

# Research and development in aircraft gas turbines at the Gas Turbine Research Establishment

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The Gas Turbine Research Establishment (GTRE) of the Defence Research Development Organization (DRDO) was started nearly three decades ago to design and develop aero engines principally for military aviation, and to establish the required test and research facilities for component and full scale engine development and to execute research and development connected with new designs.

In essence, this charter was the cornerstone of applied research and associated development in many aspects of gas turbine technology. To name a few areas, aerothermodynamics and internal flows in turbomachinery, combustion of liquid fuels and combustion chamber development including after-burners, rotor dynamics, vibration, stress analysis using finite element methods, experimental stress analysis, heat transfer in air cooled turbine blades, structural integrity and mechanical behaviour of engine components.

R&D work in the field of gas turbines demands availability of a continuous large air mass flow around  $30 \text{ kg s}^{-1}$  at reasonably high pressures (6 to 15 atm.). The air would cater for aerothermodynamic studies on engine components such as turbines, combustors, after-burners, propelling nozzles and for heat transfer studies in full size turbine blades and vanes. The compressors, on the other hand, need a large shaft power (around 8000 kW) for testing them at their design speed which is normally above 10,000 rpm. As part of its charter, GTRE settled on an unique way of providing such a facility at relatively low cost. The method was to expand the exhaust energy, obtained by running a grounded aircraft's gas turbine engine, through a free (i.e. unconnected) power turbine which provided shaft power in two modes—(i) to a plant compressor of the centrifugal type for high-pressure air supply or (ii) to the research com-

pressor undergoing development.

The efforts on aerothermodynamic experimental investigations have to be matched with facilities that allow investigation of structural integrity and life of components before a prototype engine is tested on its test stand. Thus, specialized rigs have been established for fatigue testing of titanium compressor blades, cyclic spin testing and overspeed burst testing of titanium compressor rotor drums and super alloy turbine discs, and torsional fatigue testing of shafts.

An aircraft gas turbine engine has to deliver its rated thrust and performance at various flight conditions. Thus, the engine has to be tested in a facility that can simulate the normally aspirated engine inlet condition, as also that at high forward speeds of the aircraft at both high and low altitudes. The high Mach No. test facility (Figure 1) caters for testing of engines corresponding to a high inlet temperature and pressure as is prevalent on a hot day at high forward speeds.

In summary, the basic infrastructure for the design and development and associated research on full-scale gas

turbine engine components has been established at GTRE. Over the years, technology demonstrator projects have been successfully completed. An engine development programme for the light combat aircraft (LCA) has now commenced. In the course of this work, some interesting technical concepts have been made use of. These are highlighted below.

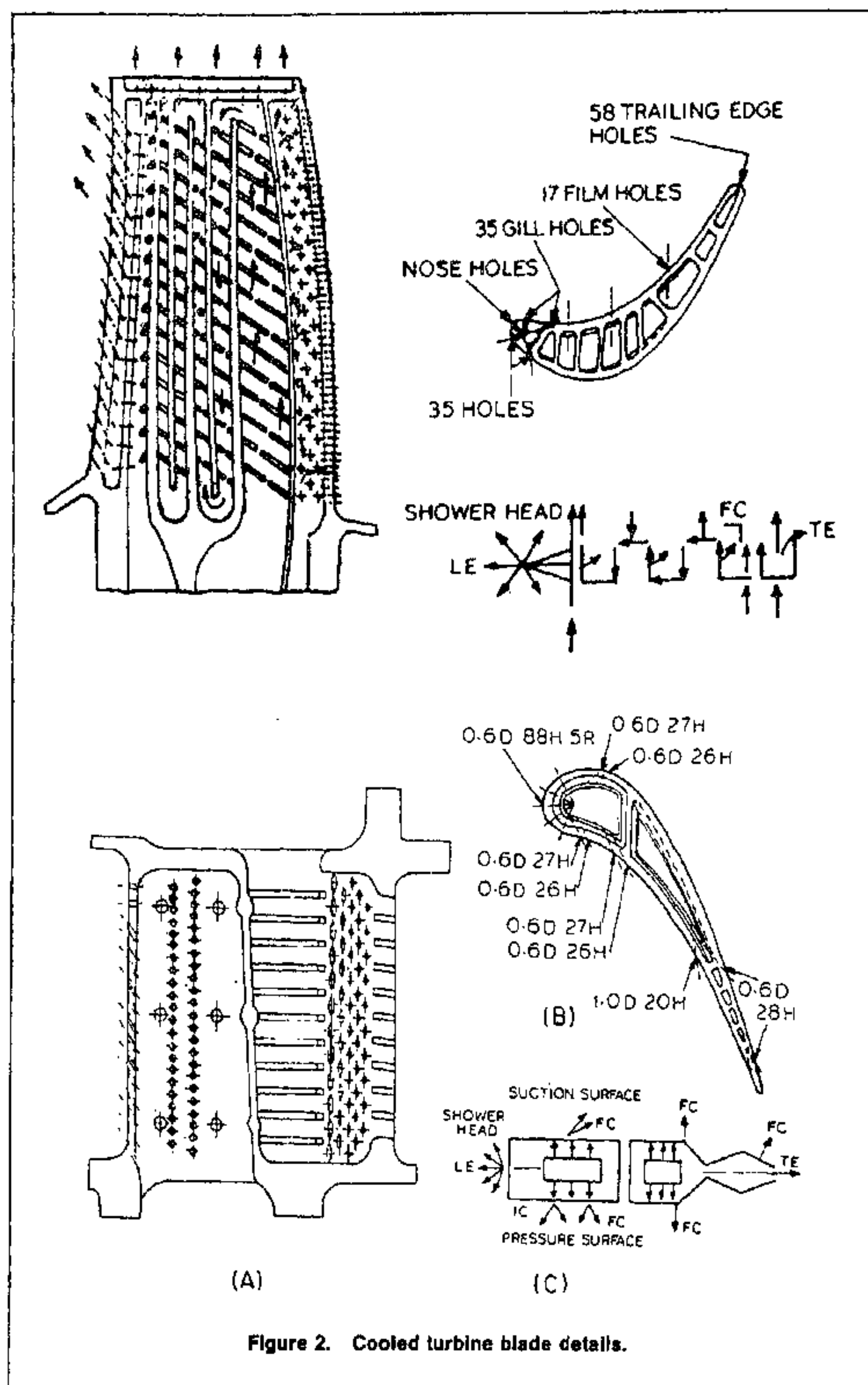
## Technical innovations

### *High-performance cooled turbine blade*

To be able to operate effectively at the high gas temperature of 1650–1700 K, the high pressure turbine vane (stator) and turbine rotor blade are designed with internal cooling passages through which air tapped from the compressor of the engine is passed. The hottest parts of the blade are the leading edge and the trailing edge. The leading edge is cooled by impinging the cooling air passing through the blade in a metered manner to cope with the external gas temperature profile. Also, the cooling air cools the root of the blade through a multipass (serpentine) zigzag route



Figure 1. Simulated full scale engine test facility.



through pin fins (Figure 2). The air effluxes through slots in the trailing edge. The design is based on a heat transfer analysis using the finite element method (Figure 3) to maintain allowable blade stresses at permissible surface temperatures at all points on the blade for a prescribed gas temperature distribution. Such a blade design has been optimized for a maximum life of 400 h in the actual rotational environment. The blades were made by precision casting, that is by shell moulding and vacuum casting with directional solidification. Validation of the design was done by testing the blade in a stationary cascade arrangement where the gas temperature was at the design value and the mass rate of flow of cooling air to gas flow was kept the same as the exact engine. This blade is now undergoing

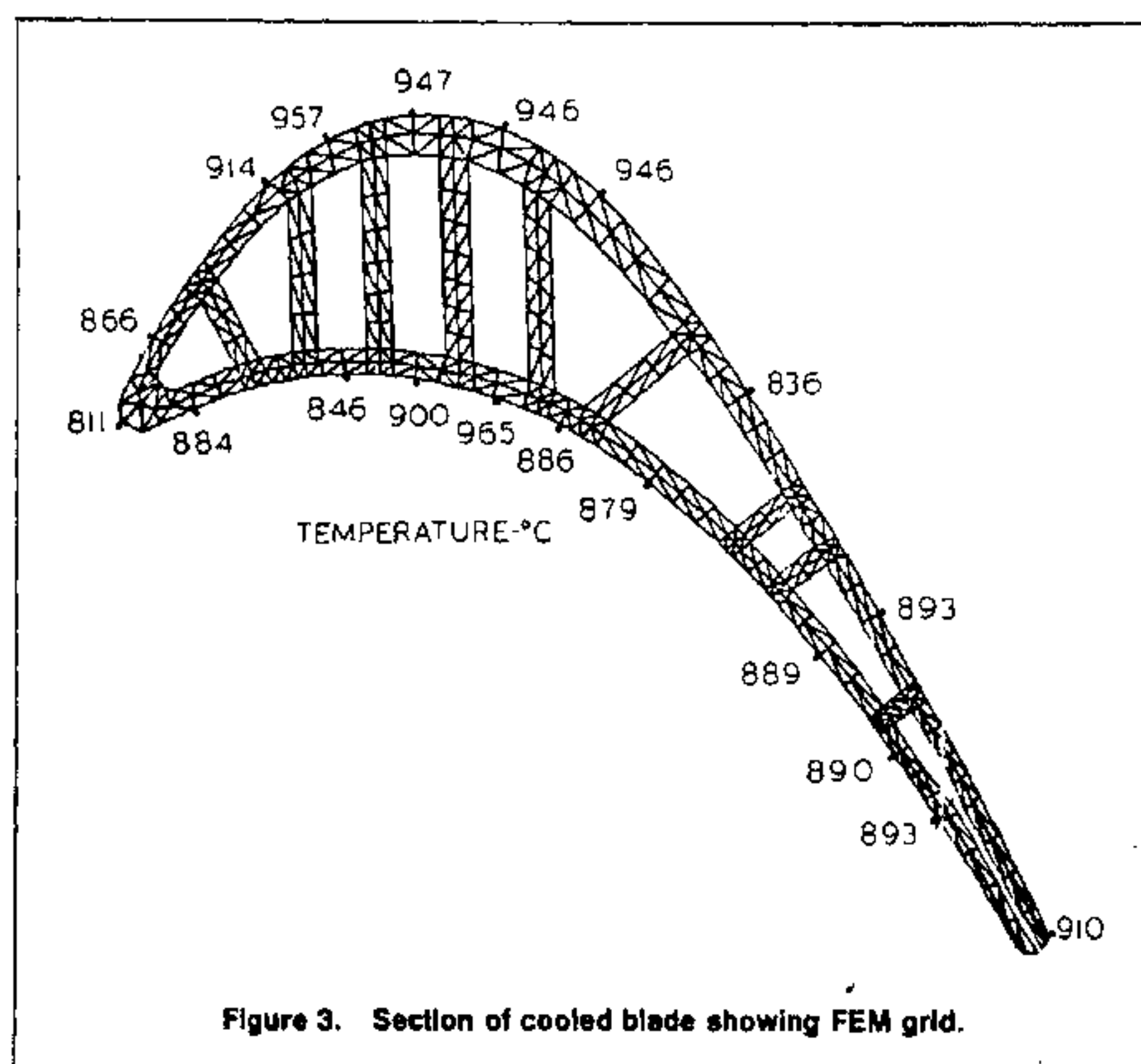
testing as the HP turbine rotor stage in an actual engine at the GTRE test stand.

#### Squeeze film dampers

The adaptation of squeeze film dampers at the bearing supports has been accepted as an effective method for alleviating high engine vibrations, thereby enhancing the life and the general mechanical integrity of the engine. In the case of the engine being developed at GTRE, the thrust bearing at support 1 (at the low pressure compressor) is subjected to radial and thrust loads. A damper was designed for this support by providing an oil film in the radial and axial directions. The rotor vibrations are caused when the rotor centre is no longer in a circular synchronous orbit around the geometric centre. The squeeze film, characterized by its land width, film thickness, end seals and oil supply pressure was optimized to successfully attenuate the vibration so as to enable the engine to accelerate without vibration and to cross over critical speeds (Figure 4). A design data bank for such squeeze film dampers is now available so that reduction in engine rotor vibration levels could be realized using the optimized damper configurations.

#### Incipient failure detection device

During the development testing of the engine, there was a need to monitor the condition of the hot parts of the engine,





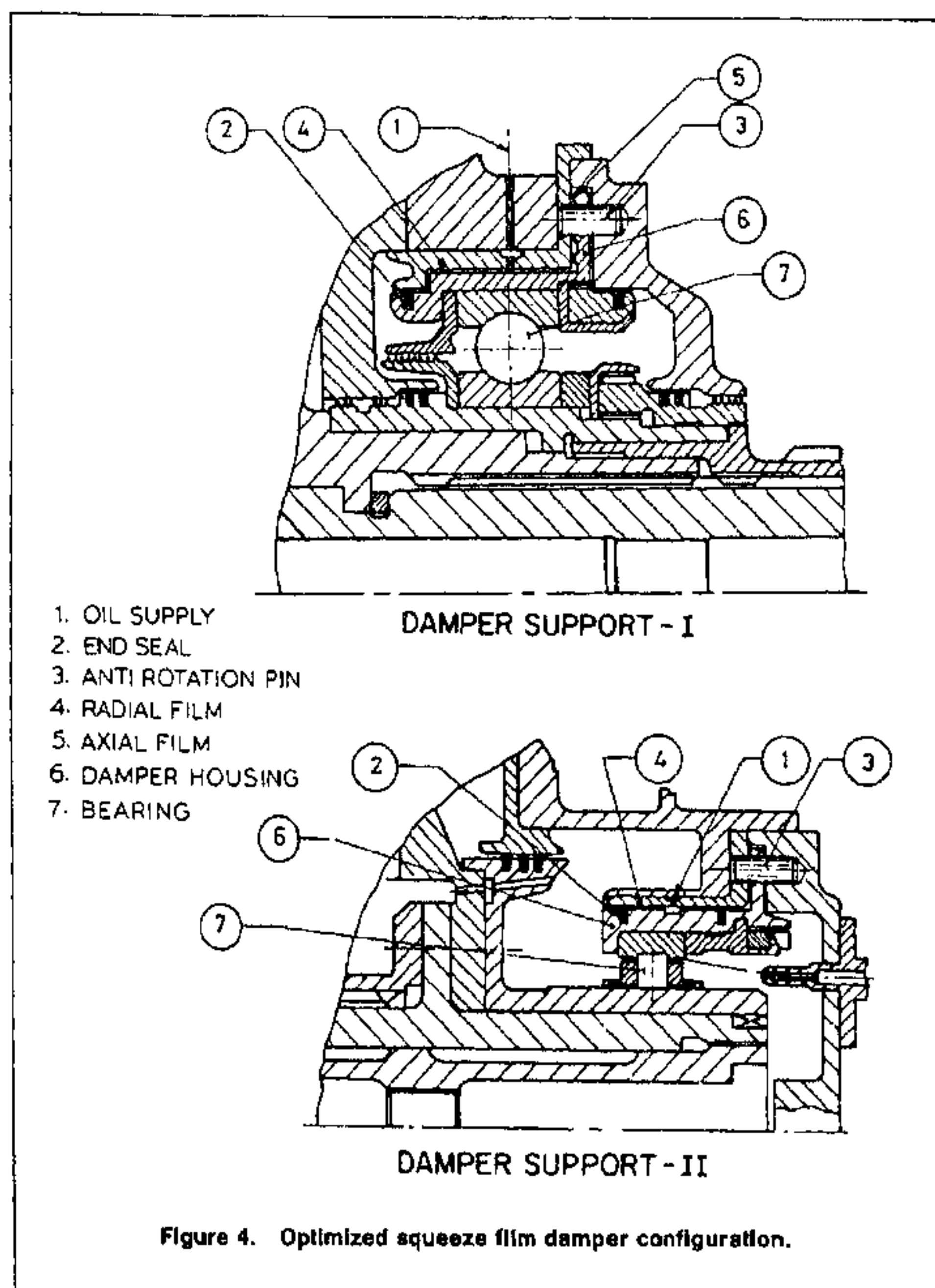


Figure 4. Optimized squeeze film damper configuration.

such as combustion chamber, turbine nozzle vanes, turbine discs, afterburner flame stabilizer and variable nozzle flaps. It was necessary to monitor and analyse any debris since structural failure of any of these components would be liable to cause a disastrous failure of the whole engine. More so, on a development engine where considerable investment has been made in instrumentation, data acquisition and innovations in components, this incipient failure detection system would predict engine distress by evaluating the increased electrical activity in the engine exhaust system, enabling the engine to be shut down before a catastrophe. It is based on the principle that, in any metal part that is on the verge of failure, particles are generated by spallation of fine metal powder when surface cracks propagate or by rubs on the seals, or rubbing of blades on casings due to reduced tip clearances. The particles are carried downstream into the exhaust pipe and nozzle. The device, in question, consists of electrostatic probes mounted on the exhaust (jet) pipe of the engine.

The probes used in this case were a triangular grid within an outer circular grid (Figure 5). The particles released from the metal while passing down the gas stream get charged electrically, and while impinging on the grid described above they cause voltage spikes in it. The voltage spikes are processed and counted on a calibrated digital counter. An increase in the rate of the counts beyond a pre-determined threshold level is an indication of an impending failure of the part within the engine. This device is being constantly used for every test run of each engine at GTRE.

#### Measurement of rotor tip clearance

In axial flow compressors the clearance between the tip of the rotor blade and the casing is a critical parameter that determines the efficiency of the compressor and its stability margin (its ability to deliver air in a smooth through flow). When the compressor is unstable the flow is highly fluctuating and results in severe vibrations of the blades causing catastrophic fatigue failure.

On the other hand, a too tight clearance for better efficiency could cause the rotor blade to rub on the casing (as described above) during rapid accelerations and decelerations due to flexure or bouncing of the flexible drum compressor rotor. Thus a precise measurement of the rotor tip clearance during transient/steady state running of the engine was required. The problem was solved by mounting on the casing a capacitance transducer, which was non intrusive in nature. A hole was drilled on the casing such that the mid chord of the blade section of the tip was viewed by the transducer. The sensitivity of the device increases rapidly as the proximity of the pick up to the blade tip increases, making it ideally suited for accurate measurement of very narrow clearances. Digital circuitry along with memory is used to accurately convert the output into clearances in engineering units. The device is calibrated, so that a print out is obtained on-line of clearance vs engine speed.

#### Measurement of rotor blade vibration

A similar non-intrusive capacitance transducer has been used to measure blade vibration. Blade vibration is expressed in terms of  $af$  value, where  $a$  is the semi peak to peak amplitude and  $f$  the frequency of vibration. The device computes  $af$  values from measurement of instantaneous blade tip velocity. The instantaneous tip velocity is the sum of tip speed due to rotation and the velocity of blade vibration. Digital circuitry extracts the modulation due to blade vibration. The blade vibration of each blade in a blade row is displayed by a raster. This method of blade vibration measurement is simpler and more direct than the conventional method of measurement through strain gauges mounted on the blade, as it relates directly to the fatigue life of the blade.

#### Monitoring compressor surge

The phenomenon of compressor surge occurs in an engine when the air flow into the compressor is distorted or when during sudden engine acceleration the blockage downstream of the compressor is so large as to cause separation of flow on the compressor air-foils. Such separated flow usually occurs in isolated



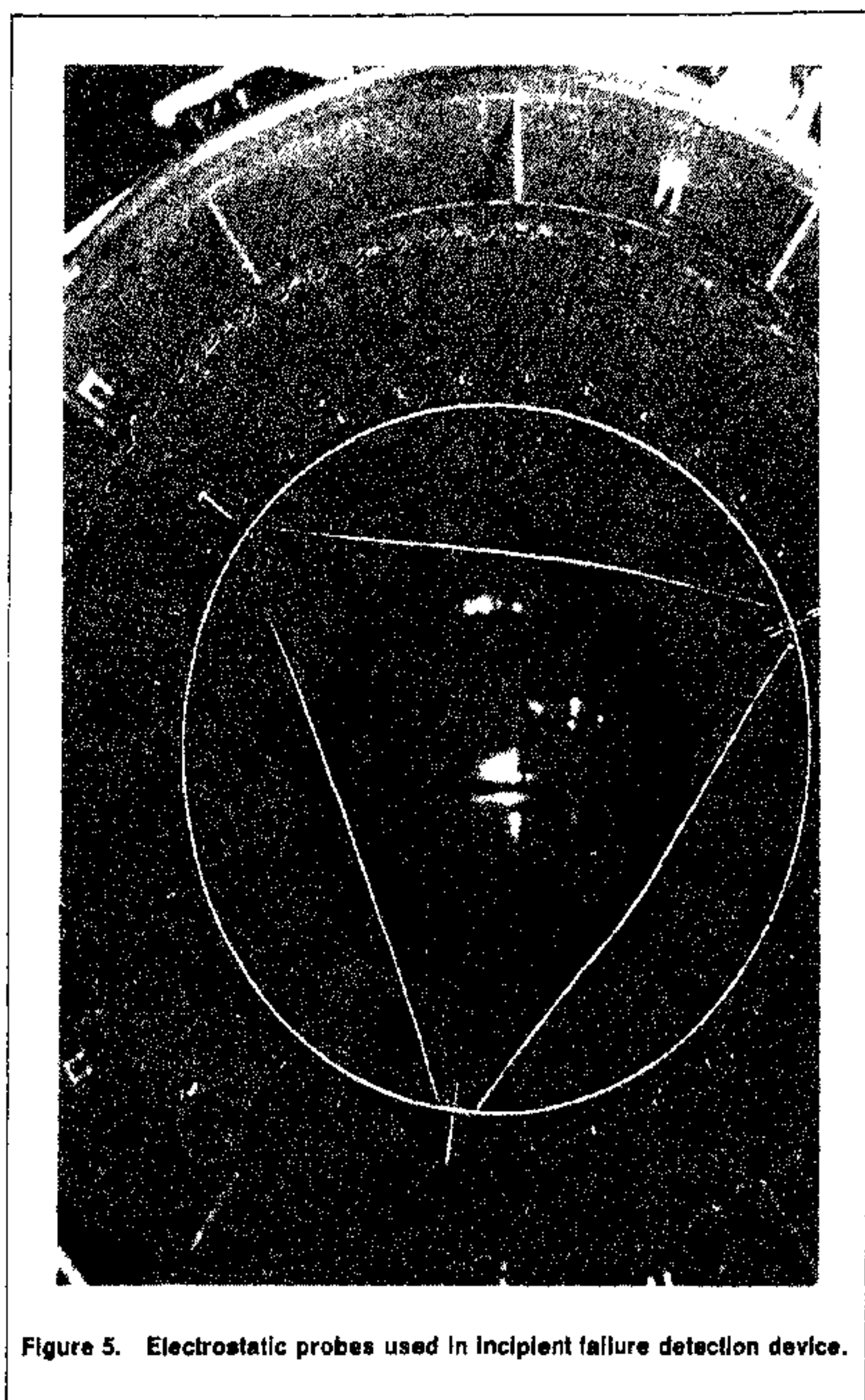


Figure 5. Electrostatic probes used in incipient failure detection device.

pockets along the compressor annulus and leads to stalling of the airfoils (rotating stall). As a result, the through put of the compressor is unsteady and if the blockage is not relieved leads to a total reversal of flow in the compressor. This is called surge. In a two spool engine (called thus because of its two concentric shafts, one driving the low pressure spool and the other high pressure spool), there is a larger susceptibility to surge the LP compressor while decelerating and the HP compressor while accelerating. Hence, an adequate surge monitor system was essential during the development testing of the GTRE engines. The system developed at GTRE consisted of a Z-80 microprocessor, 16 K RAM, 4 K ROM, diskette drive, analog input subsystem and a colour graphic terminal.

The analog input subsystem caters for 8 channels of input parameters and consists of transducers, signal conditioners, multiplexer, sample and hold amplifiers and analog to digital converter (ADC). The signals from these tempe-

rature and pressure transducers are fed to signal conditioners whose output is about 10 V for maximum range of each parameter. The outputs of the signal conditioners feed into a solid state multiplexer which scans them sequentially. The ON resistance of the multiplexer is about 300 ohm and transfer accuracy is 0.01%. The output of the multiplexer goes into a sample and hold amplifier. This amplifier samples each signal and holds it till the ADC converts the analog signal into a 12 bit digital signal. The acquisition time of the sample and hold amplifier is  $1\mu\text{s}$  and the droop rate  $1\mu\text{Vs}^{-1}$ . The conversion time of the analog to digital converter is  $25\mu\text{s}$ . The output of the ADC is fed to the microprocessor system. The rig tested compressor characteristic (called compressor map) is drawn on the colour graphic terminal which is connected to the microprocessor through the RS-232 C interface. The compressor characteristic shown in Figure 6 has the engine operating line also traced on it. The transient excursions of the

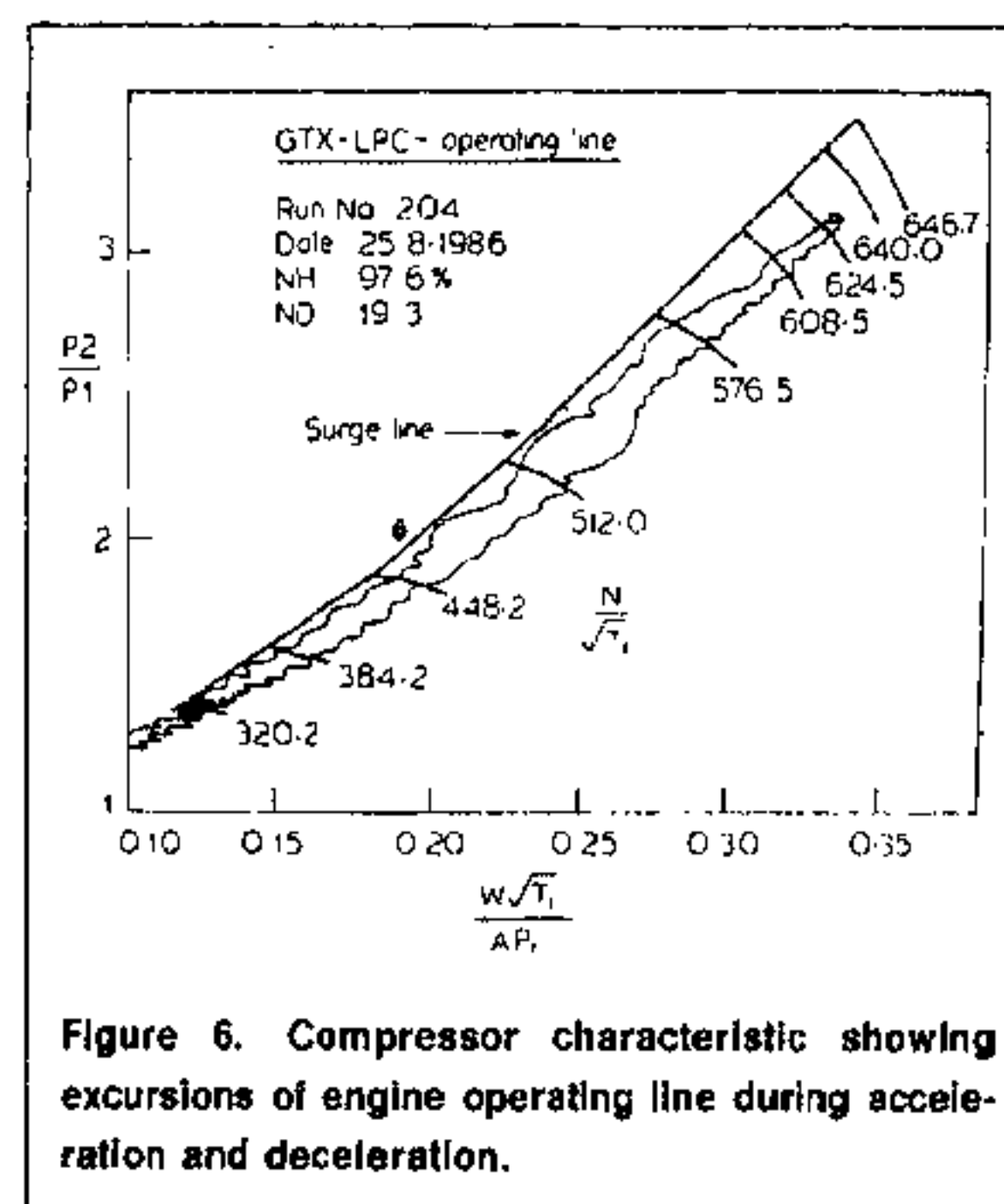


Figure 6. Compressor characteristic showing excursions of engine operating line during acceleration and deceleration.

operating line during engine acceleration and deceleration should not cross the surge line for smooth surge-free operation. The data of the compressor characteristic are stored in the read only memory (ROM). On command from the keyboard of the colour graphic terminal, this characteristic is drawn on the display. During the engine test, the compressor delivery pressure is averaged from readings taken along its annulus. Values of the inlet pressure, compressor speed (NL or NH) and compressor air inlet temperature are measured, averaged for five scans and processed. The parameters  $P2/P1$  (pressure ratio across the compressor) and mass flow  $W\sqrt{T}/AP$  are computed from a look up table. These values are computed for every cycle, repeated once in 100 ms. The display print out in Figure 6 shows a trace during a typical acceleration and deceleration of the engine. Such a system is a very useful tool in optimizing the fuel injection rates of the engine as also the jet nozzle area, so that the typical 5 s or less acceleration rate for a military engine can be met without any harmful effect on the engine.

In conclusion, the work on research and development in gas turbine technology has resulted in the development of excellent analytical and experimental tools. The benefits of innovations in computer science, electronics and mechanical engineering have been exploited towards a better understanding of the complex technology.

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