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1. *Introduction*.—The report describes experiments with a small scale model of a ram jet burning hydrogen in the 1-ft diameter Circular High-speed Tunnel of the National Physical Laboratory adapted to run at a (nominal) Mach number of 1·4. The purpose of the tests was twofold. First, to examine how far it was practicable to test such a small scale model in a wind tunnel. Secondly, to determine to what extent the external drag of a model duct tested hot would differ from that of the same model tested cold.

The design, development and construction of a suitable model was carried out by R. P. Probert and the staff of Power Jets (Research and Development) Ltd., (now National Gas Turbine Establishment) whilst the testing was done jointly with the staff of the National Physical Laboratory.

2. *Description of Model*.—Photographs of the model are shown in Figs. 1(a) and (b) and a drawing in Fig. 2. The size chosen was a matter of compromise; it was rather too large for the tunnel since the bow shock reflected from the tunnel wall struck the model near the wing root (Fig. 3); on the other hand it was definitely too small to allow of the burning of liquid fuel. However, the burning of hydrogen proved a fairly simple matter and it is possible that hydrogen would burn successfully in an even smaller model.

2.1. *Mechanical Details*.—The model consists of a steel combustion chamber (Fig. 2) mounted on a bi-convex wing section $7\frac{1}{2}$ per cent thick of 2-in. chord and 12-in. span with end fittings to suit the 1-ft Circular High-speed Tunnel balance. Interchangeable entry diffusers and exhaust nozzles were attached to the combustion chamber by bayonet joints. Hydrogen was led through tubing let into a recess in the wing to burners in the form of six jets pointing upstream.

The connection of the hydrogen supply piping to the ends of the wings were made through flexible joints designed to leave the balance free to swing about its pivots; this proved quite satisfactory. After leaving the supply cylinder, the hydrogen passed through an orifice meter which was calibrated to give a measure of the fuel flow.

Electric leads for igniting the fuel were carried along a similar recess in the other wing ending in a spark gap; the igniting current was provided by a hand operated dynamo.

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The earliest model had brazed joints between combustion chamber and wing roots, but these failed from overheating on a trial run, and it was necessary to construct a new model with wings integral with the body. During construction considerable skill and care were required from the machinist and the fitter.

2.2. *Thermodynamic Design.*—The model was fitted with two alternative detachable entry diffusers to each of which corresponded three detachable exit nozzles; the diameters and areas at entry and exit are given in Table 1. The inlet areas were calculated to give speeds of 100 and 200 ft/sec respectively at the entry to the combustion chamber; the outlet areas were calculated to correspond to rises of stagnation temperature of 700, 850 and 1,000 deg C. respectively. A typical combination is identified by the notation 100/700. The design calculations are based on the following assumptions:—

(1) Normal shock just attached to the nose.

(2) Entry conditions: Mach number 1·4; stagnation temperature and pressure equal to standard free atmosphere values.

(3) Adiabatic efficiency in entry diffuser of 70 per cent.

(4) Baffle loss in flow past the burners taken as three velocity heads at entry to combustion chamber.

(5) Temperature rise completed in parallel combustion chamber.

(6) Zero loss in exit nozzle.

(7) Choking at exit. It seems reasonable to assume that in practice if the fuel flow is increased till the bow shock is just not forced off the nose, the exit will thus be choked.

In practice, it was found that the design conditions could be attained with the 100 ft/sec entry with all three exit nozzles, but that for the 200/850 and 200/1,000 combinations, an increase of fuel sufficient to give a shock just on the nose forced the main tunnel shock (*i.e.* the recompression shock after passing the tunnel working section) forward on to the tail of the model. With the 200/700 combination, conditions were critical but by careful manipulation of the tunnel valve it was found just possible to keep the tunnel shock sufficiently far back.

3. *Experimental Difficulties.*—3.1. *Free Steam Mach Number.*—One of the chief limitations of this experiment was the complete lack of uniformity in the supersonic flow-stream. This is partly due to the poor effuser, which was made too short owing to limitations of space, and also to moisture condensation. This is further aggravated by the circular section of the tunnel which tends to concentrate the disturbances on to the axis of the tunnel. The original tunnel design was for a Mach number of 1·52. Before return ducts were fitted an average Mach number of 1·4 was obtained in the working section and the model was designed for this Mach number. Using return ducts with the consequent lessening of condensation troubles, the average tunnel Mach number became 1·45 and the local Mach number in the vicinity of the model's nose was about 1·52.

3.2. *Mechanical Difficulties.*—Various mechanical difficulties frequently held up work. Distortion of the interchangeable exhaust nozzles caused them to jam. Overheating, following a failure in the burner made a complete refitting of the model necessary. Vibration caused cements used as fillings in the grooves milled in the wing for the ignition lead to come out, until a cement consisting of alumina mixed with glycerol was suggested by Materials Department of the Royal Aircraft Establishment.

4. *Details of Experiment.*—4.1. *Hot Tests.*—Owing to the high temperature reached by the casing it was impossible to measure gas pressures in the interior of the model. Attempts to measure stagnation temperature in the wake were only partially successful because the smallest size of stagnation temperature tube that could be used was rather large in comparison with the outlet area while the temperature gradients were very steep (Fig. 4). Some traverses of total head were also taken, though spot readings were sufficient.

The following procedure was therefore adopted. The model was first mounted in a transparent (perspex) working section (Fig. 3). A mercury vapour lamp shining through the tunnel gave an image of the shock pattern as a direct shadow on a screen. The hydrogen was ignited at subsonic speed and the tunnel speed was then raised to its full value with the tunnel shock well behind the tail of the model. The fuel pressure was then increased until a normal shock appeared just in front of the inlet. The fuel pressure was then decreased until the normal shock was just on the point of disappearing into the inlet and the fuel flow as measured by the orifice meter was recorded. Observations were made with the four combinations 100/700, 100/850, 100/1,000 and 200/700 but were not possible with 200/850 and 200/1,000.

It was not possible to see the nature of the mixing of the hot exhaust with the air stream on the shadow screen.

At high temperatures the model glowed a bright red around the position of the hydrogen burners, but became cooler towards the tail. Very little flame was visible.

The transparent working section was next replaced by the balance section in which it was possible to measure the overall drag of the model and its supporting aerofoil. The above procedure was then repeated with the fuel flow adjusted to the same value as in the transparent section and it was assumed that the same conditions would obtain of plane shock across the inlet orifice and choking at the exit nozzle.

The high-speed tunnel electric balance determines the drag of a model as the difference between the moments of the aerodynamic forces about two parallel axes, and the moment axis situated at the quarter-chord point of the supporting wing and the drag axis displaced one chord from the moment axis in a direction perpendicular to the chord. The reading about the moment axis should be zero for a symmetrical model; in the present experiments it was found to be negligibly small so that readings only about the drag axis were necessary.

4.2. *Cold Tests.*—A number of internal baffle plates of varying resistance were made for use when testing the model cold. By a suitable adjustment of the baffle resistance it should be possible to imitate the entry conditions of the model when burning any amount of fuel. The exit Mach number in the cold case would be considerably less than unity for any degree of baffling. It would be expected that the external drag would be independent of entry conditions provided that the shock wave was not actually off the nose so that extra baffling additional to that of the burners was not really necessary. However, it was found by trial and error that baffle 4* just pushed the shock off the nose while baffle 3 did not. A number of tests were made with baffle 3 and also with no baffle. Complete traverses of both static and pitot pressure were taken in the exit plane of the model. Once again conditions were assumed to be the same when the model is placed in the balance section of the tunnel.

5. *Reduction of Results.*—5.1. *Cold Case.*—In order to determine complete conditions at exit it is necessary to know three independent quantities, *e.g.* mass flow Q through the duct, the total temperature T_T and the total head H. Alternatively T_T, H and the static pressure p could be used. If we assume the entry conditions to be known, the unit mass flow can be calculated since there was no mass loss or gain through the duct. The total temperature also was constant through the model and so the measurement of one quantity, total head, should be sufficient. It was found that the total head over the exit nozzle was not constant, and that a spot reading on the axis did not give a reliable value. Moreover, the variation over the nozzle was not the same in all cases, so that no correction applied universally. For this reason traverses of total pressure were taken (Fig. 5), and the average value of the traverse used in each case. As a check static traverses behind the exit nozzle were also taken (Fig. 6) and the mass flow calculated using H, p and T_T and the geometrical exit area. In two cases out of eight this agreed with the flow calculated from entry conditions—in the other six it gave a flow about 10 per cent higher. It is unlikely that this is due to constriction of flow in the exit nozzles, since the discharge coefficient which would have to be introduced to modify the exit area, would certainly

* See notation at end of report.

not vary by so large an amount as 10 per cent. However, there is great difficulty in obtaining an accurate assessment of air mass flow on so small a scale under the conditions obtaining, and air mass flow from entry conditions was considered to be the more reliable. A speciment calculation in a cold case is shown in the Appendix and results are tabulated in Table 2.

5.2. *Hot Case.*—A typical pitot traverse of the exit nozzle for the hot case is given in Fig. 7. It is seen that the total head is almost constant across the wake. In consequence of this it was possible to use spot readings throughout together with a discharge coefficient of 0·95. As only the one reading needed to be taken in the hot stream an ordinary steel pitot tube was used, since the reading was taken and the tube withdrawn before it became overheated.

The exhaust conditions were calculated from the known air flow, the assumed conditions of choking and the spot reading of total head. A specimen calculation for a hot case is shown in the Appendix and the results are tabulated in Table 3.

6. *Temperature Considerations.*—Temperature traverses were taken using a stagnation thermometer, but the temperature gradients involved were too large in relation to the size of the thermometer to be able to place any reliance on the results. Typical traverses are shown in Fig. 4. The traverses were made in the same plane along lines at 60 deg to each other.

From the known conditions of exhaust it is possible to calculate the total temperature. Using this value and the known fuel flow together with a figure for the calorific value of hydrogen, it should be possible to calculate the combustion efficiency. In this particular instance, however, it was found that the combustion efficiency came to be of the order of 100 per cent. This is certainly not the case and so the calculations have not been included in the report. Since the fuel flow has not been used in any other calculations the measurement has only been used as a guide to fuel consumption. This does not invalidate the rest of the calculations.

7. *External Drag.*—7.1 *Balance Measurements.*—Having calculated the internal thrust of the model, the external drag was tested by subtracting this value from the measured force on the balance. In the cold case the balance drags were measured on different runs from those on which the pressure measurements were taken, consequently the average value of the balance drag was used for each rig. In the hot cases the balance drag was measured on the same run as the pressure measurements and consequently no such average was necessary.

7.2. *Calcnlation from components.*—The external drag was also calculated from its three main components, and tabulated in Table 4.

(1) The drag of a two-dimensional wing of the given form was known, and so the drag D_W of the two supports themselves was known and constant. The interference drag of the wing junctions was not calculable.

(2) A longitudinal static traverse was taken alongside the model fore and aft of the wing. One is shown in Fig. 2. As a number of Mach lines were visible it was possible to relate the pressure on the model to those on the traversing line, by the assumption that the pressure is constant along a Mach line. Over the rear of the model the pressure is seen to rise steeply—this being due to bad distribution in the tunnel. The kink is due to a disturbance from the wing root on the tunnel wall. The flow has almost certainly separated over the tail of the model—this would certainly happen with such an adverse pressure gradient, though not necessarily in flight—and so an average value of the pressure on the surface of the separation was assumed. By integrating these pressures up, the form drag D_F was found.

(3) To find the skin friction drag D_f, a skin friction coefficient of 0·0036 was assumed, based on Schlichting's approximate formula $C_{Df}=0{\cdot}455/(\log 10^R)^{2.58}$, which yields a cold skin friction drag (assuming separation) of 0·75 lb in the 100/700 case. The effect of heating the surface to about 550 deg C. is not calculable, but a rough estimate can be formed assuming the sole effects to be a change of Reynolds number which increased C_{Df} to 0·004, coupled with an appropriate reduction of density. This suggests that the hot skin friction drag would be about 0·3 lb less than the cold for the 100/700 case.

The external drag $D_E = D_F + D_f + D_W$.

8. *Discussion of Results.*—The comparative results for the external drags in the two cases are as follows:—

		100/700	100/850	100/1000	200/700
Drag					
	Cold	7·01 to 7·24	6·97 to 7·06	6·93	6·90
	Hot	6·84 to 6·91	6·42	6.21	6·21
Drag Coefficient					
	Cold	0·365 to 0·368	0·358 to 0·368	0·359	0·352
	Hot	0·355	0·327	0·317	0·317

It is thus seen that there is a reduction in external drag which increases with the temperature of the body for the 100/entry nozzle. With the possible exception of the 100/700 case, the reduction in drag may be considered to be greater than the experimental error, and is therefore real. There are two possible reasons for such a change.

(i) A decrease in skin friction as mentioned above. An approximate calculation for the 100/700 case gives a decrease of the order of 0·3 lbs. in the drag or of 0·015 in the drag coefficient.

(ii) A change in exhaust conditions causing separation of a different kind over the tail.

The external drag of the model does not vary beyond the limits of experimental error for different intakes and nozzles.

The drag deduced from pressure plotting (Table 4) for the 100/700 case gives fairly close agreement with that deduced from the air flow and balance readings.

9. *Future Experiments.*—Undoubtedly many of the difficulties experienced are due to the extremely bad velocity distribution in the tunnel. When a larger tunnel is available with a really good velocity distribution, the experiment could be repeated with more success, particularly since the bow wave reflection would no longer affect the model. By the use of plane glass walls it should be possible to view the model throughout the whole experiment, and in particular to study the flow near the tail in the two cases. It would also be advisable to pressure plot the exterior of the model, at any rate in the cold tests.

10. *Conclusions.*—The present experiment has been of value in giving experience of the experimental technique and in giving some verification of the assumptions commonly made in designing a ram jet. There appears to be experimental evidence of a reduction in the external drag of the model due to heating, and this must be allowed for in future work.

Satisfactory accuracy could be obtained on all points by the use of a large tunnel of square or rectangular section, with a good velocity distribution and transparent walls large enough to view the whole model.

11. *Acknowledgement.*—The authors wish to thank Mr. Lock of the N.P.L. and Mr. Probert of the National Gas Turbine Establishment for their help and criticism of the paper.

NOTATION

A	Cross-sectional area of duct in sq. in.
A_F	Frontal area of model and supports in sq. in.
B	Balance force in lb.
C_D	External drag coefficient.
C_{Df}	Skin friction drag coefficient.
D_E	External drag in lb.
D_f	Skin friction drag in lb.
D_F	Form drag in lb.
D_I	Internal drag in lb.
D_W	Drag of wing supports in lb.
F_M	Exit momentum in lb $(= C_4 V_4^2 A_4)$.
F_P	Pressure component of internal thrust in lb $(= (P_4 - P_1) A_4)$.
H	Total head in inches of mercury (Hg).
I_M	Inlet momentum in lb $(= p_1 V_1^2 A_1)$.
M	Mach number.
p	Static pressure in inches of mercury (Hg).
Q_A	Air mass flow in lb.
T_T	Total temperature in deg K.
V	Velocity in ft/sec.
γ	Ratio of specific heats of air.
π	Factor converting inches of mercury (Hg) to lb/sq in. $(= 0 \cdot 492)$.
ρ	Density.
Suffix '0'	indicates stagnation conditions.
'1'	,, entry ,,
'4'	,, exit ,,
100/700/3	means Design velocity at entry to combustion chamber = 100ft/sec. Design temperature rise of 700 deg C. Baffle '3' in position.

REFERENCE

Author	*Title, etc.*
R. P. Probert	The Development of a Model Propulsion Duct for Wind Tunnel Experiment. Power Jets Report No. R.1122.

TABLE 1

Combination	Entry Diameter (in.)	Entry Area (sq ft)	Exit Diameter (in.)	Exit Area (sq ft)
100/700	0·53	0·00153	0·76	0·00315
100/850	,,	,,	0·785	0·00336
100/1,000	,,	,,	0·81	0·00357
200/700	0·75	0·00306	1·08	0·00635
200/850	,,	,,	1·125	0·00690
200/1,000	,,	,,	1·20	0·00780

TABLE 2 (Cold Tests)

Combination Test number		100/700/0 93	100/700/3 86	100/700/3 70	100/850/0 82	100/850/3 89	100/1,000/0 83	100/1,000/3 84	200/700/0 87
Entry Total Head (Hg in.) ..	H_1	28·80	28·87	28·31	28·59	28·98	28·59	28·59	28·97
Static Pressure (Hg in.)	p_1	7·59	7·66	7·40	7·48	7·67	7·58	7·58	7·66
Mach number	M_1	1·520	1·520	1·528	1·524	1·518	1·518	1·518	1·522
Air flow (i) Slugs/sec $\times 10^{-3}$ ..	Q_A	1·90	1·91	1·86	1·88	1·92	1·89	1·89	3·82
(ii) lb/sec $\times 10^{-2}$	..	6·12	6·15	5·98	6·05	6·18	6·09	6·09	12·29
Momentum lb	I_M	2·94	2·635	2·89	2·64	2·96	2·93	2·93	5·45
Total Temperature deg K	T_T	288	288	288	288	288	288	288	288
Exit Total Head (Hg in.) ..	H_4	15·31	14·47	14·67	13·39	14·58	13·59	13·59	14·77
Mach number	M_4	0·522	0·570	0·535	0·567	0·517	0·510	0·512	0·546
Static Pressure (Hg in.)	p_4	12·72	11·59	12·08	10·74	12·18	11·39	11·38	12·03
Momentum lb	F_M	1·082	1·174	1·079	1·152	1·083	1·048	1·053	2·259
Pressure component of thrust (lb)	F_p	1·147	0·876	1·047	0·779	1·074	0·963	0·960	1·974
Internal drag (lb)	D_I	0·71	0·58	0·76	0·79	0·80	0·92	0·92	1·22
Balance drag (lb)	B	7·73	7·77	7·77	7·77	7·77	7·75	7·85	8·12
External drag (lb)	D_E	7·02	7·19	7·01	7·06	6·97	6·93	6·93	6·90
External drag coefficient lb ..	..	0·365	0·368	0·367	0·368	0·358	0·359	0·359	0·352

TABLE 3 (Hot Cases)

Combination Test number			100/700 36	100/700 95	100/850 96	100/1,000 104	200/700 106
Entry	Total Head (Hg in.)	H_1	28·58	28·81	28·52	28·85	28·85
	Static Pressure (Hg in.)	p_1	7·47	7·56	7·72	7·64	7·74
	Mach Number	M_1	1·528	1·528	1·514	1·523	1·514
	Air Flow (slugs/sec $\times 10^{-3}$)	Q_A	1·89	1·90	1·92	1·91	3·84
	Momentum (lb)	I_M	2·64	2·73	2·70	2·71	5·27
Exit	Total Head (Hg in.)	H_4	24·35	23·51	23·91	23·85	21·01
	Static Pressure (Hg in.)	p_4	13·22	12·78	13·00	12·98	11·43
	Momentum (lb)	F_M	3·93	3·60	3·91	4·15	6·49
Pressure Component of Thrust (lb)		F_p	1·25	1·13	1·22	1·31	1·61
Nett Thrust (lb)			2·54	2·00	2·43	2·75	2·83
Balance Drag (lb)		B	4·30	4·91	3·99	3·46	3·38
External drag (lb)			6·84	6·91	6·42	6·21	6·21
External drag coefficient			0·355	0·355	0·327	0·317	0·317

TABLE 4

Cold Case: Combination 100/700

Form Drag	1·98
Skin Friction	0·75
Wing drag	4·31
External Drag	7·04
External Drag Coefficient (for $p = 7{\cdot}55$ $M = 1{\cdot}52$)	0·364

APPENDIX

Specimen Calculation

1. COLD CASE Test number 93 Combination 100/700/0

Entry observation $H_1 = 28{\cdot}80,$

and $p_1 = 7{\cdot}59,$

whence $M_1 = 1{\cdot}520.$

Now $Q_A = \gamma\,\pi\,H_1 A_1 M_1 (1 + 0{\cdot}2\,M_1{}^2)^{-3}/a_0$ (a_0 = stagnation velocity of sound)

$$= \frac{1{\cdot}4 \times 0{\cdot}492 \times 28{\cdot}80 \times 144 \times 0{\cdot}00153 \times 1{\cdot}52\,(1 + {\cdot}462)^{-3}}{1120}$$

$= 0{\cdot}00190$ slugs/sec.

Entry momentum $I_M = \rho_1 V_1{}^2 A_1 = \gamma\,\pi p_1\,M_1{}^2 A_1$

$= 2{\cdot}94$ lb

Exit From observation H_4 (exit total head) $= 15{\cdot}31$

therefore $M_4\,(1 + 0{\cdot}2\,M_4{}^2)^{-3} = \dfrac{a_0\,Q_A}{\gamma\pi\,H_4\,A_4}$

$$= \frac{1120 \times 0{\cdot}00190}{1{\cdot}4 \times 0{\cdot}492 \times 15{\cdot}31 \times 144 \times 0{\cdot}00315}$$

$= 0{\cdot}445$

Using a curve of $f\,(M) = M(1 + 0{\cdot}2\,M^2)^{-3}$ we obtain $M_4 = 0{\cdot}522$

For $M = 0{\cdot}522,\ H/p = 1{\cdot}202$, hence $p_4 = 12{\cdot}72$

Exit momentum $F_M = \gamma\pi p_4 M_4{}^2 A_4 = 1{\cdot}082$ lb

$F_p = \pi(p_4 - p_1)A_4 = 1{\cdot}147$ lb

therefore Internal drag $= I_M - F_M - F_p = 2{\cdot}94 - 1{\cdot}082 - 1{\cdot}147$

$= 0{\cdot}71$ lb

From balance drag $B = 7{\cdot}73$

therefore External drag $= 7{\cdot}02$ lb

Total frontal area of model and supports $A_f = 0{\cdot}0223$ sq. ft.

$$\text{therefore } C_D = \frac{\text{External Drag}}{\frac{1}{2}\,\rho_1 V_1{}^2\,A_f} = \frac{7{\cdot}02}{\frac{1}{2} \times 1{\cdot}4 \times 0{\cdot}492 \times p_1\,M_1{}^2 \times 0{\cdot}0223 \times 144}$$

$= 0{\cdot}365$

2. HOT CASE

Test number 36 Combination 100/700

Entry From observation $H_1 = 28{\cdot}58$,

and $p_1 = 7{\cdot}47$,

whence $M_1 = 1{\cdot}528$.

As above $Q_A = 0{\cdot}00189$

and $I_M = 2{\cdot}64$ lb

Exit From observation $H_4 = 24{\cdot}35$

Since $M_4 = 1$, $H_4/p_4 = 1{\cdot}840$ assuming $\gamma = 1{\cdot}33$

Then $p_4 = 13{\cdot}22$

Whence, $F_M = \gamma \pi p_4 M_4{}^2 A_4 = 1{\cdot}33 \times 0{\cdot}492 \times 13{\cdot}22 \times 144 \times 0{\cdot}00315 \times 0{\cdot}95$

$= 3{\cdot}93$, using a discharge coefficient of $0{\cdot}95$.

$F_p = \pi(p_4 - p_1)A_4 = 1{\cdot}246$

Therefore, Thrust $= F_M + F_p - I_M = 3{\cdot}93 + 1{\cdot}246 - 2{\cdot}64$

$= 2{\cdot}54$ lb

Balance force $B = 4{\cdot}30$

Therefore, External drag $= 6{\cdot}84$

$C_D = 0.355$

N.B. It is worth noting that for finding the internal thrust the only variables actually needed at entry and exit are p and M. Since for the hot case these are directly known it is unnecessary to find the mass flow through the duct. For calculations of combustion efficiencies, etc. we do need the mass flow, and furthermore have to include the mass of the fuel in our calculation.

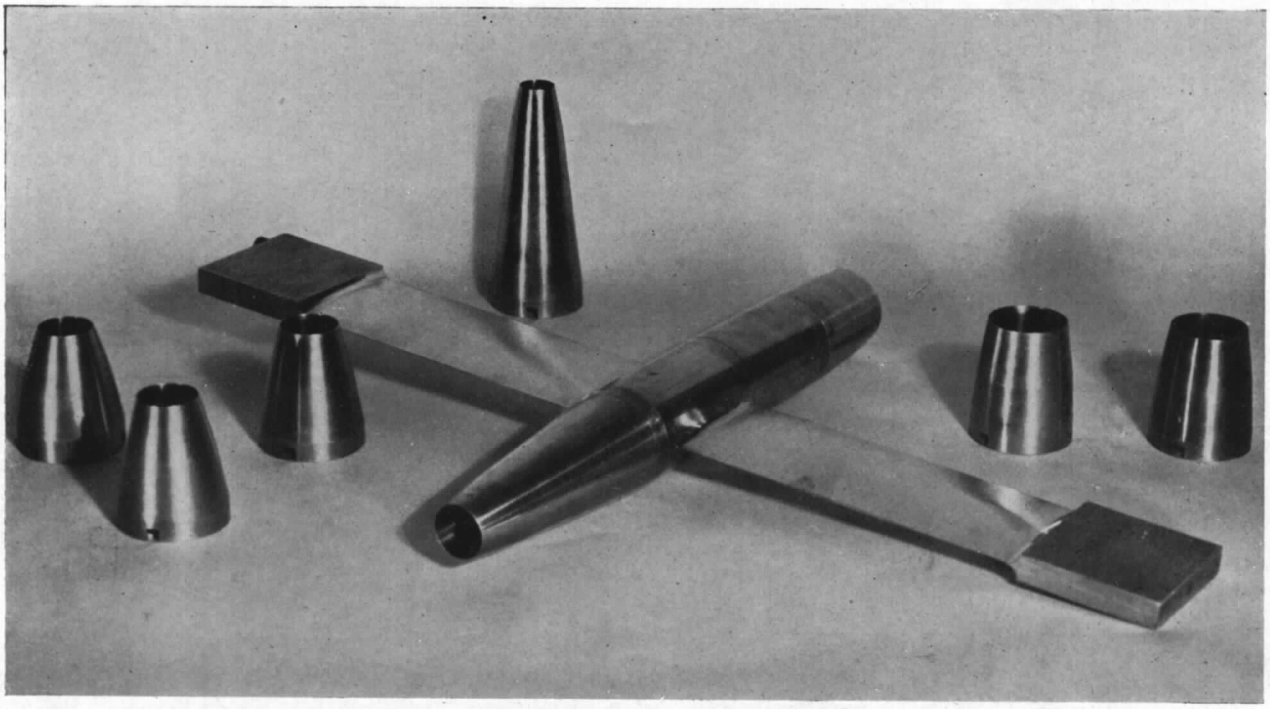

FIG. 1 (*a*).

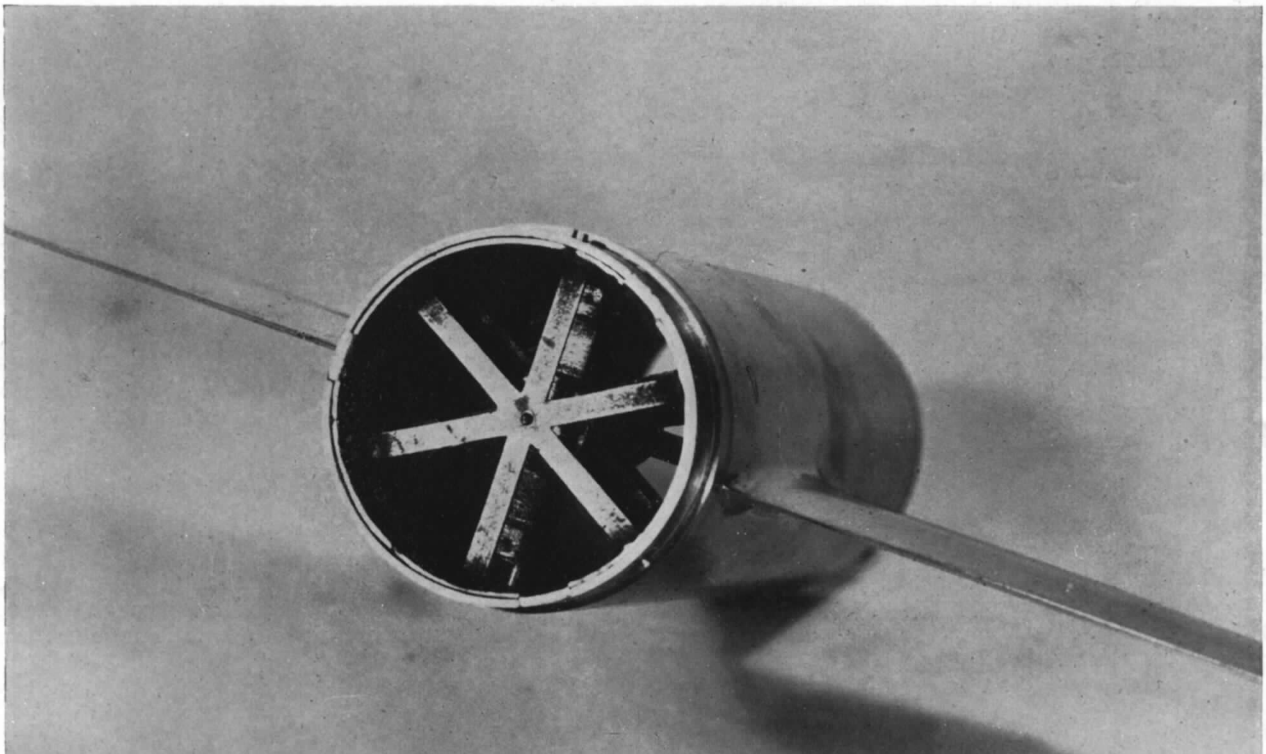

FIG. 1 (*b*).

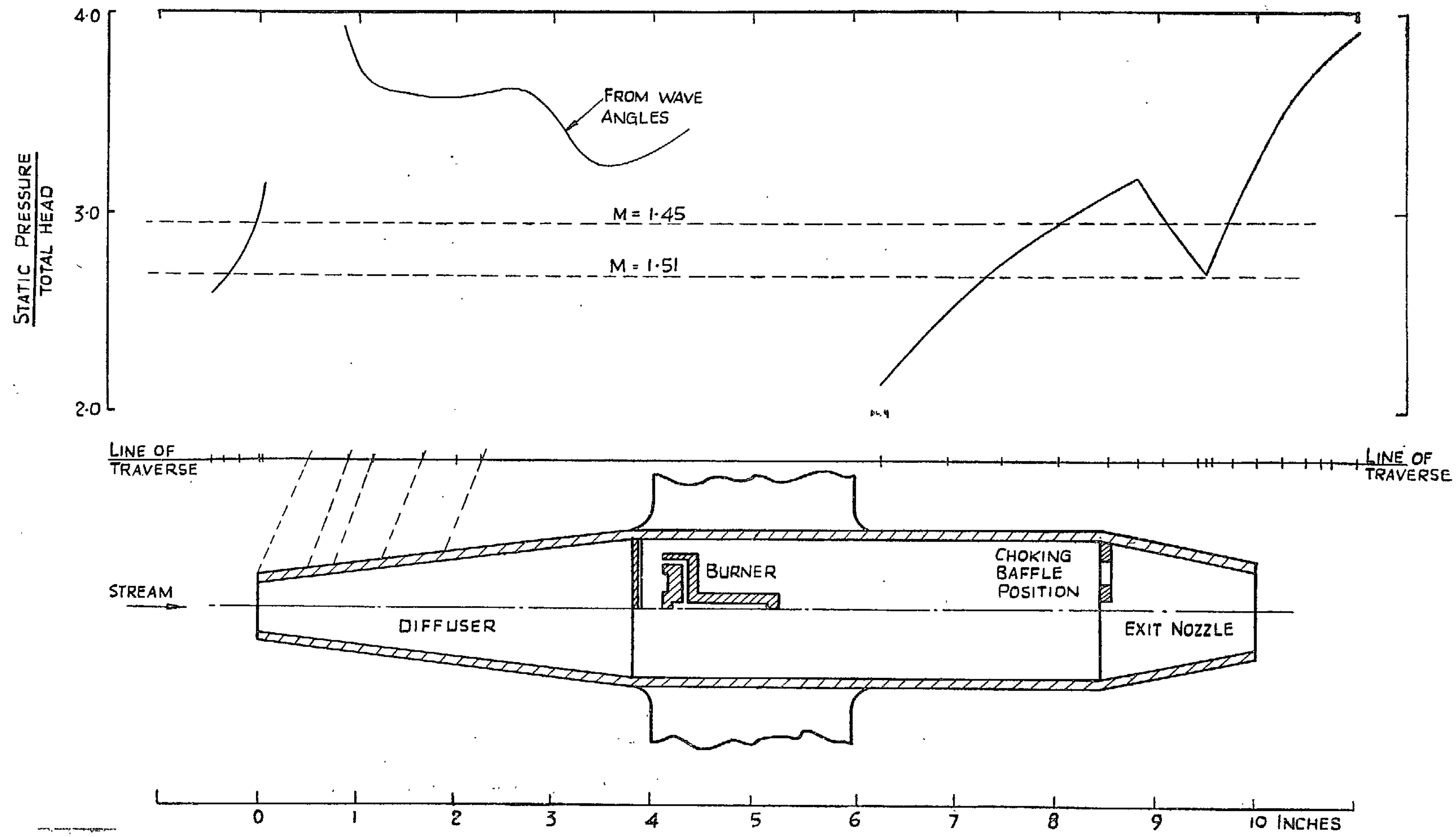

FIG. 2. Drawing of model with static pressure distribution for 100/700.

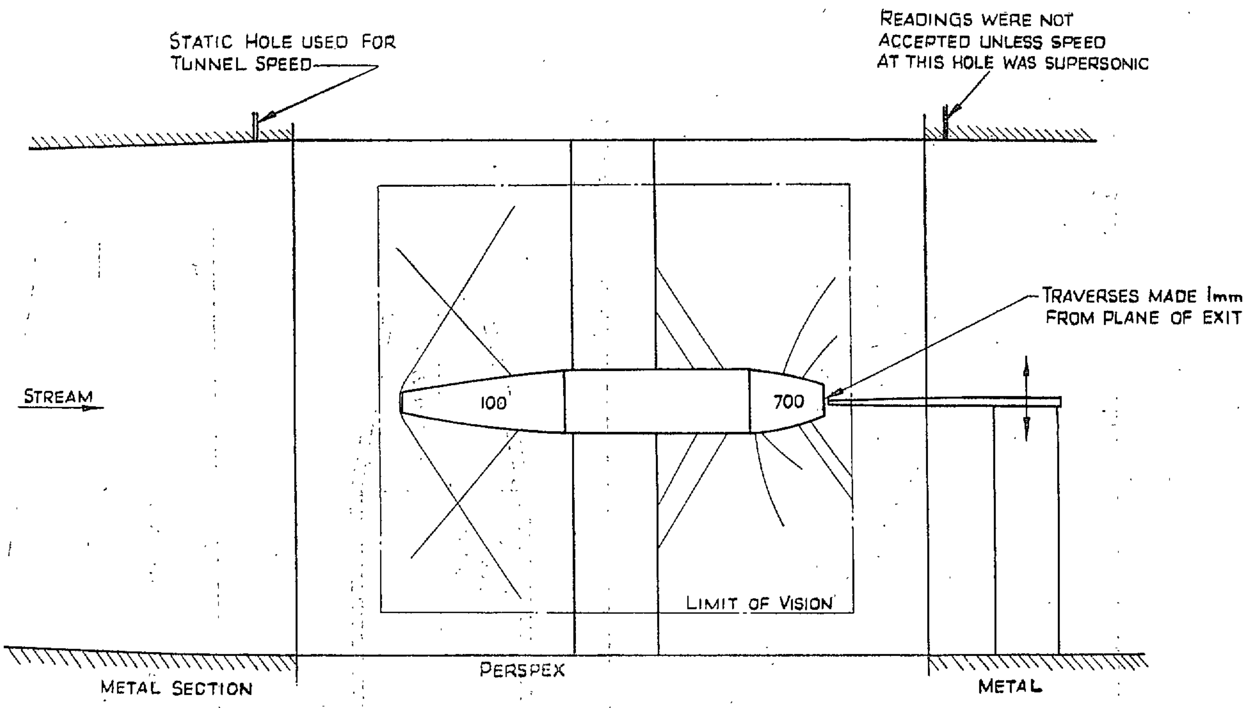

FIG. 3. Diagram showing shock waves.

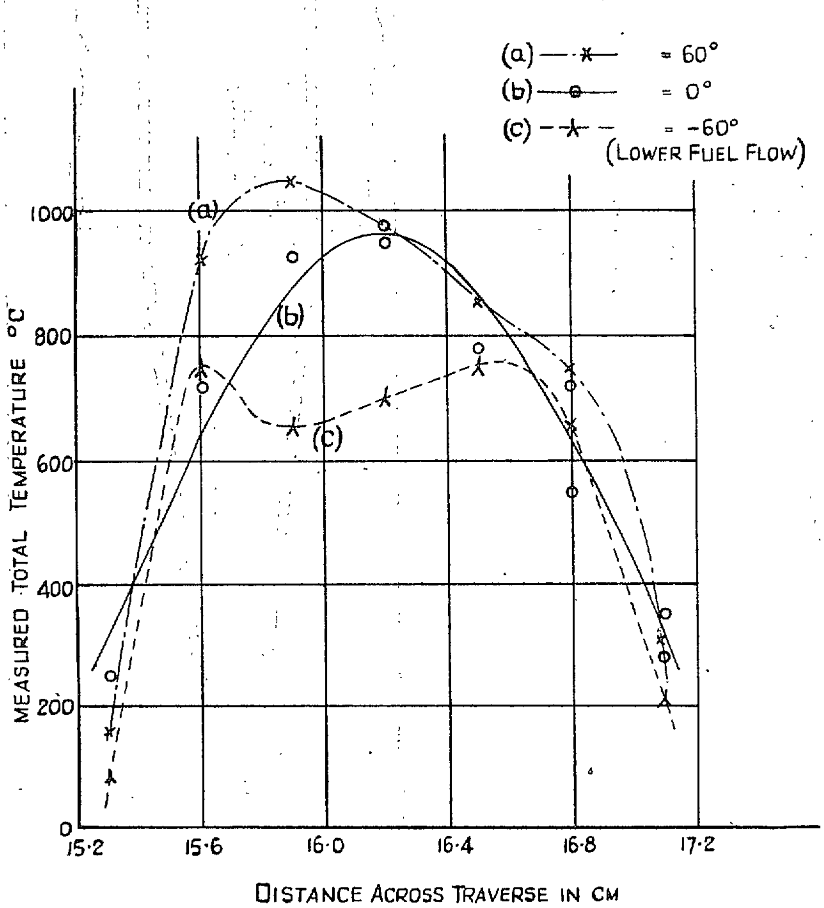

FIG. 4. Temperature traverses in wake for different directions of traverse.

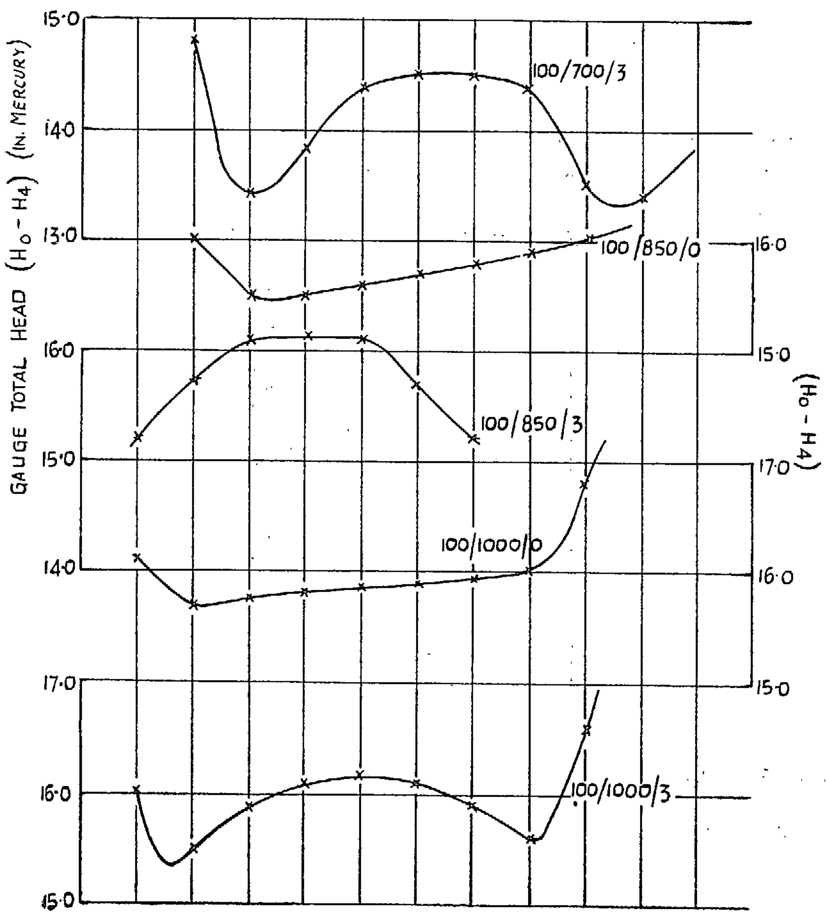

FIG. 5. Pitot traverses at exit (cold).

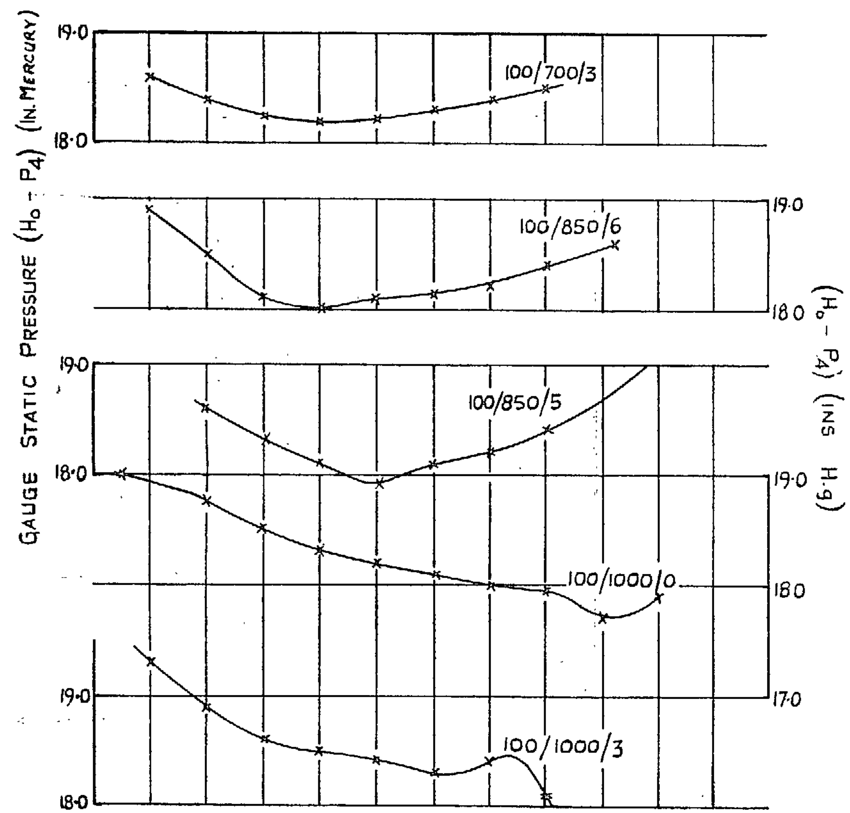

FIG. 6. Static traverse at exit (cold).

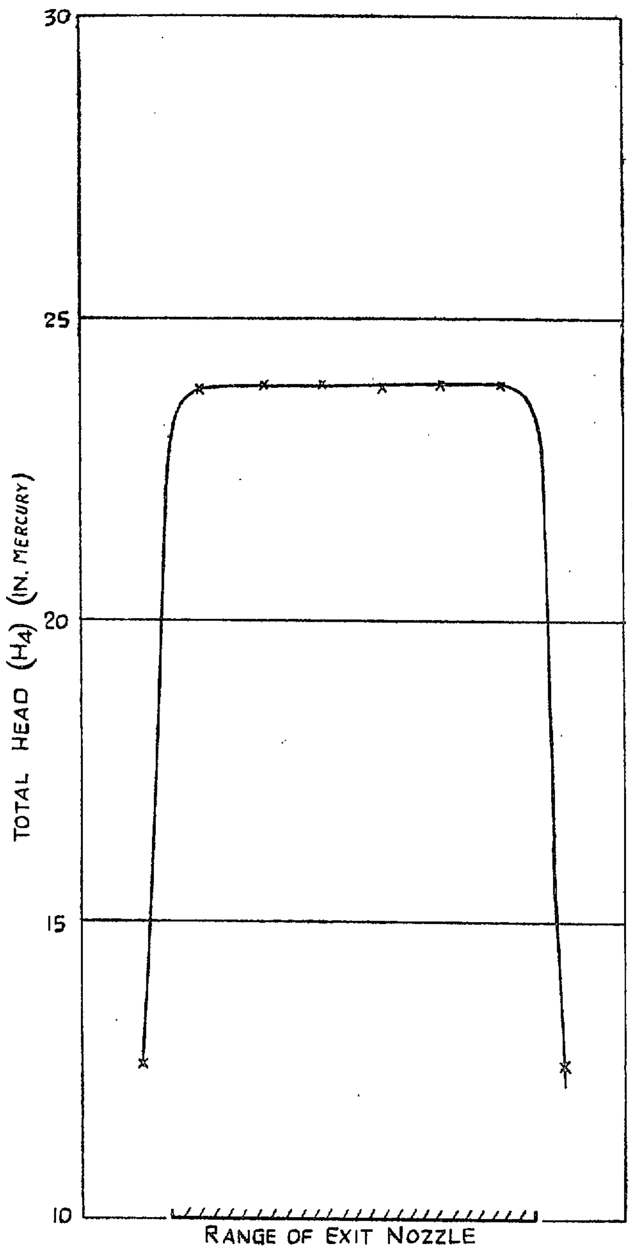

FIG. 7. Total head traverse at exit for 100/700 (hot).

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