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Combustion in the Gas Turbine

A Survey of War-time Research and Development

By

PETER LLOYD, M.A., F.R.I.C.

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COMMUNICATED BY THE PRINCIPAL DIRECTOR OF SCIENTIFIC RESEARCH (AIR),
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Introduction.—The present report attempts a general survey of the whole field of gas-turbine combustion. The report covers both research and development, and while it is mainly concerned with British work, some mention is also made of German work on the same subject. The related processes of combustion in propulsive ducts are briefly touched on. The report is based on a paper to the Institution of Mechanical Engineers, but with much fresh material, including a comprehensive bibliography.

There have been many groups of investigators concerned in this work at the Royal Aircraft Establishment, Power Jets, Joseph Lucas & Co., the Asiatic Petroleum Co., Metropolitan Vickers Ltd., Rolls-Royce, Armstrong Siddeley's, De Havillands and the City and Guilds College.

In preparing the present report, full use has been made of the work of all these groups and of the Combustion Panel of the Ministry of Aircraft Production's Gas Turbine Collaboration Committee through which they co-operated ; this debt is gratefully acknowledged. On the other hand the interpretation and assessment of the work are the author's, and for these full responsibility is taken.

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* Power Jets Special Report No. 510, received 6th May, 1948.

1. *Historical Summary.*—The following survey marks some of the milestones in the progress of gas turbine combustion. Of necessity most of these milestones are achievements in development. The less spectacular record of research must be laboriously sought in the pages that follow.

1930	Discussions of Griffith's proposals in the Engine Sub-committee of the A.R.C. envisaged the likelihood that combustion requirements for the gas turbine would raise special problems.
1937	Gas turbine projects at Power Jets and the R.A.E. supported by the A.R.C. Engine Sub-committee.
1937–1939	Early experiments by Power Jets at Rugby mainly with vaporised fuel. Similar work by the R.A.E. using upstream injection of liquid fuel.
1940	Intensification of Power Jets work on the combustion problem. Adoption of the Asiatic Petroleum Company's system of liquid fuel injection. Continuation of work at the R.A.E. and further development of the R.A.E. system by Metropolitan-Vickers for application to the F.2 engine. Exploratory work on the combustion requirements of propulsive ducts used as auxiliary power plant for aircraft.
1941	First flight of Whittle engine in E.28/39 and solution of the main combustion problems on this engine. Beginning of combustion development by J. Lucas & Co. First run of F.2 engine. Start of Halford project.
1942	Beginning of combustion research at the R.A.E. with the main objective of improving experimental techniques. Considerable difficulty with combustion of F.2 engine leading to a lengthening of the combustion chamber. First run of H.1 engine with a straight-through combustion system based on Power Jets' technique. Formation of the Combustion Panel of the G.T.C.C.
1943	Active combustion work by Power Jets, R.A.E., Lucas and Asiatic Petroleum Co. Substantial progress in F.2 combustion problem following adoption of longer chamber. Development of straight-through chambers by J. Lucas & Co. for De Havilland and Rolls-Royce engines. First run of A.S.X. engine with a vapour combustion system.
1944	Start of a fresh attempt to solve the combustion problem of the propulsive duct. Development of a system of jet pipe reheat. Application of Lucas straight-through combustion systems to further Rolls-Royce engines, including B.41.
1945	Tentative investigations of alternative fuels. Perfection of a method of spray particle-size measurement by Asiatic Petroleum Company. Examination of German gas-turbine combustion systems and propulsive ducts. Evaluation of German combustion research. First firing of a duct-propelled missile. First wind-tunnel test of a fuel-burning model duct. Development of a system for burning fuel in the wake of a ducted fan.

2. *The Problem.*—The emergence of the gas turbine as a power plant for aircraft depended in the main on the achievement of such component efficiencies in the compressor and turbine as were required to give an adequate overall efficiency, and on the availability of special materials to ensure a reasonable life, especially in the highly stressed turbine. But while these were the main factors, it was foreseen at a very early stage of the development that the combustion processes would raise special problems. This expectation was realised, and in spite of the technical effort which was devoted to it, the combustion process was for a time the limiting factor in the development of turbine engines, both centrifugal and axial. The requirements called for were in many ways extreme. In particular, since the purpose to which the turbine engine was to be applied was that of aircraft propulsion, limitations of space and weight were over-riding considerations, and the possibility of using such large refractory-lined combustion chambers as are generally

used in continuous industrial heating processes was ruled out. The space limitation is illustrated in Figs. 1 and 2, which show two early designs, the first a Whittle engine with centrifugal blower, the second an R.A.E.–M.V. design with axial compressor. In the Whittle engine, the combustion space was sub-divided into 10 separate tubular chambers and, by the use of a contra-flow principle, a combustion space of about half a cubic foot was obtained for each separate chamber. In this space, *i.e.*, in each separate chamber, it was necessary to burn up to 150 lb/hr of fuel. Even this combustion volume was only obtained by the deliberate sacrifice of pressure associated with the changes in direction of air flow. In the annular design of Fig. 2 the volume available was even less.

As to weight, the centrifugal engine as a whole was planned to weigh some 600 lb only, and of this the main proportion was necessarily allotted to the compressor and turbine, so that a light sheet-metal construction became essential for the combustion chamber. The axial-flow engine was necessarily heavier but the weight available for the combustion system was no greater than in the first case.

Other main requirements were:—

(a) The attainment of a high combustion efficiency, necessary because the fuel consumption was in any case bound to be high and a further increase due to combustion loss could not be tolerated.

(b) The smallest possible loss of total pressure in flow through the combustion system. The need for this will be apparent when it is considered that pressure loss at this stage obviously reduces the thermodynamic efficiency of the cycle.

(c) Operation was required over a wide range of air/fuel ratios and fuel flows, and it was essential that stability of combustion and efficiency should be maintained over the whole range. In terms of air/fuel ratio, the normal range requirement was from about 60 parts of air by weight per part of fuel to 120 parts by weight, but to avoid the hazard of flame extinction in flight, it was considered necessary to maintain stable burning up to air/fuel ratios of about 300 : 1. In terms of fuel flow, the range required was about tenfold, the maximum corresponding to the full-speed ground-level condition and the minimum to the cruising condition at altitude.

(d) A substantially uniform outlet temperature was required, implying a high degree of mixing of the combustion gases at the turbine entry.

Secondary requirements were:—

(e) Reliable starting on the ground and in flight.

(f) Absence of carbon deposition in the chamber.

(g) Such freedom from mechanical failure or scaling of metal parts as would give the combustion chamber a reasonable life between overhauls.

The predominant factors were, perhaps, the need for a high rate of heat release per unit of volume and the low temperature level required in the end gases, namely 600 to 800 deg C. The significance of the high combustion intensity is illustrated by Table 1, which records the intensities characteristic of some common heating operations. This shows that the level of combustion intensity required in the gas turbine is much in excess of that obtained in industrial fuel burning processes. It is comparable with that obtained in the reciprocating engine, and is only exceeded in very special processes such as the rocket, in which the absence of atmospheric nitrogen is a fundamental difference, and in premixed gas-air flames.

TABLE 1

Combustion Intensities of Representative Processes

Process	Operating Conditions	Combustion intensity, C.H.U. / ft³ hr atm	Reaction time in seconds (approx.)
Bunsen flame, town gas	Low pressure, maximum aeration (based on visible flame volume).	$0 \cdot 7 \times 10^6$	—
Premixed town gas-air, high-pressure flame.	Burning in the open	6×10^6	$0 \cdot 005$
Fuel oil flames from commercial atomisers.	Burning in the open	$0 \cdot 05 - 0 \cdot 11 \times 10^6$	—
Pulverised coal flames in open air	Particle size, 100μ 20μ	$0 \cdot 025 \times 10^6$ $0 \cdot 042 \times 10^6$	—
Oil-fired boilers	—	$0 \cdot 02 - 0 \cdot 1 \times 10^6$	—
Locomotive boiler	Based on fire box volume	$0 \cdot 22 \times 10^6$	—
Aircraft engine (4-stroke) . .	2,400 r.p.m. 110 octane fuel	About $2 \cdot 2 \times 10^6$	$0 \cdot 01$
V1 engine	Normal running condition	About 1×10^6	—
Cordite rocket	—	About $4 \cdot 5 \times 10^6$	—
V2 rocket	Maximum thrust	About 6×10^6	—
Aero turbine engines	Design conditions	$0 \cdot 8 - 2 \cdot 5 \times 10^6$	$0 \cdot 01$ to $0 \cdot 025$

The significance of the low temperature level of the end gases is that this is below the temperature at which rapid burning of the original fuel or of its decomposition products can be expected to take place. This introduces the possibility that combustion loss may be incurred without the appearance of exhaust flame, such losses being associated either with fuel which has passed through the chamber without being ignited or with reaction products which have been chilled out by excess of air.

These requirements are fundamental to the gas-turbine cycle, and the emphasis has not changed appreciably in the more recent engine designs in spite of the use of other mechanical arrangements and of more complex cycles. Even in the propulsive duct with its higher air speeds and its need for much lower pressure losses the differences are in degree rather than in kind, and in fundamentals the requirements are very similar.

The first problem was that of developing a combustion system to suit the essential requirements, and it was inevitable that this should in the first instance be dealt with empirically. But the turbine engine as a whole was no mere empirical invention and a second obvious need was to gain a sufficient understanding of the combustion process to make possible the prediction of changes in performance of given systems and the rational design of new systems. The second problem was therefore that of research. Work in both fields was facilitated by the fact that the combustion process could be isolated from the mechanical complications of the compressor and turbine (contrast the reciprocating engine) and could be examined on a reasonable experimental scale. Furthermore, it had the advantage of being a continuous process.

3. *The Development of a Technique.*—It is clear from what has been said that there was no ready-made technique available to satisfy these requirements. Carburation followed by ignition and flame stabilisation in the combustion chamber was ruled out by the large excess of air involved, which would have given a fuel/air ratio outside the normal range of inflammability. Industrial fuel burning methods were too bulky and were unsuited to high air speeds and high dilution. The method which Whittle used in most of his early work, and which is illustrated in its fully developed form in Fig. 3, was based in some measure on the principle of the Primus stove. The fuel (kerosene) was vapourised in preheating tubes which passed through the flame zone, the vapourised fuel being injected into a primary combustion zone to which only a part of the total air flow was admitted. The air flow into this zone was such as to create a region of flow reversal, and the function of this was, of course, to make the flame self-piloting by recirculating to the fuel vapour jets a supply of gas hot enough to maintain continuous ignition. This is a feature which is common to all subsequent designs, though different methods can be and have been used for obtaining the reversal. The use of an isolated primary zone followed by later addition of secondary air has also, and for obvious reasons, become a standard technique. The vapourising system was extensively tested (using an independently driven blower to provide sufficient air for an isolated combustion chamber working at atmospheric pressure) and gave reasonably good performance over a rather narrow range of conditions; but there was always a tendency when the system was working at conditions removed from the optimum for deterioration to set in, either because of cracking of the fuel to form carbon deposits in the vapourising tubes or because of priming with liquid fuel.

At this stage, an alternative method of fuel injection was proposed (by the Asiatic Petroleum Co.) using liquid fuel injection through a pressure jet atomiser, after the style of fuel oil burning techniques. The use of atomised fuel injection soon showed advantages over the vapour injection method, and this technique has since been generally adopted, the great majority of gas-turbine combustion systems now in use being based on pressure jet atomisers.

The problem of maintaining the effective operation of the system over the required range remained of fundamental importance, for with a simple fixed-component jet a flow range of 10 : 1 would obviously have implied a working range of burner pressures of 100 : 1, and with a pump delivery pressure limited as it then was to 400 lb/sq in. this would have meant a working pressure at the bottom end of the scale at which atomisation of the fuel would have been quite inadequate. The difficulty was dealt with by adopting the type of burner in which a moving plunger is forced back by fuel pressure against the opposing pressure of a spring, so uncovering the slots through which the fuel enters the swirl chamber. The use of this method* gave the required flow range at a minimum working pressure of 30 lb/sq in.

The use of liquid injection gave a greater freedom in the choice of fuel, but kerosene was retained as the standard fuel, not because of any special merits in the combustion process, but for secondary reasons.

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* The development of fuel jets for gas-turbines is a story in itself, and can only be touched on here. The moving piston or Lubbock burner, while it solved the immediate problem of widening the flow range, was always a very difficult manufacturing proposition, and the matching of a set of burners remained a problem. A need therefore came to be felt for a fuel jet capable of giving the necessary fine atomisation over a wide range of flows but without moving parts. This need is now being met by two different designs. The first design is the 'spill control' burner (Fig. 14) in which the flow through the discharge orifice is controlled by removing from the swirl chamber through a spill line a controlled amount of fuel. The second is the 'duplex' burner having two sets of swirl ports feeding a common swirl chamber and discharge orifice (Fig. 15). By using one set of swirl ports only when small flows are required the velocity of the fluid in the swirl chamber and burner the fineness of atomisation can be maintained. Fig. 16 shows the fuel-flow delivery-pressure relationships for the burners referred to. The enormous increase in flow range given by the duplex and spill control systems is obvious. Petrol was rejected in order to reduce the fire hazard, while gas oil and Diesel oil were open to the objection that their freezing points and pour points lay above the temperatures which might be expected to exist in the fuel system at extreme altitude.

Sheet metal of light gauge was used for the construction, cooled by the large excess of air which was always available.

The system thus produced, which was developed by W. R. Hawthorne and others at Power Jets in 1941, and was used in the first flight trials, is shown in Fig. 4. The device for producing the flow reversal was a vane-type swirler used in conjunction with a constricting orifice through which the flow was accelerated. Secondary air mixing was brought about mainly by injection through holes in the flame tube wall, but to secure better penetration some air was introduced through stub tubes projecting radially inwards. The performance of this system is summarised in Table 6, which also records data for other and later designs, but the test results then recorded were much less exact than those obtained by the improved experimental methods which were later developed.

At this stage the essential requirements had been met; combustion was stable, and under most operating conditions the efficiency was high, ignition was effective and the outlet temperature traverse showed only small variations. The chief weaknesses in the system were the manufacturing difficulties, especially in connection with the burners, the short service life of the flame tube, and a high pressure loss. Subsequent work on the contra-flow system under D. G. Shepherd was aimed mainly at overcoming these defects and at improving the efficiency at extreme conditions (especially at high altitude).

It is clearly impossible to record in any detail the subsequent development even of this single system, but Fig. 5 shows a later design developed by M. L. Nathan, in which the projecting stub tubes have been eliminated, a modified swirl system has been introduced and the secondary mixing arrangement has been redesigned. Performance data for the system are summarised in Table 6, and the mode of operation is briefly as follows. Atomised fuel is injected into the primary combustion zone, in which the flow pattern is determined by the air entering through the tangential swirl ports. The swirling air stream causes a flow reversal along the axis of the chamber, and here hot combustion products pass back to and kindle the entering fuel. A small air bleed round the fuel jet checks the flow reversal sufficiently to prevent deposition of carbon in this region. This primary zone receives about 20 per cent of the total air supply so that it works at a very high temperature, and most but not all of the heat release takes place here. The rest of the air is admitted more or less gradually through a system of swirl ports and plain holes. The design of these air admission ports was worked out empirically, but it may be explained that the swirl ports in this part of the system have several functions. Firstly, the opposed swirl from the alternate ports creates a high degree of turbulence, which accelerates the mixing of the entering air with the hot combustion gas. Secondly, the use of swirling air streams reduces the tendency for velocity gradients in the entering air to create non-uniform conditions in the exhaust. Thirdly, a film of relatively cool air is kept near the wall of the flame tube, which therefore runs at a lower temperature.

Before going on to consider the development of other systems and the performance achieved by competing designs, it will be well to summarise the essentials of the technique which has been evolved in the last five years. These essentials, which are almost universal in their application, are as follows:—

(a) The use of a fuel in the kerosene range (preferably low in aromatics to minimise coking), so avoiding the secondary disadvantages of petrol, namely fire hazard, and of heavier fuels, namely high freezing point.

(b) Liquid fuel injection by pressure jet, using special devices to maintain atomisation at low flows.

(c) Stabilisation of flame in a primary zone to which only a part of the air is admitted and in which a flow reversal is created so as to make the flame self-piloting.

(d) Subsequent mixing of secondary or diluent air by injection through ports in the flame tube, so arranged as to interleave the hot and cold air streams.

(e) Sheet metal construction relying on air cooling.

The development of a combustion system remains a matter of ad hoc experimentation, a process of test followed by modification and further test. The various requirements listed in section 2 of this paper are often mutually conflicting; thus high efficiency and uniform outlet temperature would be easy of attainment were it not for the pressure loss limitation. The problem is, therefore, to obtain the highest standard of general performance with the right balance of characteristics. A complicating factor of some importance is the non-uniform velocity and pressure distribution of the entering air. The work of the last four years has rationalised the process to the extent that available methods of test can now be relied on to give a sound diagnosis, and greater understanding of the mechanism has made prescription more certain; but the method remains empirical.

4. *Other Lines of Development.*—It would not be possible in the compass of the present paper to give any comprehensive account of the alternative lines of development which have been followed during the course of this work. It must, therefore, suffice to describe very briefly the more important of the many systems which have evolved.

4.1. *Metropolitan Vickers Annular Combustion Chambers.*—In parallel with the early work at Power Jets described in the last section, similar exploratory work on the combustion process was undertaken by the Royal Aircraft Establishment, starting in 1937. This work was based on a method of upstream injection of liquid fuel through low dispersion jets, and, after further development by Metropolitan Vickers Ltd., this provided the combustion systems for the early axial compressor engines B.10, D.11 and F.2. In 1941, however, when the F.2 engine was first run, using the annular combustion chamber of Fig. 7, the state of development was far behind that of the contemporary Power Jets engine, partly because of the greater intrinsic difficulties of the annular design, partly for other reasons. The main defect of the system was the very bad outlet temperature distribution; the pressure loss was not excessive.

In spite of much development effort the defects of this system have never been entirely overcome, and no annular chamber has achieved the same performance as is expected of tubular chambers. Considerable progress has nevertheless been made, first by increasing the combustion chamber length by 6 inches and later by modifying the primary flow pattern and the mixing technique (*e.g.*, Fig. 8). In the most recent system the air entry has also been redesigned to give symmetrical flow in the two sides of the annular flame zone, a change which has so far failed to give the expected improvement.

A point of general interest about the recent M.V. designs is the fact that the method of fuel injection depends on a form of air blast atomisation (Fig. 8). The fuel jets are still of the low dispersion type used in the first experiments, and their function is simply to spray the fuel on to the perforated entry plate from which it is blown by the entering air.

The history of the F.2 engine raises the general question of the comparative merits of tubular and annular designs. The present answer to this question is as follows:—

(a) The annular chamber is potentially better than a group of tubular chambers, because it fits in better with the geometry of axial flow engines and makes more efficient use of available space. The advantage is most marked on small engines.

(b) This potential advantage has not yet been realised because of three main difficulties. Development of annular chambers means working on the scale of the complete engine instead of with an isolated flame tube. Also annular chambers are intrinsically prone to maldistribution of fuel and air, leading to extreme temperature gradients. Finally, and most important of all, the swirl atomiser on which the best of the flame tubes are based does not conform to the annular symmetry.

(c) The ideal solution to this problem would be one based on an annular system of fuel injection (rotary burner or multiple vapour jet).

4.2. *Lucas Straight-through Tubular Chambers.*—The most effective and most highly developed of present systems are the Lucas straight-through designs. Lucas' started development for the W.2.B engine in 1941 and modified the existing Power Jets design by introducing the bulk of the primary air through a double conical baffle, 'the colander', giving the effect of peripheral air admission. The other essential change was the admission of a greater proportion of air to the primary zone, which improved the ground level performance of the system at the cost of some deterioration at altitude. Essentially the same system has since been applied with outstanding success to a number of straight-through designs for Rolls-Royce, De Havilland and other engines, the only fundamental change being the elimination of the stub tubes. A typical Lucas design is the chamber for the Derwent II engine shown in Fig. 6.

4.3. *Vapour Combustion. The Armstrong Siddeley System.*—Although some research studies of vapour combustion have been undertaken by the R.A.E. and Power Jets, the only system using this technique which is being developed is the Armstrong Siddeley design (Fig. 9). Flow reversal in the primary zone is created by peripheral air entry, and into this reversal is fed a rich carburetted air stream generated in a small vapouriser directly heated by the primary gases. The air/fuel ratio in this vapouriser is not known with any certainty but is thought to be 3 or 4 : 1 at typical conditions. Injection of diluent air is by scoops located in the secondary air stream, and there is a long length of ducting for the completion of secondary burning.

4.4. *German Systems.*—The two German designs which had reached a reasonable state of development were the Jumo 004 and the B.M.W. 003 systems. The Jumo engine (straight-through axial) has six tubular chambers with symmetrical air entry to a zero incidence swirler and upstream spray injection (Fig. 10). Secondary air mixing is obtained rather by outward deflection of the hot gas than by injection of cold air, this result being achieved by means of an air-cooled disc mounted in the centre of the flame tube. No alloy steel is used in the construction, the only protection against scaling being derived from aluminium spraying.

This design, in fact, has several interesting features, but its performance is indifferent in spite of a pressure-loss factor lower than that of any fully developed British design. The chief defects are instability leading to flame extinction at altitude, low combustion efficiency and rapid deterioration of the mixer. A further limitation is that the chamber has to be started on petrol; relighting in the air is therefore impossible with the normal fuel system.

The B.M.W. engine (also a straight-through axial) has an annular combustion chamber with sixteen swirl jets spraying downstream and stabilised by baffles having a circular symmetry (Fig. 11). Secondary air is injected in a single stage through mixers of the sandwich scoop type. This design represents an interesting adaptation of a swirl jet and circular baffle to an annular system, but in detail the performance is second rate, the defects being the same as those of the Jumo 004.

4.5. *The Rolls-Royce Rotary Burner.*—Although this system has not reached a state of development comparable with that of the designs so far mentioned, no survey of this field would be complete without mention of the Rolls-Royce rotary burner. The principle of this design is illustrated in Fig. 12, which shows the rotary burner fitted in a contra-flow chamber and mounted with three contra-rotating wheels. The burner may be driven either by its own air turbine or by mechanical drive from the last compressor stage; fuel is fed onto one or more of the burner rings and is thrown off from these in the form of finely atomised sheets of spray intimately mixed with air. The flame is stabilised by a flow reversal in the region indicated, and in the design shown in Fig. 12 there is no separate primary zone. The system is of interest as offering a new form of fuel atomisation and distribution.

4.6. *Jet-pipe Reheat.*—Thrust boosting by burning additional fuel in the jet-pipe involves combustion at high velocity with low pressure loss but with the advantage of high inlet temperature. A partial solution of the problem has been found by the simple expedient, illustrated in Fig. 13, of injecting fuel from a ring of plain orifice jets located around an inverted piloting space in the wake of which the necessary flow reversal takes place.

The main difficulty is one of flame stabilisation, and the system shown, though satisfactory over a limited range of burner pressures at low altitude, becomes unstable at 15,000–20,000 ft, largely because of the unfavourable effect of the sub-atmospheric pressure.

5. *Comparative Performance of the Alternative Systems.*—So far we have considered these alternative systems only in general and qualitative terms. The essential design and test data are recorded in the tables, 3—Design conditions, 4—Dimensions and weights and 5—Test Results.

The information contained in these tables is, however, not such as to allow any simple comparison of the different systems. As a first step towards such a comparison Table 6 shows some of the main performance criteria for four representative designs, two contra-flow chambers, a modern straight-through chamber and a German annular chamber. The performance quoted for these combustion systems is, however, a function of the design conditions of the engine to which they are fitted, and direct comparison is still difficult.

In order to make a general assessment of combustion chambers we must first decide to ignore the secondary factors and to pick out those parameters which have predominant importance. These are three in number:—

(a) Combustion intensity, a 'loading factor'.

(b) Combustion loss, a measure of heat loss due to unburnt constituents.

(c) Pressure loss.

For simplicity it is desirable to combine (b) and (c) to give a single efficiency factor, and this is best done by considering the effect of these separate losses on the efficiency of a typical engine at a chosen condition. There is a certain arbitrariness about the choice of this typical engine, but for present purposes it is legitimate to make an assessment on the basis of a simple jet engine, of 4 : 1 pressure ratio and with representative component efficiencies, working at ground-level static conditions, *i.e.* basing the comparison on test-bed performance. On this basis, and with certain simplifying assumptions, it is possible to work out the increase in specific fuel consumption resulting from the combined losses associated with any known combustion system.

This increase in specific fuel consumption L per cent (made up of L_p per cent, the pressure loss component and L_c per cent, the combustion loss component) can further be related to the combustion intensity so that the performance of a chamber can finally be characterised by a curve of loss against loading.

Fig. 17 shows the effect of pressure loss on the performance of the engine, and Table 7 applies the results of Fig. 17 to deduce L_p at a standard intensity, maximum intensity corresponding to 10 per cent pressure loss, and other essential criteria for some of the more important designs. A selection of these results is replotted in Fig. 18 to show the variation of combined losses with intensity. These curves illustrate the progress that has been made from the early contra-flow chamber to the contemporary straight-through designs.

Although the contra-flow chamber has now been surpassed by the straight-through type, Whittle's early decision to use this design was fundamentally right. This layout gave the long flow path and the flexibility which were needed at that stage of development and avoided the delays suffered by the axial engine as a result of the inadequacy of the space allotted for combustion in the early designs.

The curves also show that the combustion process, with an overall efficiency at normal intensity of about 95 per cent, is the most efficient of the main component processes of the turbine engine. As an indication of future possibilities Fig. 18 includes a curve for an experimental chamber of low pressure loss, scheme T.

The German chambers can only be characterised by single points, since in this case the variation of combustion loss with intensity is indeterminate. The position of these operating points and the data of Table 3 bring out one of the characteristics of German combustion chamber designs, the use of high combustion intensities accompanied by high losses.

In considering these results it must, of course, be remembered that the comparison is based on a particular test condition, and that it ignores many secondary, but nevertheless important, characteristics.

6. *Combustion Systems for Propulsive Ducts.*—The essential differences between the combustion requirements of proplusive ducts and those of gas turbines are:—

(a) *Higher air speeds through the system.*—In gas-turbine chambers though the entry air speed may be high, 200 to 300 ft/sec, the mean speed reckoned on the full cross-section is generally 50 to 100 ft/sec only. By contrast, propulsive ducts require speeds in the region 150 to 300 ft/sec.

(b) *Lower pressure losses.*—Especially at low (near sonic) flight speeds the propulsive duct cycle is most sensitive to the effect of pressure loss. In practice, the limit of permissible pressure loss may be taken as $\phi^* = 2$ to 4 according to the design requirements.

(c) *Outlet temperature distribution.*—So long as overheating of the shell and propelling nozzle is avoided, uniformity of outlet temperature is of no account.

(d) *Range of gas temperatures.*—With no limitations imposed by turbine blading, the propulsive duct operates at higher gas temperatures than the gas turbine. The range is roughly from the upper limit of gas turbine operation, *i.e.*, 1100 deg K up to the maximum flame temperature.

(e) Though the combustion intensitives required are similar (roughly

$$2 \times 10^6 \text{ CHU/hr/ cu ft/atm.})$$

the higher temperature level implies shorter reaction times, about 10 millisecs.

As a result of the requirements (a) and (b) the main difficulty is that of flame stabilisation. (d) effectively eliminates flame chilling and (c) provides a relaxation of the mixing standard.

The problem has been tackled in various ways. In the first place a solution has been sought by simply modifying a gas turbine combustion system so as to reduce its pressure loss and increase its throughput. This method has been used with success to provide a combustion system for the augmentor of a ducted fan engine (Power Jets engine for M.52 high-speed aircraft), *i.e.*, to burn fuel in the wake of the fan, a requirement involving essentially the same conditions as a subsonic propulsive duct. The resulting design is shown in Fig. 19.

The second method has been to retain self-piloting as the essential mechanism but to seek new means of producing the flow reversal. The main effort in this country has been along these lines, and the technique which has resulted from this approach is illustrated in Fig. 20. Three principles are involved:—

(a) The elimination of the primary zone separated from the secondary air stream by a flame tube.

(b) Upstream injection of atomised fuel, giving a measure of intrinsic stabilisation as a result of gas entrainment in the spray.

(c) Multi-stage injection of fuel, the second and third stages of fuel injection being ignited by the pilot flame formed further upsteam.

The stabilising baffle which has been found most effective is shown in Fig. 21. The hemispherical shield establishes a flow reversal at the axis which is reinforced by the entrainment of air in the fuel spray; the perforated cone feeds fresh air into the region of ignition.

This technique seems able to satisfy all the essential requirements. By suitable choice of baffle size the pressure loss can be reduced to one velocity head or less, and except on a miniature scale the stability requirement can be comfortably met, even when this involves such velocity and temperature ratio as will choke the flow in a parallel pipe, this being the condition required to give the maximum thrust. The chief problems remaining are those of scaling up on diameter

* ϕ is the ratio of pressure loss to inlet dynamic head reckoned on the maximum cross-sectional area.

without increase in length and of correcting fuel distribution so as to achieve higher combustion efficiency. Flame stabilisation at low pressure corresponding to extreme altitude may also need further investigation. A special development based on the same technique is the miniature duct, for wind-tunnel test, shown in Fig. 22. In this case the small scale adds to the other problems of the propulsive duct the need for a very high combustion intensity, about

$$20 \times 10^6 \text{ CHU/hr/cu ft/atm.}$$

This problem was solved by the use of a multiple hydrogen burner, flame stabilisation being obtained by upstream injection at high velocity.

The corresponding technique developed in the U.S.A. is base on precarburation with a volatile fuel followed by flame stabilisation on conical or other baffles. This provides an attractively simple system, which is very well suited to work with mixture strengths approaching the theoretical; its disadvantage would seem to be a limited range of stability. A number of alternative techniques had also been developed in Germany; none of them seems to be of outstanding performance.

All the methods so far mentioned depend on self-piloting. Another method, which has achieved little success in this country, though it has been widely used in the U.S.A., is to ignite a carburetted air stream by means of a pyrotechnic 'tracer' or an oxyhydrogen torch. The tracer method is obviously limited to very short endurances, as for projectile applications; the torch method would hardly seem justified unless the self-piloting technique were to break down in some unforeseen way.

Comparative test results on propulsive duct combustion chambers are lacking, but Fig. 23 shows the operating conditions and stability limits of some chosen systems.

7. *Experimental Methods.*—Before any effective research could be started, and indeed before the empirical development work could be put on a firm basis, it was necessary to develop an experimental technique. The basic test on a combustion chamber is the measurement of efficiency by heat balance of a single chamber mounted on a test rig and supplied with air by a plant compressor. This test is essential both in development and in research; in development it is necessary to know the effect of design modifications on efficiency, in research the variation of efficiency in response to other variables is important. The heat balance requires measurement of the quantities air flow, fuel flow, calorific value, inlet air temperature and outlet gas temperature. Each of these measurements has its problems, but one in particular, gas thermometry, limits the accuracy of the heat balance. We have recently learned that the German investigators facing the same problem had, in effect, given it up, preferring to rely on inferential methods, such as one based on the thrust produced when the gas is expanded to atmospheric pressure. In this country we have persevered with our attack on the problem and can claim to have solved it satisfactorily.

The main sources of error were found to be radiation losses and conduction effects along the thermocouple leads and mountings, and after a good deal of investigation a number of thermocouple designs have been produced which reduce these to negligible proportions. Two typical designs are shown in Fig. 24. The first is a fully shielded couple intended for use in rigs working at atmospheric pressure, the second is a combined total-head tube and thermocouple for use at high pressure. With couples of this kind, providing that a sufficient number of measurements is taken across the outlet section to give a true average, temperature rise can be measured with a probable accuracy of $\pm$ 5 deg C. Radiation errors, however, increase rapidly as gas temperatures rise, and when higher turbine temperatures come to be used further work on methods of gas temperature measurement may be necessary.

Even when all possible precautions have been taken there remains a certain inevitable error in the heat balance measurement resulting from the difficulties of air flow measurement and of thermometry. This residual error is estimated to be about $\pm$ 1·5 per cent, and with efficiencies in the region if 90 to 100 per cent this implies some uncertainty about the magnitude of combustion loss. For this reason considerable effort has been directed to applying methods of gas analysis so as to get more direct measurement of chemical losses.

But gas analysis also involves special problems because of the high degree of dilution. The volumetric concentrations of typical unburnt constituents corresponding to 1 per cent combustion loss in a 150 : 1 mixture are:—

Unchanged hydrocarbon	0·0015 per cent
Methane	0·0095 ,,
Hydrogen	0·032 ,,
Carbon monoxide	0·027 ,,
Formaldehyde	0·016 ,,
Formic acid	0·038 ,,
Carbon	(0·108 mg/litre)

These figures give an indication of the accuracy of measurement required; conventional methods, by which volume or pressure changes in a large sample of gas following absorption of a particular constituent in liquid reagents, are not good enough. Apart from this chemical problem, there is also the physical one of collecting a representative sample from a stream having varying velocity, temperature and concentration of exhaust products. This requires the use of an automatic traversing gear carrying a multiple sampling tube.

The two most successful methods of analysis are:—

(a) Direct measurement of minor constituents isolated by liquefaction* (or by liquefaction of their combustion products); the sequence of operations is illustrated diagramatically in Fig 25. The method is a laborious one involving a high vacuum technique, and the analysis of a single sample is a two-day job.

(b) Absorption or reaction train methods in which minor constituents are dealt with by chemical methods and measured gravimetrically (*see* Fig. 26).

This is a new method which promises the same order of accuracy as (a) with much less labour. The present accuracy is such as to give an error of less than 1 per cent in combustion loss measurement at 150 : 1 (with correspondingly greater accuracy at high concentrations), which makes this the best method for efficiency measurement.

Apart from its use for measuring combustion efficiency, gas analysis has various other applications. In research it can throw light on the mechanism of combustion, and in development it can be used to check distribution of fuel and air in the system. Both these applications are still in their infancy.

Comprehensive testing of a combustion chamber calls for much more than efficiency measurement over a range of conditions, other essential measurements being pressure loss, flame stability, which is limited at the weak end by extinction and sometimes at the rich end by resonance, observation of exhaust flame, coke deposition, and temperature variation at outlet. But while these measurements have all their special problems, none of them has called for the application of new methods.

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* This method was developed in Prof. Norrish's laboratory at Cambridge.

8. *Research.*—The main research objective in the field of gas-turbine combustion has been to gain a sufficient understanding of the process to provide a theoretical foundation for development work and a basis for predicting changes in performance. Past academic research and investigations of related processes, while providing some general guidance, have found only limited application. The problem is a difficult one, because of its complexity; because, in short, combustion embraces not one process but many. On the other hand, the continuity of the process and the possibility of isolating it from the mechanical side of the power plant both facilitate detailed investigation. Four main lines of attack have been followed, and these are illustrated in Fig. 27.

(a) Detailed exploration by pyrometry, gas analysis and pressure traverse of conditions within the system. Relatively ineffective as a research method, this remains an essential technique in rational development work.

(b) Investigation on a given combustion system of the effects of a number of controlling variables such as fuel/air ratio, fuel characteristics, throughput, pressure, air temperature, linear scale, etc. In spite of the intrinsic weaknesses of this method, such as the difficulty of isolating fundamental variables, it has given a great deal of information and remains a useful technique.

(c) Examination of the isolated component processes. This must be the ultimate research method, and it has already given invaluable results. The processes of atomisation, mixing of gas streams, and vaporisation of fuel/air mixtures have been fairly thoroughly dealt with. Ignition, flame stabilisation and the burning of the individual droplet have been only briefly examined.

(d) Detailed study of special effects in combustion. The most important problem which has so far come up under this head is coking. Others which have not yet had detailed investigation are resonance effects and behaviour of sheet metal work.

It would be impossible in the available space to give any adequate summary covering the whole of the researches which have been made in this field; instead, an attempt will be made to interpret some of the constituent processes of combustion in the light of this research. The following processes will be considered.

(i) Flow in the swirl atomiser.

(ii) The process of atomisation in still air.

(iii) Vaporisation of hydrocarbon fuels.

(iv) Ignition and burning of fuel sprays injected into a hot stream.

(v) Flame stabilisation and the primary combustion zone.

(vi) Mixing and the secondary zone.

(vii) The performance of the complete chamber.

(viii) Coking and the effect of fuel characteristics.

(ix) The mechanical characteristics of combustion chambers.

8.1. *Flow in the Swirl Atomiser.*—At first sight flow in the swirl atomiser seems to be one of the few constituent processes in combustion which may be suitable subjects for analytical treatment. A few investigators have attempted to provide a working theory of the flow through a nozzle and so a rational basis for design, but the problem has proved to be most intractable.

E. A. Watson of J. Lucas & Co. has produced a theory based on vortex flow and constant angular momentum in the swirl chamber, making use of the experimental observation that in actual atomisers the diameter of the air core, which is at the pressure of the atmosphere at discharge, is constant. Unfortunately, the assumptions of simple vortex flow and constant core diameter are inconsistent, and the results cannot therefore be accepted in detail, but it is found that the theory agrees with experiment over a range of flow numbers and atomiser dimensions. As a working theory this solution is useful.

G. I. Taylor has also examined this problem, both neglecting and including the effects of fuel viscosity. In the former case, his treatment allows for variation of the core radius, *i.e.*, his assumptions are self-consistent, but the solution obtained is involved and too difficult for practical use. The inclusion of viscosity is perhaps even more confusing, since it is proved that most of the fluid leaving the nozzle must then flow through the boundary layer around the walls of the swirl chamber.

Though its fundamental limitations must be remembered, the simplified theory of Messrs. Lucas may be accepted so long as it is not applied beyond the range in which it has been checked by experiment.

8.2. *The Process of Atomisation in Still Air.*—Fig. 28 shows the process of atomisation as revealed by flash photography; the pictures show the same jet passing kerosene fuel at delivery pressures of 16, 30 and 100 lb/m². Pictures of this kind were first made by Fraser at Imperial College just before the war, the present ones were taken by the Photographic Department, R.A.E. The pictures illustrate the change in the atomising mechanism as the pressure is raised. In the first pair of pictures, where the operating pressure is low, the expanding fluid film is breaking up under the influence of surface tension. In the second and third pairs the disintegration of the film is accelerated and modified by internal turbulence. The main interest in fuel atomisers must, however, be centred less on the mechanism of atomisation than on the degree of subdivision achieved, and for this reason attempts were made at an early stage of this work to measure particle sizes in fuel sprays. After various unsuccessful attempts using photographic techniques, a method was finally evolved by J. R. Joyce of the Asiatic Petroleum Co. which used molten wax preheated to a temperature which gave the same physical properties as the fuel, and collected the solidified particles for examination. Even this method was unsatisfactory in the early stages, when average size was deduced by counting and measuring particles collected on a slide; to obtain consistency, it was found essential to collect the total spray and separate it into size groups by sieving. This method, which enabled a sample of some millions of droplets to be handled, showed that the size distribution could be conveniently represented by the Rosin-Rammler law

$$\frac{R}{100} = e^{-(x/\bar{x})^n}.$$

where R is the residue per cent over size x.

Knowing the terms $\bar{x}$ (size factor) and n (diversity factor), it is possible to deduce any of the various particle size averages which may be required. Up to the present, results have generally been expressed in terms of the so-called 'Sauter mean diameter', which is the size of particle having the same surface/volume ratio as the spray, and this is probably the most significant term when we are dealing with spray formation, in the course of which work is done against surface tension. But when we come to consider the processes of evaporation and combustion, surface may no longer be the predominant term, since rate of evaporation for droplets has been shown to be a function of diameter to the first power. Another mean diameter must therefore be recorded, that given by the droplet with the same diameter/volume ratio as the spray ($\Sigma d/v$). Table 2 shows the variation of $\bar{x}$, n, specific surface and mean diameters with operating conditions for a typical fixed orifice jet.

TABLE 2

Spray Particle Size for a Simple Pressure Jet Atomiser over a Range of Pressures

(Power Jets atomiser 0·96 flow number, 50 deg cone angle, from measurements by J. R. Joyce)

Fuel pressure, lb/in.^{2}	20	100	250	400
Fuel flow, lb/hr	32	77	125	160
$\bar{x}\mu$	233	133	90	74
n	3·29	2·45	2·36	2·44
Specific surface, cm^2/cm^3	336	683	1034	1224
Sauter mean diameter, μ	179	88	58	49
Mean diameter based on diameter/volume relation μ	155	59	37	33
Size range from 10 per cent residue to 90 per cent residue μ	177 to 295	53 to 190	35 to 128	28 to 106

This table brings out the progressive improvement of atomisation as pressure is increased. It will also be noticed that the type of size distribution, as indicated by the diversity factor n, is almost constant as between 100 to 400 lb/in.^{2}, but is markedly different at the low-pressure condition. (The higher the diversity factor n the more uniform is the size of distribution.) This difference is presumably associated with the different mechanism of atomisation at the low pressures.

An empirical but useful correlation of these particle size measurements is given in Fig. 29, which shows mean diameter plotted against a function of fuel flow and pressure for a number of atomisers. For the spill-control burner (narrow cone angle) atomisation is defined by

$$\text{Sauter mean diameter } (\mu) = 315 \frac{Q_F^{0.3}}{P^{0.5}},$$

Q_F = fuel flow lb/in., P = fuel pressure lb/in.^{2})

and the plotted results cover a flow range 15 to 94 lb/hr and a pressure range 5 to 200 lb/hr.

For the Simplex burners the best generalisation is

$$\text{Sauter mean diameter } (\mu) = 200 \frac{Q_F^{0.3}}{P^{0.5}}$$

and in this case the plotted results cover eight different atomisers of four different makes (Lucas, C.A.V., Monarch, P.J.), implying significant differences in design, and covering a range of flow numbers 0·24 to 8·7 with pressures from 6 to 150 lb/in.^{2}. All available results for Simplex jets having cone angles between 80 to 90 deg have been included. Considering all things, the agreement is surprisingly good.

Here then is a component process about which we have some knowledge, and using this data we can go a stage further by investigating theoretically the burning of a cloud of droplets conforming to the Rosin-Rammler law. It seems legitimate to apply to this process the relationship which holds for evaporation of droplets, and on this assumption the idealised combustion process can be calculated; Fig. 30 shows the results.

It is interesting to see how the diversity factor affects the progress of combustion. In the early stages a high diversity (low n) accelerates burning, but in the final stages it is the more uniform spray (high n) that shows the smallest unburnt residue. The time required to reach a given degree of combustion is a function of $\bar{x}^2$, a consequence of the rule giving evaporation rate.

8.3. *The Vaporisation of Hydrocarbon Fuels.*—In the 'conventional' combustion systems vaporisation takes place in a narrow shell surrounding each individual droplet; in other possible systems vaporisation might be isolated from combustion. The equilibria in fuel-vapour-air systems have been systematically examined for some typical gas turbine fuels and the results summarised in the form of multiple nomograms, of which one is attached as Fig. 31. This nomogram permits estimation of the following data, all over the range 1 to 10 atmospheres, 0 : 1 to 20 : 1 air/fuel ratios.

(a) Bubble point.

(b) Dew point.

(c) Heat content of the vaporised fuel at the dew point referred to a liquid phase temperature 15 deg C or to a preheated liquid phase.

(d) Minimum air temperature for vaporisation starting with liquid fuel at 15 deg C or with preheat.

(e) Minimum fuel temperature for vaporisation starting with preheated air.

8.4. *Ignition and Burning of Fuel Sprays Injected into a Hot Gas Stream.*—The processes of ignition and combustion of liquid fuel droplets may be illustrated by the experiment of injecting a fuel spray into a stream of hot gas. Under these conditions the flow reversal, an uncertain element in all primary zones, is eliminated and ignition takes place under definable conditions.

A typical experiment of this kind is illustrated in Figs. 32 and 33. Fig. 32 shows in diagrammatic form the experimental duct into which the fuel spray is injected. Three zones can be distinguished:—

(a) The region of preflame reaction. Here the characteristic intermediate products of hydrocarbon combustion reactions are formed. From this region, under appropriate temperature conditions, cool flame radiation is given off.

(b) The flame front. Here exothermic reaction starts. The ignition delay is a function of temperature, pressure, fuel characteristics and other variables.

(c) The main flame. Here the burning of the liquid or vaporised fuel goes on. The rate of burning is largely a function of atomisation, but fuel characteristics (physical) also play a part.

Fig. 33 shows the variation of ignition lag with temperature for kerosene, and kerosene doped with tertiary butyl peroxide. In the case of plain kerosene three different lines are shown corresponding to three different degrees of atomisation. Here two delays are being measured, a physical delay associated with mixing and evaporation and a chemical delay associated with the preflame reactions and the production of chain carriers. The test condition giving best atomisation is one in which the chemical delay is the predominant term.

Each zone in this experiment bears a certain relationship to effects observable in actual combustion chambers. The intermediate products formed in the first zone show what can happen to fuel which passes through the primary chamber without igniting. The critical temperature required for ignition at a given delay illustrates the requirement for flame stability, and to a less extent that for efficient combustion. The process of burning in the ignited flame illustrates the basic process of burning around individual droplets. Up to the present, this part of the system has been studied qualitatively rather than quantitatively, and there is as yet no experimental check on the theoretical results of Fig. 30.

8.5. *Flame Stabilisation and the Primary Combustion Zone.*—In a real combustion chamber, the hot gases which ignite the entering fuel-air mixture come not from an external source but from freshly burnt gases recirculated in the flow reversal to the base of the spray. This flow reversal can be produced in a variety of ways. A fuel jet directed upstream against the entering air flow will in itself create a reversal strong enough to stabilise flame at low air speeds. If the fuel be vaporised and if a high jet velocity be used then the limiting air speed is raised to an appreciable value, and if a fast burning gaseous fuel is used (hydrogen is the best example) then stable burning can be got up to 200 ft/sec and over. More usually the flow reversal is obtained by manipulating the air flow, using either swirl or peripheral air admission to create a back flow at the axis of the chamber. Figs. 20 and 21 illustrate systems in which a combination of the two principles is used, *i.e.*, upstream spray injection and peripheral air admission, with an auxiliary air supply through a central slotted cone to feed fresh oxygen to the ignition region. In such systems stability may be limited in two ways. In the first place, the flame will blow out if insufficient heat is brought back in the flow reversal to ignite the entering fuel. This 'weak' limit, is the only one which has much practical significance in present designs of gas turbine chamber. As the above definition indicates, the weak limit is adversely affected by anything which tends to lower the temperature of the flow reversal, *e.g.*, falling air-entry temperature, or to raise the ignition temperature, *e.g.*, falling static pressure. In the second place, ignition may fail if the mixture in the piloting region becomes so rich as to reach the upper inflammability limit or pulsating combustion excites resonance effects. Fig. 23 shows typical stability curves for the stabilising systems of Figs. 20 and 21, and indicates the critical conditions for blow-out.

The stabilising systems on which this work was done are of the type designed for use in a propulsive duct rather than a gas-turbine system, but the mechanism of stabilisation is essentially the same in the two cases.

8.6. *Mixing and the Secondary Zone.*—The isolation of a primary zone in which the flame is stabilised, and in which most of the burning takes place, implies the need for a later mixing process between the hot primary gases and the secondary air. This process, in an idealised form, is more easily studied than any other of the component processes, and a good deal of investigation has been done using small-scale apparatus in which air streams at two different temperatures are mixed. This work has led to a number of generalised conclusions which are summarised below, and has also provided useful information on the effectiveness of some mixing devices. The main generalisations are as follows:—

(a) Mixing by molecular diffusion across the main interface bounding the two streams is a negligible factor in 'temperature mixing'.

(b) Mixing of two parallel streams by random turbulence is also too slow a process to be of significance in typical gas-turbine chambers, *i.e.*, it requires too long a mixing path.

(c) Devices for producing ordered turbulence may have a limited application. Examples are the use of a vortex trail in the wake of a cylinder placed at the interface and the use of opposed swirl in two concentric streams. Systems of this kind are, however, limited in their application; the mixing cylinder, for instance, is only effective with air streams flowing at similar velocities and without excessive turbulence.

(d) Rapid mixing requires the 'inter-leaving' of the two air flows, either by some form of cross-stream injection or by ducting one stream across the other. All effective mixing systems make use of this principle.

(e) All the mixing techniques investigated are independent of linear scale, *i.e.*, the flow path required to give similar mixing is proportional to the linear scale of the mixing device.

(f) The penetration of an air stream across another and the pressure losses associated with mixers are functions of the momenta of the two streams and may be correlated in terms of the momentum equations. The pressure losses involved in typical mixing devices have been found amenable to a complete theoretical treatment. Penetration of a cold air jet across a hot gas stream has been investigated experimentally.

The perfection of a mixing device on an actual combustion chamber remains a matter of ad hoc experiment, partly because our understanding of the penetration of the cold air stream is limited and partly because the primary air flow is not uniform in either temperature or velocity.

So far, mixing has been regarded simply as a process of temperature equalisation in two inert air streams, but this is an idealised condition remote from the practical case. In the actual combustion chamber, two other effects occurring in the secondary zone must be considered, namely, mixing by molecular diffusion and chemical reaction in the secondary zone. Mixing by molecular diffusion across the boundaries of the individual eddies has to be taken into account because this is the only process capable of producing a chemically homogeneous system, and until this homogeneity is achieved we may still have flame at outlet from the chamber. This process has not been systematically investigated but is likely to become increasingly important as absolute working pressures rise, since the rate of mass transfer across an interface under a given concentration gradient remains roughly constant as static pressure rises.

It is in the secondary or dilution zone that conditions least favourable for combustion occur, for the mean temperature level is necessarily low, and in regions of excessive dilution it will be far too low to allow continuance of combustion. This is the so-called flame-chilling effect which is responsible for much of the combustion loss experienced on gas-turbines; such understanding of this effect as we have gained has come from tests on complete combustion chambers, the results of which are summarised in the following section.

8.7. *The Performance of the Complete Chamber.*—As our understanding of the component processes is itself incomplete it goes without saying that we are not yet in a position to synthesise anything approaching a comprehensive theoretical treatment of the system as a whole. The most we can do, in the present state of knowledge, is to discern ways in which such a synthesis may presently be attempted. The first opportunity of this kind will be for a general aerodynamic treatment of flow in a combustion chamber; the second possibility may be the application of the theory of spontaneous ignition to the problems of combustion efficiency and flame stability. These developments, however, lie in the future, and for the present our understanding of the complete combustion system and our knowledge of the variation in its performance at differing conditions is mainly based on empirical tests on representative combustion chambers. It seems worth while to attempt a summary of the general conclusions derived from this work.

In considering variations in combustion chamber performance we can distinguish two kinds of controlling effect, design factors and variables independent of design such as air flow and fuel characteristics. In this brief review, only the second group of variables can be examined.

Under the general heading 'variation in performance', we must include pressure loss, combustion loss and stability. The first of these is easily dealt with by elementary theory, since over the relevant flow range neither compressibility nor Reynolds number effects are significant. The pressure loss of a given system is completely defined by the relation

$$\frac{\Delta P}{Q_A^2/2g\rho_2 A^2} = K_1 + K_2\left(\frac{\rho_2}{\rho_3} - 1\right),$$

where ΔP is total head loss,

Q_A is air mass flow,

ρ_2 inlet total-head density,

ρ_3 outlet total-head density,

A an area, generally the maximum cross-section of the chamber,

K_1 and K_2 characterisation factors.

Fig. 34 shows typical plots of $\dfrac{\Delta P}{Q_A^2/2g\rho_2 A^2}$ against $\dfrac{\rho_2}{\rho_3}$ for some of the systems already mentioned.

The high losses of the contra-flow type of chamber which were brought out in Fig. 18 are also shown by this method of plotting. The Jumo 004 is included instead of the B.M.W. 003 as representative of German systems, and in this respect it is seen to have excellent characteristics.

The line showing the fundamental pressure loss resulting from the density change in combustion shows how insignificant is this effect in comparison with the other losses in the system. The main sources of loss are, in fact,

(a) Entry losses involved in slowing down and evening out of the flow of the fast air stream from the compressor,

(b) Turbulence losses associated with the primary zone (flame stabilisation) and the secondary zone (mixing).

Variation of combustion efficiency is much less easy to analyse. Combustion loss can occur in two distinct ways, either by persistence of flame at outlet with burning going on in the turbine and jet pipe, or in the form of inert unburnt constituents in the gas stream. This second kind of loss is by far the more important of the two, since turbine designers have insisted on an absolute minimum of exhaust flame; it may arise either through the failure of part of the injected fuel to reach its ignition temperature or through ' chilling ' of intermediate products to a temperature at which reaction is negligibly slow. This ' flameless ' combustion loss is increased by the following factors.

(i) Weak mixture.
(ii) Low inlet air temperature.
(iii) High throughout.
(iv) Small linear scale.
(v) Low absolute pressure.
(vi) Poor atomisation.
(vii) Chemically unreactive fuel.

The effects of the first two factors can be roughly combined by treating the mean temperature of the outlet gases as the controlling factor. High throughout and small scale both imply high combustion intensity, though it is uncertain whether the two effects are equivalent. Low pressure and chemically unreactive fuel are both associated with the spontaneous ignition requirement. Atomisation is a factor on its own. Re-grouping these factors, we can therefore sum up the case by saying that high combustion losses are associated with high combustion intensity (*i.e.*, short reaction time), low temperature level in the system, conditions unfavourable for spontaneous ignition of the fuel and its degradation products, and poor atomisation.

Fig. 35 illustrates the variation in performance of a combustion chamber in response to some of these variables.

In terms of flight requirements all this means that combustion efficiency is less at altitude than at ground level and less at cruising conditions than at full speed. With contemporary short endurance fighter aircraft this fall-off in efficiency is unimportant, indeed it seems likely that a rather excessive importance has been attached to combustion efficiency. With future power plants for long-range flight, however, the cruising performance at high altitude will be crucial.

' Long flame ' combustion loss is associated with rich mixtures, coarse atomisation and high absolute pressures. It is an effect of which we have no quantitative knowledge.

The effective stability limit on gas-turbine combustion chambers is the blow-out which occurs at high air/fuel ratio. This is subject to exactly the same influences as flameless combustion loss, *i.e.*, the limit is lowest at altitude and at low engine speed.

A special effect which can sometimes impose a limit on the performance of a combustion chamber is the separation of solid carbon leading to the building up of deposits in the flame tube. This effect is associated with aromatic fuels, over-rich primary zone conditions, non-uniform air flow into the primary zone and coarse atomisation. The effect of static pressure is uncertain, though there is little doubt that the related phenomenon of smoky exhaust is adversely affected by increase in pressure.

8.8. *Coking and the Effect of Fuel Characteristics.*—A secondary combustion product in turbine engines is carbon. This may either blow through the system with the gas stream, giving a smoky exhaust trail, or it may deposit in the combustion chamber. The formation of a smoke trail is not very serious, though it has obvious tactical disadvantages by night as well as by day, since the solid matter in the exhaust gives a readily visible glow. The building-up of carbon in the chamber may be much more serious, since it may cause damage to the flame tube or turbine or both. Though the effect has been exhaustively investigated it is still not perfectly understood, there is no laboratory test which gives any reliable indication of the coking tendencies for fuel, and the agreement between rig tests and engine runs is also bad. General conclusions about coking tendencies are as follows.

(a) The coking tendency of a fuel is mainly determined by its aromatic content, or more generally by its carbon/hydrogen ratio. Sulphur compounds have a secondary effect in promoting carbon deposition.

(b) The cracking reactions which lead to the separation of carbon naturally take place in regions which are starved of air. Any change which raises the fuel/air ratio in the primary zone will promote coking.

(c) Carbon formation is increased by increase of reaction time (increase in linear scale or decrease of throughput).

(d) Coarse atomisation and non-uniform air distribution promote coking.

Though fuel characteristics also influence flame stability and combustion efficiency, it is the coking effect that really decides the range of fuels on which a given combustion system can operate (fuel system problems are here ignored). This fact is brought out in Table 8, which records rig test results on a number of alternative fuels applied to a P.J./102 combustion chamber. The table shows (and engine tests confirm) that among the commercial fuels kerosene, gas oil and diesel oil give reasonable performance in this combustion chamber; of the special fuels, which were mostly chosen for their high bulk calorific value, only decalin is satisfactory.

8.9. *The Mechanical Characteristics of Combustion Chambers.*—Although the combustion chamber is one of the most lightly stressed of the chief engine components, it is also true that failures in service present a much greater problem in this component than any other. These failures arise from the combined effects of high temperature, asymmetry, and pressure pulsation in the flow through the chamber; they are most severe in centrifugal engines, but are present in lesser degree in axial engines, too.

The common materials are inconel and nimonic, both high nickel chromium alloys, for flame tubes, and mild steel for outer casings. Flame tube failures are usually fatigue cracks associated with intercrystalline corrosion for inconel and transcrystalline cracking for nimonic. Cracks usually begin at the edges of punched holes or other work-hardened parts. Casing failures are usually due to bad seam welding, but weld failures occur in flame tubes also.

Secondary troubles are the fretting of locating pins, bosses, etc. as the separate parts flex under variations of pressure, and buckling of the flame tubes due to uneven heating and cooling caused by asymmetry in the flow.

On fully developed engines a life of two hundred hours is sometimes possible, but failures are haphazard and the position is by no means satisfactory. For the future, it seems that only slight improvements in materials are possible, although materials of higher thermal conductivity will help, provided this is not accompanied by poorer fatigue properties. The most promising lines of attack will be to feed the chamber with uniform air flow and to study the natural frequencies of the chamber under different modes of vibration in a manner similar to that used for centrifugal impellers, and to examine the forcing frequencies in the chamber. Using this information, it may be possible to reduce or even exclude damaging resonances. These improvements, however, must be backed by continued refinement of design, and strict care during manufacture.

9. *The Future.*—The future holds many problems. The chief weakness of contemporary chamber designs is the short life between overhauls, and there will be urgent demands to improve performance in this respect. Improvement in heat resisting materials cannot in itself overcome this difficulty, and combustion chamber design may have to be reconsidered. There is also a growing need for certain specialised types of gas-turbine combustion system, *e.g.*,

(a) A good annular chamber.
(b) A design with very low pressure loss.
(c) A design with the shortest possible axial length.
(d) A chamber for interheat between turbine stages.
(e) A chamber to burn residual oils and pulverised coal.

As to propulsive duct combustion, the work that has so far been done has indicated the most promising lines of attack, but an enormous amount of development work remains to be done to perfect this form of power plant.

On the research side the need is to build up and strengthen the slender theoretical foundation on which our present development work is based. The work that has so far been done in this field has given us a fair measure of qualitative understanding of the underlying processes, but there remains the harder task of working out an exact treatment. Every step that is taken in this direction facilitates development and makes it more economical.

TABLE 3

Design Conditions for some Gas Turbine Combustion Systems

Layout	Straight-through Tubular				Contra-flow	Annular	
Engine	A.S.X. (Python)	H.2 (Ghost)	B.37 II (Derwent)	Jumo 004	W2/700	F2 CC/31	B.M.W. 003
Air mass flow, lb/sec	4·55	5·89	5·46	6·5	4·03	47·5	41·5
Fuel flow, lb/hr	223	338	346	455	228	2,794	2,460
Fuel/air ratio	0·0136	0·01591	0·01752	0·0182	0·01571	0·0163	0·01682
Inlet pressure, lb/sq in. abs.	74·5	58·8	54·6	44·0	62·9	59·9	39·7
Inlet temperature, deg K	492	469	480	413	486	460	450
Inlet velocity, ft/sec	242	335	357	250	153	237	595
Outlet temperature, deg K	1,000	1,060	1,125	1,050	1,070	1,073	1,018
Temperature ratio	2·04	2·26	2·34	1·88	2·20	2·34	2·26
Combustion intensity, CHU/hr/cu ft/atm	0·78	1·54	1·28	2·06	0·92	1·58	2·30
$Q_a^2/2g\rho_2A^2$, lb/sq in.	0·138	0·101	0·041	0·19	0·72	0·096	0·229
Maximum fuel flow per lb combustion chamber weight lb/hr/lb	8·64	—	9·11	10·70	10·28	12·65	8·20
Air velocity based on inlet density and max. diameter, ft/sec	75	70	47	100	—	68	122

TABLE 4

Dimensions and Weights of some Gas Turbine Combustion Systems

Layout	Straight-through Tubular				Contra-flow	Annular	
Engine	A.S.X. (Python)	H.2 (Ghost)	B.37 II (Derwent)	Jumo 004	W2/700	F2 CC/31	B.M.W. 003
Air casing diameter, in. ..	7·0	9·0	11·16	8·66	8·0	(outer) 29·125 (inner) 13·75	(outer) 25·20 (inner) 12·91
Flame tube diameter, in. ..	5·38	7·650	9·187	7·28	6·5	(outer) 27·719 (inner) 17·93	(outer) 23·85 (inner) 13·79
Maximum cross-sectional area of air casing, ft^2	0·226	0·441	0·680	0·410	0·354	—	—
Flow path, ft	5·2	2·46	2·54	2·25	2·86	2·36	1·81
Flow path/Air casing diameter or width of annulus	8·9	3·28	2·74	3·12	—	3·7	3·53
Air entry area, ft^2	0·0827	0·0925	0·0895	0·165	0·136	1·029	0·525
Gas outlet area, ft^2	0·1178	0·1479	0·1358	0·340	0·088	1·535	1·625
Primary zone volume, ft^3 ..	0·11	0·31	—	0·24	0·19	1·5	1·83
Flame zone volume, ft^3	0·59	0·565	0·746	0·76	0·60	4·47	3·98
Weight of combustion equipment, lb	284	—	266	255	222	221	300

TABLE 5

Test Results on some Gas Turbine Combustion Systems

Layout	Straight-through Tubular				Contra-flow	Annular	
Engine	A.S.X. (Python)	H.2 (Ghost)	B.37 II (Derwent)	Jumo 004	W2/700	F2 CC/31	B.M.W. 003
Air mass flow, lb/sec	4·55	5·89	5·29	2·26	1·409	16·2	41·5
Inlet pressure, lb/sq in. abs ..	71·0	58·8	55·6	15·5	17·18	20	39·7
Fuel/air ratio	·01300	·01591	·01549	·01962	·01360	·01515	·01649
Combustion efficiency, per cent ..	97	95	—	92	98	83·3	90–95
Combustion intensity, CHU/hr/cu ft/atm $\times 10^{-6}$	0·75*	1·54	—	2·05	1·01	1·24	2·12–2·22
Pressure loss, lb/sq in.	2·38	3·01	1·99	0·67	1·34	0·76	4·25
Pressure loss factor†, φ	16·6	29	45·7	12·5	62·0	20	18·5
K_1	12·8*	22	—	11·05*	45·4*	—	—
K_2	1·92*	5·5	—	1·75*	8·4*	—	—

* Based on temperature ratio.

† All pressure loss figures are for the combustion chamber along, loss in entry bend having been subtracted. The pressure loss factors are defined by

$$\varphi = \frac{\Delta P}{Q_A{}^2/2g\rho_2 A^2},$$

$$= K_1 + K_2 \frac{\rho_2}{\rho_3},$$

where ρ_2, ρ_3 are inlet and outlet densities and A is the maximum cross-sectional area of the air casing.

TABLE 6

Performance of some Combustion Chambers

Combustion	Contraflow chamber, 1941. P.J./75 in W.1	Contraflow chamber, 1945. P.J./102 in W.2/850	Straight-through chamber, 1945. Lucas B37, Mark II	German annular chamber, 1945. B.M.W.003
Combustion intensity at design point, B.Th.U/cu ft/hr/atm	$0{\cdot}95 \times 10^6$	$0{\cdot}92 \times 10^6$	$1{\cdot}28 \times 10^6$	$2{\cdot}3 \times 10^6$
Pressure loss at the design point, $\Delta P/P$ per cent	7·2	11·3	(2·33)	10·7
Approximate combustion efficiency at the design point, per cent	96	97	98	95
Combustion efficiency at altitude, taken as atmospheric pressure, per cent 50 deg C air temp., 90 : 1 air/fuel ratio	80	98	not known	78
Air/fuel ratio at blow-out ; condition as above	350 : 1	600 : 1	not known	not known
Combustion chamber weight per pound of fuel burnt per hour at design point	0·166	0·085	(0·11)	0·122

TABLE 7

Performance of Various Combustion Chambers

Combustion Chambers	1 P.J. C.C. 75	2 W2/700 C.C. 102T	3 A.S.X.	6 F.2/4	4 B.37 Mk. II	4 Ghost I (Flower Pot)	3 Jumo	5 B.M.W.	5 T Scheme
Pressure-loss factor, ϕ at $T_3/T_2 \doteq 2 \cdot 3/q1$	105	66·8	15·3	48·6	47·5	29·1	13·3	10·8	10·65
Flame zone volume, B_F ft^3	0·446	0·60	0·59	6·9	0·746	0·565	0·76	3·98	0·304
Air casing cross-section, A ft^2	0·349	0·354	0·266	5·41	0·68	0·320	0·41	2·56	0·266
$\phi B_F{}^2/A^2$ ft^2	187	192	75·3	89	55	90·8	4·75	26·1	13·9
$\Delta P/P$ per cent for $I = 10^6$	8·5	8·72	3·41	4·03	2·5	4·12	2·07	1·2	0·63
L_p per cent for $I = 10^6$ ..	7	7·2	2·6	3·0	1·8	3·1	1·5	0·85	0·45
L_c per cent	4	2·0	2·0	2·0	2·0	5·0	(7·5)	(5·2)	2·0
I for $\Delta P/P = 10$ per cent ($L_p = 8 \cdot 6$ per cent)	1·085	1·07	1·7	1·57	1·85	1·45	2·2	2·88	4·0
V ft/sec for I_{max}	—	—	137	77	79	99	148	162	164
L per cent for $I = 10^6$ ..	11	9·2	4·6	6·0	3·8	5·1	—	—	2·45

1. These results are less well established than those for later designs and should be regarded as very approximate. They include loss across the half-moon baffle but not those in the elephant's trunk.
2. Based on the data in Power Jets Memorandum 1113.
3. Based on the data in Power Jets Memorandum 1136.
4. Based on Lucas Report B.40492.
5. Based on unpublished work at Power Jets.
6. Based on verbal report from Metropolitan Vickers Ltd.

TABLE 8

Tests on Various Fuels in W2/700 Combustion System

Heavy fuels are tested with such preheat as gives the same physical properties as Reference Kerosene

Fuel	Gas Oil	Diesel Oil	Light Fuel Oil	Heavy Fuel Oil	Tetralin	Graphited Kerosene	High Flash Naphtha	Decalin
Preheat deg C..	60	85	153	140	34	Ambient	Ambient	Ambient
Carbon/hydrogen ratio of fuel	6·48	6·87	7·29	7·1	9·06	—	7·81	6·88
Change in combustion efficiency relative to S.R. 312 ..	Equal	Equal	1 per cent lower	Not determined	2 per cent lower	Not determined	2 per cent lower	Equal
Carbon deposition relative to S.R. 312	× 2·7	× 4·6	About 100 times	Excessive	About 100 times	Excessive	About 100 times	× 1·6
Stability	Normal	Normal	Burns badly below 150 deg C	Rough	Red hot spots in primary zone	Rough	Normal	Normal
Condition of exhaust	Very little flame	Fairly short flames	Fairly short flames	Long flames around elbows	Short flames	Long flames	Smoky exhaust	Very little flame

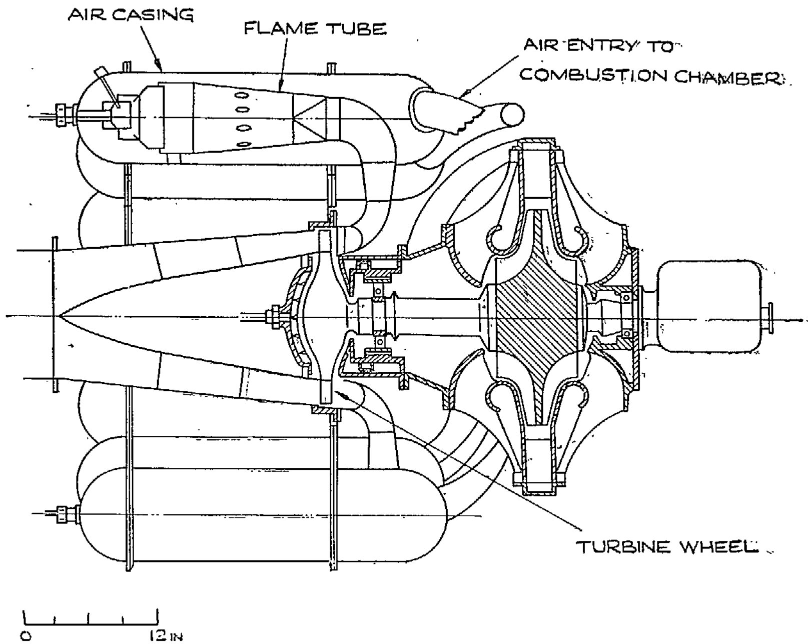


FIG. 1. Whittle engine W.1.U., 1939.

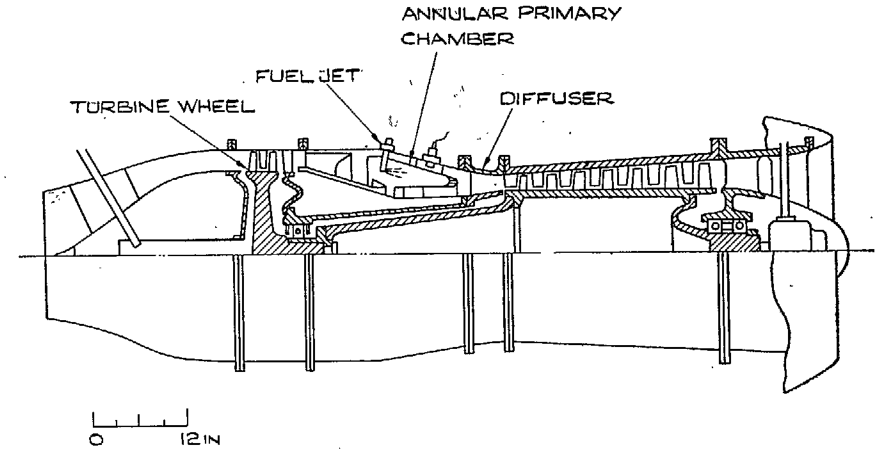


FIG. 2. R.A.E.-Metropolitan Vickers F.2 engine, 1940.

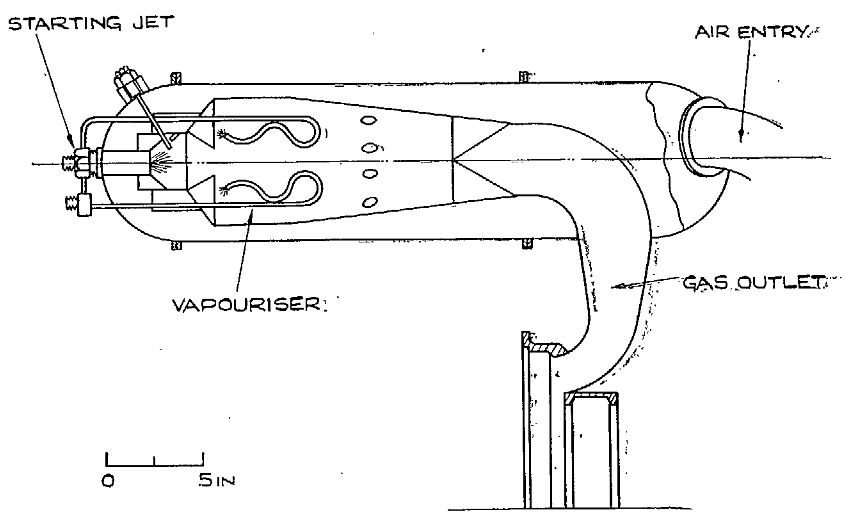


FIG. 3. 'Vap' chamber.

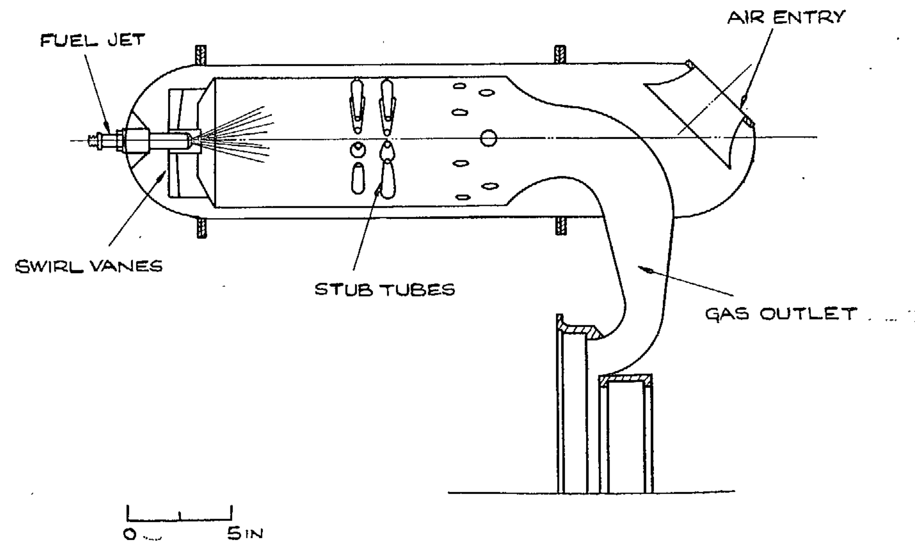


FIG. 4. Chamber type 75.

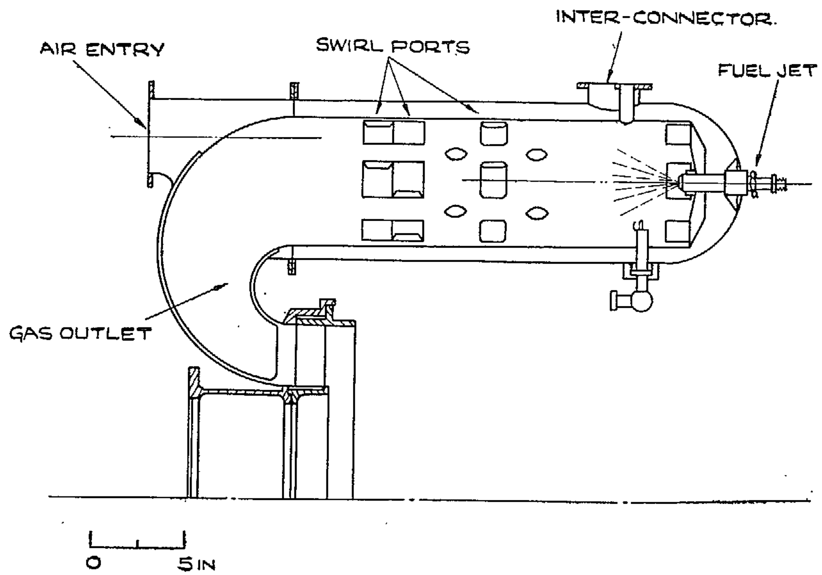


FIG. 5. Chamber type 102T.

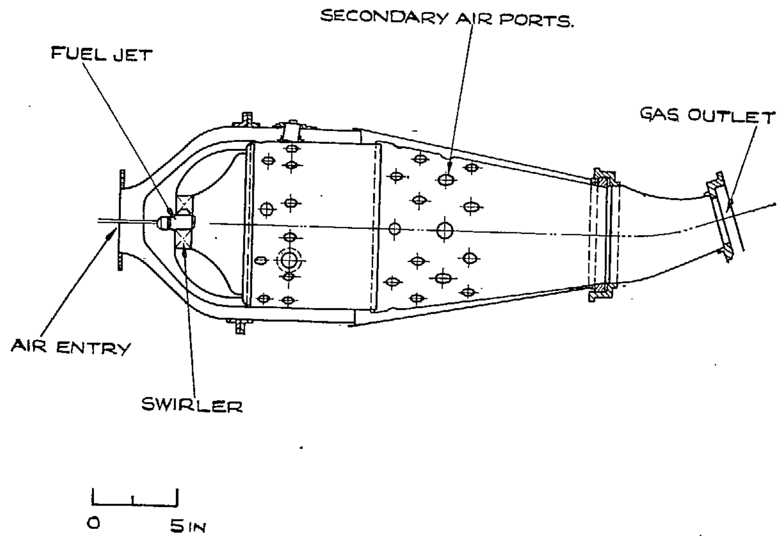


FIG. 6. Lucas combustion chamber B.37, Mk. II.

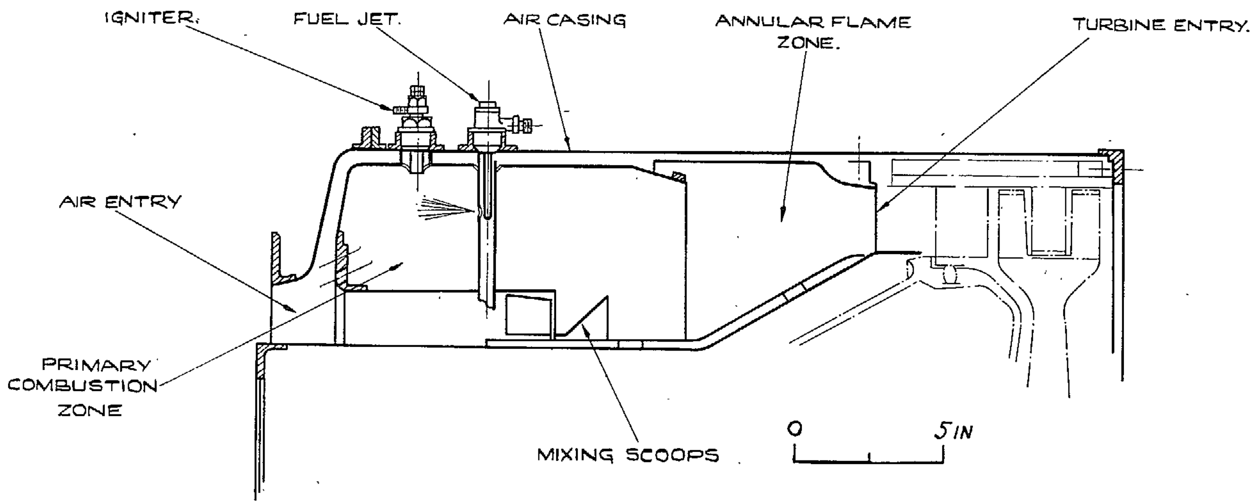


FIG. 7. Assembly of combustion chamber C.C.21. Modern lightened design. F.2 turbo-compressor.

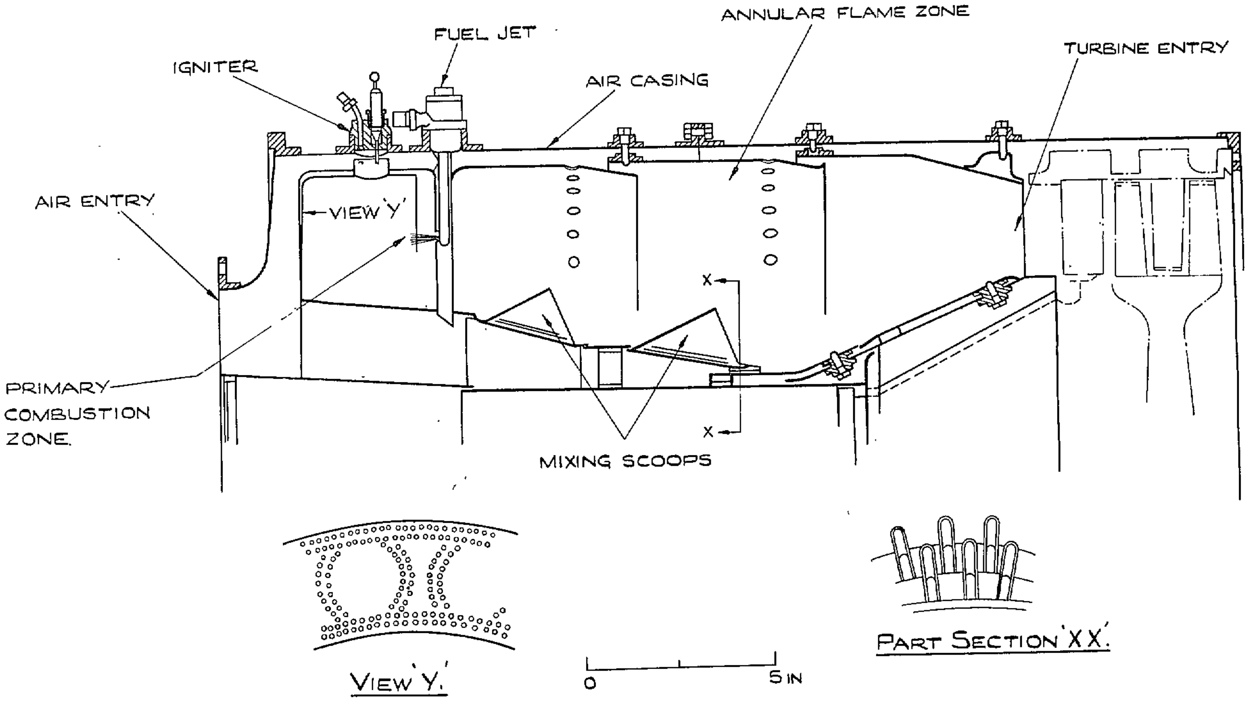


FIG. 8. F.2 unit No. 5 combustion chamber, C.C.41.

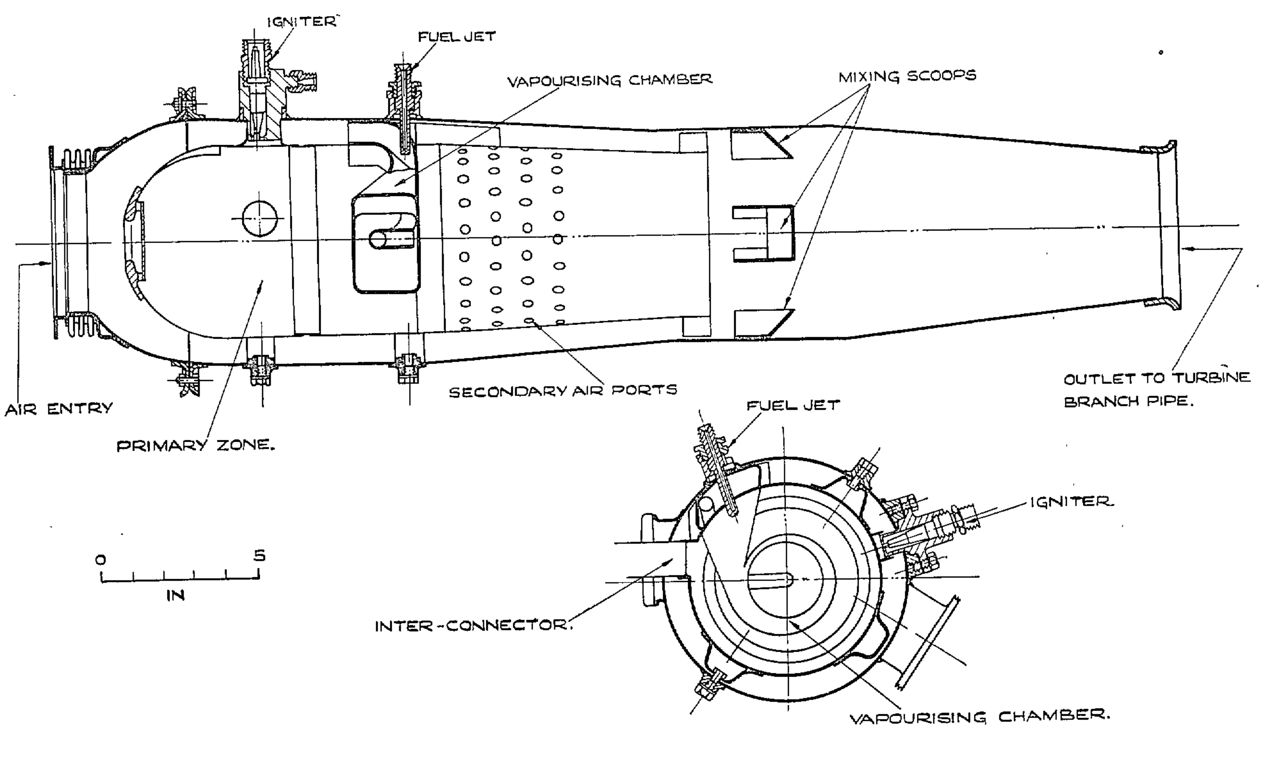


FIG. 9. A.S.X. combustion chamber.

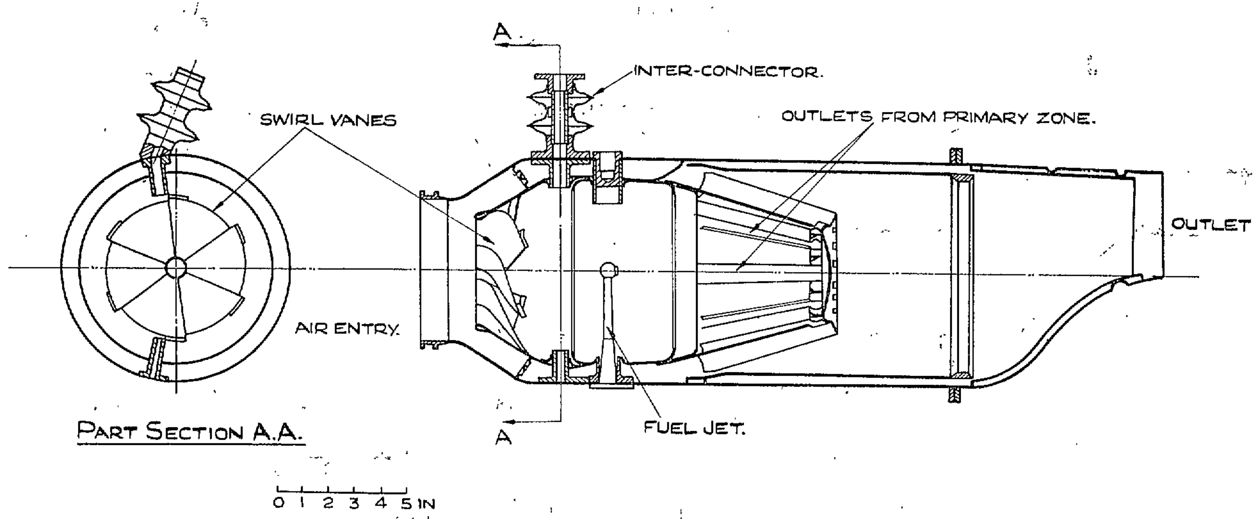


FIG. 10. Jumo 004 combustion chamber.

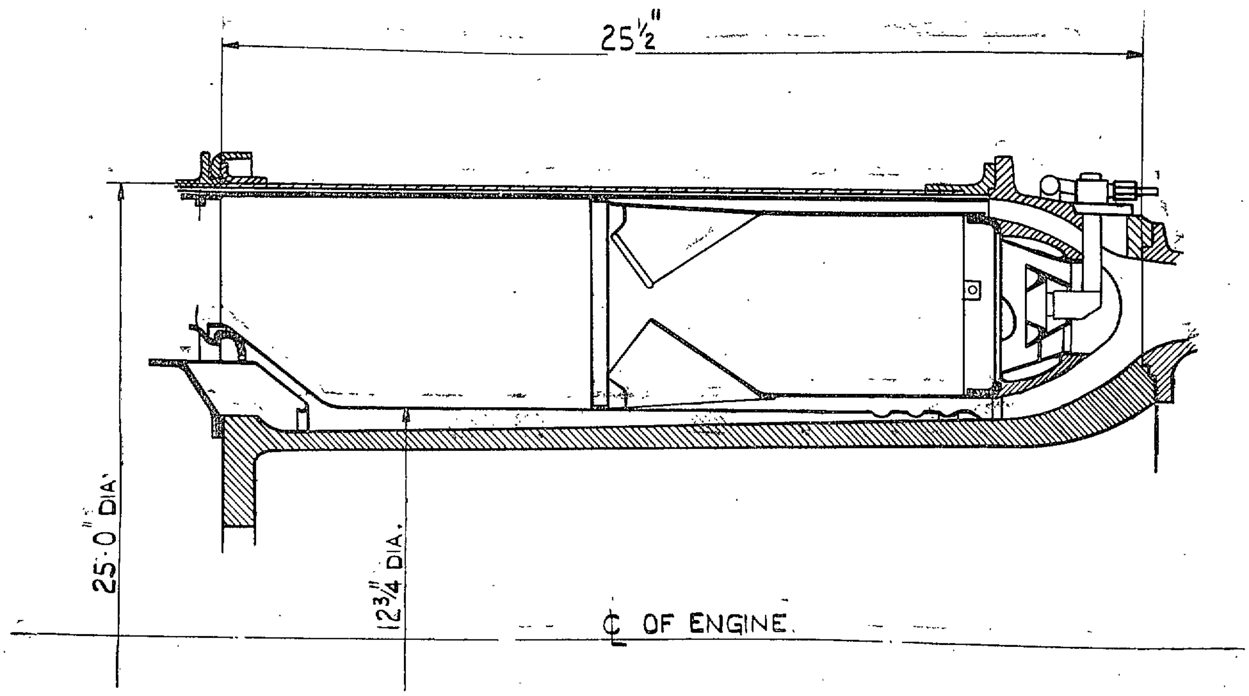


FIG. 11. B.M.W. 003 combustion chamber.

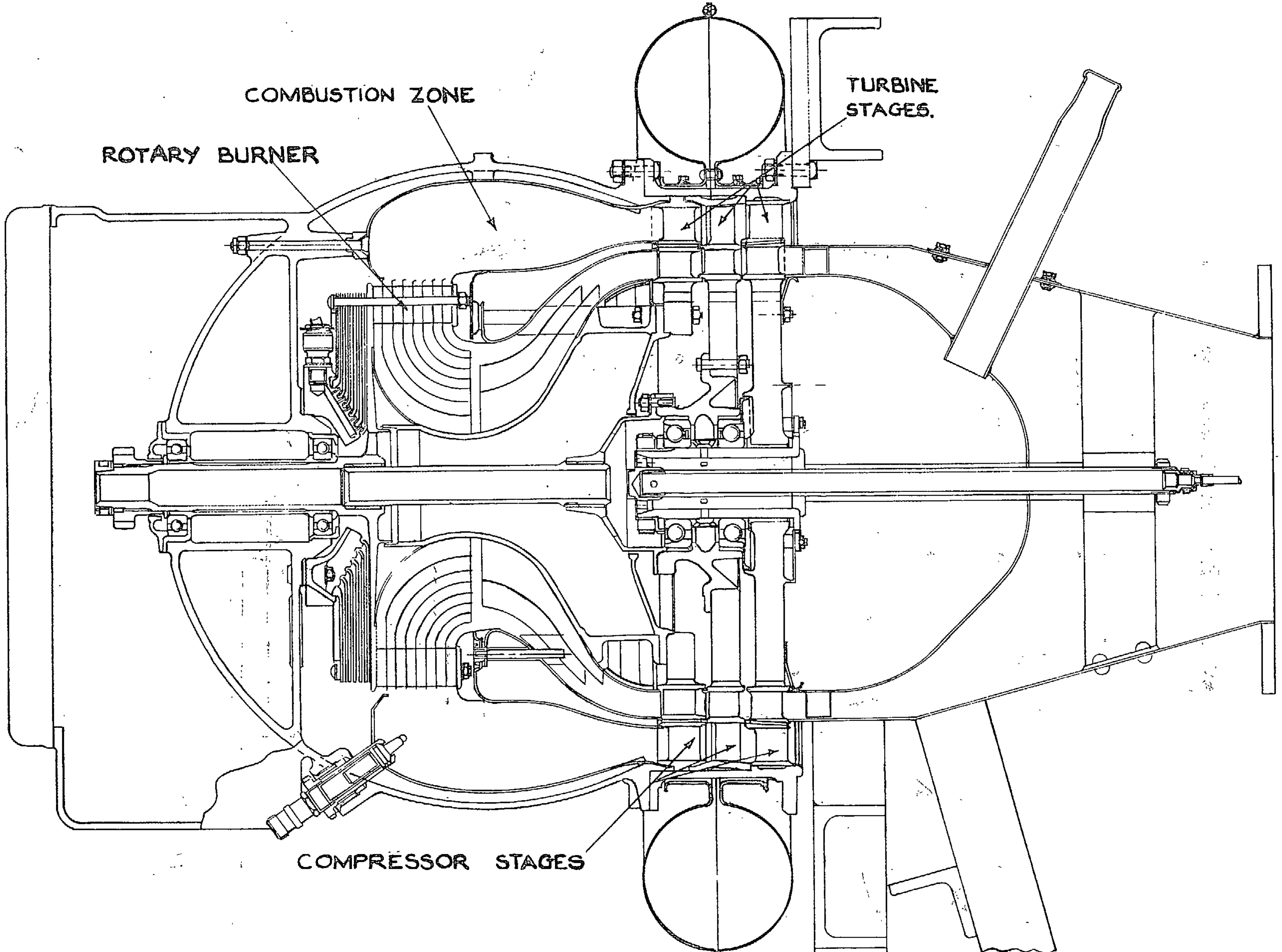


FIG. 12. C.R.1 combustion rig.

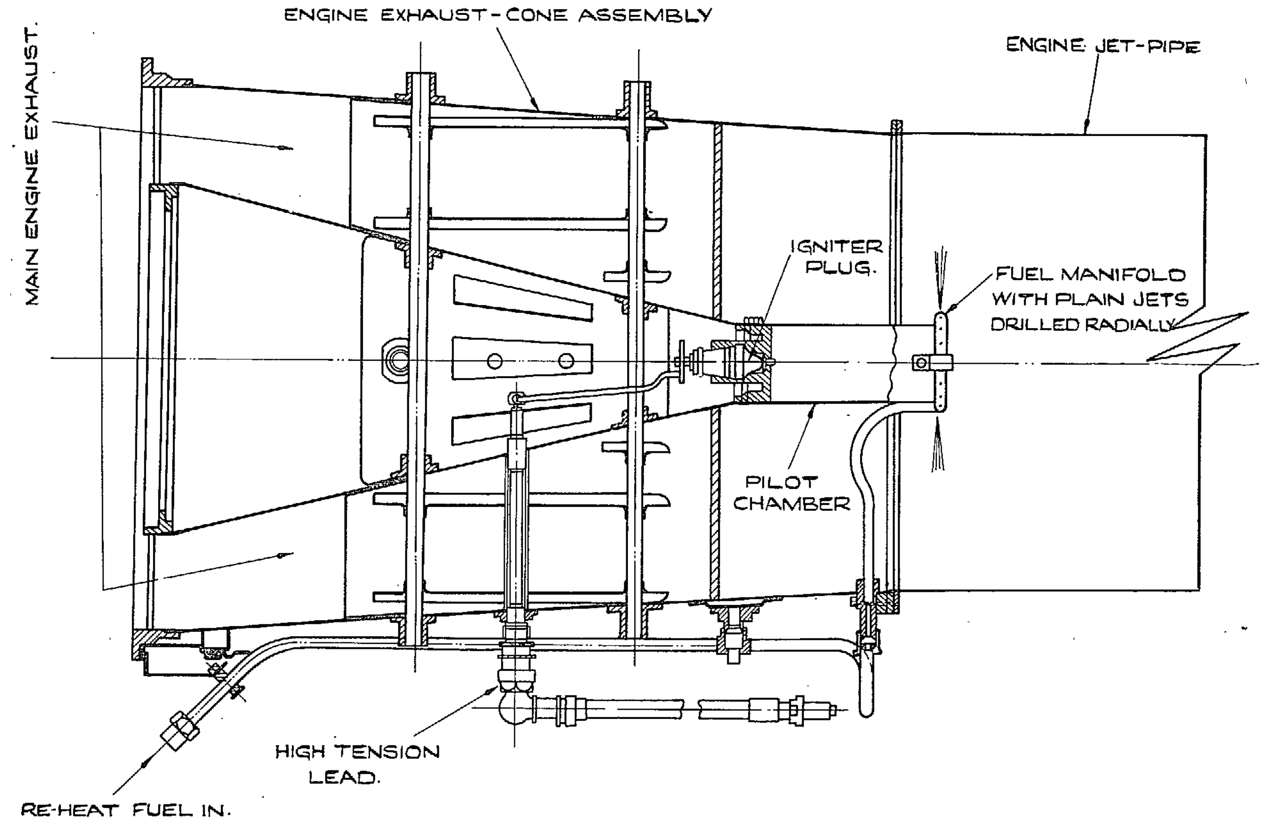


FIG. 13. Reheat system.

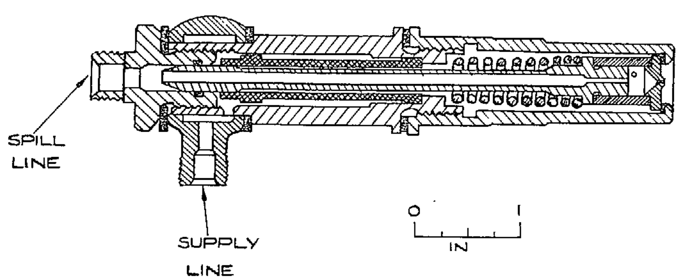


FIG. 14. Spill-controlled burner.

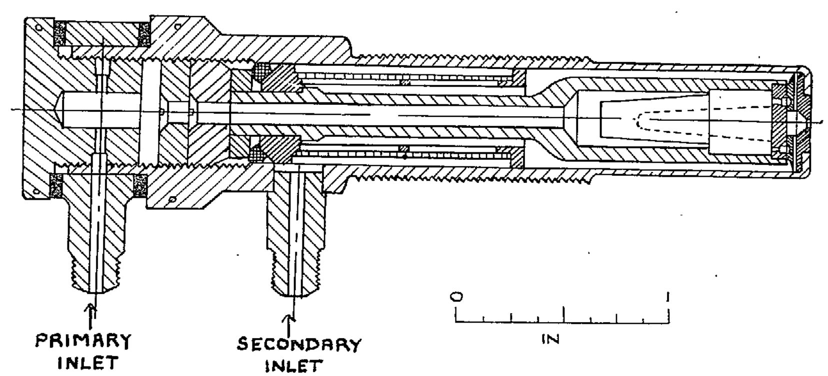


FIG. 15. Duplex burner.

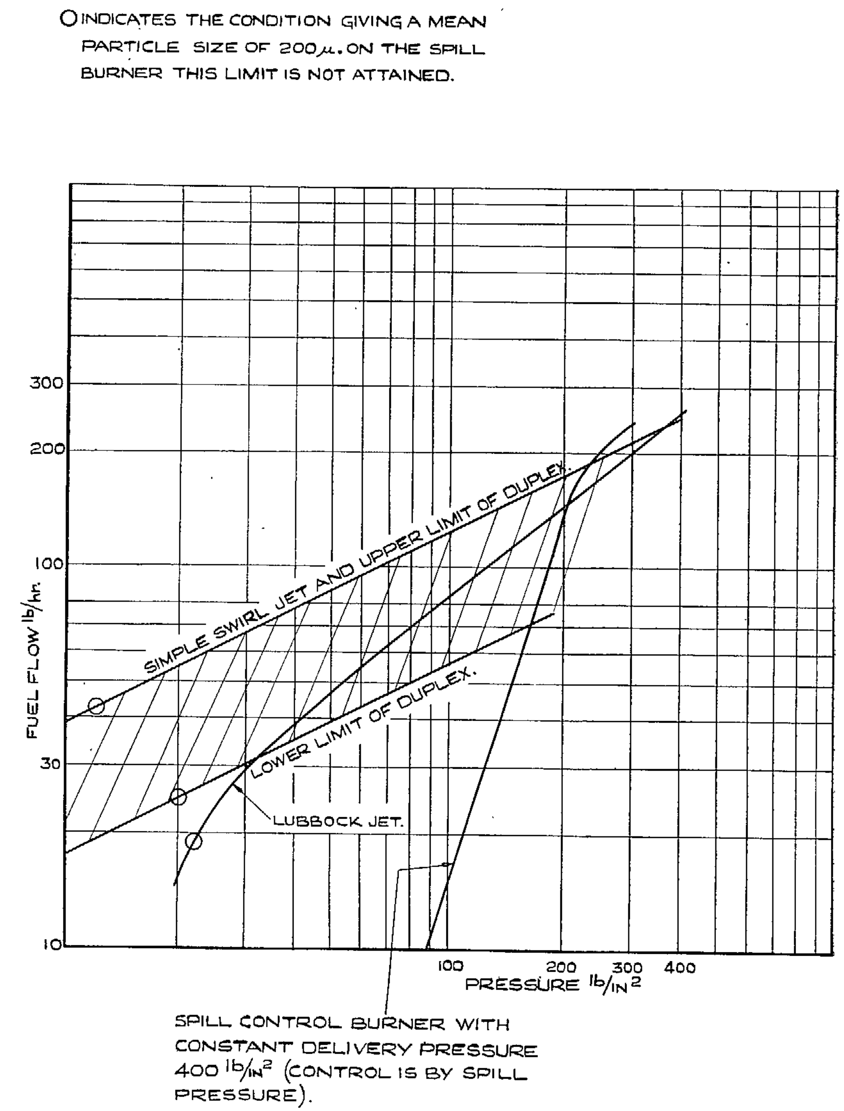


FIG. 16. The fuel flow *vs.* pressure relation for some burners.

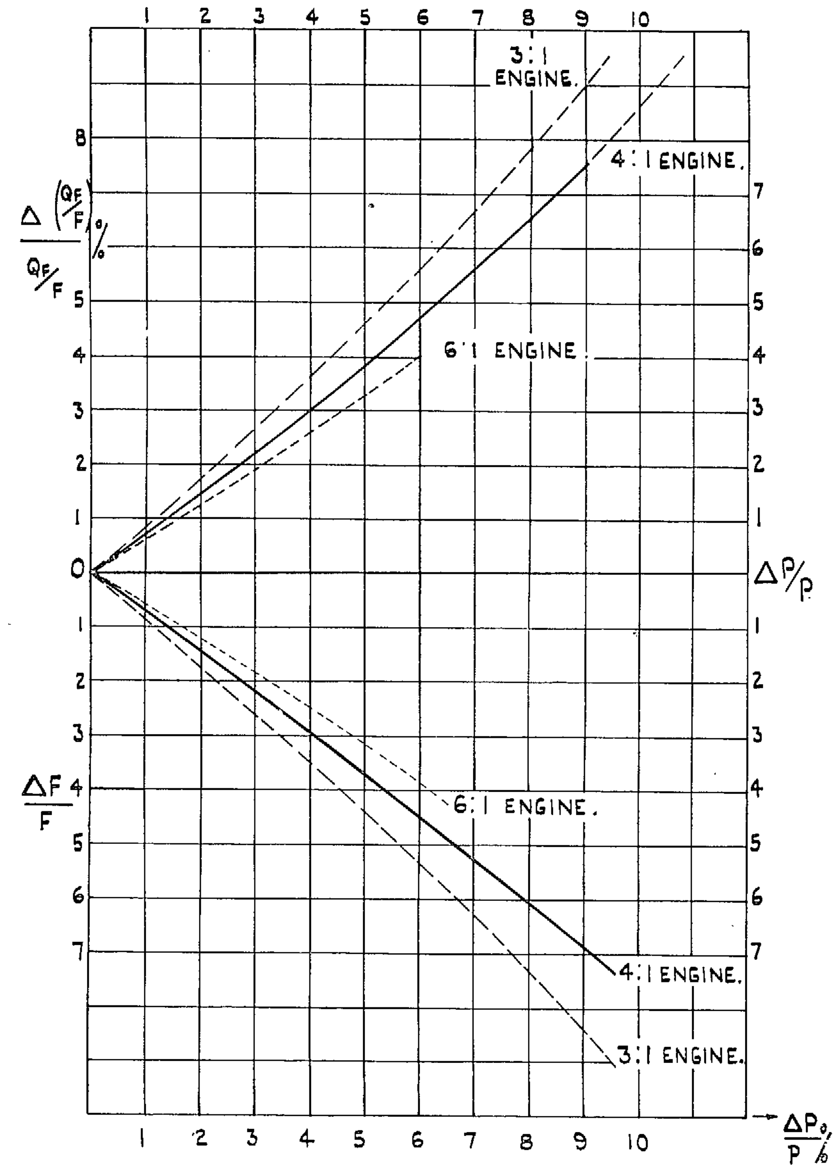


FIG. 17. Effect of pressure loss on engine performance.

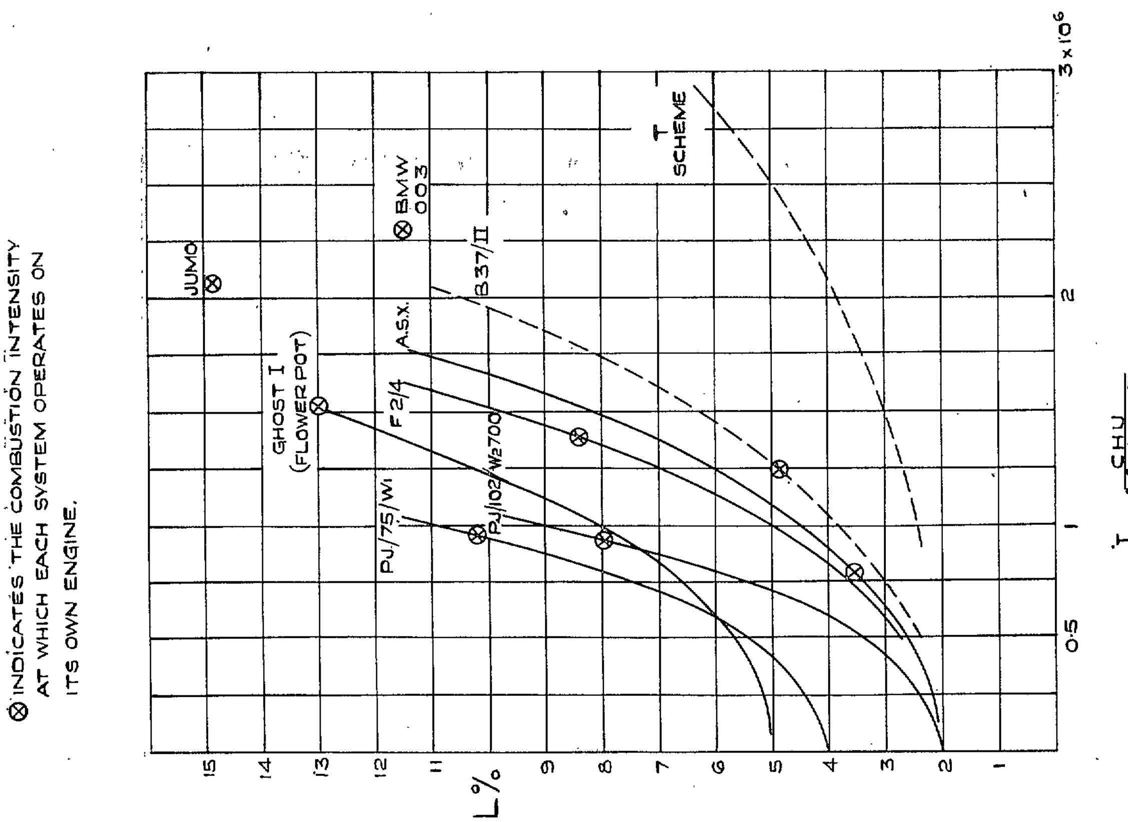


FIG. 18. Loss factor as a function of combustion intensity for various combustion systems (based on 4 : 1 jet engine, g.l.s.).

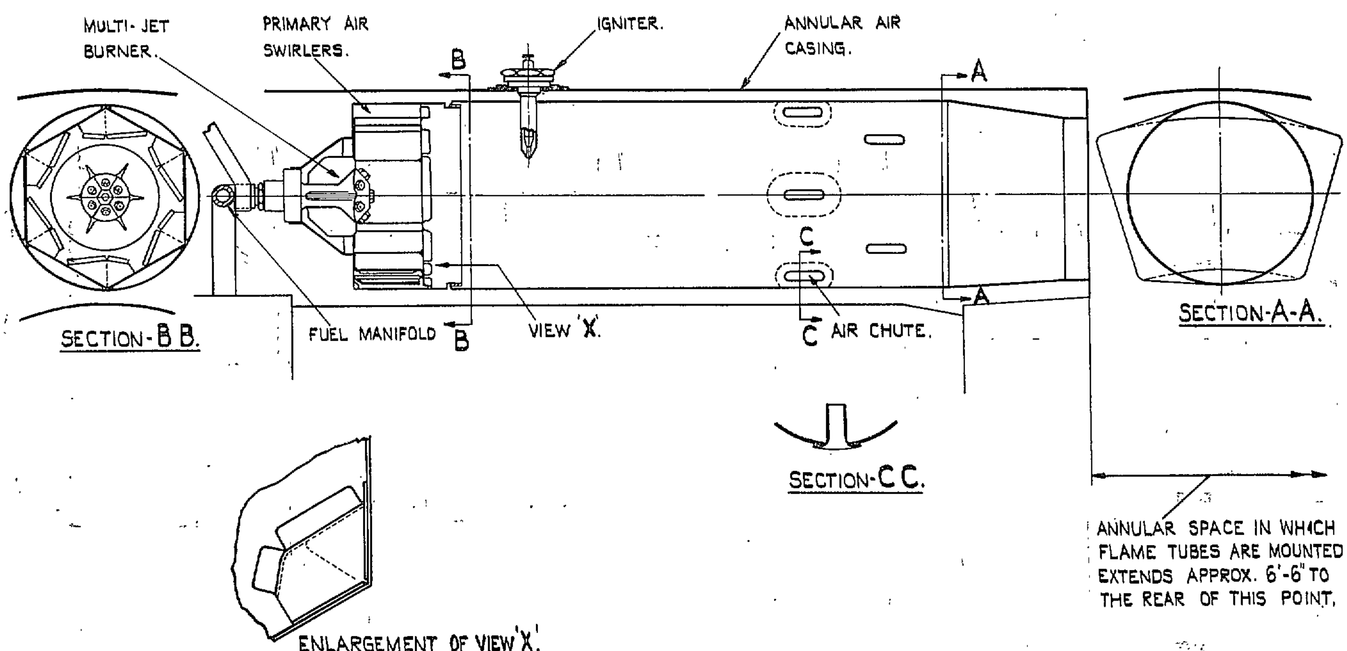


FIG. 19. Combustion chamber for augmenter (M.52 project).

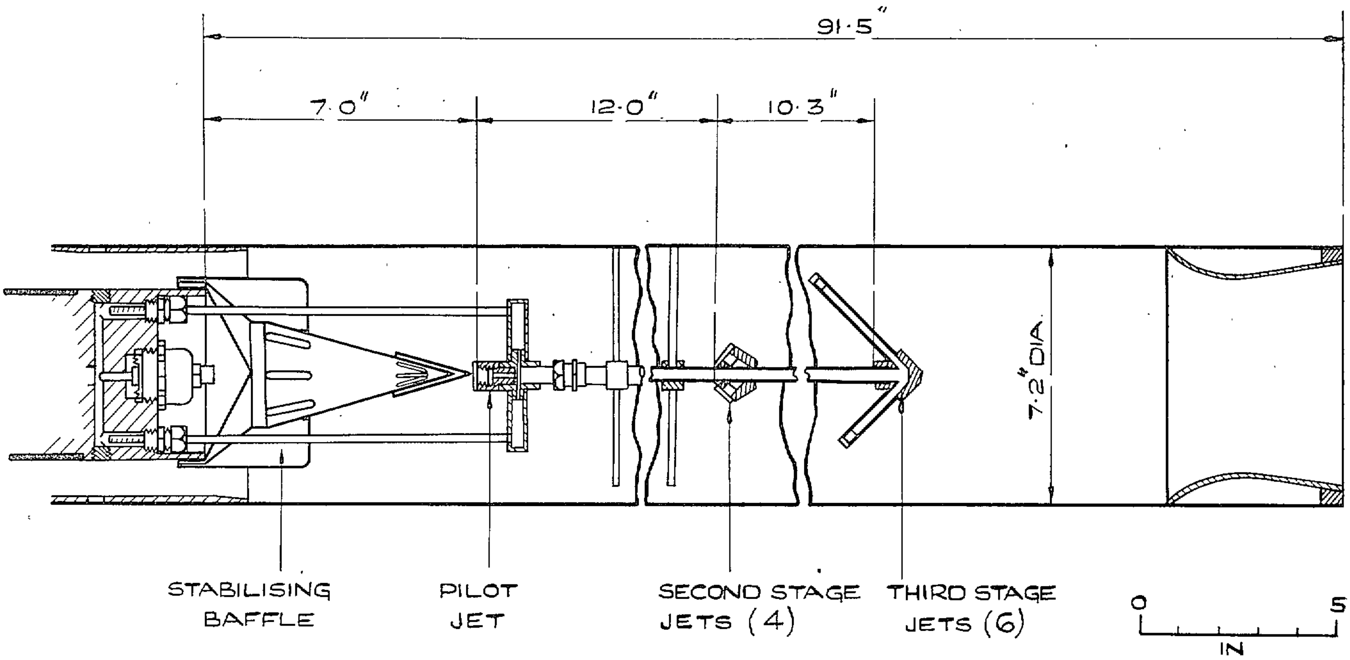


FIG. 20. G.A.P. propulsive duct.

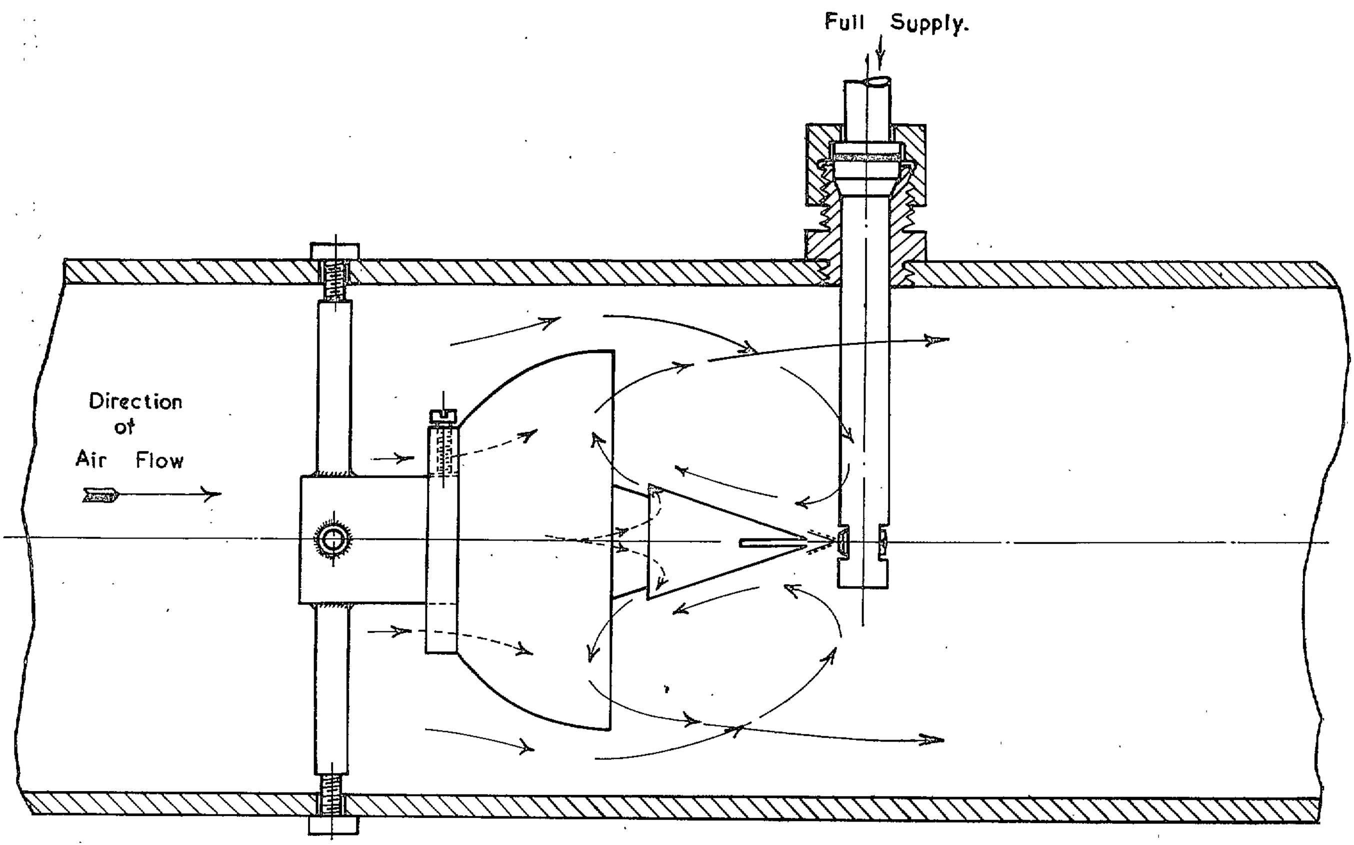


FIG. 21. Arrangement of baffle system.

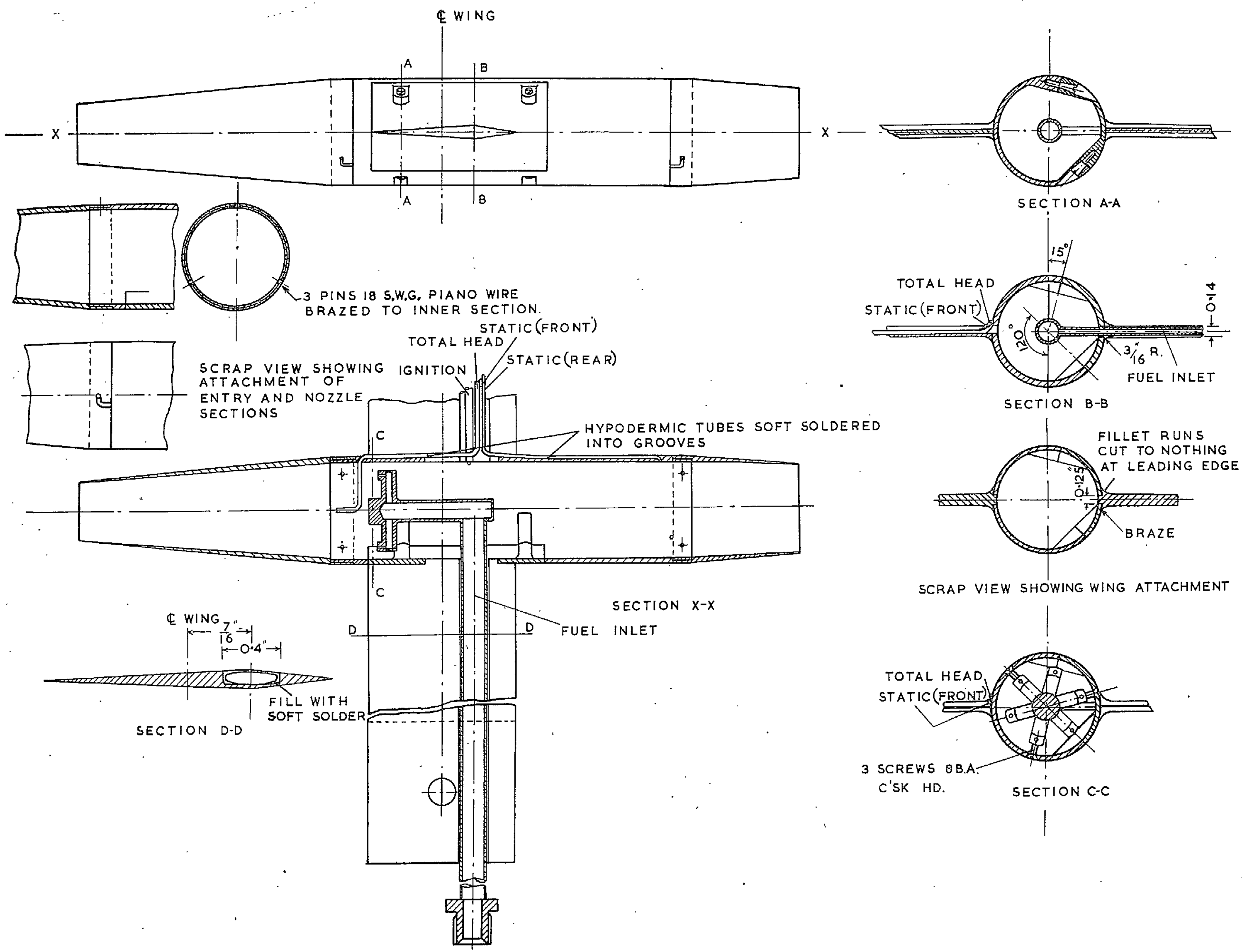


FIG. 22. General arrangement of model method—Scale 2 : 1.

(98656)

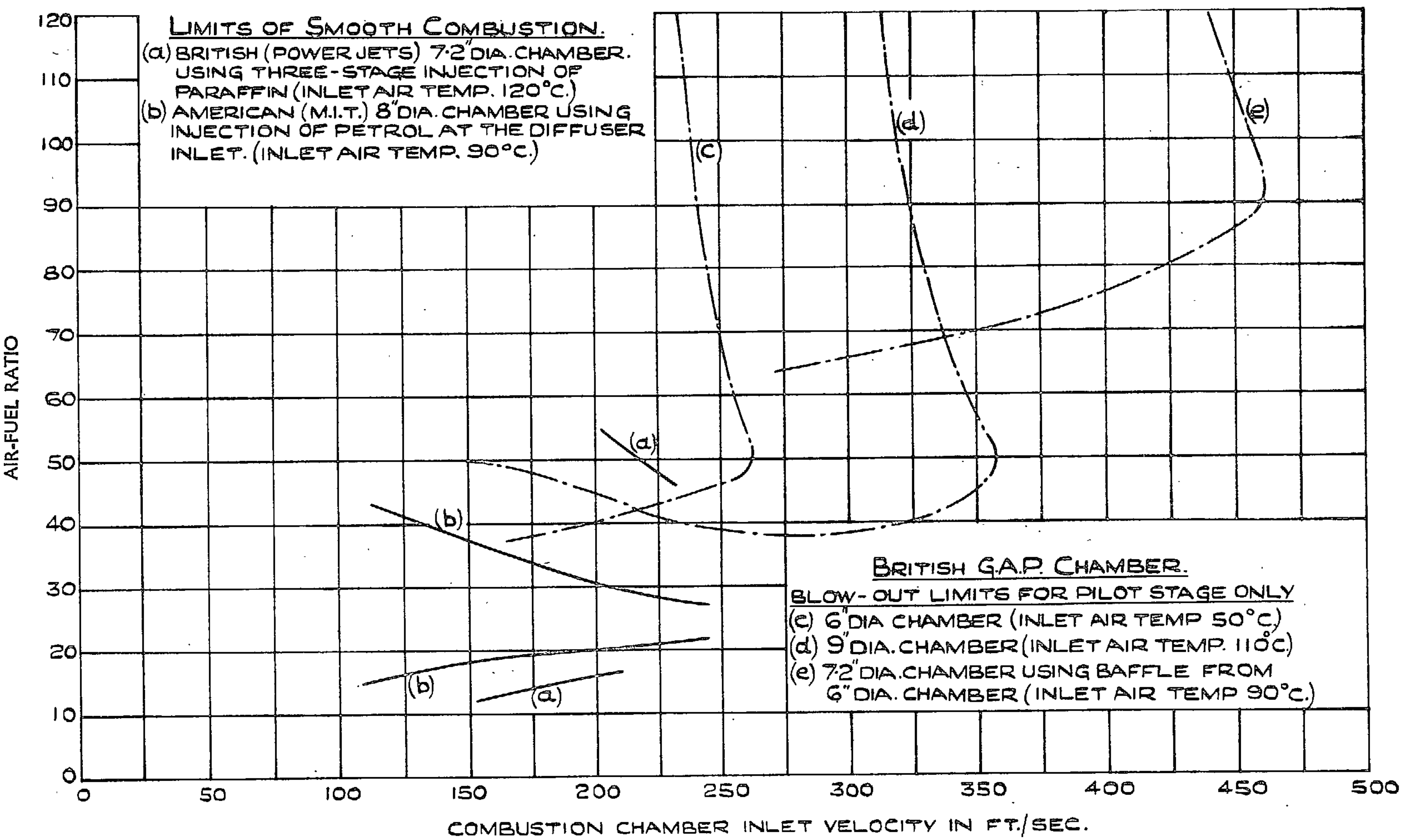


FIG. 23. Stability limits for various propulsive duct combustion systems.

C*

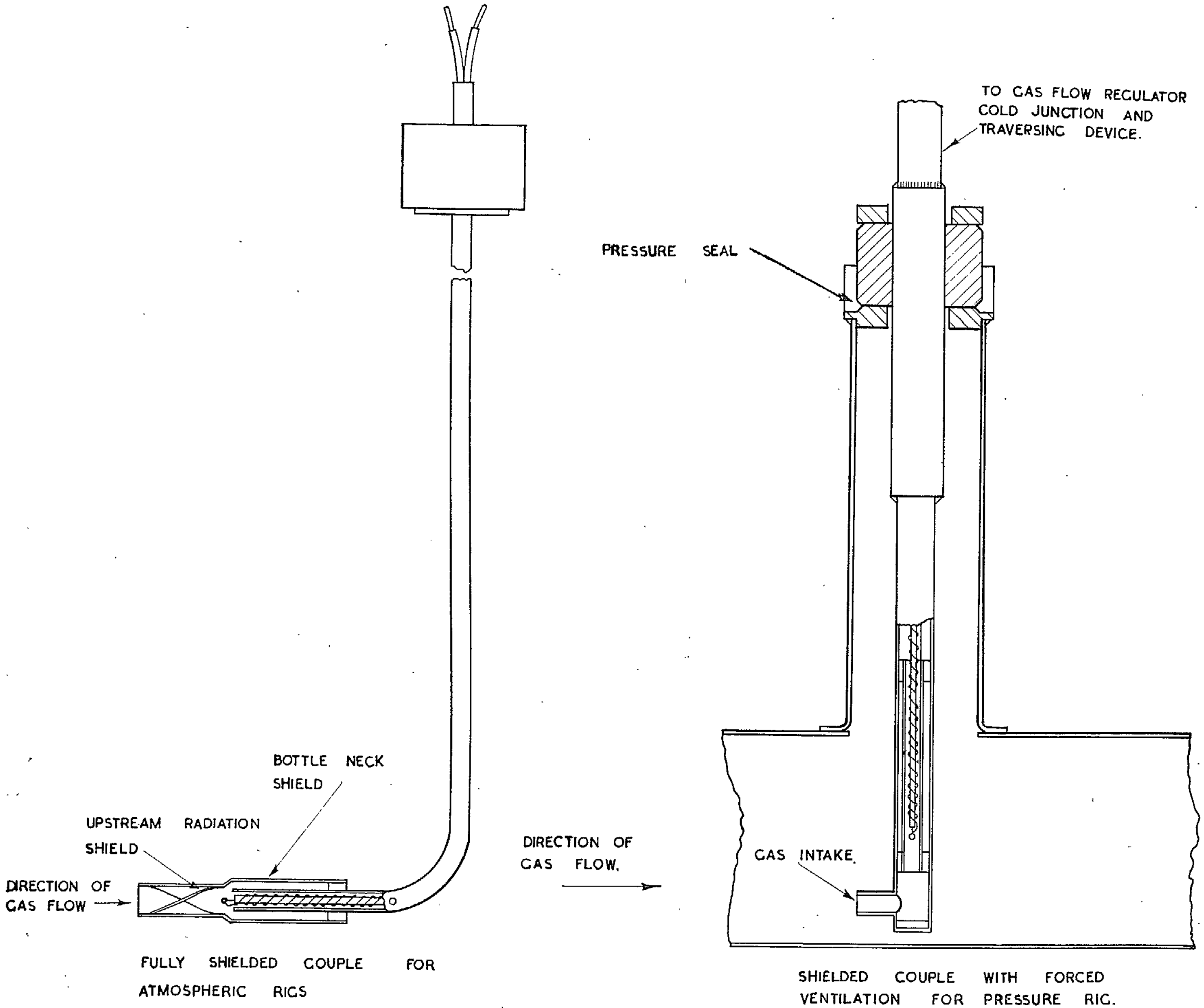


FIG. 24. Thermocouples for gas temperature measurement.

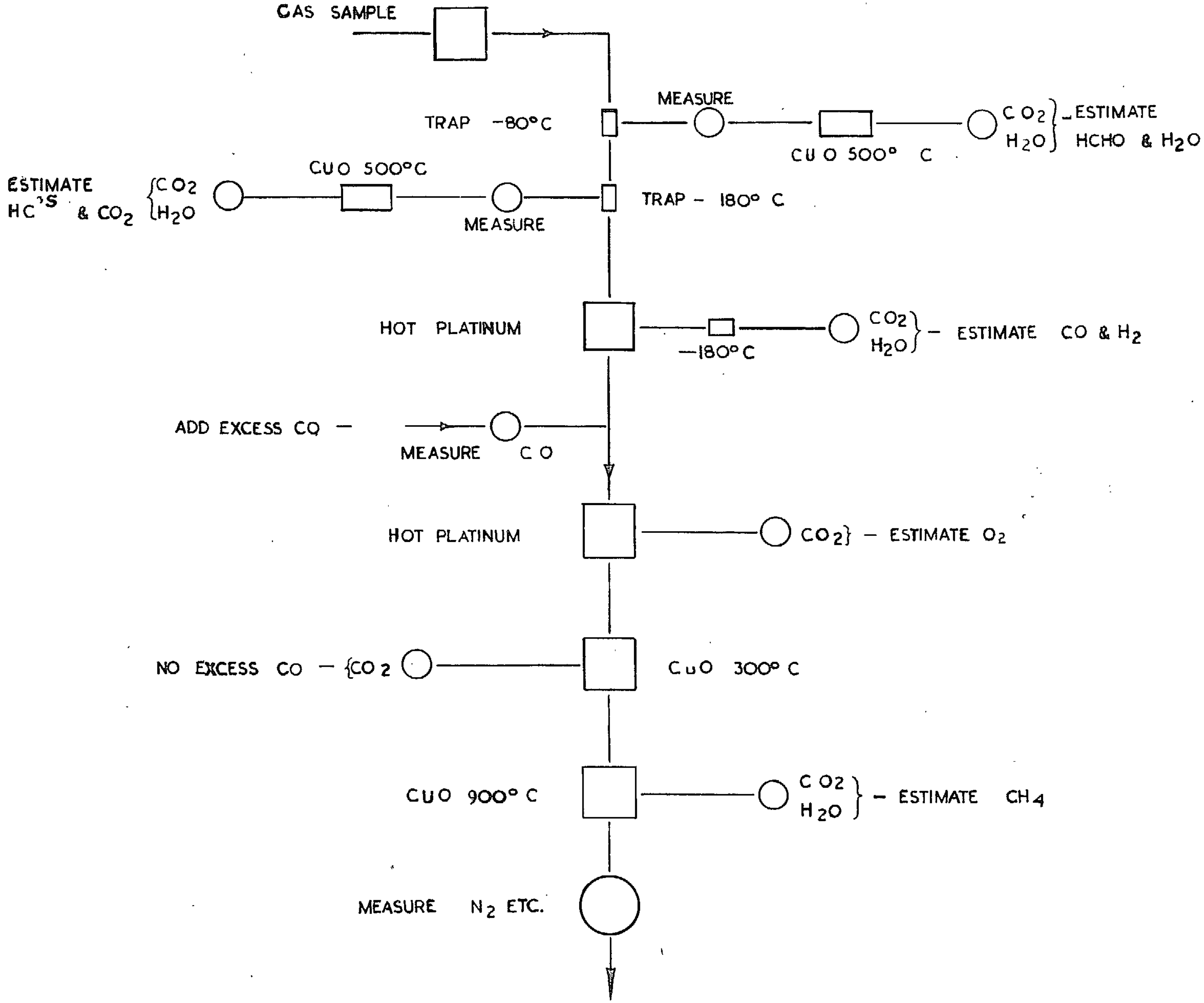


FIG. 25. High-vacuum gas analysis.

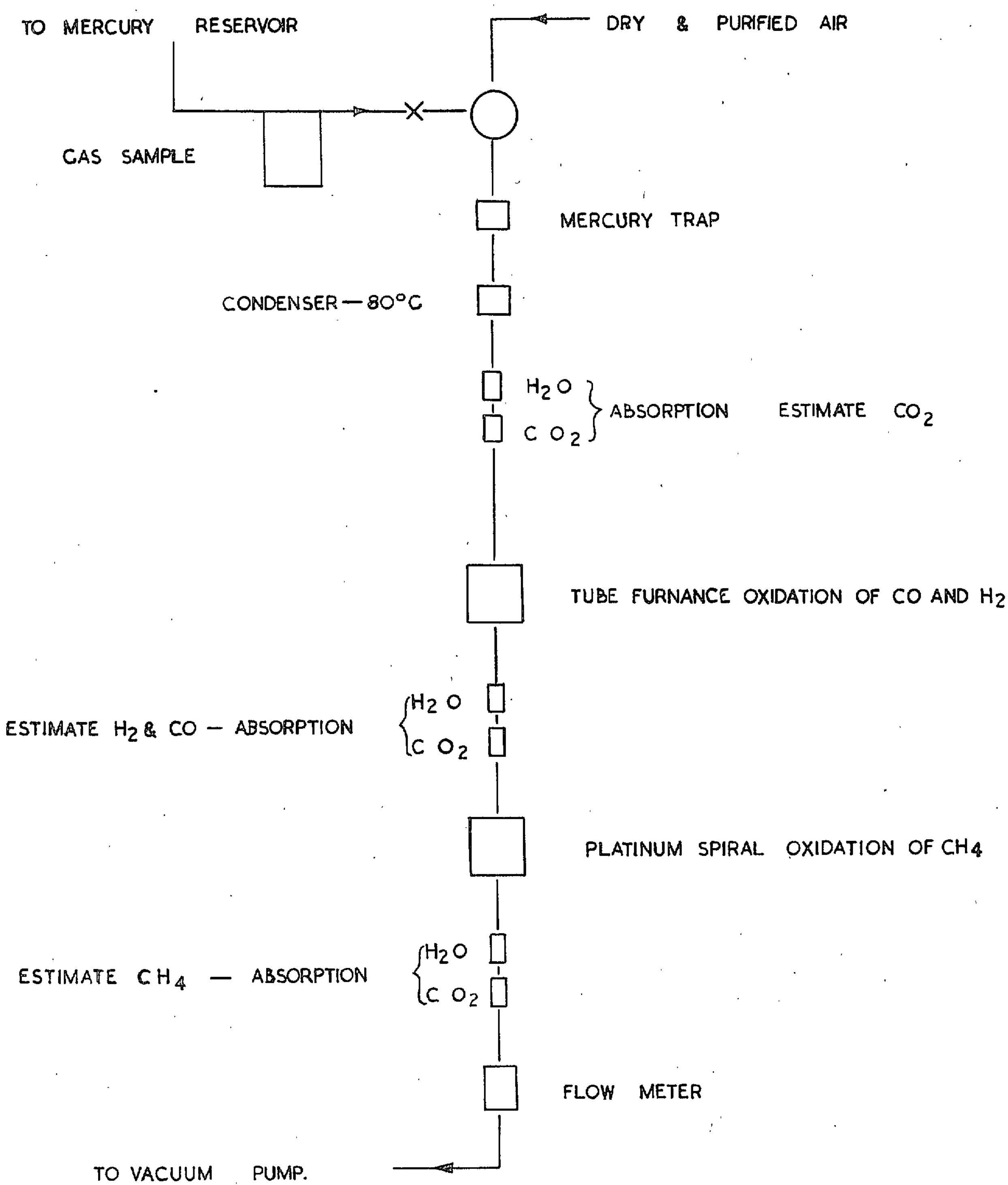


FIG. 26. Gravimetric gas analysis.

(94988)

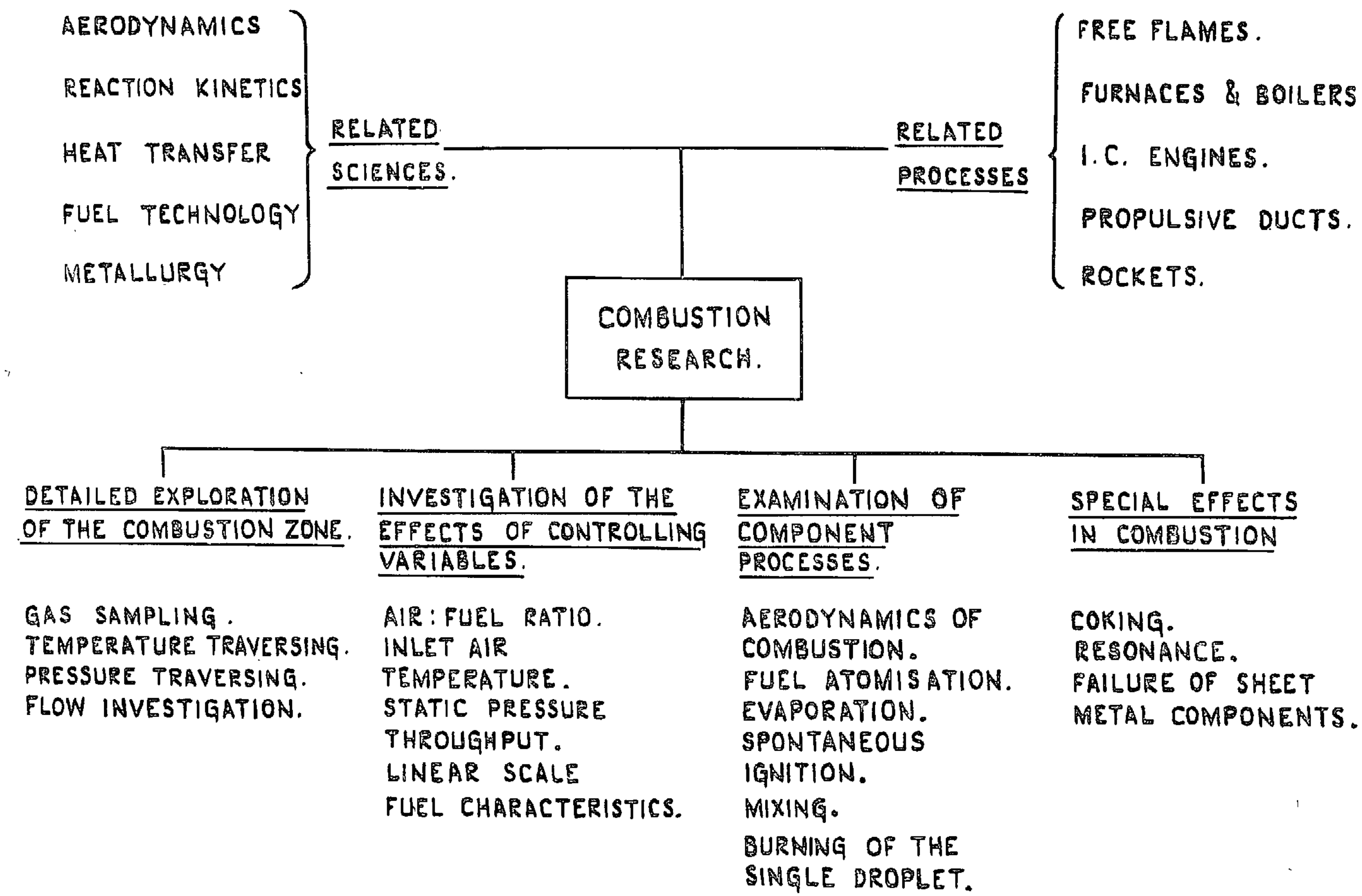
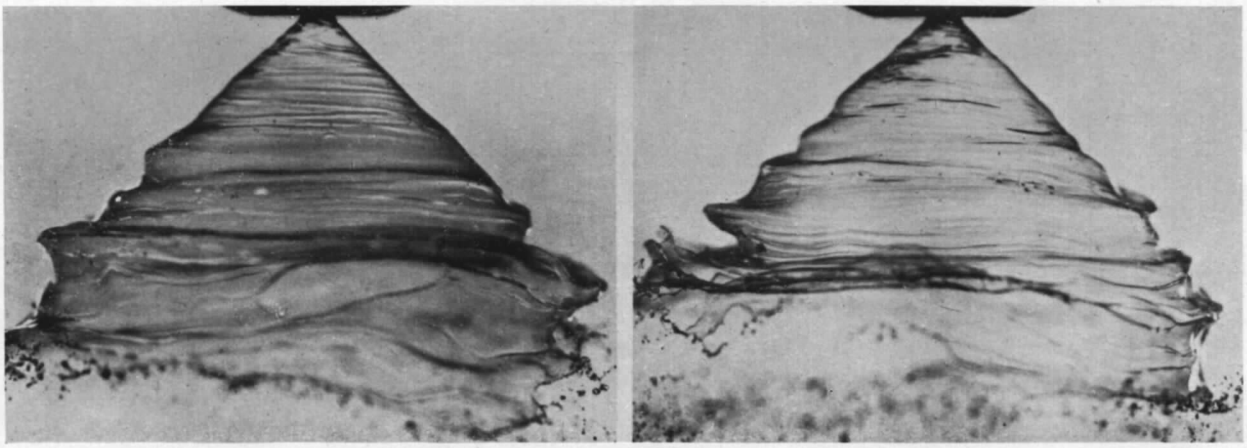


FIG. 27. Gas-turbine combustion ; the field of research.

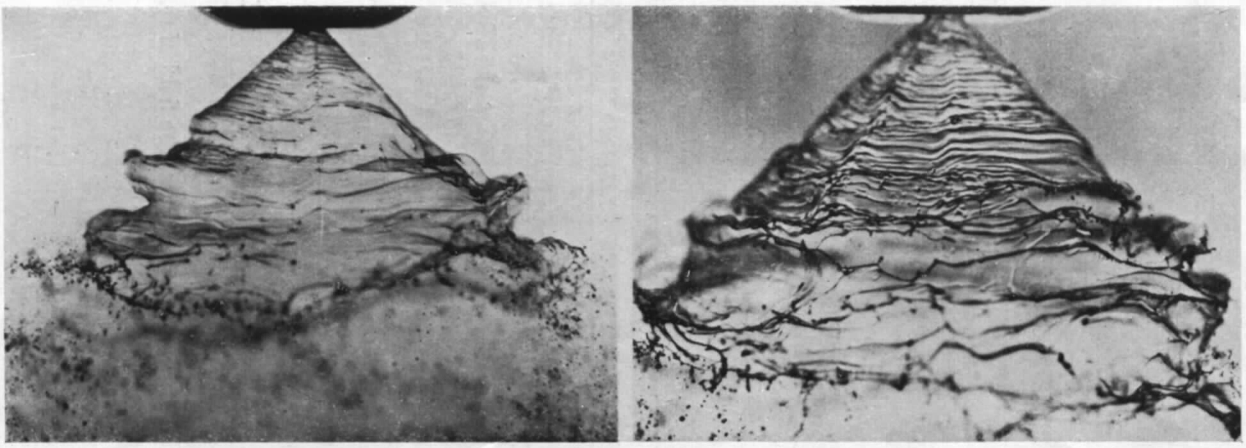
C**

Liquid: P.B.O.

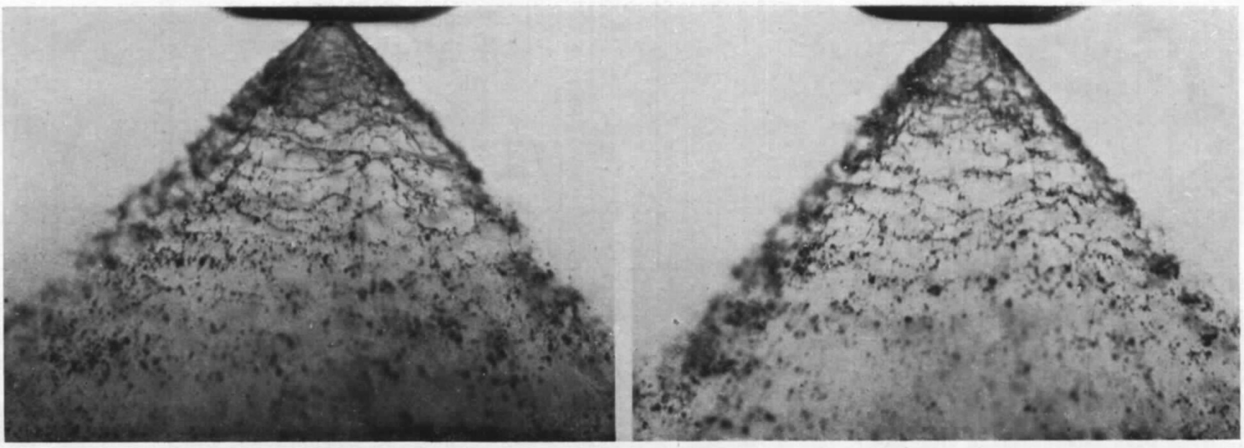
Viscosity 2·4 Centistrokes



Inlet Pressure 16 p.s.i.



Inlet Pressure 30 p.s.i.



Inlet Pressure 100 p.s.i.

FIG. 28.

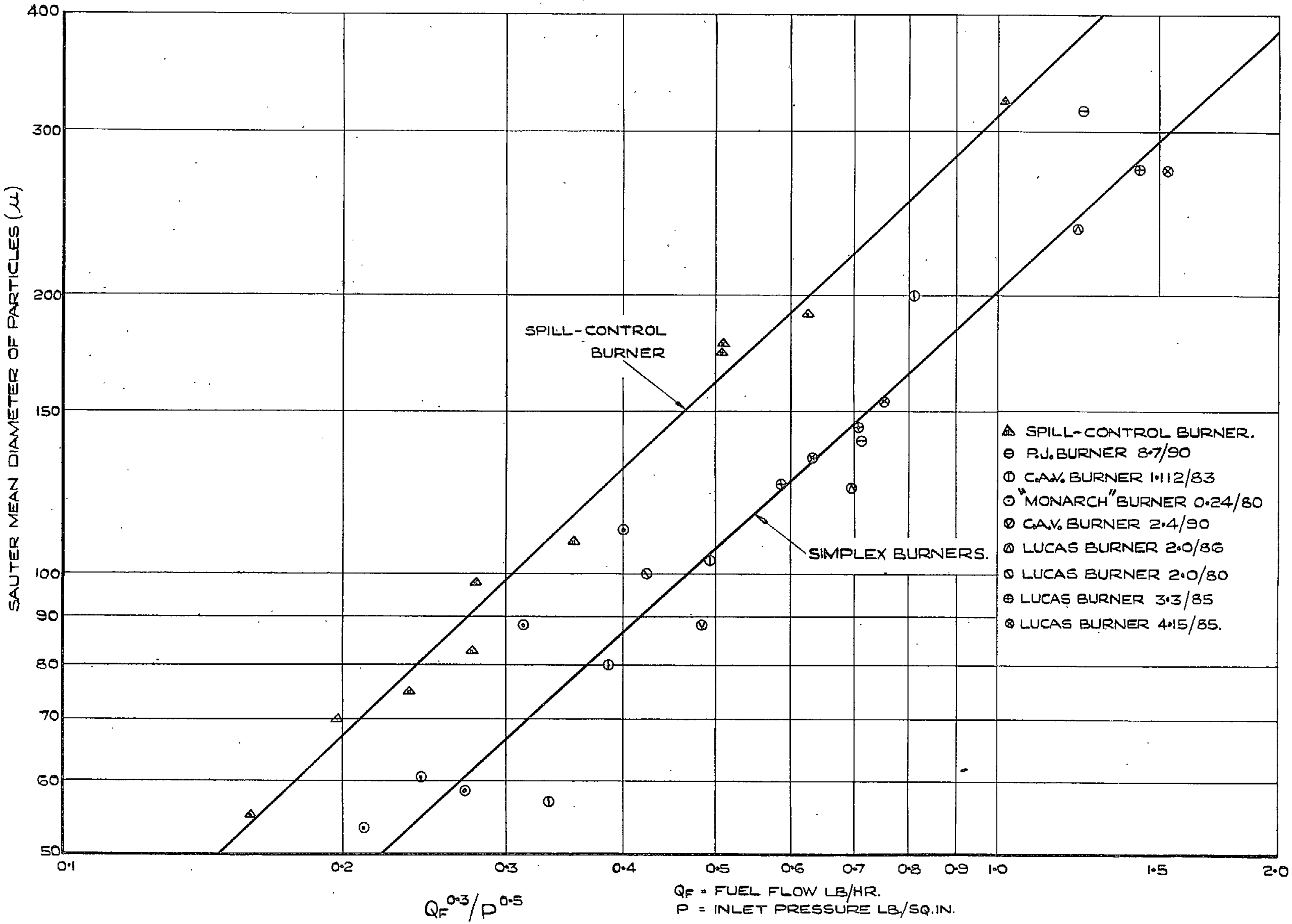


FIG. 29. Variation of spray particle size with fuel flow and pressure.

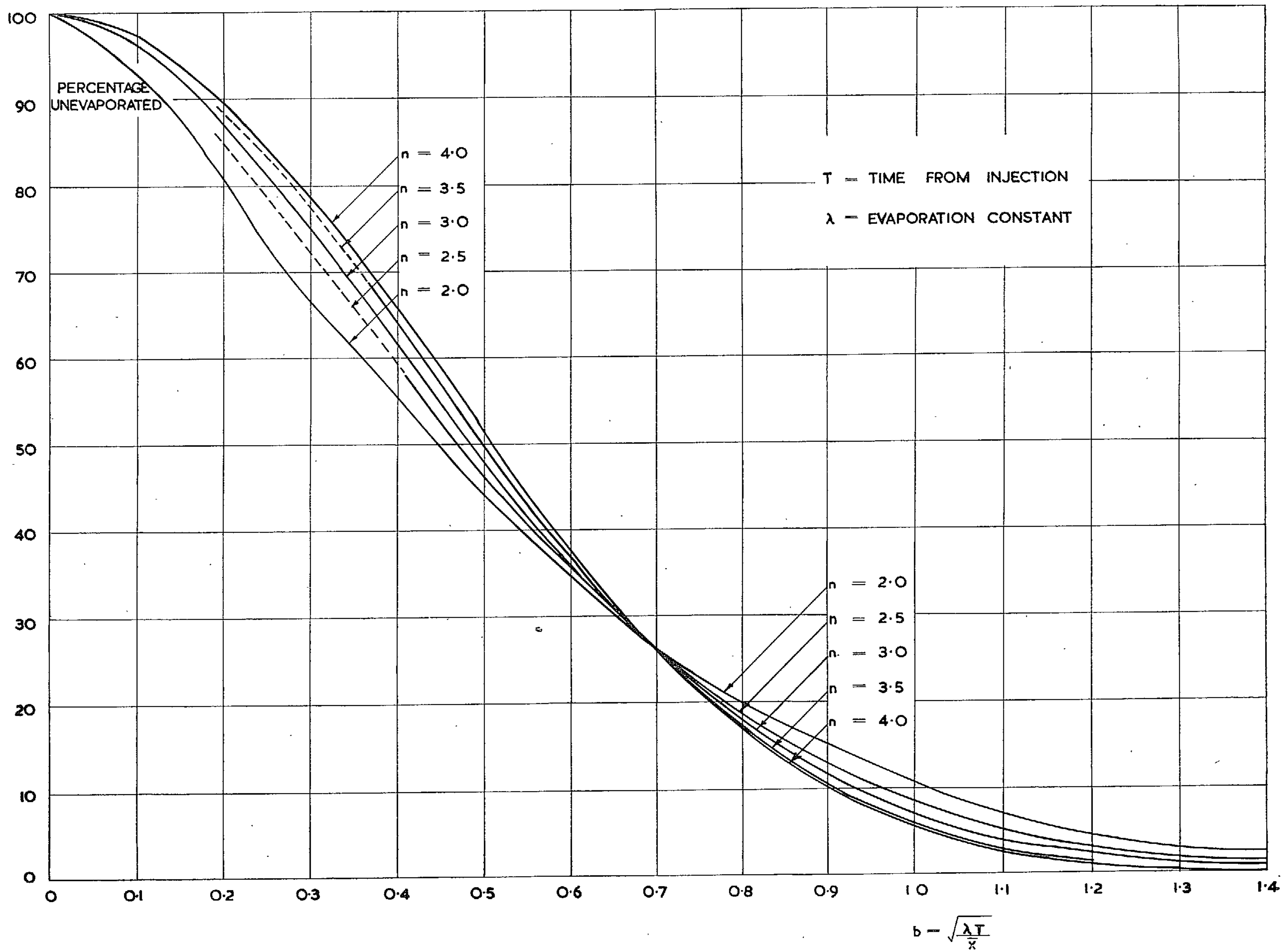
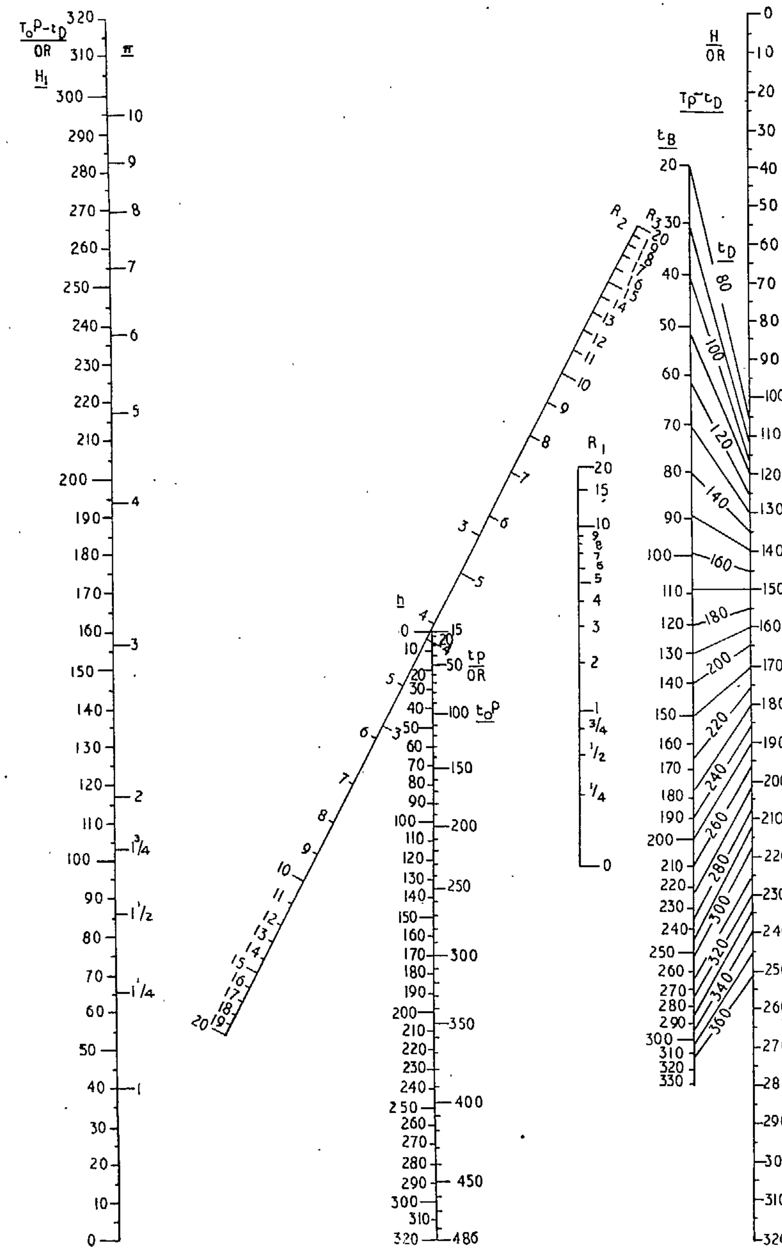


FIG. 30. Evaporation of a spray.



KEY TO VOFAM CHART

FOR A GIVEN AIR-FUEL RATIO R, AND TOTAL PRESSURE, π, EQUILIBRIUM VAPORISATION CHARACTERISTICS ARE OBTAINED FROM THE VOFAM CHART, WITHIN THE AIR-FUEL RANGE 5 1 TO 20·1 AND TOTAL PRESSURE RANGE 1-10 ATMOS. AS FOLLOWS.

CHARACTERISTIC	SYMBOL	OPERATIONS	RELEVANT SCALES.	EXAMPLE R=10 π =4
BUBBLE POINTS (=NORMAL BUBBLE POINTS WHEN R=0)	t_B	CONNECT π BY STR EDGE THRO' R ON THE R_1-SCALE TO t_B AND READ	π R_1 t_B	t_B=93°c
DEW POINTS (=NORMAL DEW POINTS WHEN R=0)	t_D	CONNECT π BY STR EDGE THRO' R ON THE R_1-SCALE TO t_B AND READ THE CORRESPONDING t_D FROM THE t_B/t_D DUPLEX SCALE	π R_1 t_D	t_D= 154°c
HEAT CONTENT OF THE VAPORISED FUEL AT THE DEW-POINT TEMP REFERRED TO A LIQUID PHASE TEMP. 15°C	H	FIRST DETERMINE t_D AS ABOVE THEN READ H CORRESPONDING TO THIS VALUE OF t_D FROM THE t_D/ H DUPLEX SCALE	t_D H	H=141½ CHU/LB
HEAT CONTENT OF THE VAPORISED FUEL AT THE DEW-POINT TEMP REFERRED TO A LIQUID PHASE PREHEAT TEMP t_P °C	H_P	FIRST DETERMINE H AS ABOVE THEN CONNECT H BY STR EDGE THROUGH t_P to H_1 AND READ	H_1 t_P H (HERE H_1 IS H_P.)	SAY t_P=100°c H_P=98½ CHU/LB
HEAT CONTENT OF THE PREHEATED LIQUID FUEL AT t_P°C REFERRED TO 15° C (INDEPENDANT OF R AND π)	h	READ h DIRECTLY FROM THE h/t_P DUPLEX SCALE	h t_P	SAY t_P=100°c h=43 C.HU/LB
MINIMUM AIR TEMP FOR VAPORISATION STARTING WITH LIQUID FUEL AT 15°C.	T_o^{15}	FIRST DETERMINE H AS ABOVE THEN CONNECT H BY STR.EDGE THRO' R ON THE R_2 SCALE TO (T_oP-t_D) AND READ ADD t_D TO THIS VALUE GIVING T_o^{15}.	T_oP-t_D R_2 H ADD t_D (HERE T_oP IS T_o^{15})	T_o^{15}= 59+154 =213°C
MINIMUM AIR TEMP FOR VAPORISATION STARTING WITH LIQUID FUEL PREHEATED TO t_P°C	T_oP	FIRST DETERMINE H_P AS ABOVE THEN CONNECT H_P ON H-SCALE BY STR EDGE THRO' R ON R_2-SCALE TO (T_oP-t_D) AND READ. ADD t_D TO THIS VALUE GIVING T_oP	T_oP-t_D R_2 H ADD t_D (HERE H IS H_P)	SAY t_P=100°c T_oP= 41+154 =195°c
MINIMUM FUEL TEMP FOR VAPORISATION STARTING WITH AIR AT T_P°C ($T_P > t_D$)	t_oP ($T_P > t_D$)	SUBTRACT t_D FROM T_P CONNECT (T_P-t_D) BY STR EDGE THRO' R ON R_3-SCALE TO H_1 NOW CONNECT THIS VALUE OF H_1 BY STR. EDGE TO t_D THRO' t_oP WHICH IS READ DIRECTLY.	H_1 R_3 t_oP t_D T_P-t_D (HERE H_1 IS A PARAMETER)	SAY T_P=180°c t_oP=162°c
MINIMUM FUEL TEMP FOR VAPORISATION STARTING WITH AIR AT T_P°c ($T_P < t_D$)	t_oP ($T_P < t_D$)	SUBTRACT T_P FROM t_D CONNECT (t_D-T_P) BY STR EDGE THRO' R ON R_3-SCALE TO H_1. CONNECT THE VALUE $(H+H_1)$ ON THE H-SCALE TO 0 ON H_1-SCALE THRO' t_oP WHICH IS READ DIRECTLY	H_1 R_3 t_oP t_D T_P-t_D (HERE H_1 IS A PARAMETER)	SAY T_P=100°c t_oP=429°c

FIG. 31. Vofam chart. Reference Kerosene S.R.312.

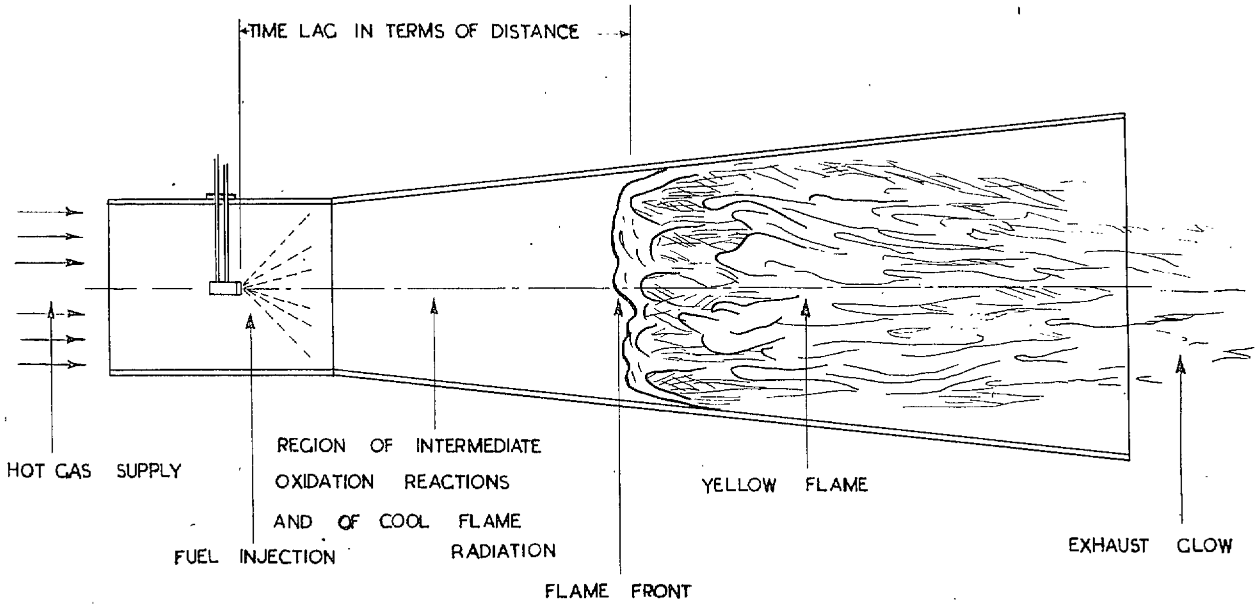


FIG. 32. Spontaneous ignition of liquid fuels.

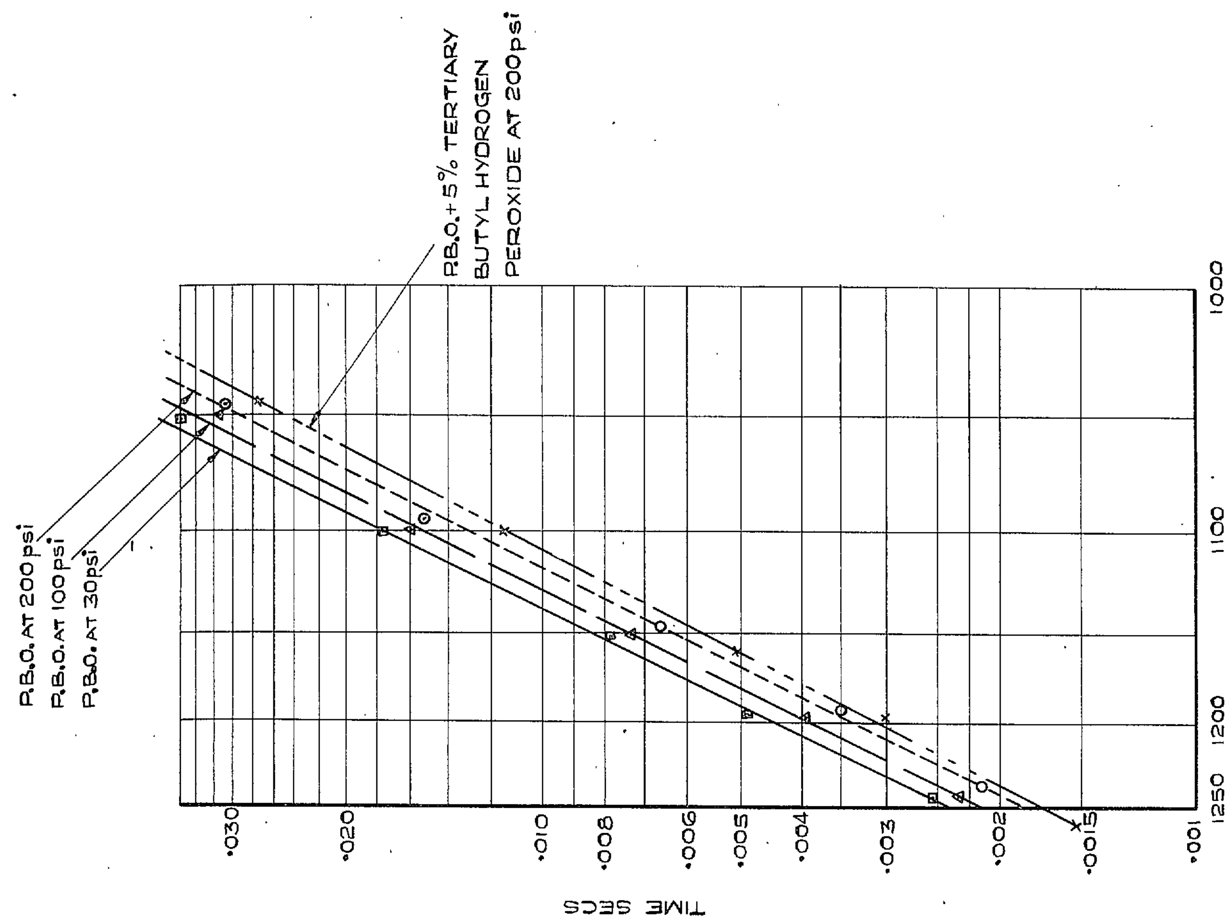


FIG. 33. Variation of ignition lag with temperature.

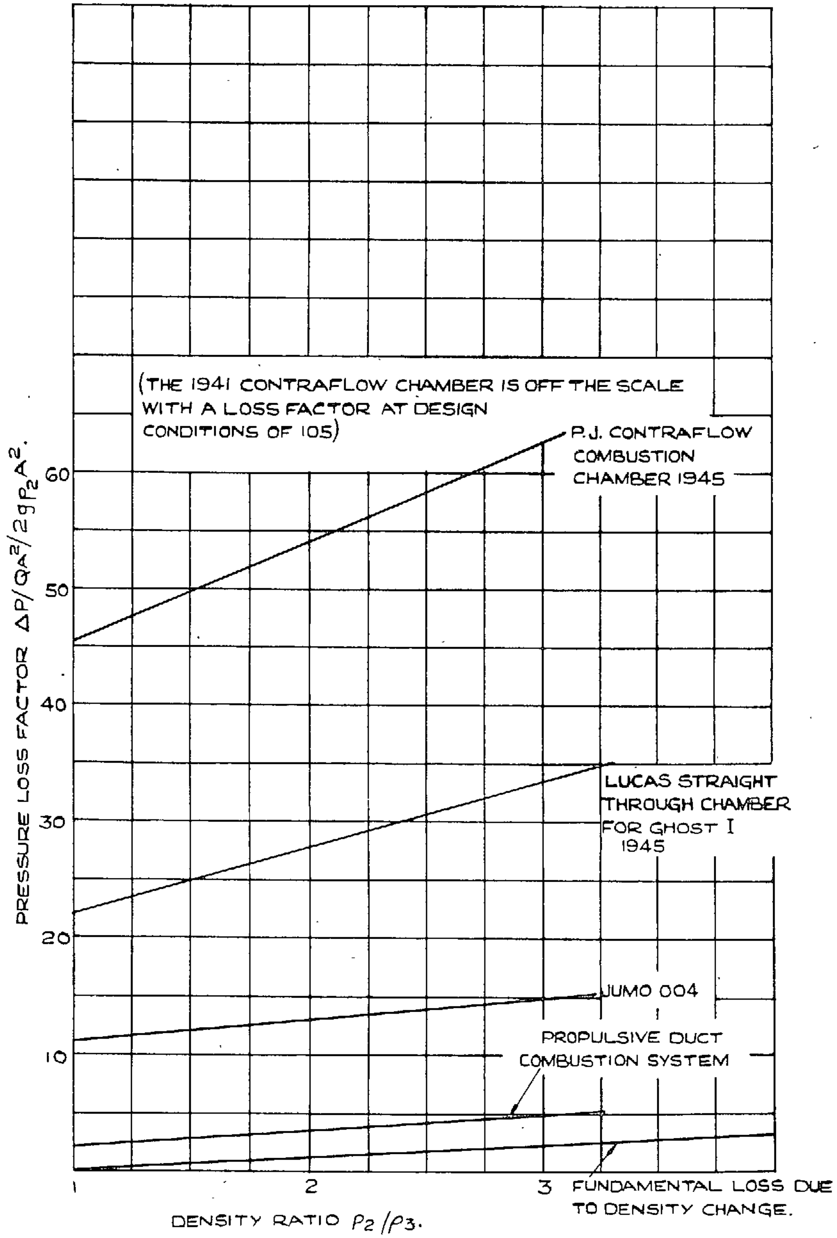


FIG. 34. Pressure losses of representative combustion systems.

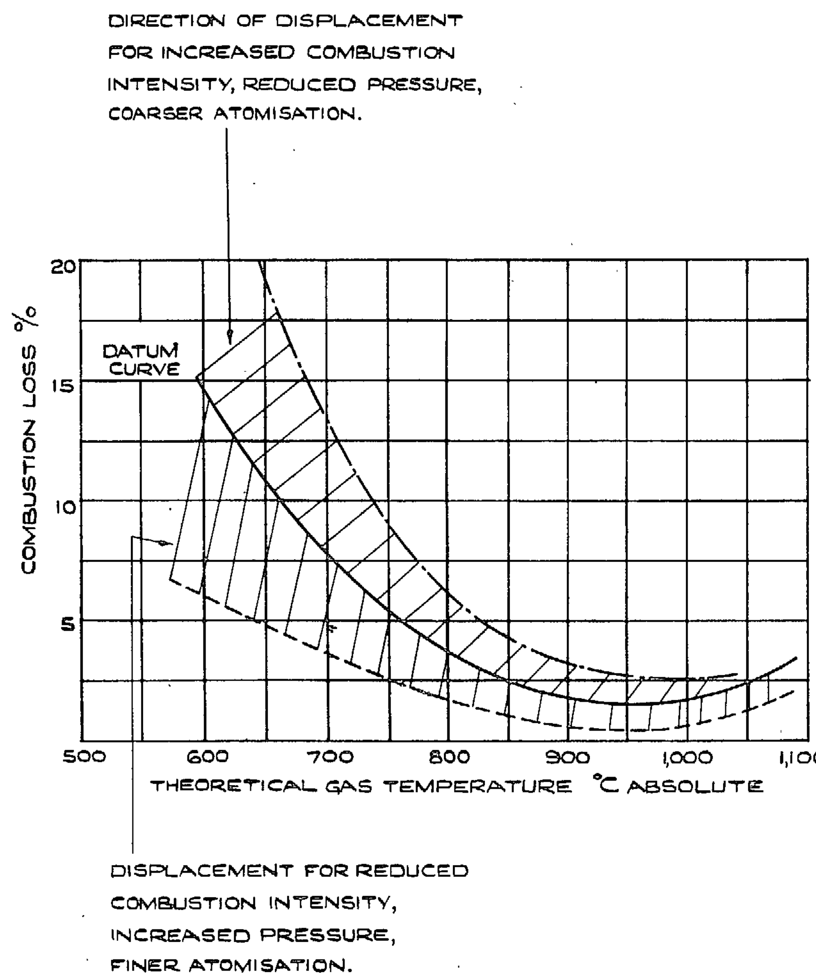


FIG. 35. Variation of combustion loss on a typical chamber.

(94988) Wt. 14/806 K5 12/52 Hw

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