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# THE DEVELOPMENT OF THE AVRO ORENDA JET PROPULSION ENGINE

by

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# THE DEVELOPMENT OF THE AVRO ORENDA JET PROPULSION ENGINE

The paper describes the development history of the AVRO Orenda jet propulsion engine. This engine is the first aircraft engine to be put into quantity production in Canada. Its rated thrust is in excess of 6000 lbs., and its fuel consumption is about 1.00 lbs./lb. thrust/hr both under sea level static conditions. The engine has an axial flow compressor, 6 combustion chambers, single stage turbine and an exhaust assembly. A number of development problems are discussed such as blade vibration, oil consumption and performance improvement. Flight testing and experimental equipment are also covered. The engine has been brought from the initial design stage to production in a period of $4\frac{1}{2}$ years, during which time 5000 hours of bench testing and 150 hours of flying have been completed.

## INTRODUCTION

The beginnings of the Avro Orenda extend much further back in history than the discussions which produced the first design layout. To appreciate the project fully it is necessary to start in 1942 when reports about the Whittle jet-propulsion engine began to reach Canada. Just prior to that time the Canadian government had seen the country's aircraft industry slowed down and sometimes halted awaiting engines from abroad for there was no native aircraft engine industry. The information received through the heavy wartime cloak of security seemed to indicate that the new engine was light, of great power, easy to design and simple to manufacture. In fact it appeared to offer the possibility of establishing a Canadian engine industry without the costly delay which would have been required to set up an internationally competitive piston engine enterprise.

Late in 1942 senior officers of the Royal Canadian Air Force investigated the situation and as a result a technical mission was sent to Great Britain early in 1943 to study the new development. Some of the enthusiastic reports proved to be groundless. However, the group recommended that Canada could make an important contribution to the British program by establishing a cold weather experimental station to test jet engines under temperature conditions which prevail at altitude or even at sea level in the Arctic. The project was assigned to the National Research Council of Canada and has been reported before the Society and elsewhere (Ref.1 & 2). The mission also suggested the creation of a gas turbine research establishment with the thought of setting up a development and manufacturing organization at a later date.

The former step was taken in 1944 when the government set up a crown company Turbo Research Limited to carry out all Canadian research work on gas turbine engines. The company took over the National Research Council's cold weather testing activities and began to recuit a research and design staff.

This team was selected and trained, some being sent to England for further training. Plans for a research establishment were started. But before they were complete the scope of the activity was broadened to include actual engine design. A provisional engine specification was agreed upon with the Royal Canadian Air Force and design work started. At the same time tentative layouts of a development and manufacturing plant were done.

This phase continued until the spring of 1946. At that time the government, in furtherance of its policy of turning over a large number of government controlled enterprises to private industry, assigned the task of design, manufacture and development to A.V. Roe Canada Limited, a member of an English group of companies which had produced many famous aircraft and engines. The allied fundamental research work was turned over to the National Research Council to whom the cold test facilities reverted.

## BACKGROUND

At the time the Turbo Research Limited project was transferred to A.V. Roe Canada Limited a small engine named the "Chinook" was in an advanced state of design. In order to confirm design assumptions, establish manufacturing techniques, get development experience, and educate subcontractors a decision was made to build this engine on a development basis. The work slowly gained momentum as precision machinery became available, manufacturing techniques were worked out, and foundries overcame their initial difficulties with complicated castings. At the same time laboratory space was allotted, test houses designed and the power plant of a wartime explosives plant at Nobel, Ontario was acquired for use as the nucleus of a full scale test plant for compressors, combustors and turbines. Much of the test equipment designed by Turbo Research Limited was manufactured in the Avro shops and installed in the experimental laboratories. This work culminated in the first running of the Chinook Engine on March 17th, 1948. In the ensuing twenty months over 1000 hours were logged on Chinook engines and the thrust was increased from the original design value of 2600 lbs. to a figure well over 3000 lbs.

## HISTORY

In the late summer of 1946, twenty months before the Chinook first ran, the design group were presented with a problem which made the current manufacturing and organizational difficulties seem insignificant. The Royal Canadian Airforce requested AVRO to develop an engine for the twin-engined long range fighter being designed by the Aircraft Division of the company. The Specification called for an engine of a thrust equal to that of the largest engines on the drawing boards of British and American companies.

With the Chinook a decision was made to name engines after Indian spirits. Following this tradition the new project was christened the "Orenda" - an Iroquois spirit whose presence in an object or person confers power. Design studies were started in September 1946, and finalized at the end of the year when detailing was commenced. Drawings began to appear in the shops in May 1947 but the drawing issue was not complete until January 1948. There were two reasons for this seemingly long time required to issue the design. The first was a policy decision to do the detail design extremely carefully in order that a minimum of development difficulties would be built into the prototype. The second was the result of a crisis which arose in the summer of 1947 when development work at the English firm commissioned to design the combustion equipment, indicated that a longer combustor would be necessary to assure reasonable performance. This discovery had a substantial effect on design progress as it necessitated a complete redesign of the backbone casting, shafting and turbine bearing to accommodate the lengthened combustion chamber.

The first engine was assembled and delivered to the test house on February 8th, 1949. Two days later after preliminary motoring trials, the first attempted start was successful. In the ensuing weeks the wisdom of the careful design policy was proven for there were no immediate operating problems, in fact the engine logged its first 100 hours in 8 weeks time and accumulated almost 1000 hours of running with only minor rebuilds in 8 months. The engine as it is now being produced does not incorporate any basic differences from the original design.

The highlights of the development history are recorded in chart form in Table I.

TABLE I.

| | | | |
|---|---|---|---|
| Layout commenced | - Sept.3rd,1946 | 500 Hrs.running completed | - July 21st,1949 |
| Layout design finalized & detailing commenced | - Dec. 6th,1946 | 1000 Hrs.running completed | - Sept.23rd,1949 |
| Detail drawing issue commenced | - May 1st,1947 | 2000 Hrs.running completed | - Feb. 10th,1950 |
| Detail drawing issue completed | - Jan.15th,1948 | First official flight clearance at design rating | - Mar. 2nd,1950 |
| First engine delivered to Test House | - Feb. 8th,1949 | | |
| First run | - Feb.10th,1949 | First flight (in Lancaster flying test bed) | - July 13th,1950 |
| 100 Hrs.running completed | - Apr. 4th,1949 | First flight in service type aircraft | - Oct. 5th,1950 |
| Engine first ran at design take-off speed | - May 3rd,1949 | First 100 hours flying | - Oct. 20th,1950 |
| Engine first delivered design performance | - May 10th,1949 | 5000 Hrs.running completed | - Feb. 5th,1951 |

ORENDA HISTORY CHART

## DESCRIPTION OF ENGINE

### GENERAL

The Orenda is an axial flow jet engine having 10 compressor stages, six combustion chambers, a single stage turbine and an exhaust cone. Under sea-level static conditions the version now in production has a thrust in excess of 6000 lbs., and a specific fuel consumption of about 1.00 lb. per hour per pound of thrust. The dry weight is about 2500 lbs. The nominal diameter is 42 inches and the overall length is very close to 10 ft.

Two mounting arrangements are possible. The first is a four point suspension with two trunnions on the turbine nozzle box and two mounting pads on the centre casting. The second is a three point pick-up having two trunnions on the centre casting and an adjustable strut on the backbone casting. A diagramatic section of the engine is shown in Fig.1 and an exterior view in Fig.2.

## COMPRESSOR

The compressor intake is a magnesium alloy casting having an annular air entry around a housing which contains the drive gear box for the engine auxiliaries and the compressor front bearing. The housing is supported by six struts which contain the auxiliary drive shafts, lubrication lines, thermocouple leads, and starter cables. The electric starting motor is mounted on the housing and is covered by the entry bullet.

The rotor is composed of discs mounted on an internal drum. The first nine stages have aluminum discs while the tenth disc is steel. A stepped sealing ring projects from the rear of the tenth stage disc into a gland mounted on the centre casting. A small flow of air is permitted to escape past this seal and is used for cooling the rear face of the turbine disc. The blades are retained in the discs by a form of "fir tree" fixing for the first three stages and dovetails for the remaining ones. The first, second, third, and tenth rotor blades are steel. The rest are an aluminum alloy. Rotor and stator blades are unshrouded. The rotor is supported on a bearing in the intake casting and on the centre bearing in the centre casting.

The compressor stator casings are of magnesium alloy. The stationary blades are mounted by dovetails in rings which are retained in place by lips on interstage spacers which in turn are bolted to the stator casing. Provision is made for bleeding air for engine and aircraft purposes at the second, fifth, and eighth stages.

## CENTRE SECTION AND BACKBONE

The centre section is an aluminum alloy casting containing the diffusing ducts leading from the compressor to the combustion chambers. The centre bearing assembly is mounted inside the centre section. The rotor is retained axially and the rotor thrust absorbed at this point. The centre bearing assembly consists of two bearings with accurately ground spacer rings between them which permit the bearings to share the thrust load. The bearing housing is spherically ground on its outer diameter to allow the bearing to accommodate angular misalignment of the main shaft due to aircraft manoeuvers. The bearing housing is spring loaded against a composite rubber thrust ring which deforms slightly with angular misalignment but maintains the axial location of the rotor assembly.

The backbone is a casting of light alloy joining the centre section and the turbine nozzle box. The turbine bearing is mounted on an internal flange at the rear of the backbone.

## COMBUSTION CHAMBERS

The six combustion chambers are bolted to the centre casting at the front and are a sliding fit in the nozzle box at the rear. They are arranged around the backbone, Interconnector tubes are provided between chambers to allow crossfiring on light-up. Torch igniters are mounted on two interconnectors for ignition purposes. These consist of a small fuel atomizing nozzle and a spark plug. The combustion chamber consists of a cast aluminum expansion section and a mild steel outer casing with a high temperature alloy flame tube mounted within. The atomizing burners are mounted on pads on the diffuser ducts and project into the combustion chamber.

## NOZZLE BOX

This assembly consists of a welded structure of steel castings and pressings. The turbine nozzle blades are mounted into it as well as the transition ducts which lead the products of combustion from each chamber to the nozzle annulus. The shroud ring which surrounds the turbine rotor blades is attached to the nozzle box.

## TURBINE AND DRIVE SHAFT

The turbine consists of an austenitic steel disc with an integral stub shaft. The blades of nickel-chromium alloy are mounted on its periphery by "fir tree" fixings. The turbine bearing is mounted on a sleeve on the stub shaft section. The stub shaft is attached to the main shaft which drives the compressor through a splined coupling near the centre bearing. The front face and rear faces of the turbine disc are cooled by fifth stage air and tenth stage air respectively. The turbine bearing is cooled by second stage air.

## TAIL CONE

This assembly is fabricated largely from stainless steel sheet. It consists of an outer cone and an inner bullet supported front and rear by four tubular struts covered by a fairing. Tenth stage air is conducted through the front struts and forward to the front of the bullet. From here it flows outward between the face of the bullet and the turbine disc escaping into the gas stream at the disc periphery. The outer surface of the tail cone is insulated by a fibre glass and foil blanket protected by aluminum covers.

## FUEL SYSTEM

The fuel system is the means of controlling engine output. The pilot's throttle is connected to an altitude-sensitive scheduling-type flow control unit which varies the delivery of two engine-driven pumps through a servo-system to maintain engine speed constant for any throttle setting irrespective of altitude. The pumps have integral overspeed governors. The remaining fuel system components are:-

1 - Solenoid operated reducing valve to supply fuel to the torch igniters.
1 - Flow distributor to meter the flow to the burners.
6 - Double orifice burners arranged to allow good atomization over a wide flow range.
1 - Dump valve.

A high pressure shut-off cock for the pilot and a low pressure filter are incorporated in the flow control unit.

## LUBRICATION SYSTEM

The lubricant is supplied by the oil pump to the rotor bearings, gearboxes, front bearing seal and drive shaft flexible coupling through a ring main. Pressure is kept constant in the main by a pressure control valve which returns excess oil to the reservoir. Separate scavenge pump elements are used to pump lubricant from the following sumps:-

1. Rear bearing
2. Centre bearing
3. Front oil drains
4. Flexible coupling

These discharge into the oil reservoir which is a tank of 13 Imperial pints capacity. The lubricant returned from the rear and centre bearings is cooled by a heat exchanger which uses incoming fuel as a coolant. The ring main system operates at 15-18 p.s.i. The engine oil consumption is about 1 pint per hour.

## COOLING AIR SYSTEM

Air is bled from the compressor at the second, fifth and tenth stages and used for cooling as follows:-

1. Second Stage Air
   (a) backbone cavity
   (b) turbine bearing
   (c) nozzle box

2. Fifth Stage Air
   (a) centre bearing
   (b) front face of turbine disc

3. Tenth Stage Air - rear face of turbine disc

## STARTING

Starting is effected by a 32 volt electric motor housed in the nose bullet. An over-riding clutch disengages the starter motor when the engine reaches self sustaining speed. The rest of the starting system consists of the booster coils for the torch igniter spark plugs and the control circuit for the torch igniter reducing valve. An external sequence control is necessary to ensure that starting current, fuel for the torch igniters, and power for the torch igniter spark plug are provided at the correct times to permit clean starts.

## MECHANICAL DEVELOPMENT

The early development running showed that the engine lived up to expectations from the performance standpoint, the predicted thrust being obtained with design speed and jet pipe temperature at an early point. Starting was excellent, and acceleration reasonable. The engine showed a slight instability in the region of 70% of full speed, however, this was not a limitation on test bed running. Oil consumption was very high. Tenth stage stator blades showed a regretable tendency to come off in quantities, and turbine blades developed cracks at the tip near the trailing edge.

Oil consumption continued to give trouble on most succeeding engines, but their performance in this respect was extremely erratic. Before this problem could be tackled it was necessary to get a proper adjustment of bearing oil flows, cooling air flows and air flow to pressurized glands. When these were corrected to give satisfactory bearing conditions the oil consumption still remained high. Initial attempts to localize the oil loss were misleading but, it was finally traced to the turbine bearing area. Several months were taken up in trying different sump and scavenge arrangements, new seal designs and flinger rings. Finally checks made with a very accurately calibrated oil system showed that the scavenge flow was equal to that supplied to the bearing. The only clue seemed to be that oil loss was small at lower speeds or until the engine had been running for sometime. About the same time an adequate bearing test machine became available. The rear bearing assembly operated with negligible loss on the test rig. A thorough check of the detailed design was then made as the evidence seemed to point to a temperature affect and this was confirmed for the only difference between rig and engine conditions was the relative temperature of the components. The trouble was apparent when it was discovered that a steel sleeve supporting the bearing, which was an interference fit in an aluminum casting, was not sealing one of the oil drillings in the casting. Under operating conditions the casting had a greater thermal expansion than had been anticipated thus relieving the interference fit and often permitting oil to escape. The end of the oil drilling was plugged and after confirmatory tests it was announced, with some embarrassment, that the oil consumption problem was solved.

During the course of development the rest of the oil system behaved very well. There were no aeration difficulties. Very little effort was required to get adequate venting arrangements for oil tanks and sumps, although pressurization of the oil tanks produced spectacular results at times before the problem was cured.

The failures of tenth stage stators were initially traced to fatigue cracks which were the result of intercrystalline corrosion Fig.3. Other blades in the engine were also found to be subject to this form of attack. The material used in the later stages of the compressor was changed to a similar material which was more resistant to this kind of corrosion. However, the failures although not as frequent, continued to give trouble, appearing now as ordinary fatigue cracks Fig.4. A study of the resonance conditions of the blade showed that it was being excited by the tenth rotor in the second flexural mode. Strain gauge tests indicated that the blade would be strong enough if made in steel. The change proved to be a satisfactory solution.

The turbine blade cracks Fig.5 originated in the blade tips near the trailing edge. As originally designed the blade had a feather edge which was provided to prevent serious damage in case of tip rubs which might occur due to the small clearances used. A survey of the nodal patterns of the vibration modes occurring in the running range showed that the second complex mode Fig .6 had an area of high bending stress extending right to the tip. Cracks started in the thin feather edge and were propogated along the line of high stress. As the behavior of turbine shroud ring under operating conditions was established it was possible to employ satisfactory tip clearances without the danger of rubs. Since the need for the feather edge had disappeared it was deleted in order to strengthen the blade tip. This proved to be an effective remedy.

As more running hours were accumulated further difficulties began to crop up. Almost all engines were inclined to heavy rubbing of the tenth stage peripheral seal with complete loss of its effectiveness. This was thought to happen on running down from high speeds when the pressure behind the tenth stage disc decreased rapidly but a high pressure could remain momentarily between the ninth and tenth discs causing the disc to flex and the stepped sealing ring to foul its gland. The interstage cavity was vented through the tenth stage disc which cured the trouble.

Considerable work was involved in the development of the flexible thrust ring of the centre bearing assembly. This feature was incorporated in the design to look after angular misalignment resulting from flight manoeuvres. A similar design had been used on the Chinook engine and proved to be an excellent method of obtaining the required degree of angular flexibility.

As originally designed the rings were of soft rubber with slotted steel corner braces. They suffered considerably from extrusion of the rubber around the edges of the corner braces, through the slots in the corner braces and from the unbraced corners Fig. 7. The resulting collapse of the ring permitted the compressor to move forward and foul the stator assembly. This was partially remedied by placing a strip of tape between the corner brace and the rubber. A further variation consisted of binding the ring with nylon tape. However, it was difficult to cure the ring to the required dimensions even under pressure as the tape binding tended to make the ring oval after curing. Under operating conditions the ring would assume the rectangular shape intended with reduction of its axial dimension and compressor fouling could still occur. The problem was finally solved by using a composite ring with a hard exterior and soft core without binding which did not alter dimensionally during operation.

The front bearing seal went through several stages of development to produce a satisfactory design. The seal in question is introduced behind the front bearing of the compressor rotor to prevent lubricant from escaping into the cavity immediately in front of the first rotor discs. This space is at a depression of 1.5-2 psi as it is connected to the compressor air passage ahead of the first stage rotors. The bearing area is at atmospheric pressure. The original seal was a carbon ring held in place against a cast iron rubbing surface by a single helical spring with a pressurized gland on the shaft as a further seal. This arrangement did not work well as the single helical spring did not bear evenly on the carbon ring. The result was uneven heavy wear of the sealing surfaces. Subsequently a carbon ring seal supported by a multiplicity of small springs was adopted. At this time it was decided to delete the gland as an unnecessary complication. The new seal worked well on rig tests but was erratic on the engine. It was thought that the differential thermal expansion of the rotor and stator casing was causing the trouble by permitting the sealing surfaces to move apart. This was checked by introducing wear plugs near the seal which showed that little differential expansion was taking place. When this was established a large number of seals were examined and it was discovered that the spring rate and travel of seals as supplied was varying considerably from specification. When these were brought under closer control the trouble disappeared.

One of the more difficult problems did not become apparent until almost a year of testing had been done and engines had logged about 2000 hours of operation. Then a single seventh stage rotor failed followed by several more within a short period, some on engines with relatively few hours of running Fig.8. Failures continued to occur in apparently random fashion thereafter. Such failures appear as fatigue cracks. This trouble will serve to show the many steps which are often necessary to get a satisfactory solution to development difficulites in aircraft engines. They are the result of the high vibratory stresses induced in the blade by the coincidence of one of the natural frequencies of the blade with an exciting force such as that caused by the passage of the blade through the wakes of the preceding row of blades.

The first step is a precautionary one taken in the design stage. The fundamental flexural and torsional critical frequencies of all blades are calculated. A check is then made to determine that these are not in resonance with known exciting frequencies within the engine operating range. As it is very difficult to calculate the frequencies of higher modes of vibration which could cause trouble these are found experimentally when the first blades are manufactured. From this information an "interference diagram" is plotted for each blade. A typical diagram is shown in Fig.9. The frequency of various exciting forces is plotted against engine speed and the critical frequencies of the blade are also plotted. At the speed where the lines cross the blade will be in resonance with the exciting frequency.

At the time the blade frequencies are determined the vibrational pattern of each mode is studied and recorded to assist in the identification of any failures. It is usual to do about 150 tests of this type for an engine such as the Orenda. A set of vibration patterns is shown in Fig.10. The lines and dark areas indicate nodes.

As the failures being considered were somewhat sporadic and as there was a great spread between the lengths of running time which produced failures it was suspected that the material might be at fault. Consequently, a survey was made of the physical properties of the batches of material from which the failed blades had been made and these were compared with the properties of batches which did not produce any failures. This threw little light on the situation. Since the experimental engines had been used for a variety of tests a survey of the running history of all engines was made by plotting the operating time in each 100 rpm speed range. It was not possible to draw any conclusions from a comparison of histories of engines which had failed blades and those which had not. The study was narrowed down to a comparison of histories 10 hours, 5 hours and 1 hour prior to failure with the idea of establishing speed ranges which caused failure. This also proved abortive.

As more failures occurred they were tentatively identified from the nodal pattern surveys as being caused by either second torsional mode or the first complex mode both of which occurred within the running range, but it was still not possible to explain how some engines could run several hundred hours without failure and others would fail in less than one hundred hours even when the known scatter of the endurance properties of the material was taken into account.

Failed blades were carefully examined for manufacturing flaws and inconsistencies such as variations in thickness and trailing edge radius without result.

As soon as it was apparent that the failures were not isolated ones a decision was made to study the problem using strain-gauge techniques. This involved the development of a slip-ring unit to transmit the minute strain gauge signals from the rotor at high speeds without electrical distortion. Several months of laboratory work and engine testing were required before an adequate slip ring unit, proper instrumentation and reliable wiring methods were established. It was then possible to determine the relative magnitudes of the stresses in the various resonances and also the width of the resonance bands.

At the same time the problem was being attacked using more ad hoc methods by running engines to a test schedule which consisted of equal operating periods at 25 rpm speed increments with frequent inspection of the blades through viewing ports in the compressor casing in order to determine the speed range in which failures occurred and the length of time required at speed to produce failures. The strain gauge tests and engine running confirmed that both suspected resonances were contributing to the trouble and established a speed range which should be avoided in order to prevent fractures until a redesigned blade could be manufactured and proven. As capacity for manufacturing new blade designs was already strained, several interim solutions were tried using the existing blades. These reduced the incidence of failure considerably. The first was a change in the number of stator blades preceding the rotors in which failures were occurring. The sixth and seventh stators originally had equal numbers of blades in each row. It was thought that this might be causing trouble as both the leading and trailing edges of the blade would receive impulses at the same time thus increasing the coupling of the exciting force. The number of blades in the sixth stator row was increased 10% to correct this condition and to move the resonance speed away from the cruising speed range. The results of this change were beneficial as far as the 7th rotor blades were concerned. However, it had the effect of causing 8th stage rotor failures after long operating periods. The second interim solution was introduced when it was determined that the relative indexing of the 7th rotor row to the preceding rotor rows had an appreciable effect on blade life. As originally built the relative positions of the blades in the various rotor stages was quite random. During the special engine tests mentioned above one engine ran for several hundred hours without failure while another failed blades consistently with a few hours of running. Both engines were carefully examined for component variations, the only apparent difference being the indexing of the rotor blades. When the method of blade indexing was reversed on the two engines their blade breaking abilities did also. This largely explained the wide variations in running time to failure. The favourable indexing was then adopted as standard for all engines. Further studies with controlled indexing variations are now proceeding using strain gauges to get quantitative information to guide future work.

The final solution of blade vibration problems requires that one of the following courses shall be adopted:-

(a) the damping action of the blade root shall be increased sufficiently to prevent the blade from being overstressed.
(b) the blade shall be strengthened to be able to withstand the vibratory stresses.
(c) the blade shall be redesigned or the exciting frequency altered so that the natural frequency of the blade does not coincide with the troublesome exciting frequency within the engine operating range.

The final method was adopted as the solution for this particular problem.

## PERFORMANCE DEVELOPMENT

As mentioned earlier the Orenda gave little performance trouble as originally designed. The starting characteristics were good. As a hedge against possible difficulties a test stand starting motor was provided which could turn the engine at idling speed and the engine itself was built with blow-off valves to permit the early stages to run unstalled at low speeds. The initial tests showed that an electrical starting motor of reasonable size would be able to accelerate the engine to pull-away speed without trouble and that the use of blow-off valves was unnecessary.

The engine exhibited a distinct change of note at about 60% of full speed and some instability at 70% of full speed. The first phenomenon was associated with the unstalling of the early compressor stages and the latter was an indication that the engine was operating close to the surge point at this speed.

The first step in the performance development was the determination of the correct turbine nozzle guide vane throat area for best performance. This can be quite critical when the nozzles are designed to operate choked as is usual with most jet engines. The optimum nozzle guide vane area is a function of the slope of the mass-flow versus pressure ratio curve of the compressor. If the rate of change of slope is large, small variations in throat area can have an appreciable affect on engine output. Since it is almost impossible to predict the shape of the curve it is necessary to adjust the nozzle area and the ratio between nozzle area and jet area experimentally to obtain maximum output within the established limits of combustion temperature and speed.

This was done quickly on the Orenda which proved to be much less sensitive in this regard than the Chinook where small variations caused considerable change in output as shown in Fig.11.

As mentioned above evidence showed that the Orenda was unstable in a narrow portion of the operating range. Consequently, it was not surprising when the engine showed a tendency to surge during rapid accelerations. It was known that this would cause more trouble at altitude so a thorough investigation of the compressor characteristics was carried out on the compressor rig and verified on the test bed. This work revealed a mismatching between the earlier and later stages of the compressor. Typical mass flow-pressure ratio curves of the first stage and a later stage are given in Fig.12 showing the working point. The same trouble had shown up on the Chinook where it was associated with very low outlet velocities at the tip of the first stage rotor as shown in the velocity traverse along the blade Fig.13. When temperature and pressure traverses were done on the Orenda it was confirmed that the situation was similar. A satisfactory solution had been worked out for the Chinook on a two-stage compressor test rig. Fig.14 shows how the pressure ratio of the first stage was increased by restaggering the inlet guide vanes and first rotors. As this was rather easy to do it was tried on the Orenda. It did achieve better matching but as the relative air velocities were higher on the Orenda, and the restaggering effected an increased angle of incidence on the rotor blades, the overall compressor efficiency was lower with a consequent serious loss of engine performance.

The first two stages of the engine were then redesigned using radial equilibrium principles to get a better velocity distribution. This had the desired effect of improving the matching of the front and back stages of the compressor considerably. In consequence the acceleration was greatly improved. For purposes of assessing the acceleration potential of an engine an overfueling curve is plotted on a speed basis. This shows the ability of the engine to accept fuel over that required for steady state operation at any speed, without surging the compressor or exceeding jet pipe temperature limitations. The extra fuel represents, of course, the energy that is available for acceleration. Typical overfueling curves for engines incorporating the original and revised compressors are shown in Fig.15.

An interesting feature of the original design was the splitter vane located in the diffusers Fig.16. These were incorporated in order to improve the velocity distribution to the combustion chambers. As shown by Lovesay (Ref.3) a poor velocity profile can lead to serious buckling in combustion chambers. The improvement in outlet velocity distribution effected by the splitters is shown in Fig.17. Considerable difficulty was experienced in designing an aerodynamically acceptable splitter which did not suffer fatigue failures due to buffeting of the air stream consequently this feature was deleted. Although some combustion chambers have failed due to buckling it has been after very long periods of engine running and is not considered to be a limitation on the engine.

## COMBUSTION DEVELOPMENT

As mentioned previously the combustion system of the Orenda engine was developed by an English firm which has provided combustion equipment for many British engines. It is not intended to cover the development work done on this form of combustion chamber as it has been done very ably elsewhere (Ref.4 & 5). Suffice it to say that this part of the engine has given very little trouble, both from the standpoint of reliability and performance.

## FLIGHT DEVELOPMENT

The flight testing of the Orenda began in an Avro Lancaster bomber suitably modified for test work Fig.18. The outboard piston engines were removed. The jet engines were mounted in their place in new nacelles Fig.19. A completely new fuel system was installed to serve the jet engines. Nose and tail fairings were added to replace the bomb sight position and the tail turret. New fuel tanks were installed in the bomb bays to allow enough fuel to be carried for long test flights. New cockpit instrumentation, controls, and starting systems were provided for the jet engines. All non-essential military equipment was deleted. An extensive revision of the aircraft services was undertaken to make the aircraft more useful as a test vehicle. A test observers panel was installed to duplicate the instrumentation used in ground testing and an elaborate automatic observer was designed and developed to record all important test measurements Fig.20.

The Lancaster has many advantages as a flying test bed. It has been extensively used for this purpose in Great Britain and its flying characteristics with jet engines are well known. The aircraft chosen was built at Malton during the last war. Consequently, the staff were familiar with it, permitting modifications to be designed, manufactured and installed rapidly. Spares were readily available and service crews knew the aircraft well. An aircraft of this type will carry enough fuel for several hours of test flying. It permits great flexibility in the installation of instrumentation due to its size. It is not dependent upon the jet engines except during take-off. However, its ceiling limits flying to about 33,000 ft. which is far short of the required operational altitude for jet engines. The airframe can only be flown at moderate speeds, and aerobatics are out of the question. It might seem that the first limitation would be a serious one, however at 33,000 ft. 93% of the temperature variation and 82% of the pressure variation between ground level and required maximum operational altitude have taken place. Consequently, test conditions are not as far from those experienced at higher altitudes as it would appear initially.

The Orenda Lancaster has proven to be a very useful development tool. Tests have been done on a number of engines, verifying the method of predicting altitude performance and the variation to be expected between engines of a type. Ignition trials have been made to establish light-up techniques and limitations. Permissible acceleration rates have been determined, the effect of altitude on the control and governing systems have been studied. Combustion stability limits have been checked. The reliability of the lubrication system has been proven.

Further flight testing has been carried out in a North American F86 Sabre aircraft. This has extended the range of testing to very much higher altitudes and flight speeds than has been possible with the Lancaster. One of the notable features of this work has been the exploration of the stability limits of the combustion chamber which has shown itself to be extremely good in this regard.

## TEST EQUIPMENT

It would require several separate papers to describe the experimental equipment and techniques used in the development of the Orenda. The design of such equipment is an art in itself for it must often withstand the same temperature and stress conditions as the engine parts being tested. It cannot be a great deal heavier or stiffer for engine conditions would not be simulated and yet test equipment must be much more reliable than the parts on test. This requires that considerable effort must be expended upon the design of test rigs and experimental plant. Most aircraft engine companies have accumulated a great variety of such equipment over a period of years. When a new problem arises it is often possible to construct an urgently required test rig from equipment already on hand. In fact some very valuable experimental work has been done on such "hay-wire" set-ups. In the case of the Orenda this was more difficult as no laboratory facilities existed initially. Their design and construction was proceeding in parallel with the engine work. This necessitated an extreme degree of improvisation at times when trouble arose. A case in point was the overnight construction of a flow checking rig for fuel system distributors which was lashed up from six modified garbage cans, an oil drum, some galvanized iron and a commercial weigh scale when the program was suddenly beset with a rash of temperature distribution difficulties.

One fortunate circumstance was the acquisition of a steam generating plant which had a large air compressor installation associated with it. The plant had been part of a wartime explosives factory at Nobel, Ontario, about 180 miles north of Malton. This permitted the early erection of compressor and combustion test facilities. The plant has now been expanded to include two compressor rigs, full pressure, low pressure, and altitude combustion test bays, a cascade wind tunnel, high speed air flow rigs, a turbine test rig, a small combustion chemistry laboratory and an experimental machine shop. While this equipment has proven to be very valuable, considerable effort was required to commission the new facilities at the same time as the engine manufacture was proceeding.

An important part of the success of the Orenda development program has been the presence of a strong instrument design and manufacturing group from the very early stages. This has permitted the development of many special measuring techniques and automatic rig controls with associated data plotting gear which have greatly increased the usefulness of some of the test facilities. One phase of this work is being described in conjunction with the presentation of this paper (Ref. 6).

## CONCLUSIONS

It has been stated that there is no device as simple in conception but as complicated in resolution as an aircraft gas turbine. The author hopes the reasons for this have been made clear to those who are unfamiliar with the aircraft industry and that they will have gained from this paper some idea of the scope and magnitude of the work required in developing a successful aero engine. The features which distinguish this branch of engineering from others are:-

1. The painstaking and often rigorous design methods employed in the thermodynamic, mechanical, and metallurgical phases of the work. In this connection it should be noted that it was originally thought jet engines would require less effort to design than piston engines. This was undoubtedly true in the case of the original engines of the type introduced by the pioneer, Sir Frank Whittle. Since then, however, great demands have been made in the way of increased thrust, less fuel consumption, lower weight and better operational qualities. This has tended to produce larger and more complicated engines requiring more engineering effort.

2. The extent of the work required during the development period. This involves the construction of a number of prototype engines to permit problems to be attacked in parallel, and the provision of a large test plant in order to develop individual components to give the maximum in the way of reliability and performance.

3. The elapsed time from the initiation of a new project to the commencement of the production. This is largely due to the two points mentioned above combined with the fact that prototype engines are produced with simple tooling by "job shop" methods and hence take considerable time to construct.

These points are elaborated and indeed the whole field of aircraft engine development covered by Banks in his paper "The Art of the Aircraft Engine" (Ref.7). This admirable treatise, written with the background of the British aircraft engine industry over the last two decades, provides a very convenient yardstick for the assessment of an engine development program.

Coming back to the Orenda project specifically

1. An engine of advanced design has been developed.

2. Major test plant and laboratory facilities have been provided.

3. Experimental and production manufacturing organizations have been created.

4. The shop staff has been trained to do work of an accuracy and complication not previously undertaken in Canada.

All of this has taken place within the space of $4\frac{1}{2}$ years during which time 5000 hours of bench testing and 150 hours of flying have been accomplished. This is the standard which Banks has set as being the desirable time for an established company to bring an advanced engine design into production. It constitutes a very real achievement on the part of Canadian engineers and manufacturing personnel. It has firmly established the Canadian aircraft engine industry in competitive world markets.

## ACKNOWLEDGEMENTS

The author's thanks are due to the management of A.V. Roe Canada Limited, the Department of Defence Production and the Royal Canadian Air Force for permission to present the information contained in this paper. He also wishes to acknowledge the help given by his colleagues in its preparation.

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7. "The Art of the Aircraft Engine" by F.R. Banks, The First Louis Bleriot Lecture, Presented before the Association Francaise des Ingenieurs et Techniciens de l'Aeronautique, Paris, May 1948; published, Journal, Royal Aeronautical Society, London, Vol. 52, No. 449, pages 527-549.

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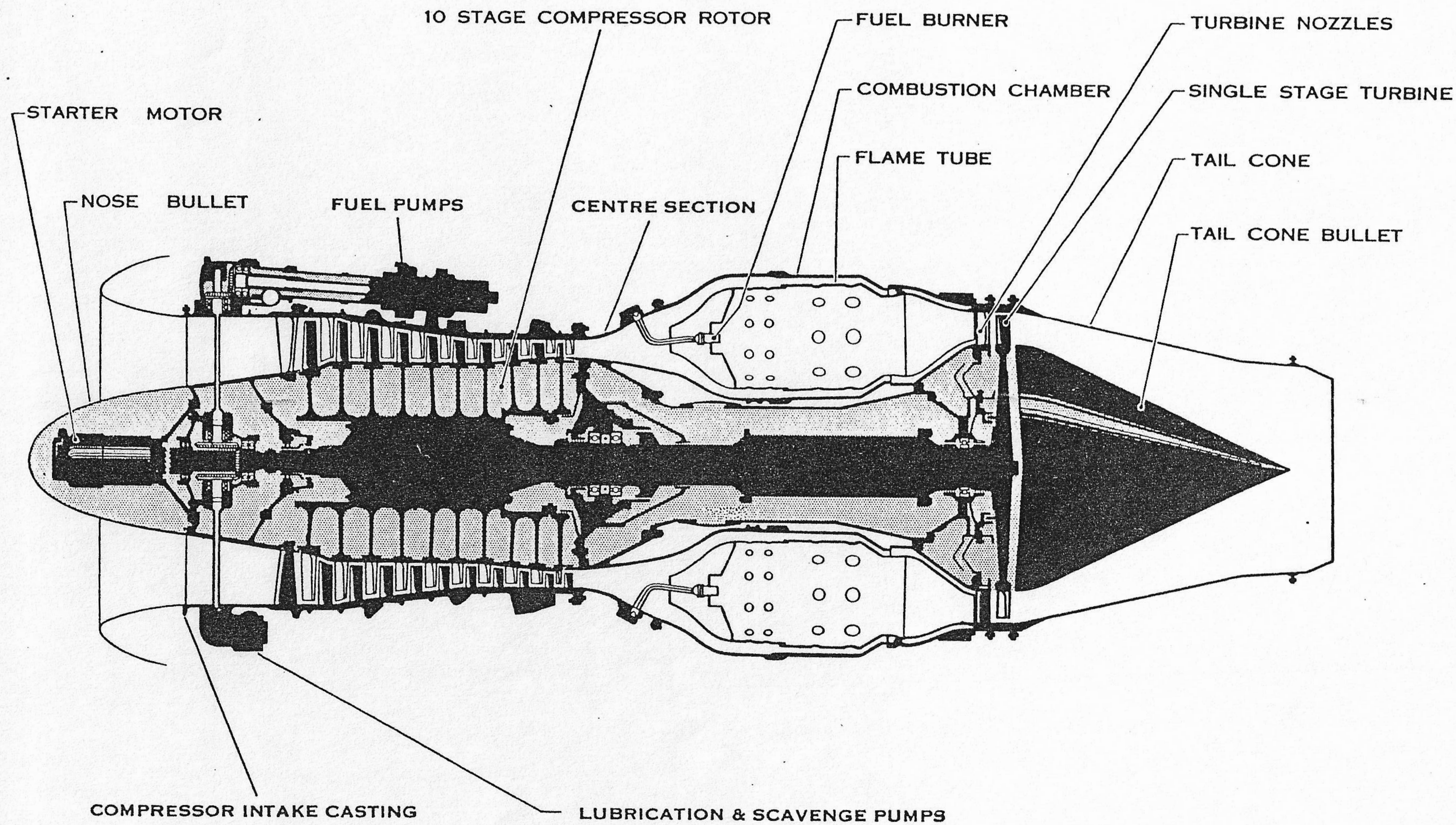



FIG. 1 DIAGRAMATIC SECTION OF ORENDA ENGINE

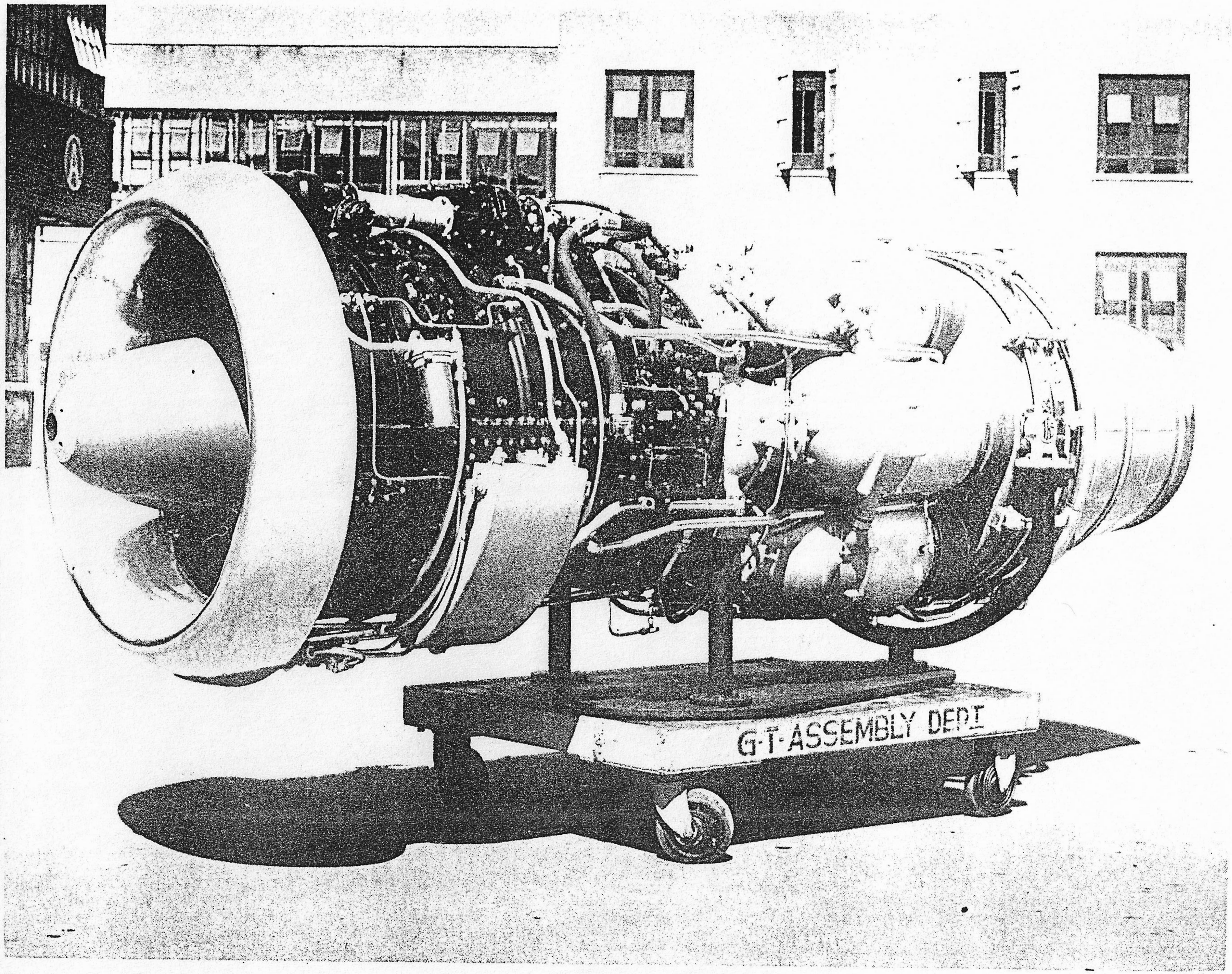



FIG. 2 EXTERIOR VIEW OF THE ORENDA

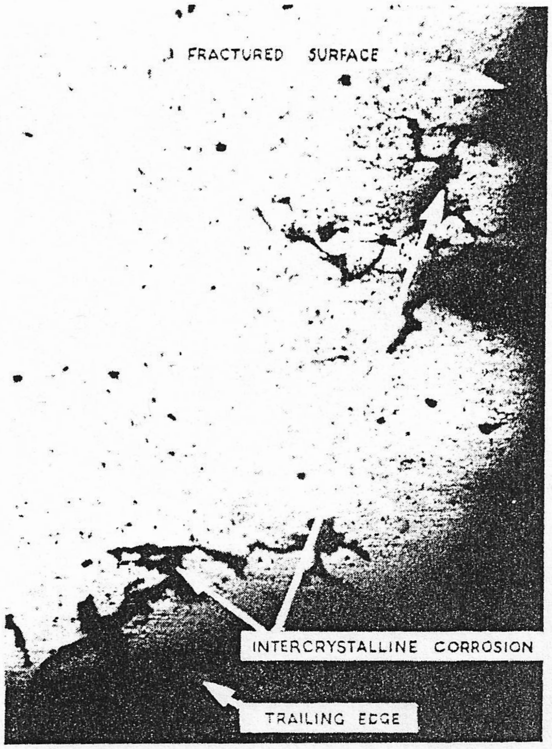



FIG. 3 FATIGUE CRACK DUE TO INTERCRYSTALLINE CORROSION

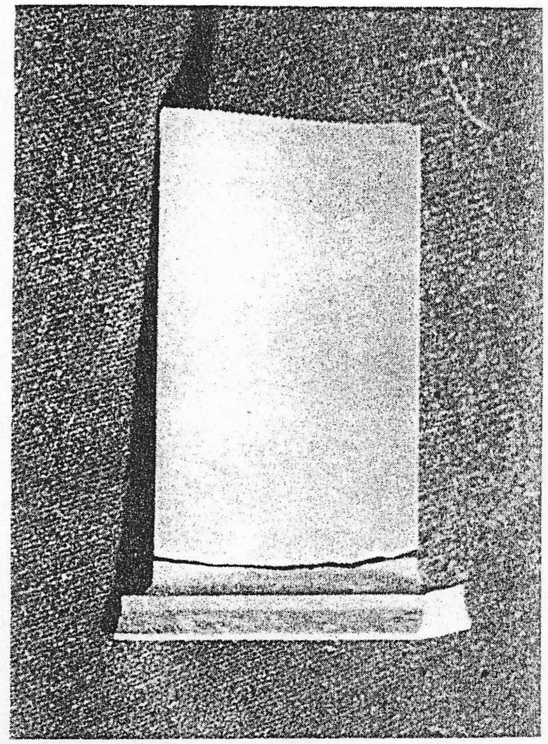


FIG. 4 FATIGUE CRACK IN TENTH STAGE STATOR BLADE

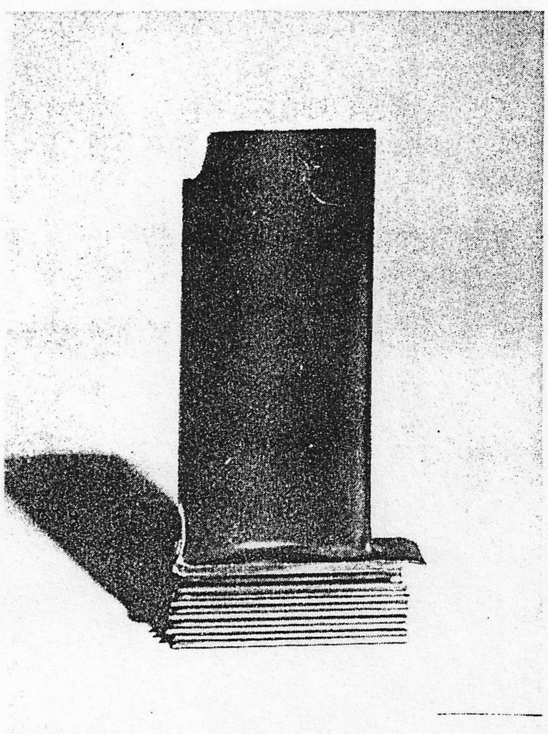


FIG. 5 CRACKED TURBINE BLADE TIP

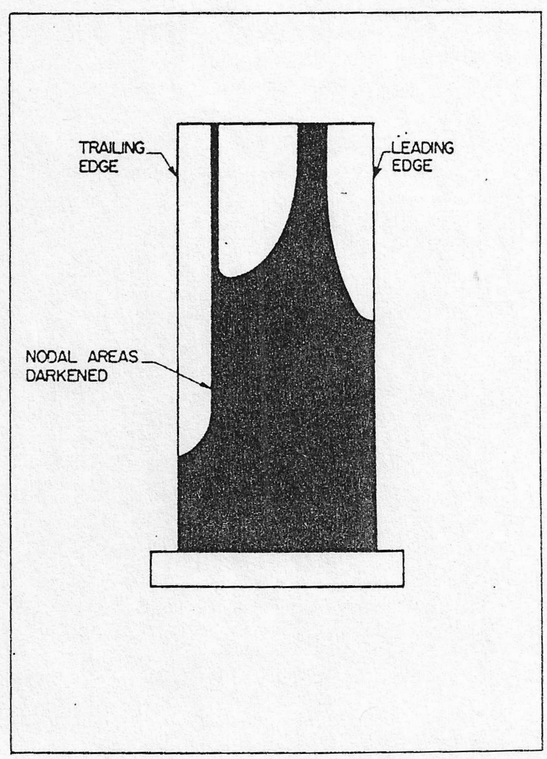



FIG. 6 VIBRATION PATTERN OF TURBINE BLADE IN SECOND

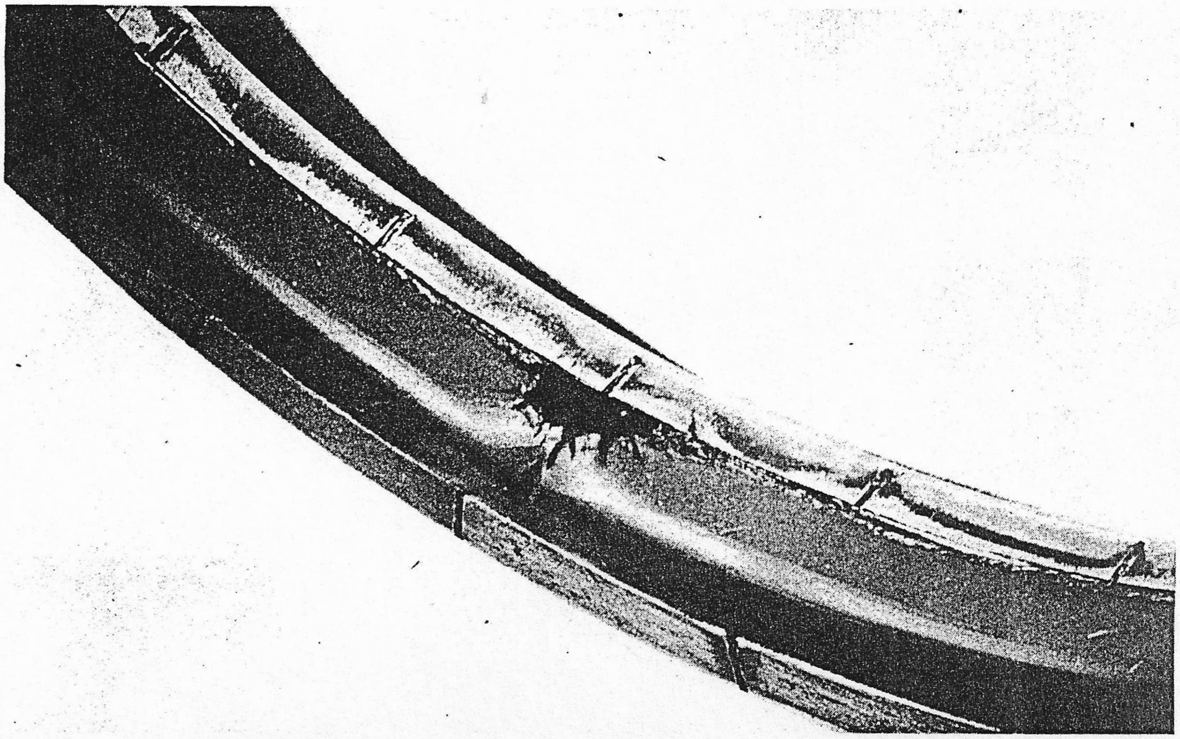


FIG. 7 FAILURE OF CENTRE BEARING THRUST RING

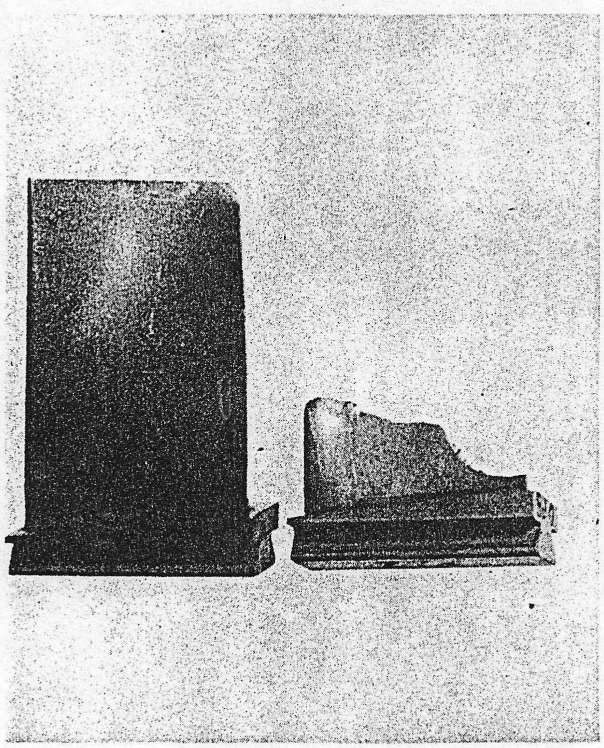


FIG. 8 FATIGUE CRACK IN SEVENTH STAGE ROTOR BLADE

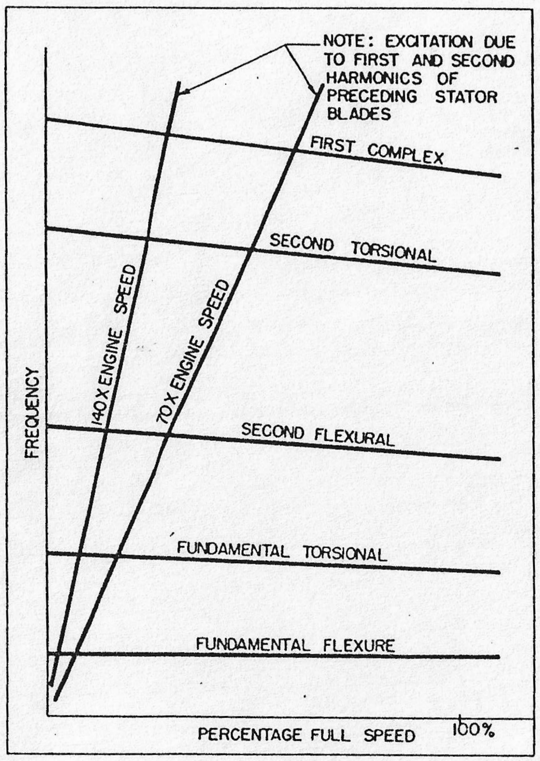



FIG. 9 INTERFERENCE DIAGRAM FOR SEVENTH STAGE ROTOR BLADE

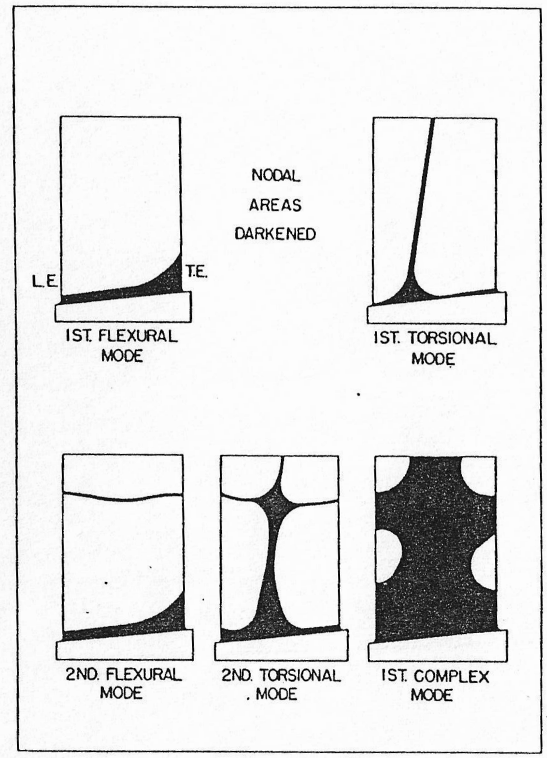



FIG. 10 VIBRATION PATTERNS FOR SEVENTH STAGE ROTOR BLADE

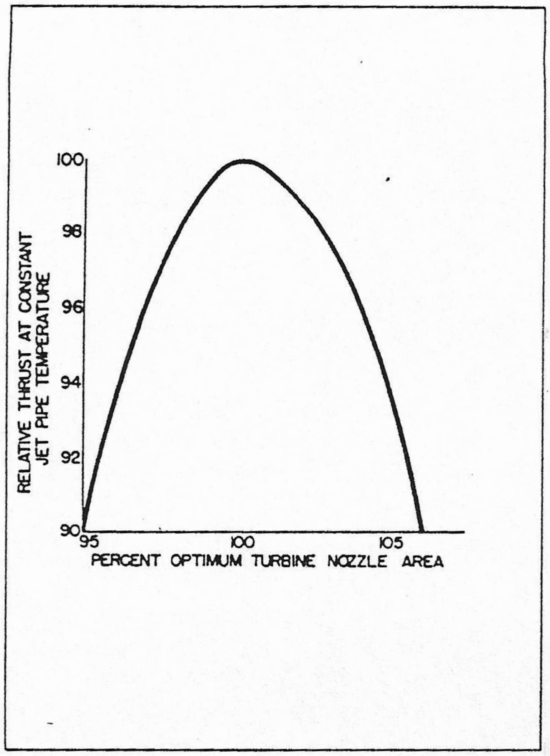



FIG. 11 VARIATION OF THRUST WITH TURBINE NOZZLE AREA, CHINOOK ENGINE

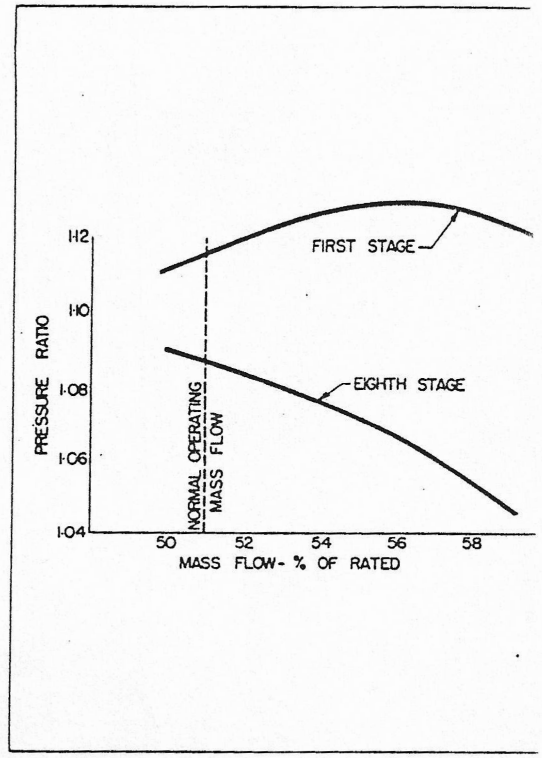



FIG. 12 TYPICAL MASS FLOW -PRESSURE RATIO CURVES

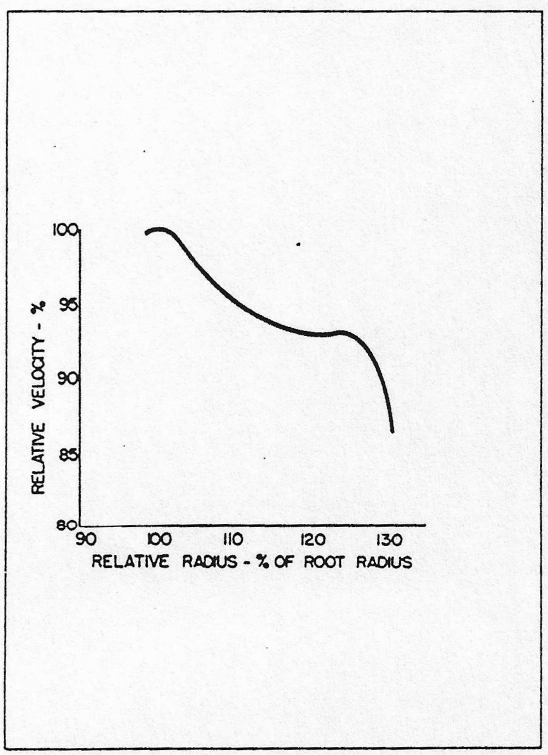



FIG. 13 VELOCITY TRAVERSE BEHIND FIRST ROTOR BLADES, CHINOOK ENGINE

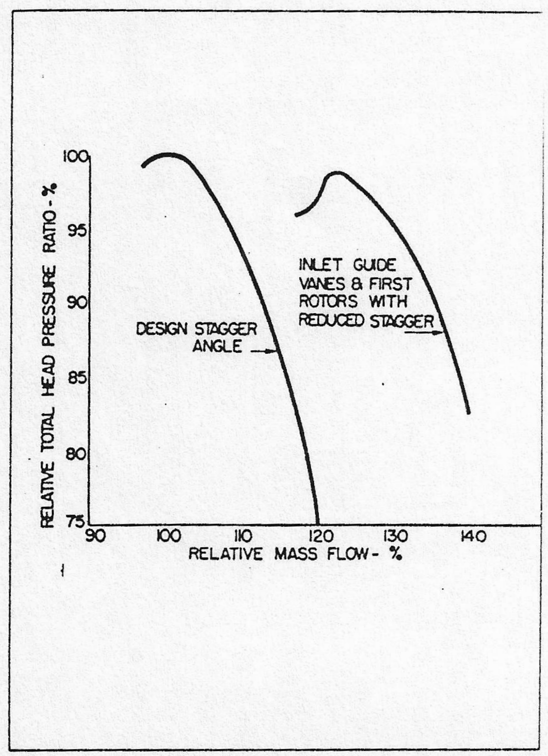



FIG. 14 MASS FLOW-PRESSURE RATIO CURVES-CHINOOK TWO STAGE TESTS

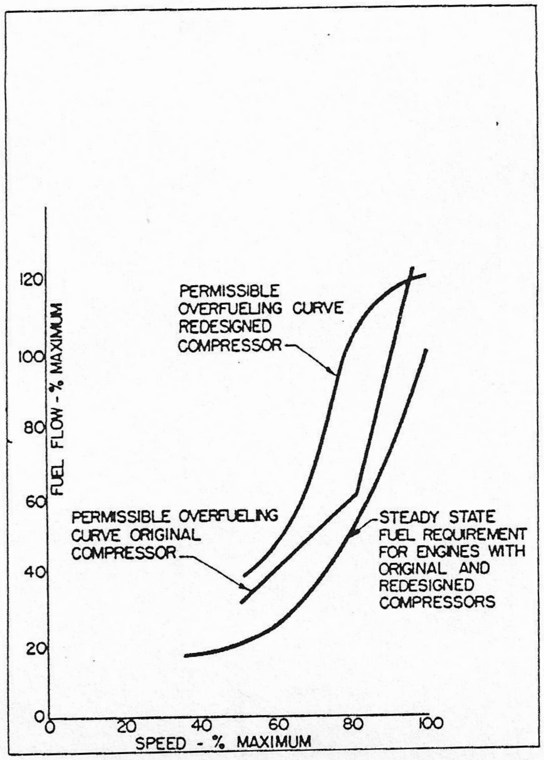



FIG. 15 OVERFUELING CURVES

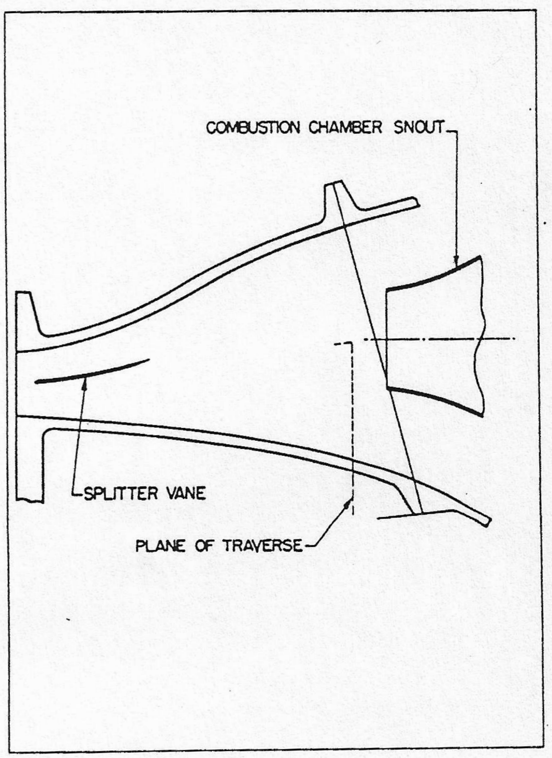



FIG. 16 DIAGRAMATIC SECTION -ORENDA DIFFUSER WITH SPLITTER

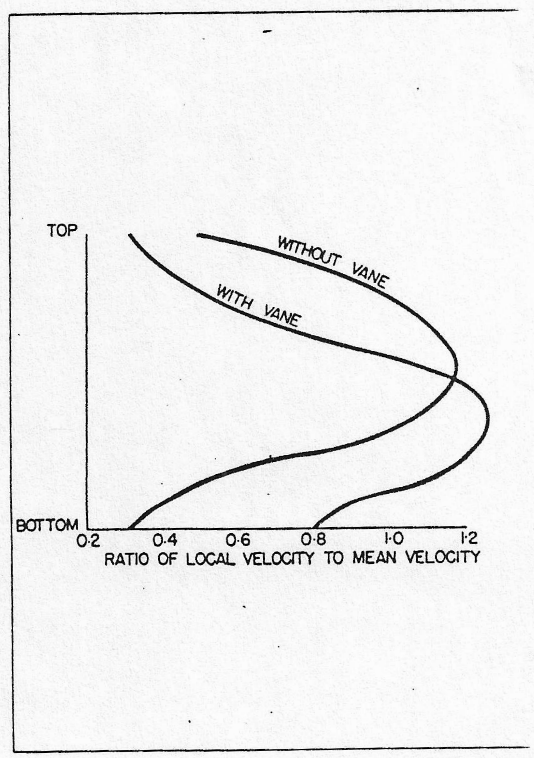



FIG. 17 VELOCITY DISTRIBUTION -DIFFUSER OUTLET

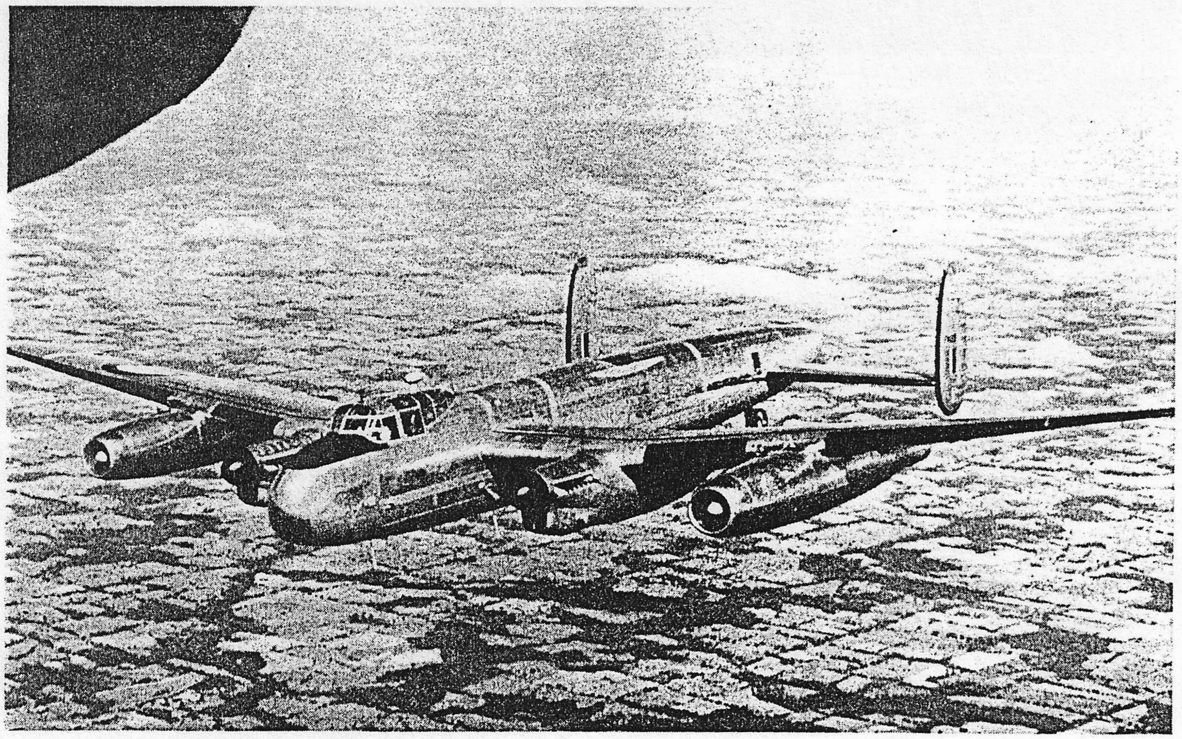
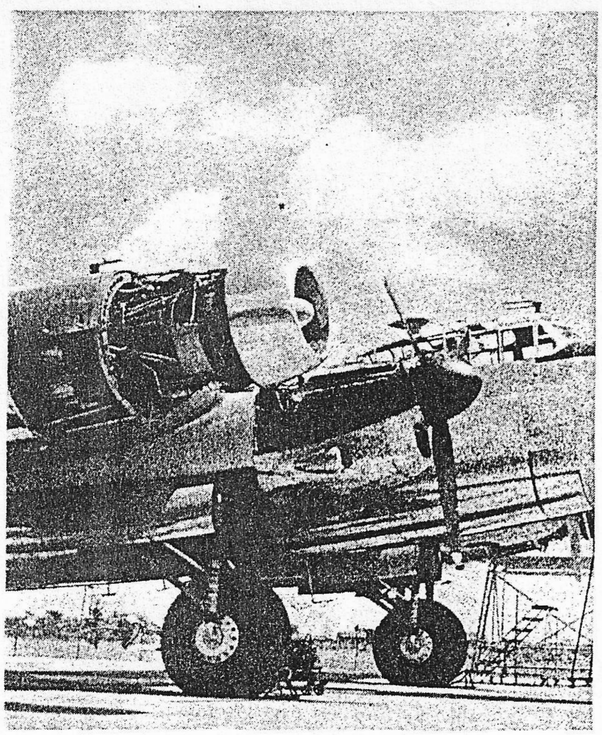


FIG. 18 ORENDA-LANCASTER AIRCRAFT



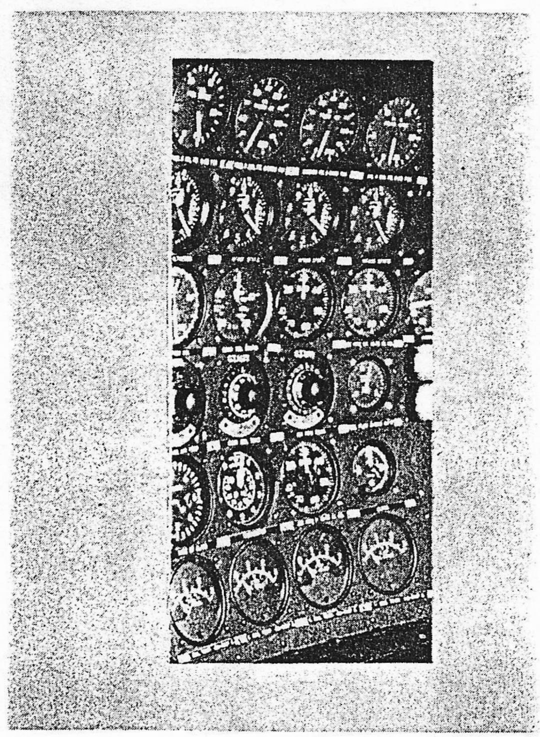


FIG. 19 ENGINE INSTALLATION
ORENDA LANCASTER

FIG. 20 AUTOMATIC OBSERVER
FOR ORENDA LANCASTE