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Summary.—The problems of engine airflow representation in wind-tunnel models are reviewed. Methods which have been used satisfactorily in low subsonic tunnels are described briefly. Special difficulties associated with testing at transonic speeds are noted. Techniques of special application to small supersonic tunnels are described in some detail. It is shown that there are reasons why the representation of jets may be more important at supersonic speeds than at subsonic speeds and a description is given of the Royal Aircraft Establishment Jet Interference Tunnel, which is designed for the study of some of the problems involved.

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1. *Introduction.*—It is conventional to think of an aeroplane as having a definite drag, characteristic of the aircraft, and a thrust, characteristic of the power plant. This is a natural and useful simplification but full of difficulties in detail, as anyone who has tried to prepare precise definitions of thrust and drag (as, for example, in Ref. 1) will readily understand. Ideally, when a wind-tunnel test is done, the drag measurement required is one which can be subtracted from a value of engine thrust obtained from engine tests and measurements of air-intake efficiency to obtain the total longitudinal force on the aircraft. The other aerodynamic characteristics measured such as lift, pitching moment, stalling behaviour, for example, are all liable to be affected by the action of the power plant to an extent which while often small is by no means always so. These facts lead directly to the idea that the propulsive system should, ideally, be represented completely in all wind-tunnel tests of aircraft. Complete representation is in most cases impracticable and we have to review the adequacy of various simplified representations. These range from the use of solid fairings over the air intake and jet exit, through correct representation of the mass flow into the intake (letting the internal flow exhaust freely at the jet exit without representing the jet velocity and temperature) to representation of the jet by hot or cold jets of the correct momentum.

In the past, the emphasis in wind-tunnel testing has moved about among these various forms of representation, being guided by the needs of a particular type of experiment, the requirements of a particular range of tunnel speed, or the stage of development of full-scale aircraft and engines. Section 2 of this paper gives a survey of past experience at the Royal Aircraft Establishment, an assessment of the present position and a suggestion of possible future developments. The later sections describe specifically a number of items of technique, equipment, and experimental and theoretical investigations, which contribute to the general study.

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2. *General Survey.*—The importance of accurate representation of an aircraft's means of propulsion has varied at different stages of aeronautical development. For example, with moderately powered, propeller-driven aircraft the effect of slipstream on the aerodynamic characteristics was generally small, except at low flight speeds, and adequate results could usually be obtained without representation of the slipstream. In the later stages of development of high-performance, propeller-driven aircraft, slipstream representation became important because the effects of slipstream were significant over a much larger proportion of the aircraft's speed range. Fortunately the techniques for representing the slipstream were not too difficult. The propeller stressing problem could be coped with and special electric motors were available.

With the advent of jet engines the position was initially eased, since the jet, which is the equivalent of the slipstream, although more intense, was always arranged so that it did not impinge on other parts of the aircraft. Further the amount of air flowing through the engines was comparatively small. This meant that for many low-speed tests adequate results could be obtained either without representation of the engine airflow, *i.e.*, using solid fairings, or by a simple representation of the flow into the air intake. To help in deciding when such representation was likely to be satisfactory and when the jet would have to be represented, special tests had to be made on the inflow induced by the entrainment of air, by mixing, at the edges of the jet. In some such tests the jet engine was represented by the use of small contra-rotating fans, driven by electric motors buried in the models, while other tests were made with an apparatus for producing a hot jet, mounted outside the tunnel. In the latter case, an investigation of whether it was necessary, at subsonic speeds, to represent correctly the temperature of the jet as well as its momentum, could be made. These techniques and tests are described in Section 3. Part of this section is taken from Squire's paper[2] on the wartime work on jet flow, which contains in addition much information on the fundamentals of the problem.

At high subsonic and transonic speeds, because of the importance of blockage corrections, it becomes almost essential to represent correctly the airflow into the air intakes, but as the jet exit often provides the only practicable place to support the model and strain-gauge balances have to be buried within the model, there are sometimes serious design difficulties. Typical examples of such models, illustrating the difficulties faced, are described in Section 4.

It is an important part of our thesis in this paper that with the advent of supersonic aircraft, and the developments of the jet engine which have made supersonic flight possible, the situation with regard to the representation of engine airflow is becoming in many ways more difficult and that a phase is now being entered in which it will be more than ever important to represent the jet engines in wind-tunnel tests. The reason for this lies in two facts. First, the cross-sectional area of the stream tube in the undisturbed flow corresponding to the air passing through the engines has risen steadily by comparison with the frontal area of the aircraft as speeds have risen. This is illustrated in Fig. 1, where a series of typical aircraft are sketched with design Mach numbers ranging from $0 \cdot 7$ to $2 \cdot 0$, and the intake stream-tube area shown as a proportion of the total frontal area. The values increase from $2 \cdot 5$ per cent for the aircraft designed for $M = 0 \cdot 7$ to 16 per cent for an aircraft designed for $M = 2$. Fig. 2 illustrates the same point graphically. Clearly an airflow corresponding to more than 15 per cent of the frontal area of the aircraft cannot be diverted round, instead of through, the aircraft without running a strong risk of altering the aerodynamic characteristics of the aircraft even if suitable entry fairings could be designed. Moreover, at supersonic speeds the shocks caused by the intake flow may fall on other parts of the aircraft some distance from the intakes and influence the aerodynamic characteristics. The second fact contributing to the importance of full representation of the engine in supersonic testing is that at high forward speeds the exit jet will often be at a much higher pressure than the outside air stream, so that the jet will expand rapidly on leaving the exit nozzle. If there is a supersonic external stream, such an expanding jet will give rise to shock waves which may affect the flow over other parts of the aircraft. Thus there is a potential interference between the jet and the aircraft in cases where the fin, rudder, tailplane, etc., extend further aft than the jet exits, which may be a whole order larger than the jet interferences, due to air entrainment, likely to be obtained at subsonic speeds. To represent fully the effects of the jet on a supersonic aircraft, it is necessary to provide a hot jet with the right temperature and pressure ratios, but the extent to which a cold jet can provide a reasonable approximation still needs investigation.

Unfortunately the means to provide full representation of the engine in supersonic tunnels, although badly needed, are not to hand. The most complete representation would be the use of scaled-down jet engines. Losses of efficiency due to low Reynolds numbers on the blades and due to the relatively larger tip clearances required in small engines limit the scaling down of jet engines but it is technically possible to make small jet engines suitable for models of the order of 5 to 10 ft span. This means that representation by using small engines is not technically impossible for the largest supersonic tunnels but of course whether engines of such a size will ever be available is another question.

It is, in general, impracticable to represent a cold jet of the required velocity or momentum by using internal fans driven by electric motors owing to the small space available for motors and the relatively high powers required even in quite small models. On the other hand, if a high temperature jet is to be represented with lower excess pressures, a ram jet would be possible for models of about half the size mentioned above and for smaller models an electrical heater seems to be not beyond the bounds of possibility, based on rough checks of heat-transfer data. In practice to date we shy away from these ideas and the development difficulties which go with them and our main approach to the problem of engine flow representation has involved separate studies of the representation of the flow into the engine and of the effect of the exit jet. In fact we are forced back on a series of partial tests, special intake tests, the general aerodynamic tests on the complete model with intake flow represented and special jet interference tests.

In the special intake tests, of which more details are given in Section 5.1, the intake and intake flow are represented accurately and measurements of drag and intake pressure recovery can be made but the interference of the intake on the rest of the aircraft cannot be included. In the main aerodynamic tests, owing to the small scale, the intake may have to be represented more crudely but its influence on the rest of the aircraft is correctly represented. It is worth noting here that a simple pipe with a suitably shaped external surface near the lip can represent adequately most types of intake as far as the external flow is concerned, providing the intake to be represented is operating at, or near, full mass flow. With small pipes representing engines it is important to know the conditions for which an attached shock wave and a supersonic internal flow will be obtained. This involves estimating the rate of boundary-layer growth in the tubes and a theoretical method of estimating when supersonic internal flow will be obtained has been developed and checked experimentally. Details are given in Section 5.3.

Turning to the representation of exit flow it can be deduced from Fig. 2 that the total pressure of the jet is required to be some $2\frac{1}{2}$ times the total pressure of the external air stream in the typical supersonic case (though this total pressure ratio will tend to fall at higher supersonic speeds, probably between $M = 2\cdot0$ and $M = 3\cdot0$, as the job of the compressor is taken over by the air intake and the engine tends to approach the ram jet). As stated earlier, it seems impracticable to provide this extra total pressure even for cold tests by an internal fan in the model, owing to the large power requirements of even moderate-sized models and the design difficulties with very small models. An external supply of jet air is therefore required and special tunnel arrangements are needed to introduce this jet air. One possible arrangement, an annular tunnel, has been developed at the R.A.E. and is described in more detail in Section 6.

We now turn to more detail consideration of specific items which contribute in various ways to the general problem of engine flow representation.

3. *Testing at Low Subsonic Speeds.*—3.1. *Use of Internal Motors and Fans.*—For complete model tests in low-speed wind tunnels, when it has been desired to check the interference effects of jets on the flow over tail surfaces, the technique of installing contra-rotating fans, driven by electric motors, inside the model, has been employed on a number of occasions. Fig. 3 shows, for a typical case, how the exit velocity ratio of a cold jet varies with aircraft lift coefficient in

steady flight (*a*) when the jet mass flow only is correctly represented and (*b*) when the jet thrust, or momentum flux, is correctly represented. Ignoring the internal losses, which make only a small modification, a velocity ratio of $1 \cdot 0$ is obtained from free flow through the ducts, and this is sufficient to represent the mass flow for a lift coefficient (C_L) of $0 \cdot 5$ at sea level and over $1 \cdot 0$ at 30,000 ft. For lower values of C_L (*i.e.*, higher flight-speed conditions), it is merely necessary to throttle the duct at some convenient station, usually the exit. If the jet momentum is to be represented, however, using a correctly scaled size of exit, a forced jet is required down to a C_L of $0 \cdot 05$, in effect for all level flight conditions. The lower curve of Fig. 3 shows approximately how the power output* of this jet varies with the velocity ratio and hence with C_L. It is seen that the power which has to be installed in the model becomes considerable. For example, if the area of the jet is 1/500 of the area of the tunnel, the power output of the jet at a velocity ratio of $4 \cdot 0$ is over 10 per cent of the power output of the tunnel.

An example of the use of internal fans is illustrated in Fig. 4 and relates to tests on a 1/5 scale model of the *Vampire*. In this case the fans were driven by a single 25 horse-power motor housed in the nose of the fuselage. A more frequent arrangement has been to place two smaller motors end to end with the pair of fans situated between them, one fan being driven by each motor.

The usual procedure, which was followed in the case quoted, is to restrict the tunnel speed to 60 or 80 ft/sec for tests at high jet velocity ratio. In the $11\frac{1}{2}$ ft Wind Tunnel, this gives a Reynolds number of about $0 \cdot 5 \times 10^6$, based on wing mean chord. Tests with free duct flow are made at this speed and also at one or two higher speeds, to check the scale effect.

A sample of results from the *Vampire* model tests is shown in Fig. 5. Curves of pitching moment at two elevator settings are given for the cases of high jet velocity ratio, velocity ratio $1 \cdot 0$ (approximately the free-flow case) and for the model with duct entries faired over. It is seen that restricting the tests to the case of solid fairings would in this instance have introduced misleading features into the pitching-moment results.

The thrust and mass-flow calibrations of internal fan systems have normally been obtained by means of pitot and static-pressure surveys at the duct exit. Unless special care is taken in traversing the nozzle boundary layer, the accuracy of this method is not high, but an error of 5 to 10 per cent in thrust may be acceptable for the purpose of investigating the jet effect on stability.

Some further discussion of the use of internal fans in complete model testing at low speeds is given by Anscombe in Ref. 3.

3.2. *A Hot Jet Apparatus.*—An apparatus for producing a heated jet, designed for and originally (1943) installed in the R.A.E. 5 ft Open Jet Tunnel, is pictured in Fig. 6. Air from the compressed-air mains is heated by passing through four heaters, each consisting of four electrically heated coils, pairs of which are connected in series on to the 500-volt mains. Three heaters operate at a constant power input and the fourth can be operated at three different inputs.

The coils are of 17g Nichrome V wire, 30-in. long and $0 \cdot 5$-in. diameter, enclosed in a 3-in. diameter pipe and held in position by a cementing compound. In order that the coils should be held in the cement and yet have air flow through them, they were first fixed on to $0 \cdot 5$-in. diameter formers and placed in position in the pipe. The annular space between the pipe and the formers was then filled with cement. The formers were then withdrawn through the coils, leaving half the diameter of the wire embedded in the cement. The pipes were finally heat treated in an oven to dry out the cement and thereby insulate the coils from each other.

The heaters are connected in parallel as shown in Fig. 6. The air mass flow is measured at an orifice plate installed in the cold compressed-air delivery line. Heaters 2, 3 and 4 have each one main switch in circuit and take $34 \cdot 2$ kW each at $64 \cdot 7$ amps. Heater 1 has a main switch and two subsidiary switches, which allow it to operate at full, half or quarter power.

* Defined as the volume flow × rise in total pressure.

The apparatus provides jet temperatures up to about 650 deg abs. at airflows of the order of $\frac{1}{2}$ lb/sec. A $1\frac{1}{4}$-in. diameter jet can be run choked at all temperatures. Similarly a 1-in. jet can be run with one-quarter atmosphere excess pressure.

The equipment has recently been transferred to the No. 1, $11\frac{1}{2}$-ft Tunnel, where it is currently in use.

3.3. *Tests with Heated Jets.*—The hot-jet apparatus described above has been used in several types of experiment. One important line of investigation is to determine equivalence relationships between hot and cold jets, so that in classes of experiment where it becomes difficult or perhaps impossible to incorporate a heated jet, the best representation by means of a cold jet may be obtained. In this connection, some work has been done on temperature effects on downwash at an aerofoil placed some distance downstream of an unchoked jet and, in another case, on the drag of a similarly placed aerofoil.

The fact that, as a hot jet develops downstream, the momentum remains constant, while the mass flow, velocity and temperature all vary, suggests that a cold jet should have the some momentum as the hot jet it represents, *i.e.*, the quantity $A\rho V^2$* at the exit should be the same in the two cases. Within the limits of the experiments made, this supposition is confirmed. Fig. 7 shows that (*a*) the downwash angle and (*b*) the drag of an aerofoil in the wake are both independent of temperature when the momentum is maintained constant.

In the case of a choked jet operating at high excess pressure ratio, it is probable that temperature and momentum effects are no longer so closely related, particularly in the vicinity of the exit nozzle itself. Further experimental work is undoubtedly required on the equivalence of hot and cold jets. At the R.A.E. a special form of supersonic tunnel has been built to carry on this work (Section 6).

4. *Testing at Transonic Speeds.*—4.1. *General.*—In the testing of complete models in high subsonic and low supersonic tunnels, considerable difficulty is often experienced in providing the simplest form of ducted model, the model with free internal flow, regulated by the size of the exit. This comes about primarily because of the great significance of support interference and tunnel-blockage effects at such Mach numbers.

When the design of air intakes is a feature of the programme, it is usually required to provide three types of information from the tests, namely:

(*a*) Internal mass flow and pressure recovery

(*b*) Model drag

(*c*) Surface pressure distributions in the neighbourhood of the intakes.

The combining of facilities for making the three types of measurement in a single model is generally a matter of considerable difficulty. The following two examples are illustrative of some of the difficulties involved.

4.2. *Ducted Model for R.A.E. High-Speed Tunnel.*—Fig. 8 shows the general arrangement of a ducted model which formed one of a series tested in the R.A.E. High-Speed Tunnel up to 0·93 Mach number. The model is representative of a single-engined aircraft having twin side intakes with boundary-layer bleeds. Photographs of the model are shown in Figs. 9 to 11.

The first problem is that of choosing a suitable method of support. A conventional rear sting passing through the duct exit falsifies the rear-end configuration and considerably complicates the internal design. A half-model mounted on the floor of the tunnel is unsuitable, unless special measures are taken to divert the tunnel boundary layer from the intake and even then is hazardous, since the detail pressure distribution in the wing–body junction is an important feature of high-speed design and might be affected by the nearness of the tunnel boundary. In the present case, the method adopted was to utilise the cabin space to house a drag balance, which was mounted on the front end of the sting. The latter then passed down through the

* ρ = Density of jet; V = Velocity of jet; A = Jet exit area.

bottom of the fuselage and by this means was kept completely clear of the ducts. The sting (Fig. 11) undoubtedly set up some interference beneath the fuselage and this was the principal objection to the method. The procedure appeared to be justified by results at subsonic speeds, but for transonic tests it would be unsuitable. A half-model of a W-wing (Fig. 12) might provide the best arrangement at transonic speeds, although strength considerations would then tend to restrict the tests to low incidences.

Certain details of the model construction are of interest in indicating the difficulties. The model was built up from port and starboard sub-assemblies. Each wing and half fuselage was manufactured integrally in light alloy. The outboard portion of the inside surface of the duct was machine-shaped from this wing–fuselage block, the duct sections being left open to provide access for the milling cutter. After machining, the duct was closed by fitting a thin blanking-off plate (Fig. 11) to the open semi-circular section. An airtight seal was obtained by treating the mating surfaces with a film of Araldite just before the plate was screwed home. To each half of the model was next added the appropriate nose portion. This (Figs. 9 and 10) was made of laminated teak to simplify the shaping of the boundary-layer bleed. The port and starboard assemblies were then bolted on to the drag balance (Fig. 11). This brought the rear ends of the two blanking-off plates together, thus sealing off the balance compartment from the ducts. Assembly was completed by attaching a wooden afterbody and an appropriate exit nozzle.

A further complicated process was the installation of pressure-plotting tubes. The method of installation can be seen in Fig. 10. The required number of copper tubes (3/32 in. O.D.) were led from the balance compartment through a slot in the body to the outer surface of the intake fairing. Each tube was fixed in place by means of a jig, the assembly being then encased in a transparent matrix of Araldite. After this latter had been worked down to a true surface, a hole 0·020-in. diameter was drilled into each tube. Close to the leading edge of the lip, very fine bore tubing has to be used. Drilling into this being impracticable, these particular tubes were instead plugged with close-fitting lengths of Nylon gut whilst the Araldite was being applied; the plugs were then withdrawn as soon as the Araldite had hardened.

Within the balance compartment the copper tubes were connected to lengths of flexible tubing, which were led out through the bottom of the fuselage and secured along the sting. The bulk of fifty-inch lines was considerable and constituted a further argument in favour of keeping the sting outside the duct. The connection between model and sting formed by these pipelines ruled out any attempt to measure forces and pressures simultaneously so the entire range of tests had to be covered twice, allowing each type of measurement to be made independently.

4.3. *Ducted Model for R.A.E. Bedford 3-ft Supersonic Tunnel.*—Fig. 13 shows the general arrangement of the centre portion of a somewhat similar ducted model built for tests in the 3-ft Supersonic Tunnel at the R.A.E. Bedford. This was designed for a Mach number of 1·4 but can also be tested transonically up to 1·03.

In this case a central sting mounting was chosen. Principal features of the model construction are as follows. The intake section is represented correctly as far as the engine-face position. This section is made from a paper-based laminated plastic and is detachable from the main fuselage and wing. A three-component strain-gauge balance is carried in approximately the engine position. The balance is integral with the main fuselage section and is shielded from the internal airflow by fairing plates, which also carry the connections to a pitot-static comb at the engine-face position. A fork joint attaches the sting to the drag-balance earth frame and the lift and pitching-moment stations on the sting are faired over with plastic resin.

The drawing gives a fair idea of the degree of complication and congestion existing within the model. It will be noted that in this case, surface-pressure plotting lines were not required.

5. *Testing at Supersonic Speeds.*—5.1. *Drag Tests of Intake Research Models.*—In research testing on the aerodynamics of the air intake itself, there is usually no problem in providing adequate representation of internal flow. It is necessary only to represent the correct mass-flow

ratio and this can be done for most conditions* by allowing free flow through the model, regulated by a suitable throttling device at the exit. This state of affairs is broadly true whether tests are being made at subsonic, transonic or supersonic speeds.

Most of the supersonic intake research of the R.A.E. is made in small tunnels of the order of 6 in. square. The method of determining the intake drag in this work is of some interest.

To obtain the external drag, the total drag force on the model is measured by a strain-gauge balance attached to the supporting sting, and from this the internal drag, base drag, and the pressure force on the end of the sting are subtracted. A model is shown mounted on the balance sting in Fig. 14. Six pitot rakes spaced equally round the circumference of the base are placed so that there is a small clearance (0·02 in. to 0·05 in.) between the ends of the tubes and the exit plane. As can be seen, three of the rakes measure the total pressure of the internal flow at exit and the other three measure effectively the static pressure on the base. This method has been checked against one using static-pressure holes in the base and found to be accurate so long as the tubes measuring base pressure are separated circumferentially from those measuring flow pressure.

The sting pressure force is measured by means of a pressure tapping from the balance housing. The balance is calibrated before and after each run, the temperature being kept approximately constant throughout the running and calibrating periods.

The above method has on the whole proved satisfactory for the measurement of the drag of intake models. It is probably less good than a method in which the balance is housed inside the intake centre body, since then it is easy to arrange for drag and internal pressure recovery measurements to be made at the same time. On the present rig the balance measurements are restricted to a relatively small number of discrete points, one for each of a series of different exit-plug throttles. This is often not adequate for a full pressure-recovery investigation; the latter is therefore normally made on a modified rig incorporating a traversing cone at the rear which gives a continuous range of mass-flow variation. However, it should be remembered that the rig was developed originally for work on pitot intakes (having no centre body) and even with centre-body intakes the smallness of the scale would lead to difficulties in the design and installation of an internal balance.

5.2. *Jets Supplied by Compressed or Atmospheric Air*.—In many small-scale tunnel experiments, a high-pressure jet can conveniently be supplied from an external compressed-air main. In a tunnel working at atmospheric stagnation pressure, a useful jet pressure ratio at supersonic tunnel speeds may sometimes be obtained merely by letting in air from the laboratory to form the jet. Furthermore, the pressure ratio may be increased by reducing the tunnel stagnation pressure, as for example by partially closing an upstream valve.

In the case shown in Fig. 15 it was required to produce a jet at varying pressure ratios (total pressure in jet ÷ static pressure in free stream) up to values corresponding to choking of the jet. The choking pressure ratio was a variable function of free-stream Mach number, as shown, owing to the particular form of jet and location of the exit.

The jet was produced by letting in air at approximately laboratory total pressure.† At a tunnel Mach number of 2·41, with atmospheric stagnation pressure, the available jet pressure ratio was more than sufficient. At $M = 1 \cdot 35$, under similar conditions, it was just possible to choke the jet. Below this Mach number, a jet produced in this way would have been inadequate, as shown. But by manipulating upstream valves in the tunnel line so as to lower the tunnel stagnation pressure, a satisfactory jet was again achieved, at $M = 0 \cdot 9$. Finally, by closing off the upstream end completely, the tests could be repeated at $M = 0$. This method gave a very useful Mach number coverage for the particular investigation.

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* An exception is the condition of very low flight speed, or in the extreme, the static or ground-running condition, when a forced internal flow becomes necessary. This case, however, hardly comes in the category of a wind-tunnel test, since in its extreme form, the tunnel speed is zero.

† Actually the jet pressure was augmented by use of a small auxiliary compressor, but it can be seen from the diagram that the net augmentation, after allowing for piping losses, was small.

Consideration should be given to the reduction in Reynolds number caused by reducing the tunnel pressure.

5.3. *Compressible Viscous Flow in Straight Pipes.*—In the design of complete models for testing in small supersonic tunnels, it may be difficult or impracticable to incorporate detail representation of the engine ducts. Nevertheless, it may be important to provide a ducted model with at least a correct representation of mass flow. In such cases the use of simple open pipes to simulate nacelles may be considered. The problem then arises as to what is the effect of internal boundary layer on the flow through such pipes. For if the effective contraction in area produced by the boundary layer exceeds a critical value, it will not be possible to establish supersonic internal flow with an attached shock at the entry. Thus the external flow pattern may be non-representative and the resulting interference effects on neighbouring surfaces different from the real case.

A combined theoretical and experimental study of this problem has been made[4]. In the theoretical treatment, it is assumed that the radius of a pipe at any section is effectively reduced by an amount equal to the displacement thickness of the boundary layer. Laminar and turbulent layers are both considered. In each case, using the best information available for the variation of displacement thickness with Mach number, an expression is derived relating the minimum Mach number M_1, at which the internal flow passes from subsonic to supersonic, to a parameter combining the Reynolds number and length/radius ratio of the pipe. The final relationships are:

(*a*) For a laminar layer:

$$R_1^{-1/2}\frac{l}{r_3} = \frac{0\cdot 575 f_1(M_1)}{(1 + 0\cdot 227\bar{M}^2}\left[1 - \sqrt{\left(\frac{A_2}{A_3}\right)}\left\{\frac{(\gamma-1)}{(\gamma+1)} + \frac{2}{(\gamma+1)M_1^2}\right\}^{1/4} \times \right.$$
$$\left. \times \left\{\frac{2\gamma}{(\gamma+1)} - \frac{(\gamma-1)}{(\gamma+1)M_1^2}\right\}^{1/\{2(\gamma-1)\}}\right] \qquad \ldots \quad (1)$$

(*b*) For a turbulent layer:

$$R_1^{-1/5}\frac{l}{r_3} = \frac{21\cdot 1 f_2(M_1)}{(1 + 0\cdot 219\bar{M}^2)}\left[1 - \sqrt{\left(\frac{A_2}{A_3}\right)}\left\{\frac{(\gamma-1)}{(\gamma+1)} + \frac{2}{(\gamma+1)M_1^2}\right\}^{1/4} \times \right.$$
$$\left. \times \left\{\frac{2\gamma}{(\gamma+1)} - \frac{(\gamma-1)}{(\gamma+1)M_1^2}\right\}^{1/\{2(\gamma-1)\}}\right]. \qquad \ldots \quad (2)$$

In these:

l = length of pipe

r_3 = exit radius (internal) of pipe

A_2 = entry area of pipe

A_3 = exit area of pipe

R_1 = Reynolds number based on l and M_1

$\bar{M}$ = mean internal Mach number.

f_1 and f_2 are functions of Mach number, defining essentially the ratio of mean Reynolds number of the internal flow to R_1.

The relationships (1) and (2) are plotted in Figs. 17 and 18 respectively.

The experiment consisted in mounting a number of straight pipes of various sizes in turn in a supersonic tunnel and noting in each case whether the internal flow was subsonic or supersonic. This was determined (*a*) by observing the nature of the front shock (*i.e.*, whether attached or detached) and (*b*) by measuring the internal static pressure at a suitable point well inside the tube. Typical plots of this pressure and the internal Mach number deduced from it are shown in Fig. 16. On each curve, the jump as between subsonic and supersonic internal conditions is noted. In

Figs. 17 and 18, for smooth and rough tubes respectively, the various test points are plotted, indicating the appropriate observed condition of internal flow. It is seen that the results in general give good support to the theory.

The experiments have been extended to include the case of pipes inclined at various angles to the stream. From the results an empirical generalisation has been deduced to cover this case.

Referring back to the initial reason for making this study, the results are reassuring in that at Mach numbers greater than about $1 \cdot 4$ it is possible to establish supersonic flow through quite small tubes. Thus at $M = 1 \cdot 5$ the critical diameter for a smooth pipe, of length/radius = 20, in a tunnel at atmospheric stagnation pressure, is approximately $\frac{1}{4}$ in. At $M = 2 \cdot 5$ the corresponding critical diameter is as low as 1/20 in.

6. *The R.A.E. Jet Interference Tunnel.*—As mentioned in the introduction, a need was foreseen some time ago for a means of carrying out researches, in some detail, on the influence of jets on the flow around the rear of a fuselage or engine macelle and on the shocks and disturbances in the main stream which arise from the jet when the static pressure in the jet, at exit, differs considerably from the static pressure in the external flow. It was judged that this need would grow and probably continue for some time. To represent the excess pressure in the jet of a typical gas-turbine engine, suitable for high-speed flight, a jet total pressure up to between two and three times the total pressure of the outside air stream is required as is shown by Fig. 2. This value is typical for flight speeds up to about $M = 2 \cdot 0$ but at higher speeds the total pressure ratio may be expected to fall off fairly sharply and a lower jet pressure may be adequate. To get a correct representation of the hot jet, the ratio of the total temperature between the jet and the outer stream should bemaintained at the full-scale value. Thus, with a normal tunnel using air at atmospheric total temperature, the jet temperature requirements are only moderate at high tunnel Mach numbers. For example, a jet engine with a jet-pipe total temperature of 1,000 deg K flying at $M = 2 \cdot 0$ in the stratosphere, can be represented by a jet of total temperature 740 deg K in the wind tunnel, while a reheated jet of say 1,800 deg K would need a jet of only 1,330 deg K to represent it. At a Mach number of $3 \cdot 0$ the jet temperature required would drop to 475 deg K and 850 deg K respectively.

It was decided that the type of research to be done called for a model of not less than 2-in. diameter and preferably about 4-in. diameter and that if at all possible a hot jet should be represented. With these needs in mind the available plant was reviewed and it was found that the needs could be substantially met by using existing suction and compression plants in use in the R.A.E. at Farnborough. The suction plant and associated driers could supply a mass flow of 16 lb/sec of dry air at near atmospheric pressure (14 lb/sq in. abs.) exhausting at a pressure of 5 lb/sq. in. absolute. This was a continuous-flow, open-circuit system exhausting to atmosphere, capable of handling exhaust gases and unburnt fuel and capable of providing a supersonic airflow of about 100 sq in. at Mach numbers in the region of $2 \cdot 0$ The compressor plant, again open circuit, could provide a supply of air at about $2\frac{1}{2}$ atmospheres absolute (35 lb/sq in.) and about 100 deg C and ample in quantity to provide a jet for the sizes of model mentioned above. With this size of model and air pressure and temperature, it was thought that the development of a kerosene burner, to allow the total temperature to be controlled to any value between 800 and 1,800 deg K, would not present any great problems. The outstanding problem was how to get the jet air into the tunnel without disturbing the main flow unduly. The solution adopted, which is believed to have been new, was to introduce the jet air through the throat of the supersonic tunnel, thus forming an annular tunnel with the body containing the jet air in the centre, surrounded by a shaped nozzle as shown in Fig. 19. The design of such a nozzle sets no fundamentally new problems but calls for some modification of the normal methods for axisymmetric and two-dimensional nozzles. It has the advantage that the loss in flow uniformity associated with the focussing of small disturbances on the axis, which can be troublesome in dealing with axisymmetric nozzles, is avoided. The fact that such annular nozzles are the general case from which the axisymmetric and two-dimensional designs are obtained as limits has led W. T. Lord to develop a general analytical method of nozzle design based on the annular nozzle.

The general lay-out of the tunnel is shown in Fig. 19. The main structure consists of a mild steel drum divided by partitions into a jet-air settling chamber, a tunnel-air settling chamber, a working-section containing the nozzle and provided with windows and a diffuser section fitted with explosion relief valves. As will be seen from Fig. 19, a uniform open jet, 11-in. (one diameter) long, has been provided between the nozzle exit and the beginning of the collector and tunnel diffuser. Uniform flow is achieved a short distance ahead of the nozzle exit, so that there is a region of uniform flow near the tunnel axis about 20 in. long, apart from disturbances caused by the afterbody and jet under test.

One of the most important variables in the type of research under discussion is the state of the boundary layer and its thickness relative to the body or nacelle under test. Fortunately the type of boundary layer to be represented will normally be a fully turbulent layer, with a surface temperature near to the thermal equilibrium value; and with the model scale used this is likely to occur naturally on the model, using a water-cooled tail pipe when a hot jet is being used. For checking roughly whether the order of boundary-layer thickness is likely to be reasonable, it is convenient and a fair approximation to consider the boundary layer on the central body as having an effective starting point somewhere in the region of the throat of the tunnel, and forgetting the somewhat complicated history of the boundary layer. With such an assumption the effective length/diameter ratio of the body shown in Fig. 19 is 5 or 6 : 1 and by use of different afterbodies, values in the range between 4 : 1 and 8 : 1 might be obtained. These compare with typical full-scale values in the region of say 5 : 1 and 12 : 1 for nacelle and fuselage installations respectively. For true representation of the ratio of boundary-layer thickness to body diameter, the ratio of the effective length/diameter of the model to the length/diameter full-scale must equal the fifth root of the ratio of model and full-scale Reynolds numbers, based on length. This means that a shorter model can to some extent compensate for a lower Reynolds number. The relation taken in conjunction with the typical figures given above means that the correct relative boundary-layer thickness can be obtained in many cases, though it may be difficult to get the boundary layer thin enough to represent short nacelles accurately. As examples, the boundary-layer thickness could be obtained corresponding to a 15-ft long, 3-ft diameter nacelle at heights of about 50,000 ft and upwards, or a 5-ft diameter, 60-ft long body at heights of 25,000 ft and upwards. It has been realised that in common with all long bodies any misalignments or small flow assymetry round the central jet pipe and afterbody could lead to exaggerated assymetries in the boundary layer, but it should be possible, with care, to achieve a high standard of flow symmetry and a tolerably uniform boundary-layer distribution.

As set up at present the tunnel is suitable for a Mach-number range from about $1 \cdot 7$ or $1 \cdot 8$ up to $M = 2 \cdot 5$, but this range is not limited fundamentally and the rig could be modified to cover a wider range, though at the expense of a smaller test body at lower Mach numbers. The ratio of total pressure in the jet to static pressure in the external stream can be varied up to $17 \cdot 5 : 1$ at $M = 2$ and referring to Fig. 2 this will be seen to give an ample margin for representing typical jet engines. The temperatures available from the kerosene burner are ample for representing jet engines with or without afterburning and, in fact, should allow some part of the problem of rocket jets to be investigated. As planned, pressure plotting of the afterbody, base, and jet, schlieren observation of the flow and overall measurements of gross thrust, less afterbody and base drag, can be made. Pressure and direction surveys in the flow, and measurements on fins, tailplanes or any other parts of the aircraft close to the jet can also be carried out when necessary. The scale of the rig is such that representation of any cooling airflow between the tail pipe and the fuselage is possible, a not unimportant factor in some installations.

One important limitation is that tests on bodies and jets at an incidence to the air stream cannot easily be made, though by use of a cranked centre body, some such tests can be covered. In this respect the use of a half-model and semi-circular jet mounted on a reflection plate may be more convenient but in most other respects the annular tunnel arrangement is expected to be preferable.

Although the preliminary planning and design of this facility was carried out some time ago, there have been some disappointments and delays in bringing it into use. However, the present situation is that preliminary tests have been made. The initial nozzle has been calibrated and found to give a less uniform flow than had been hoped for, and a revised design has been prepared. The diffuser throat design has been investigated and suitable shapes and diameters chosen. The kerosene burner and water-cooled jet pipe have been developed and the thrust-measuring rig has been checked under running conditions. Thus the bulk of the preliminary proving tests have been completed.

7. *Concluding Summary.*—In this note the problems of engine representation in wind-tunnel tests have been rather cursorily surveyed. For low-speed tests, representation of the engine by simple ducts, throttled at exit to give the correct flow, or even by solid fairings over the entry and exit, has generally proved sufficient but techniques of representing the engine by internal fans in the model, electrically driven, have been developed for use in special cases. Research tests have suggested that the cold jet produced in this way can be arranged to be aerodynamically equivalent to a hot jet for the normal purposes of low-speed tunnel tests.

At high subsonic and transonic speeds, representation of the flow into the air intake becomes almost essential and can generally be arranged, though with some complications in the design and construction of the model. No suitable means is generally available for representing the exit jet in such tests but the lack of such a technique has not proved too embarrassing.

At supersonic speeds there are reasons for suggesting that full representation of the engine, including both intake air flow and hot exit jet, may be more important, and no satisfactory means have been developed, nor can any technique suitable for routine tests be foreseen. In this rather unsatisfactory situation the best that can be done is to make separate investigations into the effects of the entry flow and of the jet. No special problems of supersonic testing stand out as far as representation of the intake flow are concerned but it has been shown that in many tests with small models, where the intake pressure recovery is not being investigated, a simple tube can be used to represent the engine. A method has been developed, and checked experimentally, for estimating the ratio of diameter to length of such a tube at which the supersonic flow into the entry will break down.

As far as representation of the jet is concerned, no method is known which does not have some drawbacks but an annular tunnel has been developed which is expected to allow much useful research to be made into the effect of the jet on the flow in the region of the jet exit and also research into the effect of a jet, at supersonic flight speeds, on any nearby aerodynamic surfaces.

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REFERENCES

No.	*Author*	*Title, etc.*
1	A.R.C. Definitions Panel ..	Report on the definitions of the thrust of a jet engine and of the internal drag of a ducted body. C.P. 190. May, 1954.
2	H. B. Squire	Jet flow and its effects on aircraft. A.R.C. 10,189. R.A.E. Report Aero. 2171. September, 1946.
3	F. B. Bradfield (editor) ..	Notes on the technique employed at the R.A.E. in low-speed wind-tunnel tests in the period 1939 to 1945. R. & M. 2556. October, 1947.
4	I. Seddon	The flow through short straight pipes in a compressible viscous stream. R.A.E. Report Aero. 2542. A.R.C. 17,870. April, 1955.

APPENDIX I

Transonic Flow through an Annular Throat

The velocity potential ϕ, which satisfies the equation of compressible flow

$$\left\{\left[q^{*2}-\phi_x^{\,2}-\mu^2\phi_y^{\,2}\right]\phi_{xx}-2(1-\mu^2)\phi_x\phi_y\phi_{xy}+\left[q^{*2}-\mu^2\phi_x^{\,2}-\phi_y^{\,2}\right]\phi_{yy}\right.$$

$$\left.+\left[q^{*2}-\mu^2\phi_x^{\,2}-\mu^2\phi_y^{\,2}\right]\frac{\phi}{(y+a)}\right\}=0\,,\, y$$

where q^* is the critical speed, $\mu^2=(\gamma-1)/(\gamma+1)$, and a is the radius of the internal cylinder, and fulfils the boundary conditions

$$\left.\begin{aligned}\phi_x &= q^*\left[1+\frac{x}{l}+0\left(\left(\frac{x}{l}\right)^3\right)\right]\\ \phi &= 0\end{aligned}\right\}\text{ on } y=0\,,$$

is given by the relations:

$$\frac{x}{l}=\varepsilon^2\frac{2}{(1-\mu^2)}X\,, \qquad X=0(1)\,,$$

$$\frac{y}{l}=\varepsilon Y\,, \qquad Y=0(1)\,,$$

$$\frac{a}{l}=\varepsilon A\,, \qquad 0\leqslant A\leqslant\infty\,,$$

$$\frac{\phi}{q^*l}=\varepsilon^2\frac{2}{(1-\mu^2)}\phi_2+\varepsilon^4\frac{4}{(1-\mu^2)^2}\phi_4+\varepsilon^6\frac{16}{(1-\mu^2)^4}\phi_6+0(\varepsilon^8)\,, \qquad \phi_2,\phi_4,\phi_6=0(1)\,,$$

$$\phi_2=X\,,$$

$$\phi_4=\tfrac{1}{2}X^2+XQ_4+R_4\,,$$

$$\phi_6=\tfrac{1}{2}X^2P_6+XQ_6+R_6\,,$$

$$Q_4=A^2\tfrac{1}{4}\{[Z-1]-\log Z\}\,,$$

$$R_4=A^4\,\tfrac{1}{64}\{[Z^2+4Z-5]-2[2Z+1]\log Z\}\,,$$

$$P_6=A^2\frac{(1+3\mu^2)}{8}\{[Z-1]-\log Z\}\,,$$

$$\begin{aligned}Q_6=A^4\,\tfrac{1}{128}\{&[(5+7\mu^2)Z^2-4(1-13\mu^2)Z-(1+59\mu^2)]\\ &+6[-2(1+3\mu^2)Z+(1-5\mu^2)]\log Z+4(1-\mu^2)\log^2 Z\}\,\text{o}\end{aligned}$$

$$\begin{aligned}R_6=A^6\,\tfrac{1}{1536}\{&[(5+7\mu^2)Z^3+9(1+7\mu^2)Z^2+9(3+17\mu^2)Z-(41+223\mu^2)]\\ &+6[-4(1+2\mu^2)Z^2-(7+29\mu^2)Z+(1-13\mu^2)]\log Z\\ &+6[(5+7\mu^2)Z+(1-\mu^2)]\log^2 Z\}\,,\end{aligned}$$

$$Z=\left(1+\frac{Y}{A}\right)^2.$$

It will be noted that in this solution the term of zero order is identically zero, and the term of first order merely corresponds to a uniform sonic stream.

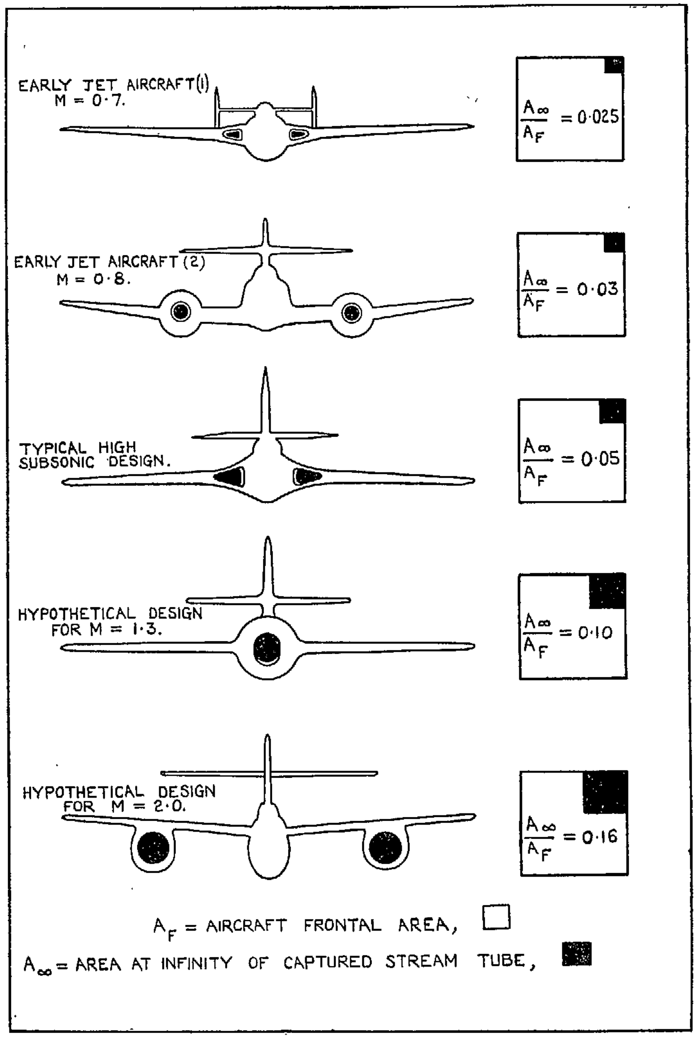
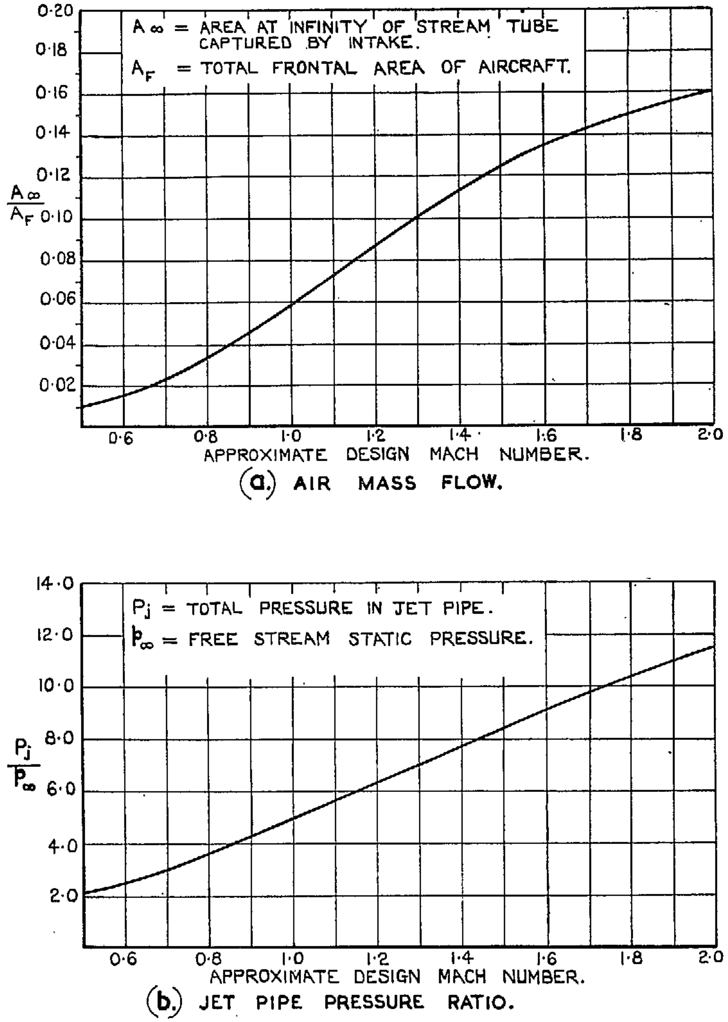


FIG. 1. Increase of air-swallowing capacity of jet aircraft with design speed.



FIGS. 2a and 2b. Increase of airflow requirements of typical jet aircraft with Mach number.

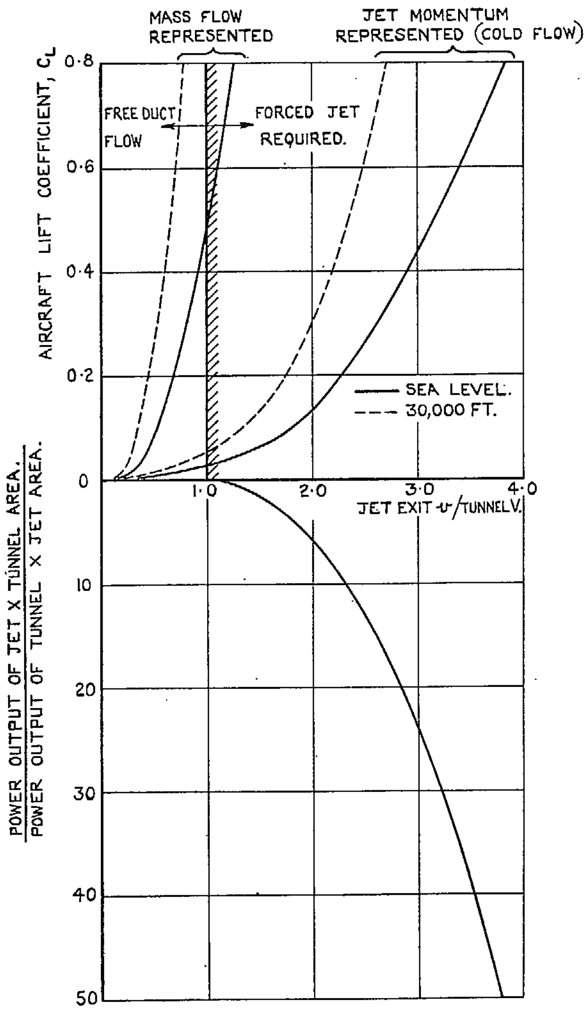


FIG. 3. Example of relationships between jet velocity ratio, jet power output and aircraft C_L for low-speed tunnel tests of a typical twin-engined fighter.

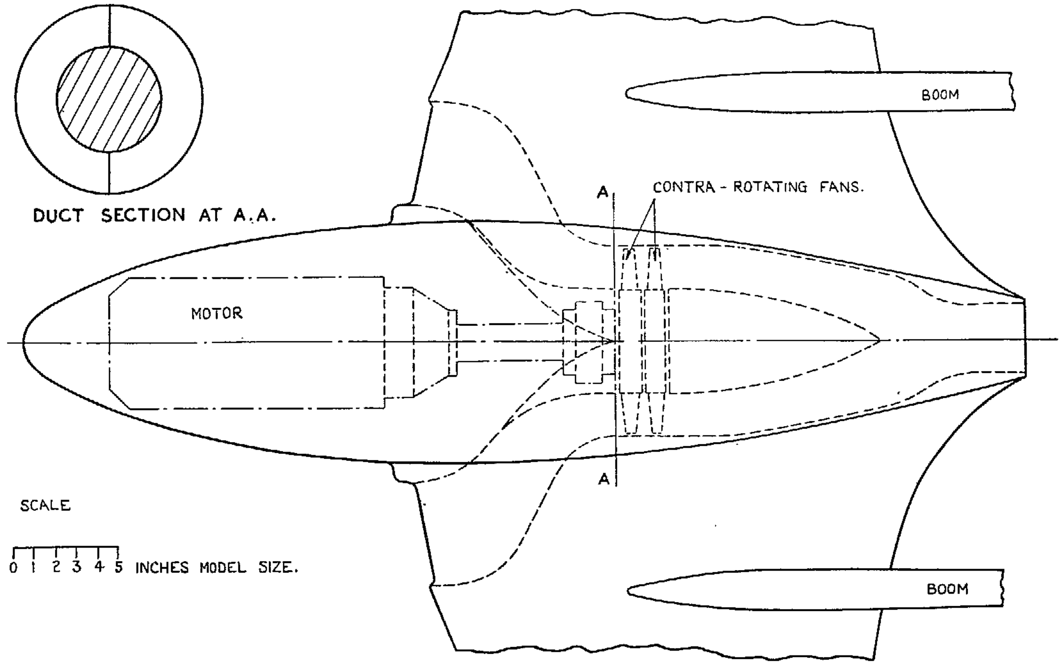


FIG. 4. *Vampire* 1/5-scale model—Fan installation.

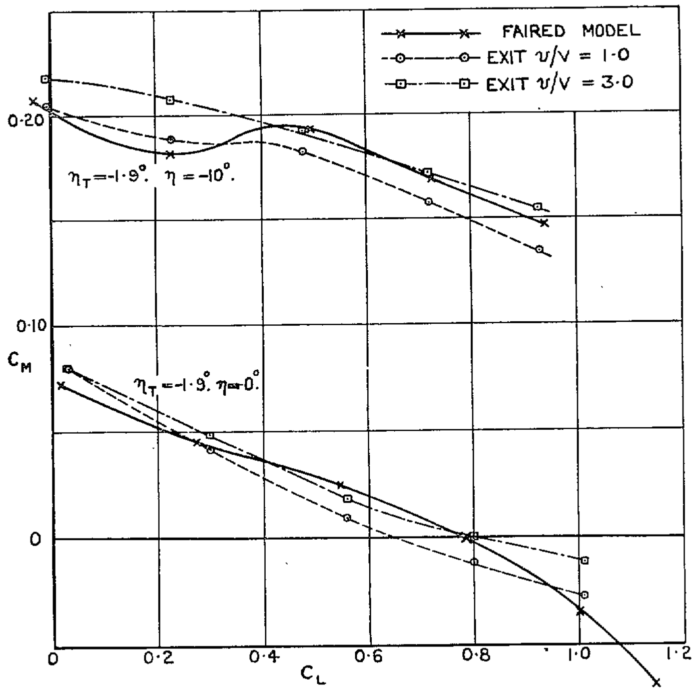


FIG. 5. Tests on 1/5-scale model *Vampire*. Effect of jet flow on longitudinal stability.

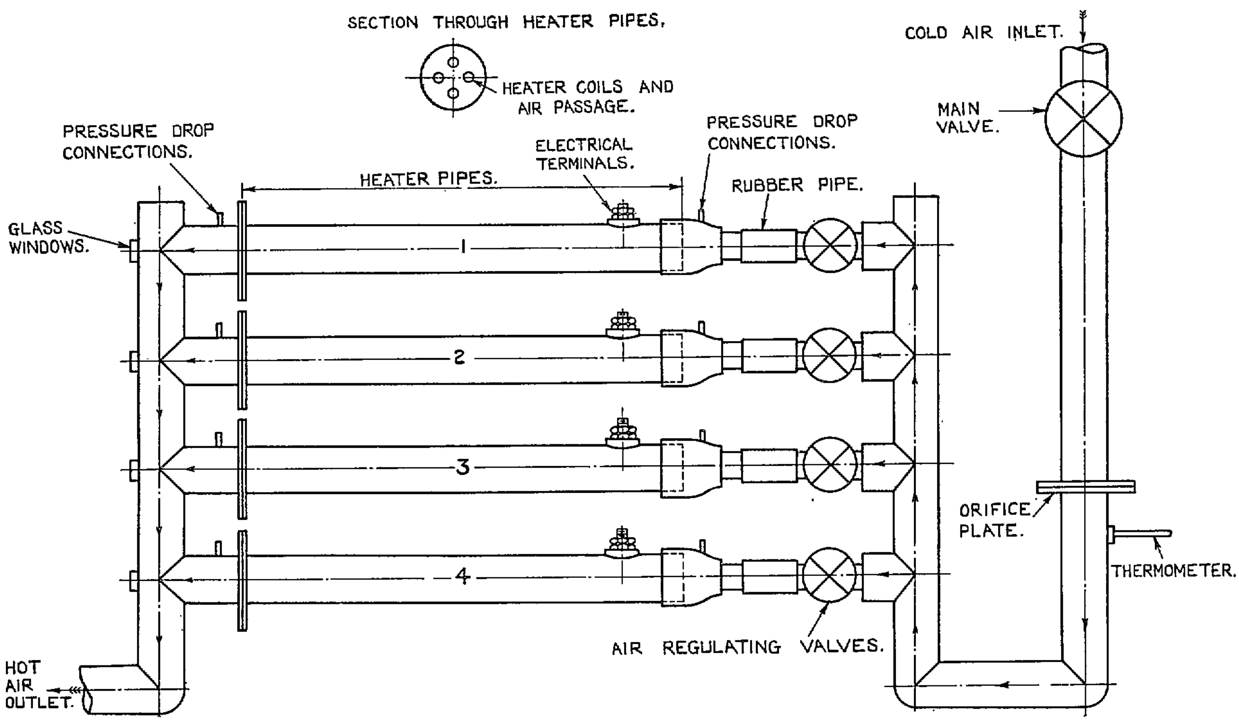


FIG. 6. Hot jet apparatus for low-speed tunnel.

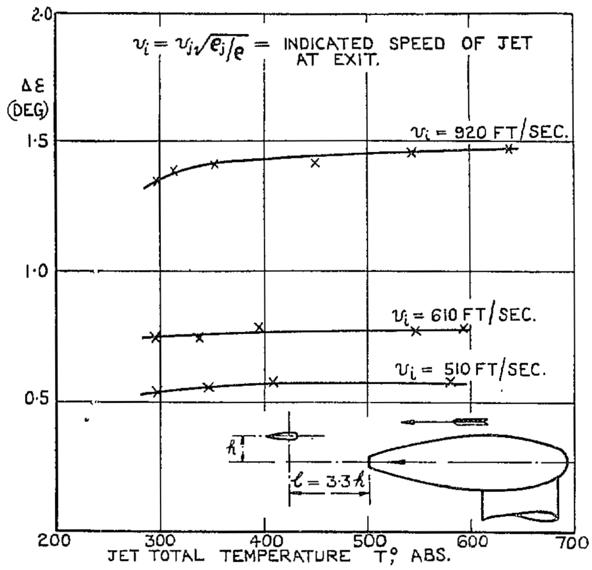


FIG. 7a. Variation of downwash with temperature (Downwash indicated by zero lift angle of small aerofoil. Wind speed 100 ft/sec).

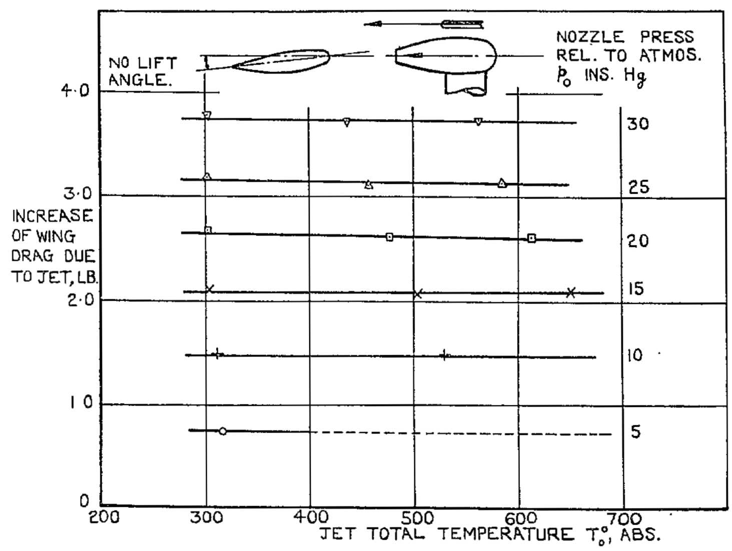


FIG. 7b. Variation of wing drag with jet temperature (Wind speed 120 ft/sec).

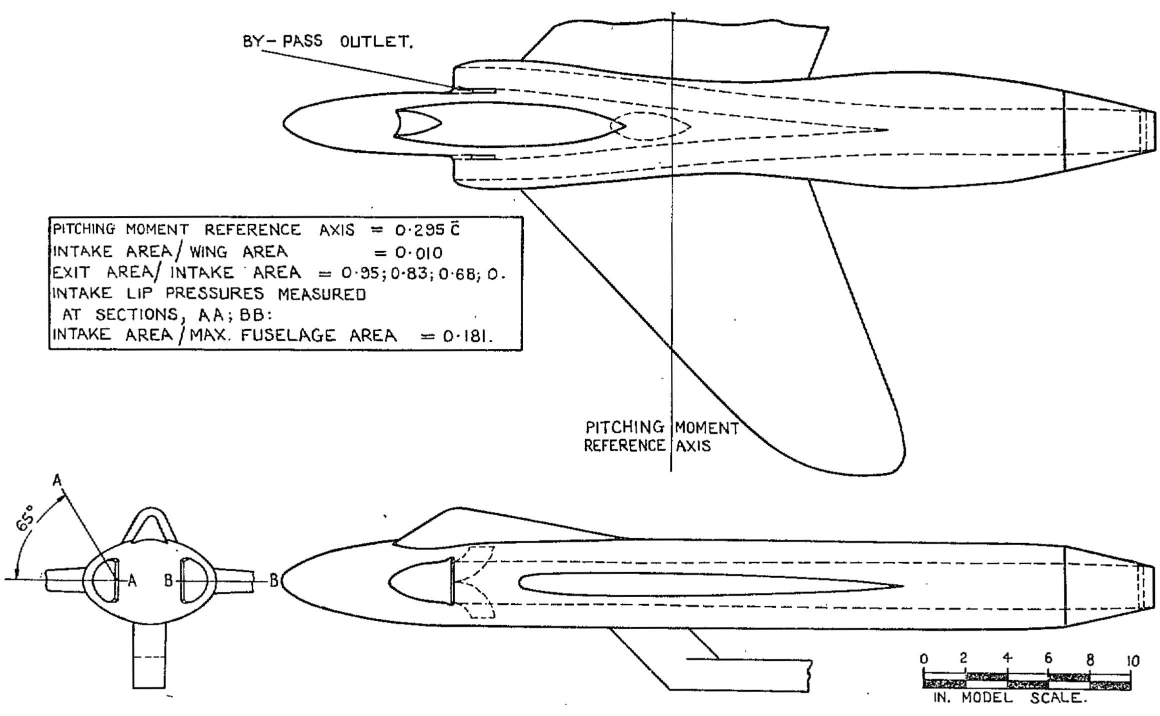


FIG. 8. General arrangement of side-intake model for high-speed tunnel test.

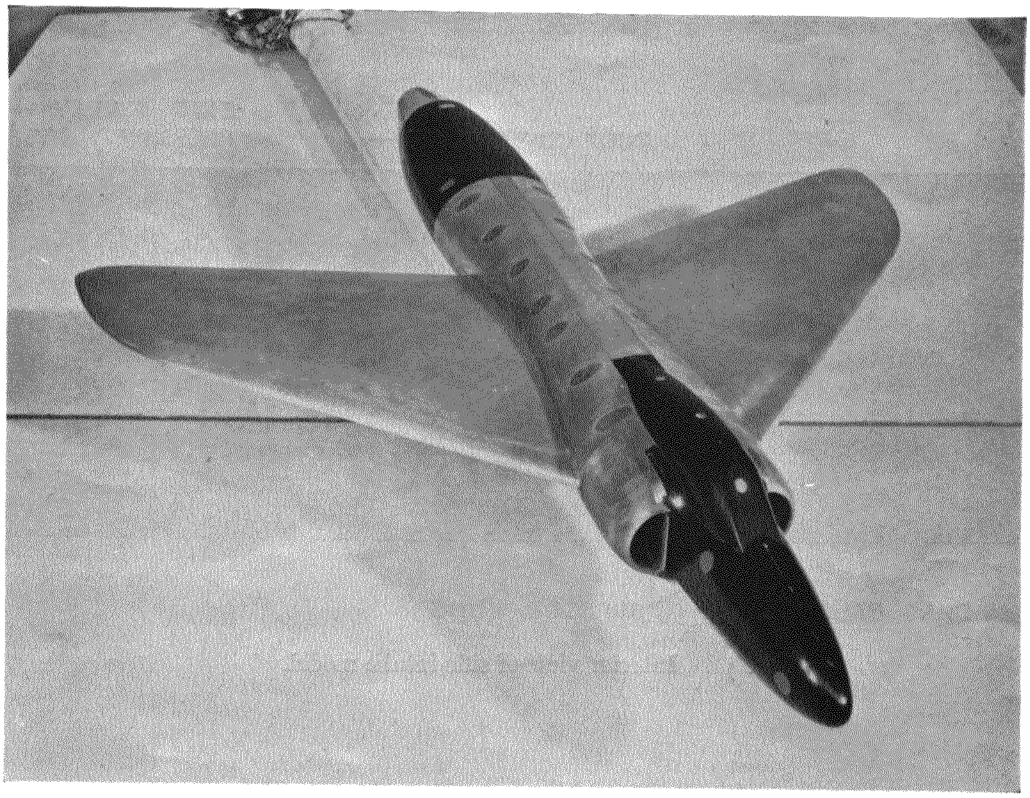


FIG. 9. General view of side-intake research model.

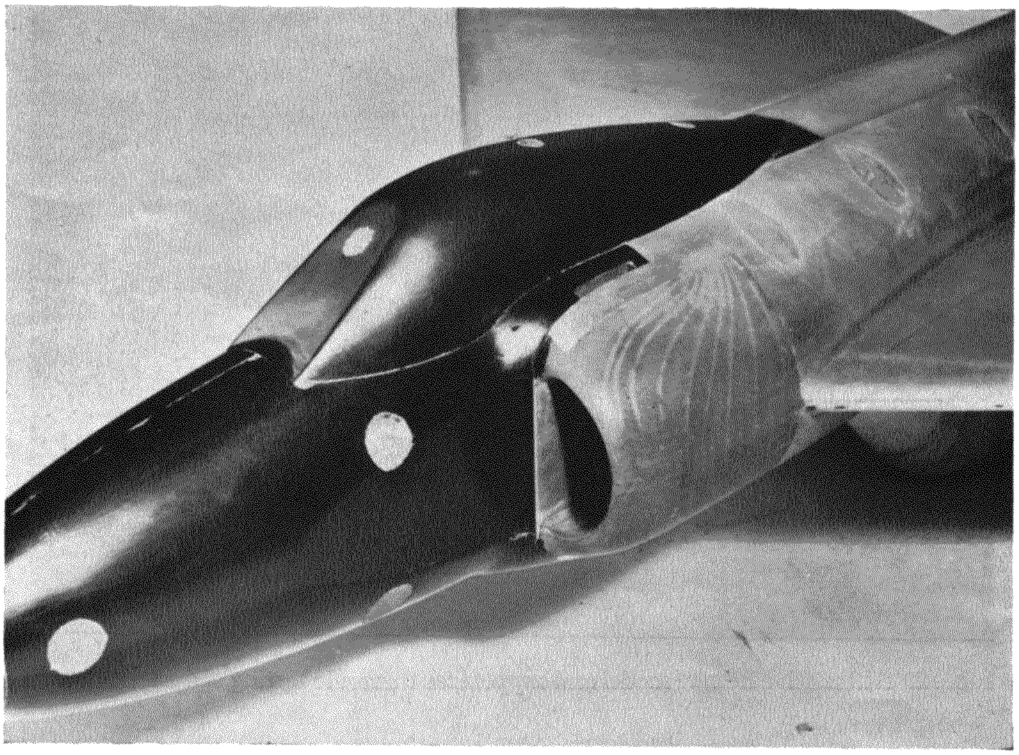


FIG. 10. Boundary-layer bleed and surface-pressure tubes.

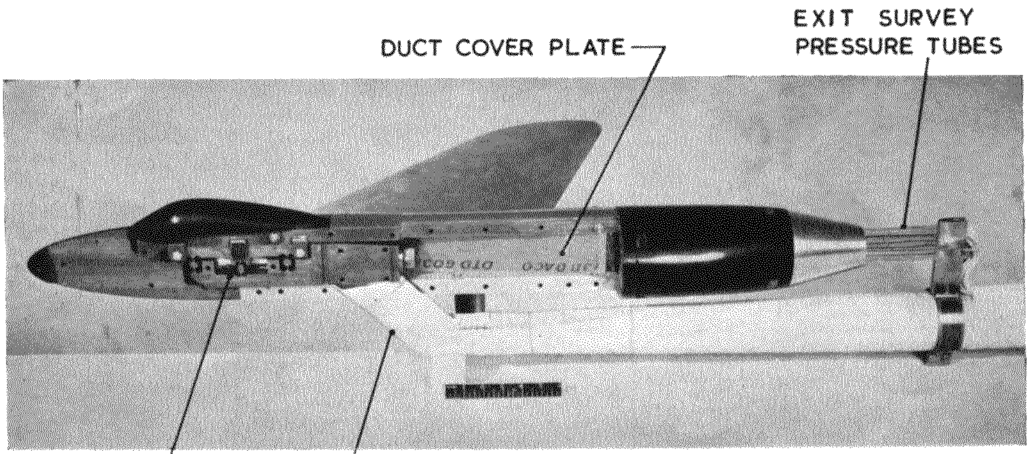


FIG. 11. Interior view of side-intake model.

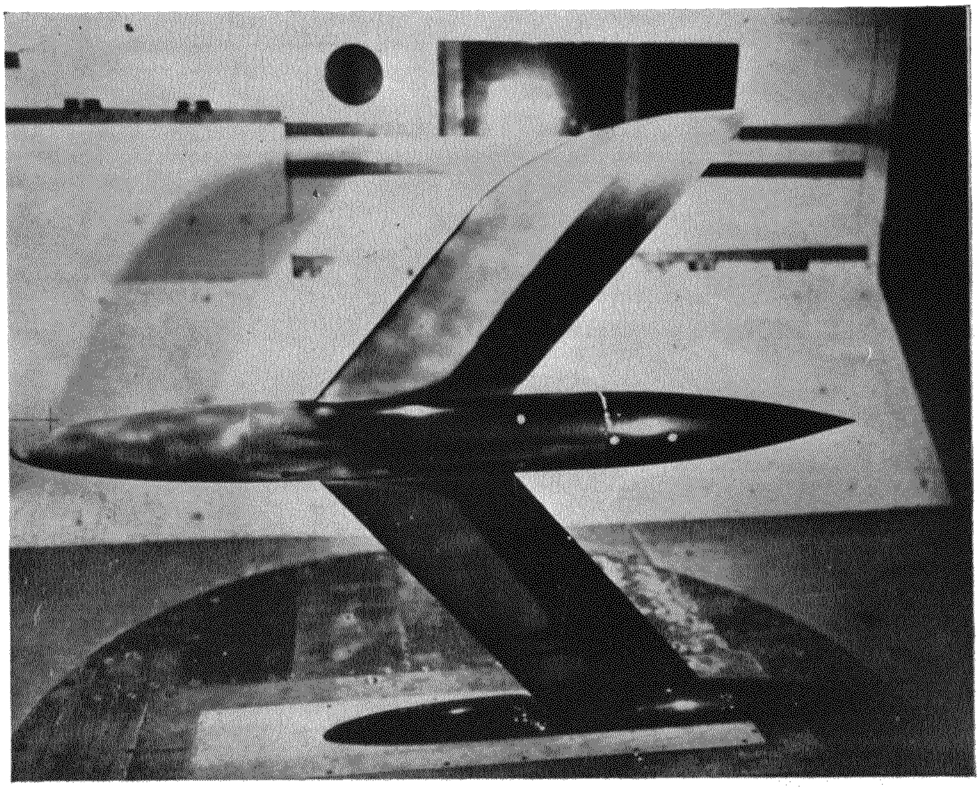


FIG. 12. Alternative form of model and support for transonic testing of ducted models.

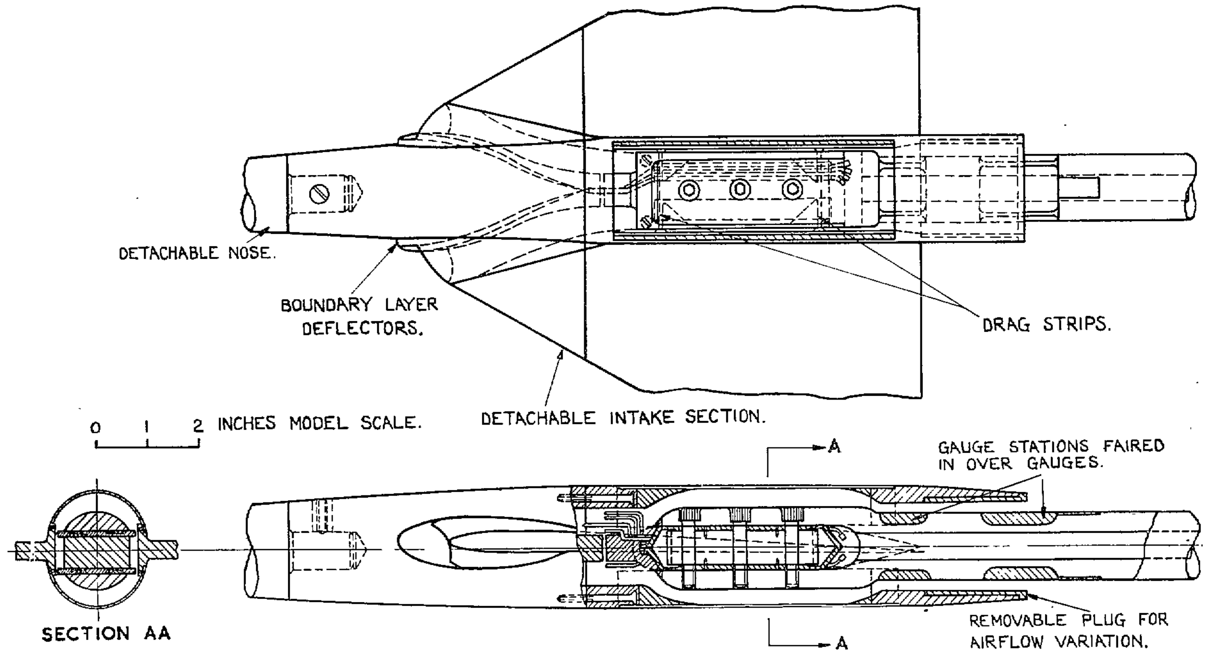


FIG. 13. General arrangement of wing-root intake model for transonic and low supersonic tunnel tests.

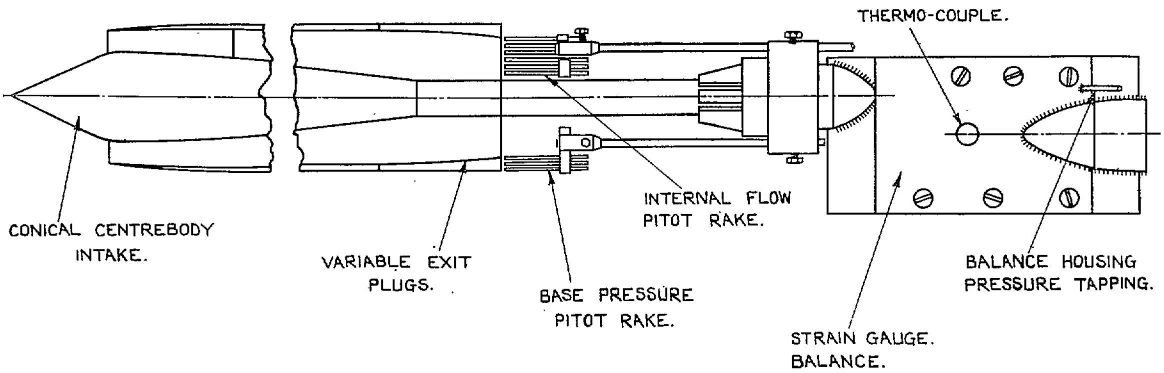


FIG. 14. Intake drag testing in small supersonic tunnel. Strain-gauge, drag balance and pitot rakes.

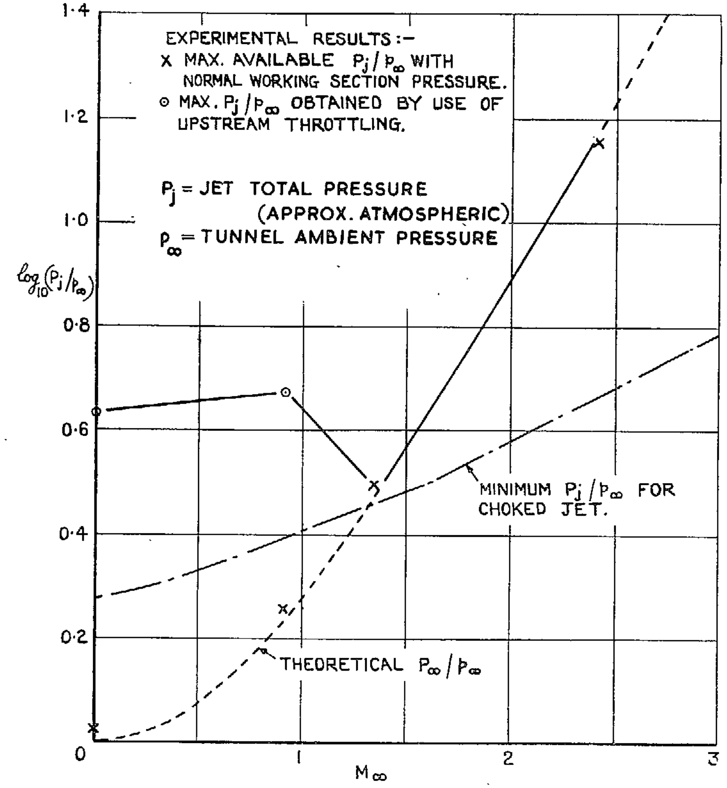


FIG. 15. Example of the use of upstream throttling to produce a choked jet at low tunnel speed.

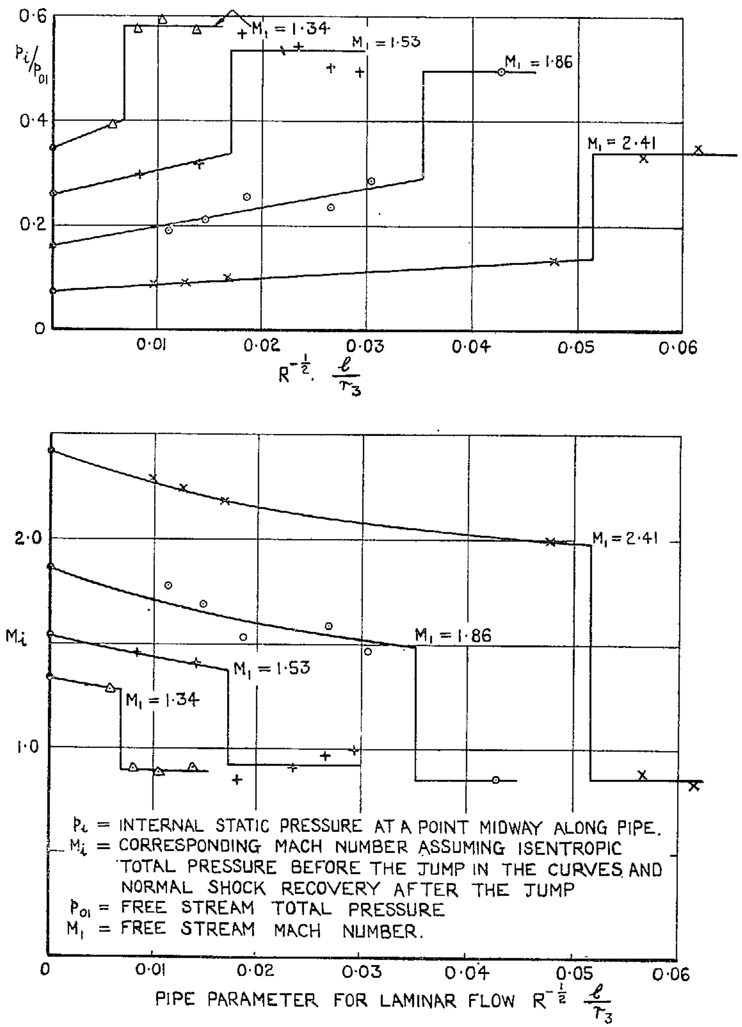


FIG. 16. Static pressure and Mach number in smooth parallel pipes.

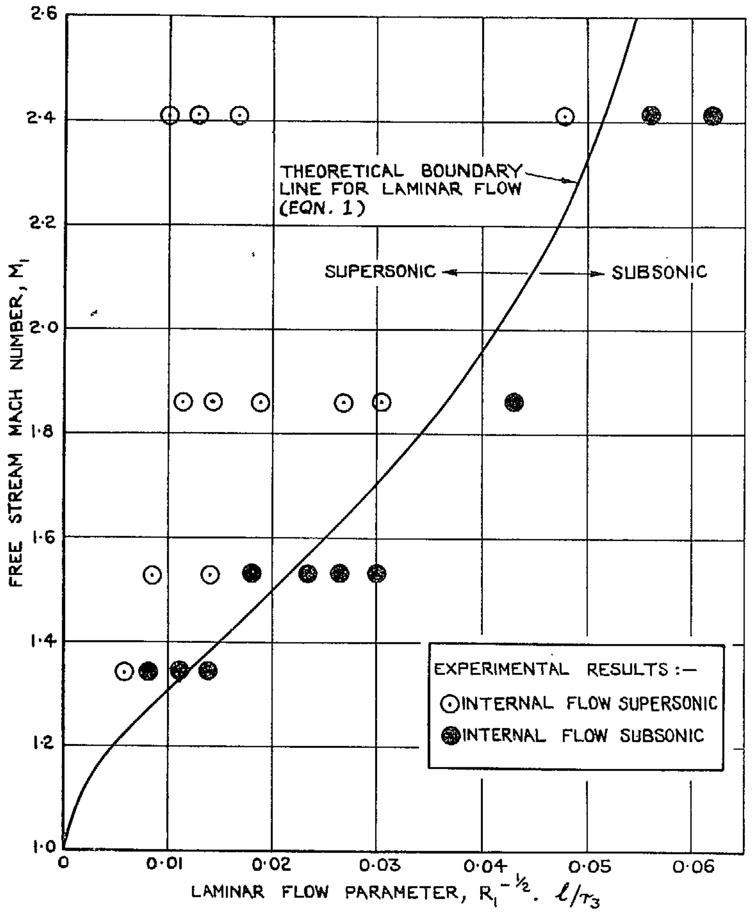


FIG. 17. Character of flow in smooth parallel pipes.

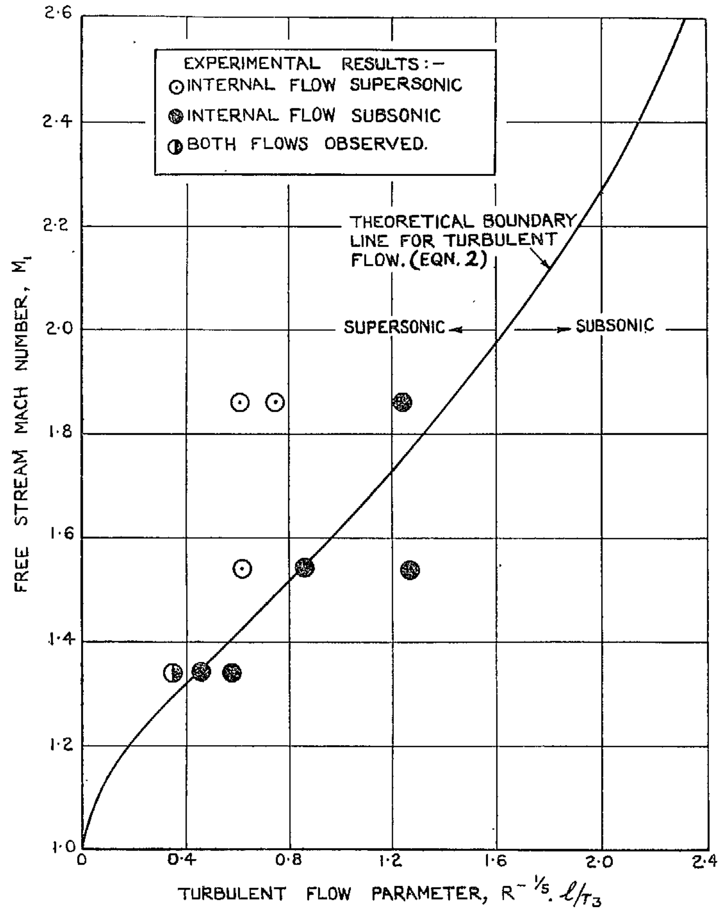


FIG. 18. Character of flow in parallel pipes with transition strip.

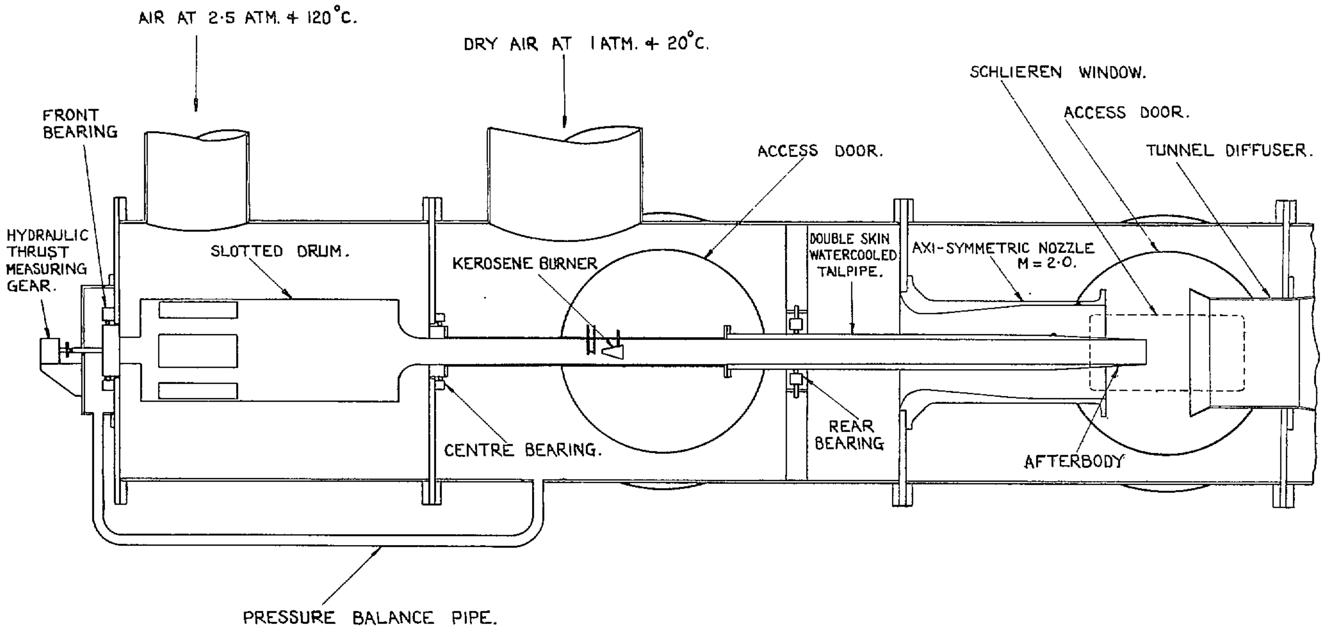


FIG. 19. Supersonic Tunnel (No. 16) for afterbody and jet tests.

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