about transients and engine handling, I kept being reminded of what I saw on the first morning, of the difference between throttle movements of the leading aircraft and the throttle movements of number four, and I hope we all keep in mind that number four is there too. I think, it is easy to think of these things as one airplane rather than as a group of airplanes. I know it is election time in Greece, it is also election time in the United States, I would like to make, what I would choose to call a paid political announcement. Our next meeting, which will be the 61st meeting will be in Copenhagen in May and June and it will deal with Auxiliary Power Systems and the other part will be dealing with Reynolds Number and Viscous Flow effects in engines. If you or someone in your company is interested in attending this meeting, I suggest that you contact your national co-ordinator who is named on that yellow piece of paper that you have got that did not give the schedule of events, even though it is reported to. Following that, next year in the fall we plan to meet in Turkey and the 67th meeting will deal with combustion and combustion problems and again I would offer you the same sort of advice. I would like at this point then on behalf of all of you, to thank the programme committee for their efforts, I think we all owe a debt of gratitude, that is hard for us to express, to the interpreters for putting up with all of us fast speakers. I think that we certainly are grateful to the Hellenic Airforce and the Greek National Authorities; not only have we had excellent facilities and excellent weather, but – you may not know it, but it has been excellent weather! But all in all the tour of yesterday, they have just done a job right and apparently they did it right the first time because I believe this is the first time P.E.P. has come to Greece. I would like also to thank those of you who did speak slowly, I was particularly impressed on a couple of occasions when the speaker was French, and indeed spoke slowly enough that those of us, who speak a language that is commonly called "*High School French*", were able to understand what they said as well. And of course, finally I think that we owe a debt of gratitude to the speakers for their presentations and for the work that they did so well in preparing for the meeting. But I also think that the audience rarely gets the credit they deserve for a meeting like this. And so I would like to thank you for your attention, for your patience, for your dedication to being here, when there were so many distractions. And so, with all of those thanks I would like to bring this meeting to a close. Thank you all very much!

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15. Abstract

The Conference Proceedings contain 28 papers and a keynote address presented at the AGARD Propulsion and Energetics Panel 60th Symposium on Engine Handling, which was held in Marathon, Greece, on 11–14 October 1982.

The Technical Evaluation Report is included at the beginning of the Proceedings. Questions and answers of the discussions follow each paper. The Symposium was arranged in six sessions: General Consideration of Design (2); Experience with Combat Aircraft Engine Handling (8); Thermal Transient Effects on Component Characteristics (4); Aerothermodynamic Interactions and Modeling in Engine Handling – I Compressor Systems (6) and II Engine Transient Behaviour (5); and Control System Concepts for Advanced Engine Handling (3). The Symposium was finalized with a Round Table Discussion.

The purpose of the Symposium was to put together and review the present experience in the field of engine handling and to provide a forum where operational experts (pilots and maintenance engineers), engine developers and research scientists could exchange information and discuss their problems.

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