INVESTIGATION OF A VARIABLE GEOMETRY SUPERSONIC DIFFUSER

Thesis by

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I SUMMARY

A flexible second throat supersonic diffuser has been tested in the GALCIT 2-1/2 inch supersonic wind tunnel at Mach numbers of 2.60 and 3.33. The theoretical minimum area relation for starting a supersonic tunnel has been checked at the two Mach numbers, and the amount by which the height of the second throat may be reduced after the tunnel has been started was determined. Minimum compression ratios for the starting and reduced second throat conditions have been determined and compared with those previously found here and theoretically estimated by Crocco. Effect of nozzle surface roughness on supersonic flow is noted, and Schlieren photographs representing the flow in a variety of conditions are presented. A series of Schlieren photographs shows the flow in the second throat at progressively greater amounts of area reduction.

The theoretical minimum second throat area relation was found to predict the second throat size for starting within 3 percent at both Mach numbers, being conservative by that amount in both cases. Only about 1/3 of the theoretically predicted reduction of second throat area after starting was realized at both Mach numbers before separation of the boundary layer broke down flow throughout the nozzle. Minimum compression ratios were reduced some 9 and 6 percent for Mach numbers 2.60 and 3.33 respectively. These decreases, though not as large as anticipated, nevertheless gave the lowest compression ratios ever measured in the GALCIT tunnel, ratios less than Crocco's estimate, heretofore never realized. Transition, though not occurring by means of a normal shock wave, was found to be stable a few inches downstream of the second throat.

II INTRODUCTION

A major portion of the energy losses occurring in supersonic wind tunnels is associated with the reconversion downstream of the test section of the flow from supersonic to subsonic. These diffusion losses make the compression ratio required to operate the tunnel at a given Mach number higher than would be necessary could they be minimized. Thus the amount of energy recovered in the transition from supersonic to subsonic velocities and subsequent diffusion to almost zero velocity is responsible in large part for the maximum Mach number attainable with given power. It is the purpose of this study to investigate methods of increasing the efficiency of a supersonic diffuser, that is to say, of lowering the overall compression ratio λ.

Previous studies of supersonic diffusers have been limited to those having fixed shapes, and hence these studies have indicated the efficiencies attainable with a particular fixed geometry condition. For reasons developed in section III, it appears from the one-dimensional theory of supersonic channel flow that some possible increases of pressure recovery are precluded by this fixed geometry condition. For this reason, and to check certain other theoretical concepts of one-dimensional channel flow, a variable geometry diffuser was conceived. This diffuser allows study of the flow under conditions in which the channel may be varied during operation of the supersonic tunnel.

Although the variable geometry supersonic deLaval nozzle has been in use for some time as a means for varying test section Mach number, such a variable diffuser as studied here has not been previously investigated. It is natural, therefore, that this study is intended more as a preliminary investigation than as a final report on variable second throat supersonic diffusers. A design criteria for the variable geometry diffuser is set up with the idea of determining experimentally some of its characteristics. Futher work may be based on the results found here.

III THEORY

The losses associated with a supersonic diffuser may be classified in three groups:

1. Friction losses on the walls
2. Expansion and separation losses
3. Losses occurring in the shock waves associated with transition from supersonic to subsonic flow.

The transition losses mentioned last here are of the most interest in a supersonic diffuser, since once the flow has become subsonic, friction and expansion losses can be estimated by the ordinary theory of subsonic diffusion. The remainder of the study, then, is devoted to the problem of an efficient deceleration of a supersonic flow to subsonic.

An isentropic deceleration from supersonic velocity to rest may be visualized as follows: The channel in which supersonic flow exists gradually contracts, reducing the Mach number to unity at a point known as the second throat. If a normal shock wave occurs exactly at this minimum area point and the flow becomes subsonic, an expanding channel will then permit subsonic diffusion to rest. In practice, completely isentropic deceleration involving a normal shock at the second throat cannot be realized. For should the shock be displaced either upstream or downstream from this second throat, the Mach number at which it occurs increases because of the area increase; and losses occur in the essentially non-isentropic shock wave at any Mach number other than unity. As is shown later in this section, losses upstream will immediately break down the flow, and consequently the transition shock must occur far enough downstream of the second throat that disturbances of steady flow conditions cannot propogate the shock wave upstream of the minimum area section. With the shock downstream at a Mach number greater than unity, losses occur, and the isentropic deceleration postulated above is impossible. Hence it appears that the best compromise will effect transition at as low a Mach number as possible. For example, at a Mach number of 3.5, only about 20% of the initial total head is regained behind a normal shock; while at $M = 1.5$, 93% of initial total head is recovered.

In supersonic wind tunnel applications, the establishment of supersonic flow in the test section is preceeded by a shock wave traveling downstream from the first throat. If this shock wave is to pass through the test section and on into the diffuser, the throat of the diffuser, hereafter referred to as the second throat of the tunnel, must be large enough to permit passage of the same mass flow as that through the first throat, but at the reduced total head existing behind the shock wave. This throat area is a minimum when the Mach number at the second throat is unity with the shock upstream. If the first throat, test section, and second throat are arranged as in Figure 1,

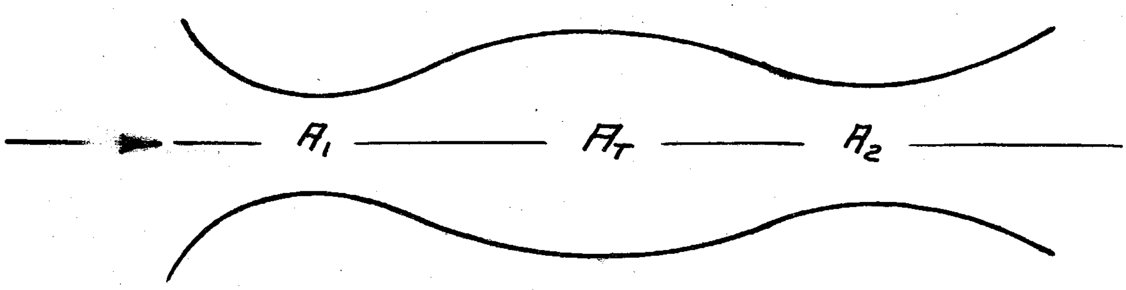

Figure 1

a contraction ratio, C_R, may be defined relating the areas of the test section and second throat:

$$C_R = \frac{A_T}{A_2}$$

This may be written,

$$C_R = \frac{A_T}{A_2} = \frac{A_T}{A_1} \cdot \left(\frac{A_1}{A_2}\right)_M$$

where $\frac{A_T}{A_1}$ is the isentropic area ratio from first throat to test section, and $\left(\frac{A_1}{A_2}\right)_M$ is the minimum area ratio required when a normal shock wave at test section Mach number is between the first and second throats. As is known from one-dimensional theory, the second throat area required is increased by the ratio of the total head behind the shock to that in front. This follows from the fact that, although the velocities for Mach number unity are the same at both throats, the density at the second throat is reduced, requiring a larger area to maintain continuity. The reduction of density is the result of decreased pressure in a constant enthalpy process.[1] We have then,

$$\left(\frac{A_1}{A_2}\right)_M = \frac{P_3}{P_0}$$

where P_3 is the total head behind the normal shock, and P_0 that before it. Using the isentropic area ratio relation, the expression for C_R becomes,

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[1] See for example: Liepmann and Puckett, "Introduction to Aerodynamics of a Compressible Fluid."

$$C_R = \frac{M_T\left(1 + \frac{\gamma-1}{2} M_T^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}}{\left(1 + \frac{\gamma-1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}} \cdot \frac{P_3}{P_0}$$

This relation is plotted in Figure 7 as a function of test section Mach number, utilizing the fact that $\frac{P_3}{P_0}$ is determined explicitly by the test section Mach number at which the normal shock is assumed to occur.

It has been found, however, that a position of the normal shock wave upstream or at the throat of the diffuser is unstable and unattainable in practice;[2] and it is observed that the shock formed on starting a supersonic tunnel with a diffuser throat of sufficient size passes on through and takes up a position downstream. The flow beyond the second throat then remains supersonic, and has increasing Mach number because of the increasing area; hence the further the shock occurs downstream, the higher will be its Mach number. As previously noted, this is to be associated with increased losses.

The reconversion from supersonic to subsonic flow may not in practice occur by means of a normal shock wave, but may take the form of a series of oblique transition waves. In fact it has been noted[3] that this oblique family transition is the common means employed by the flow for reducing its speed from Mach numbers in excess of 1.5. But in general the losses associated with the transition will bear a relation to the Mach number at which they occur, and the position of this reconversion region will be determined by the overall compression ratio, λ, from supply tank, P_0, to stagnation downstream

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[2] Kantrowitz and Donaldson, "Preliminary Investigation of Supersonic Diffusers"

[3] Puckett and Schamberg, "Final Report: GALCIT Supersonic Wind Tunnel Projectile Tests," and Puckett, "Final Report on Model Supersonic Wind Tunnel Project,"

of the diffuser, P_D.

One-dimensional flow theory and the flows observed in practice thus indicated that a diffuser employing a variable-geometry second throat might allow the establishment of flow patters giving higher energy recoveries than possible with the fixed throat. After the initial shock waves formed on starting the tunnel had passed through, a flexible wall would allow the second throat to be reduced to such an area that the local Mach number was unity. If the transition to subsonic flow occurred closely downstream of this reduced second throat, it would occur where the channel had considerably smaller area than in the case of a fixed wall. Hence if a position of the transition just downstream of the second throat is stable, losses should be smaller in accordance with the reduction of Mach number.

Transition to subsonic flow is followed by ordinary subsonic diffusion to rest at the compressor inlets. And it has been previously noted that this subsonic diffusion is not of interest here.

From the previous development of theory, the objectives of the study may be outlined as follows:

1. To check the theoretically computed minimum second throat area for starting,
2. To determine the amount by which the second throat area may be reduced (choked) from this starting value once the tunnel is in operation,
3. To compare the minimum compression ratios, λ_{min}, attainable with the unchoked and the choked second throat, and
4. To investigate the stability of the transition zone in the diffuser.

IV DESCRIPTION OF APPARATUS

The flexible second throat apparatus to be described was designed for test in the GALCIT 2-1/2 inch square test section with 30 inches of channel length visible through glass side plates. The details of the piping, compressors, and drying apparatus of the tunnel are described in other references.[4] The Schlieren apparatus of the 2-1/2 inch tunnel[5] was used to take the flow photographs appearing later in the study. Figure 2 is a photograph showing the general layout of the test section, optical bench, and second throat apparatus.

In general the design was carried out with the objectives of simplicity and of making the flow visible by means of the Schlieren system. To this end the converging-diverging nozzle, as well as the short test section, second throat, and a portion of the subsonic diffuser were placed within the 30 inches of parallel glass side walls. An existing set of rugged base blocks was utilized to mount the nozzle and flexible throat apparatus between these glass plates. These bases were ones previously used for study of interchangeable channel shapes, and hence were easily adaptable to the present project.

A set of wooden nozzle blocks fitting the upstream portion of these bases and designed for a Mach number of 2.68 was in existence and was used. These blocks gave a test section 2.075 inches high and 2.562 inches wide at the diffuser inlet, while the test section itself was approximately four inches long. Another pair of nozzle blocks was designed for a Mach number of 3.47

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[4] Supra p.7, Puckett, "Final Report: GALCIT Supersonic Wind Tunnel,etc."

[5] Ibid.

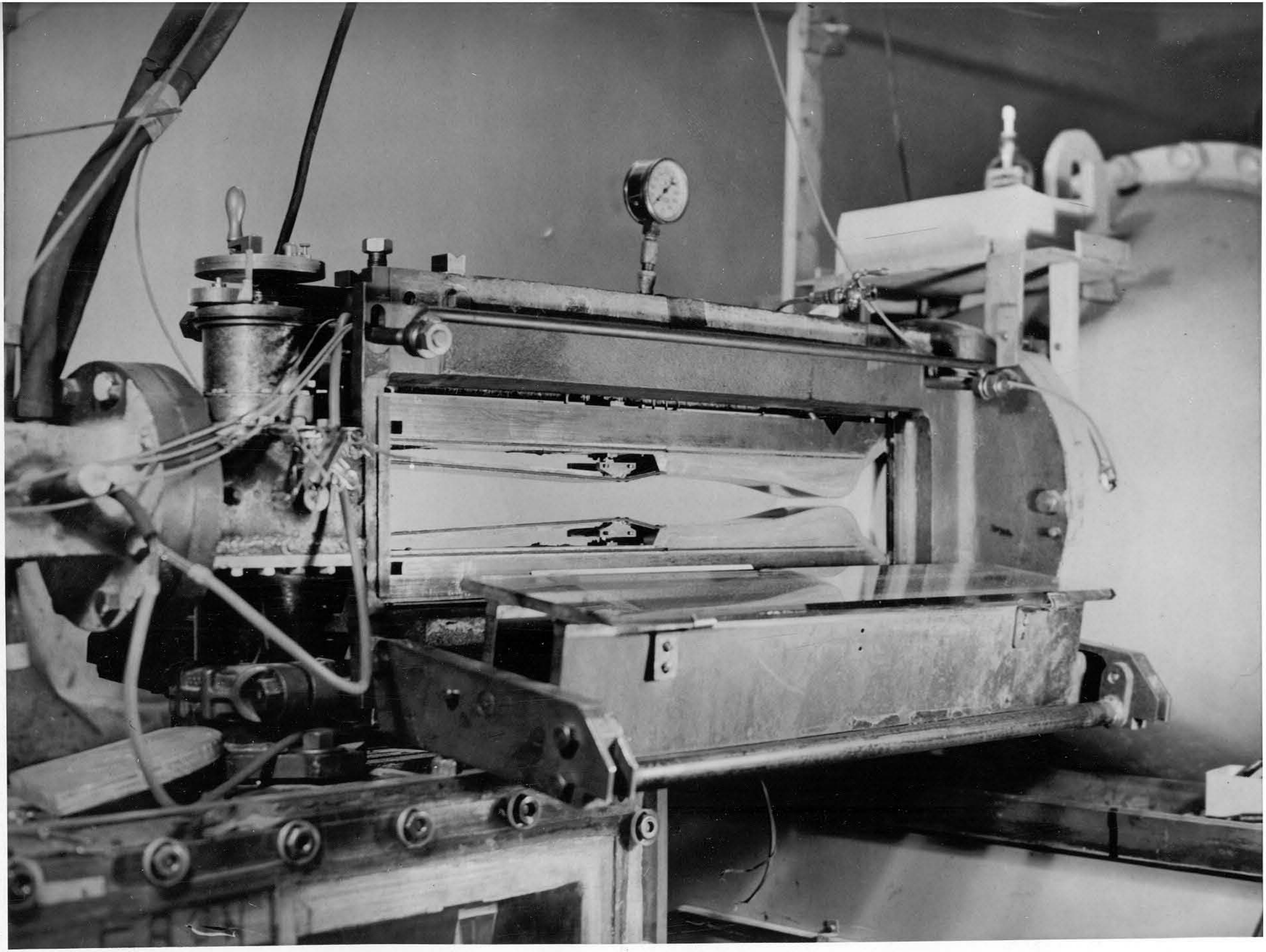

Figure 2 - General Arrangement

employing the same basic test section dimensions. Thus similarly designed nozzles for the two Mach numbers 2.68 and 3.47 were interchangeable upstream of the diffuser. Figure 3 shows a side view of the working section with the 3.47 nozzles in place ahead of the flexible throat.

The $M = 3.47$ nozzle blocks were carefully designed by the characteristics method[6] and constructed to close tolerances. The flow experienced in both nozzles was smoothly supersonic in the test section as the Schlieren photographs testify; and pressure measurements later indicated that the actual Mach numbers in these nozzles were 2.60 and 3.33. Hence both nozzles gave Mach numbers about 4% below the design values.

At the outset the mechanical design of the second throat presented problems affecting the remainder of the study. It was decided in the interest of simplicity and facility of operation to use a channel wall consisting of a number of hinged flat plates instead of a continuously curved one. This angular wall would then give rise to oblique shock waves; and these were the basis for the design.

The shock wave pattern set up in the flow by angular walls was assumed to be that of Figure 4 . If the flow deflection angle, θ , and length, **L**, are chosen correctly for the particular test section Mach number involved, the wave pattern would just be cancelled at the second throat, and the Mach number, M_2^* , will be unity. The flow solution is obtained from the shock polar diagram.[7] This solution was made for the two Mach numbers 2.68 and 3.47,

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[6] Puckett, "Supersonic Nozzle Design."

[7] Op. Cit. p.6

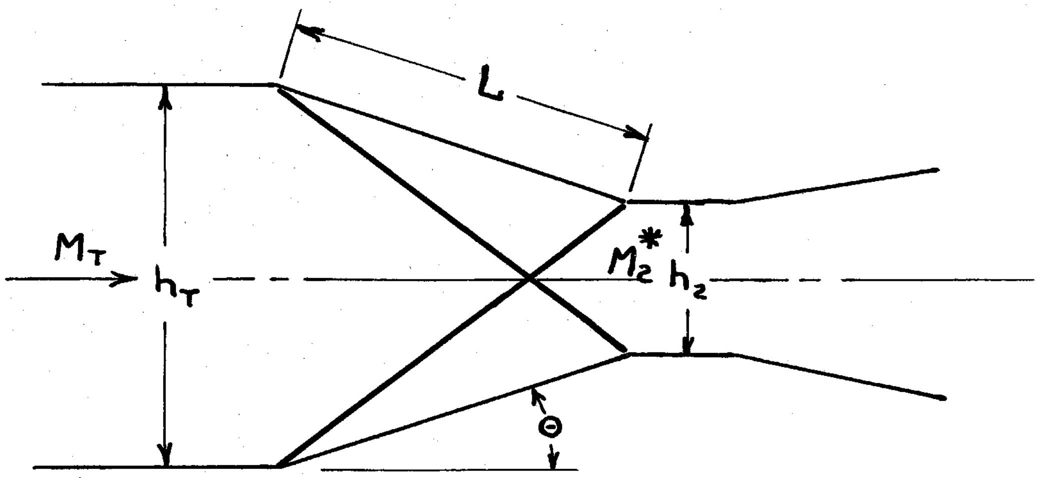


Figure 4

and the results are given in Table 1. It was apparent that a design incorp-

M_T	θ Degrees	L Inches
2.68	19.3	1.774
3.33	23.7	1.759

Table 1

orating a compromise length, **L**, and the largest deflection angle, θ, encountered would suffice for both Mach numbers. Hence this fact was used as a simplifying element in the design.

Once the size and angular range of the flexible mechanism was determined, the remaining problems were those of ease of operation, smooth channel walls, and adequate pressure sealing of the edges of the moving plates. Previous work had indicated the importance of preventing edge leakage.

The channel wall plates were made of 1/8 inch spring temper phosphor bronze. They were stiffened to give rigidity and a groove milled the length of the plates on both edges to admit a 1/8 inch diameter rubber sealing tube. In operation, this tube was flattened against the glass side walls, but slipped across them easily. The desired wall shape was obtained by milling three 1/4 inch wide slots in the back of each plate, leaving 0.005 inch flexures at the hinge points. These flexures assured ample angular freedom and a smooth channel surface. Toggle linkage from the drive mechanism controlled the motion of the individual plate segments completely so that no major loads could come onto the delicate flexures. Figure 5 is an assembly drawing of the apparatus.

The drive and position-indicating device used was a pair of standard Starett micrometer heads. These heads were so calibrated that each read half the second throat height, and their readings were made simultaneously visible to a single operator by means of mirrors. Thus the second throat height could be easily determined or adjusted during operation of the tunnel. Figure 6 shows the top micrometer in position in the tunnel.

The second throat length was 1/2 inch, and the channel plates then provided a straight-walled subsonic diffuser to the point where the channel joined the downstream tunnel structure. Because of the geometry of this structure, the total diffusion angle in all cases was less than 8 degrees.

Static pressure orifices were located in the channel as follows:

1. One at upstream stagnation, P_0, to indicate the initial total head,
2. Three in the test section at intervals of 3/4 of an inch. They were designated P_1, P_2, and P_3 progressing downstream,

and indicated test section pressure, and

3. One downstream at diffuser stagnation.

The upstream stagnation pressure, P_0, was read from a Bourdon gauge, while the diffuser pressure, P_D, was read on a U-tube mercury manometer. The test section pressures were read on a large manometer panel in conjunction with a reference pressure. A mercury barometer provided atmospheric pressure indications for each test.

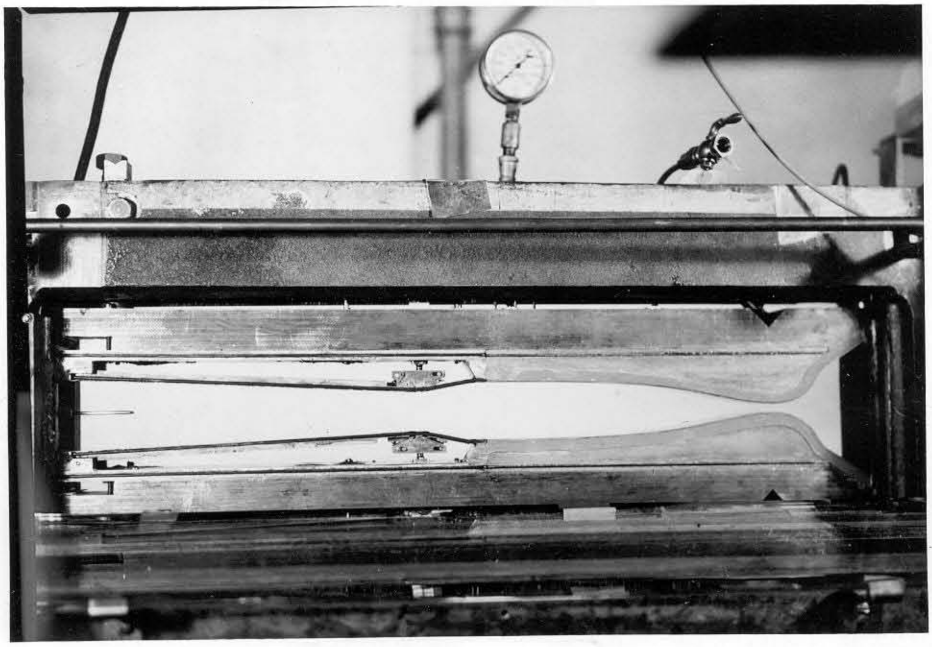


Figure 3 - Side View of Working Section

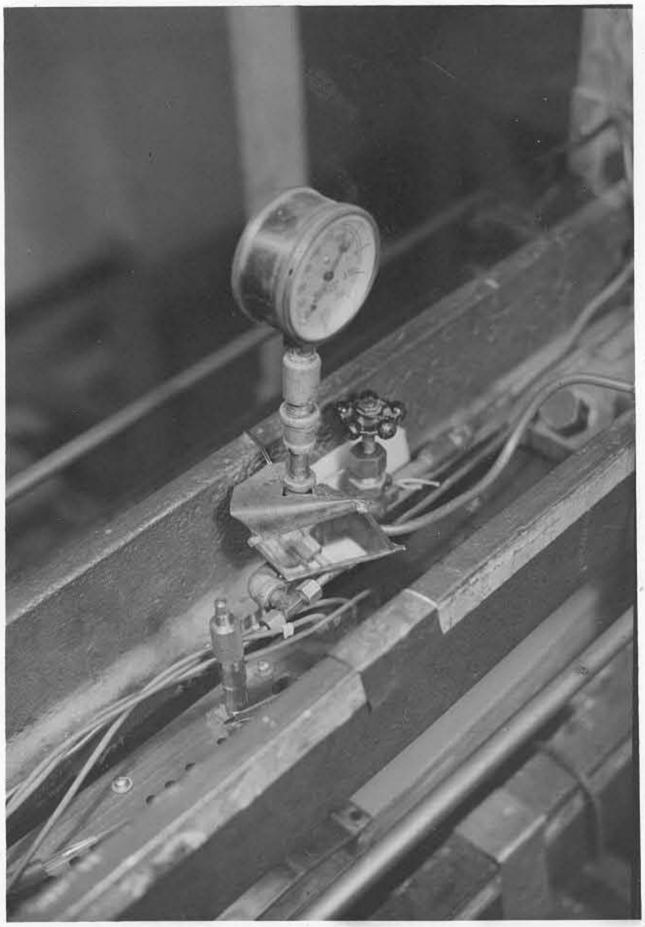
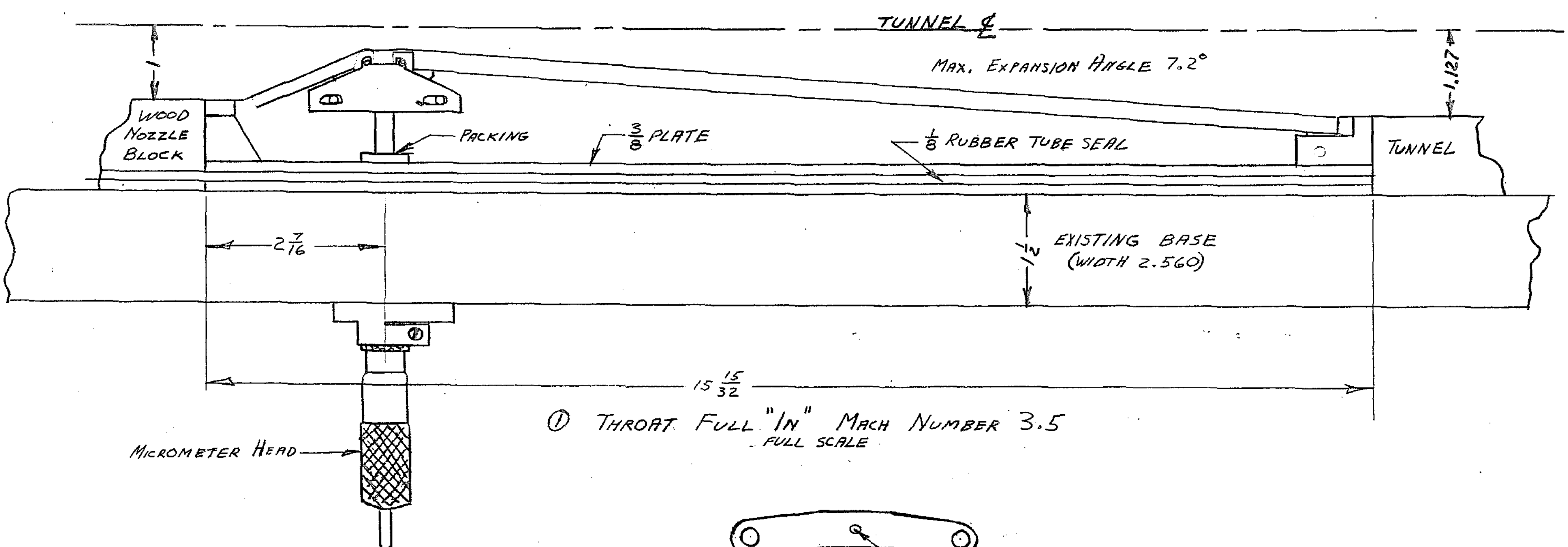
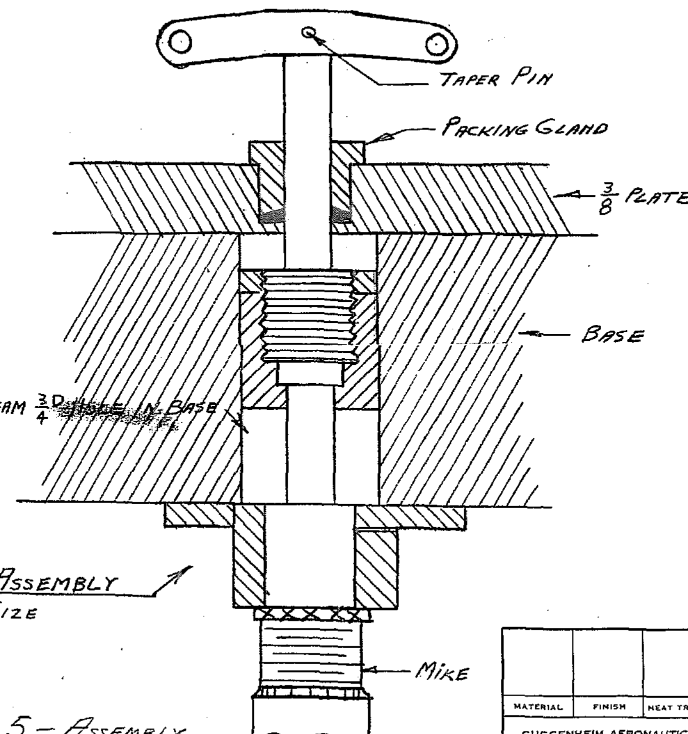


Figure 6 - Top Adjusting Micrometer



① THROAT FULL "IN" MACH NUMBER 3.5
FULL SCALE



② PUSH ROD ASSEMBLY
DOUBLE SIZE

FIGURE 5 – ASSEMBLY

			RRH			RRH	TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED	SCALE As Noted
MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	REF.	
GUGGENHEIM AERONAUTICAL LABORATORY CALIFORNIA INSTITUTE OF TECHNOLOGY			FLEXIBLE THROAT ASSEMBLIES				4-258-297	
			NAME				DRAWING NO.	

V PROCEDURE

The flexible second throat apparatus and wooden nozzle blocks were installed in the tunnel, care being taken to center the independent pieces of apparatus. Continuous rubber pressure seals were placed along the sides of the channel, and no pressure leaks were experienced during operation with the glass plates bolted securely up on each side.

The tunnel was put into operation for all tests by a fixed operating procedure. First the tunnel was opened to the atmosphere through the dryer, and the compressors started. Pressure in the large upstream stagnation tank was allowed to rise to 35 psig. before the tunnel was closed to the atmosphere and circulation through the closed circuit started.

The procedure for checking the value of the minimum second throat area for starting was simply to start the tunnel by the above procedure with various settings of the micrometers. The second throat height was gradually decreased until the transition shock system remained just downstream of the first throat. Under these conditions, smooth supersonic flow was never established in the test section. The last value of second throat height for which supersonic flow was obtained was considered to be the minimum size for starting.

Once the minimum starting area had been determined for each Mach number, Schlieren photographs of the length of the stream were taken to record this initial flow. Since the tunnel had excess power for the two Mach numbers tested and the channel area employed, a pressure difference in excess of that necessary to maintain the flow existed across the test section and diffuser initially. Under this condition of excess compression ratio, the transition shock system was well downstream of the visible section of the diffuser. This compression ratio was determined from P_0 and P_D as:

$$\lambda = \frac{P_0}{P_D}$$

Static pressures P_1, P_2, and P_3, in the test section were then read and the test section Mach number obtained from the ratio of P to P_0 by the relation,

$$\frac{P_i}{P_0} = \left(1 + \frac{\gamma - 1}{2} M_T^2\right)^{-\frac{\gamma}{\gamma-1}}$$

on the assumption of isentropic flow from the supply tank to test section. In all tests the compression ratio and Mach number were obtained in exactly this fashion.

When flow had been satisfactorily established through the minimum second throat, the area of this throat was reduced by adjusting the two micrometers. Readings were made identical top and bottom every .050 inches decrease of second throat height. Thus the channel was exactly symmetrical at stations every .050 inches, and only departed from symmetry at any time by the amount the two heads were not turned in unison by the operator. This was never more than .005 inches. Schlieren photographs of the second throat at each of these stations were taken, and the second throat height continuously reduced until the flow broke down. The breakdown is characterized by separation just after the first throat and highly turbulent flow throughout the channel. The second throat height for which breakdown occurred could be read at any time, since the micrometer readings were in no way affected by the disturbance in the flow accompanying failure of the flow.

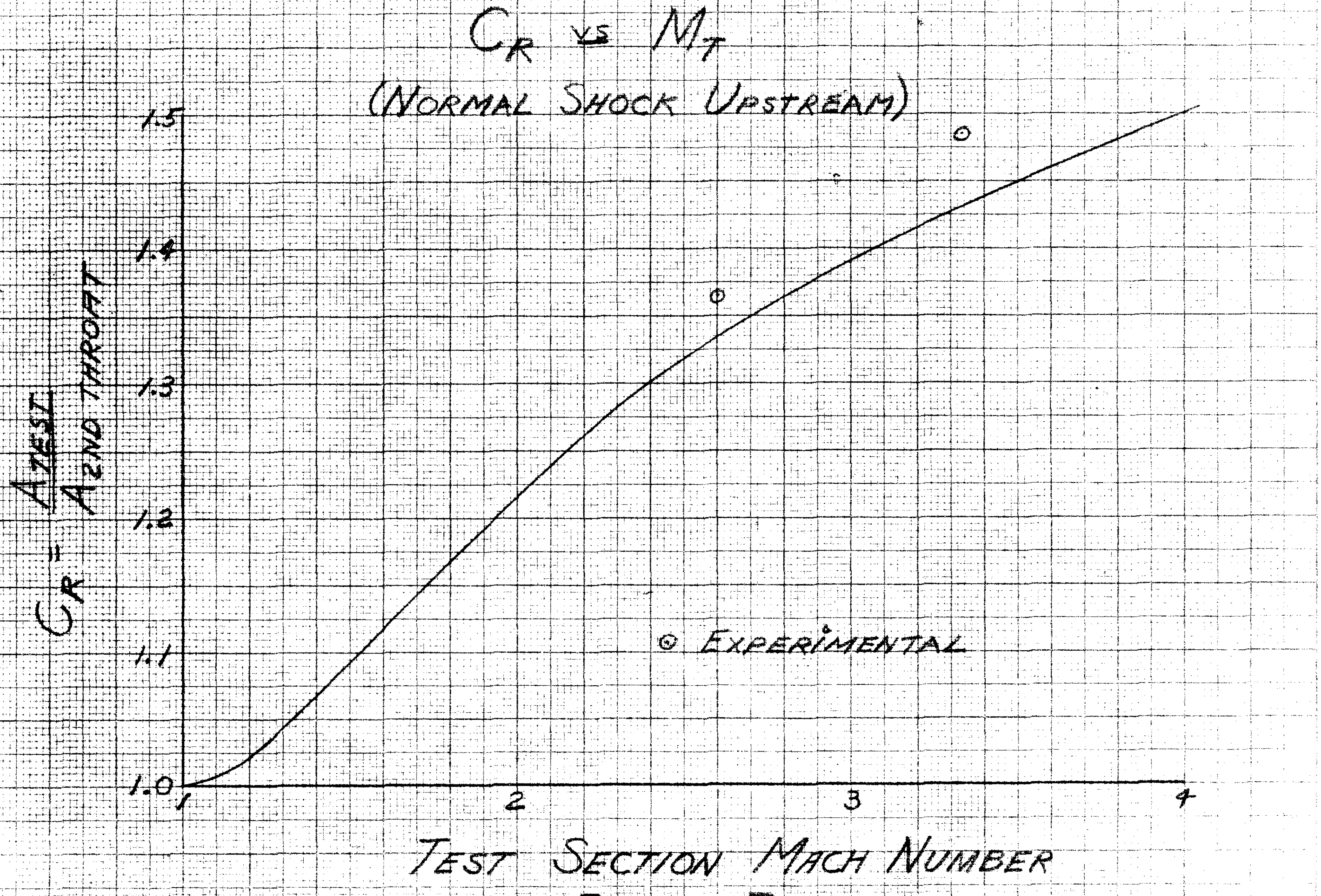
The minimum values of the compression ratio for which flow in any configuration was sustained were determined by gradually reducing the λ until the transition shock system moved ahead of the second throat and separation took place. This was accomplished by varying the amount of compressor delivery air by-passed around the test section in a series of discreet intervals in order to assure that equilibrium conditions existed at all times. The values of P_0 and P_D were taken at the breakdown, and λ_{min} computed from them. Schlieren photographs of the flow were taken with λ just in excess of λ_{min}.

Normally the second throat area was reduced before λ_{min} was measured, but some runs were also made in which λ_{min} for the starting condition was set up and the second throat then reduced. Also the values of λ_{min} and the critical second throat settings were checked a number of times to determine their repeatibility.

VI <u>RESULTS</u>

1. <u>Theoretical minimum starting area</u>: The values of the experimental second throat height for starting obtained as outlined in section V agree very closely with those predicted by the theory. Contraction ratios, C_R, obtained from Figure 7 and applied to the actual test section height give theoretically predicted second throats of 1.553 and 1.450 inches for the two Mach numbers 2.60 and 3.33. The experimentally determined minimums were 1.520 and 1.400 inches respectively. Thus at Mach number 2.6 flow was successfully established through a second throat area 2.1% less than predicted; while at $M = 3.33$, it was established with an area 3.4% less than predicted. The theoretical contraction ratio curve and the experimental points are plotted in Figure 7. As noted in section V, these values were checked and found to be repeatable at various times within one percent. It appears, therefore, that the theoretical considerations of minimum second throat area for starting are valid, and that design on this basis should give satisfactory results in practice.

2. <u>Reduction of second throat area during operation</u>: Reduction of second throat area during operation was carried out as indicated in Section V. Schlieren photographs of the second throat taken at intervals during this choking process appear for the two Mach numbers in Figures 8 and 9. Each is a series of six views of the second throat at progressively greater amounts of choking. Table 2 presents the second throat heights to which choking would have to have been carried in order to realize the design



C_R vs M_T
(NORMAL SHOCK UPSTREAM)
$C_R = \frac{A_{TEST}}{A_{2ND\ THROAT}}$
1.0
1.1
1.2
1.3
1.4
1.5
1
2
3
4
EXPERIMENTAL
TEST SECTION MACH NUMBER

FIGURE 7

condition, and compares them with the minimum heights obtained experimentally. The design condition made no allowance for boundary layer. It is evident from this table that the design choking was not achieved; in fact, in both cases about one third of the theoretical choking* was possible before the flow broke down. Therefore the design condition of the oblique shock waves

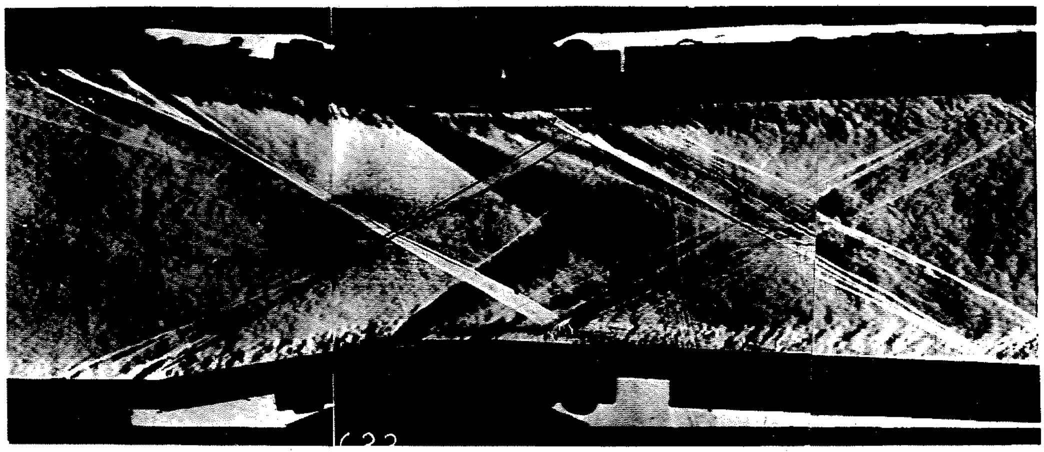
M_T	Theor. Starting 2nd Throat	Design 2nd Throat	Experimental 2nd Throat	Design Choking*	Experimental Choking	% of Design Choking
2.60	1.553"	.828"	1.290"	.725"	.263"	36
3.33	1.450"	.588"	1.132"	.862"	.318"	37

Table 2

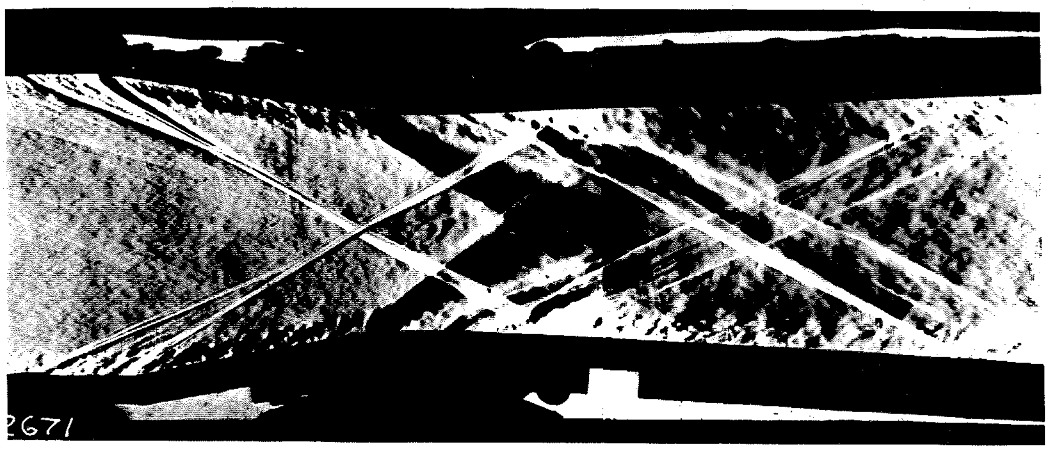
cancelling out at the second throat entrance was never realized, although the approach to this condition is evident in the Schlieren photographs, Figures 8 and 9. The steepening of the oblique shock waves and extension of the expansion zones in the second throat are seen immediately. The flow separated in the test section for choking just in excess of that shown in the last photograph of each series. Thus the process of steepening the channel walls to the design value could not be completed before breakdown of test section flow occurred.

It appeared on first consideration that the rather thick boundary layer, clearly visible in the Schlieren photographs, was responsible for an effective decrease of channel area at the second throat, such that the combination of

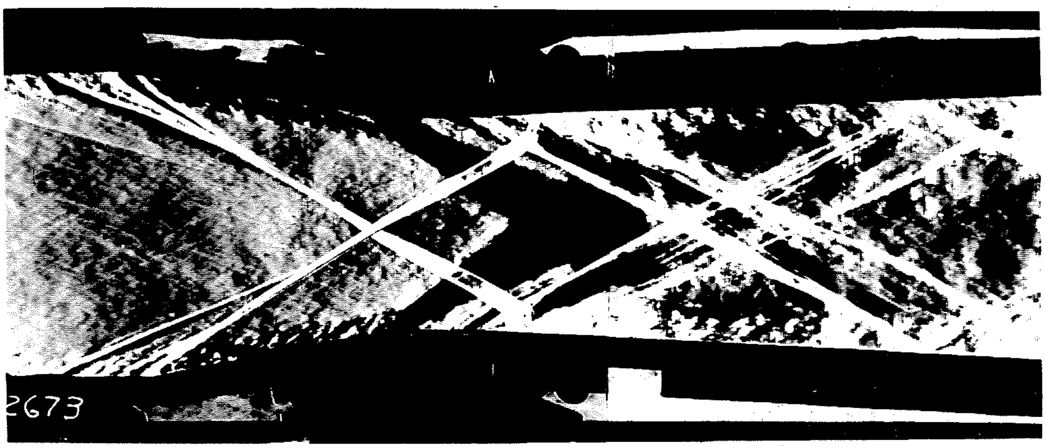
* Choking is defined as reduction of second throat height.



Second Throat Height 1.550

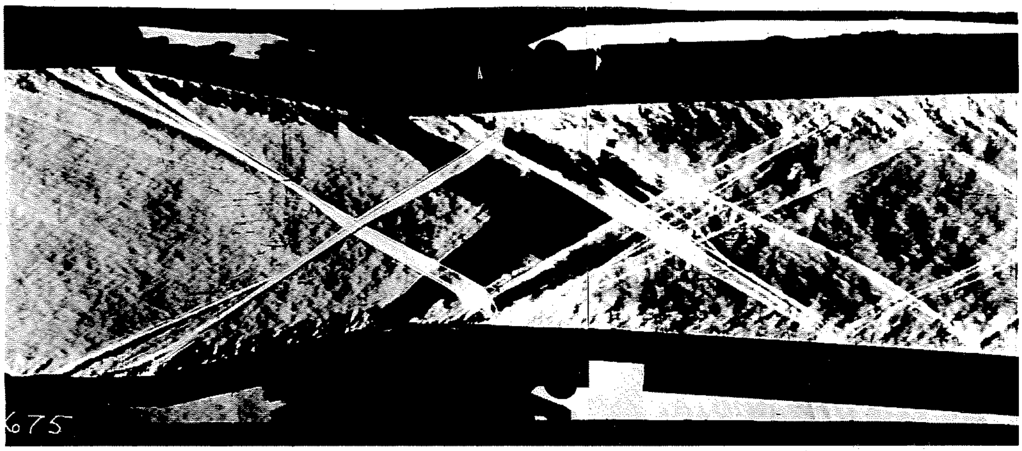


Second Throat Height 1.500

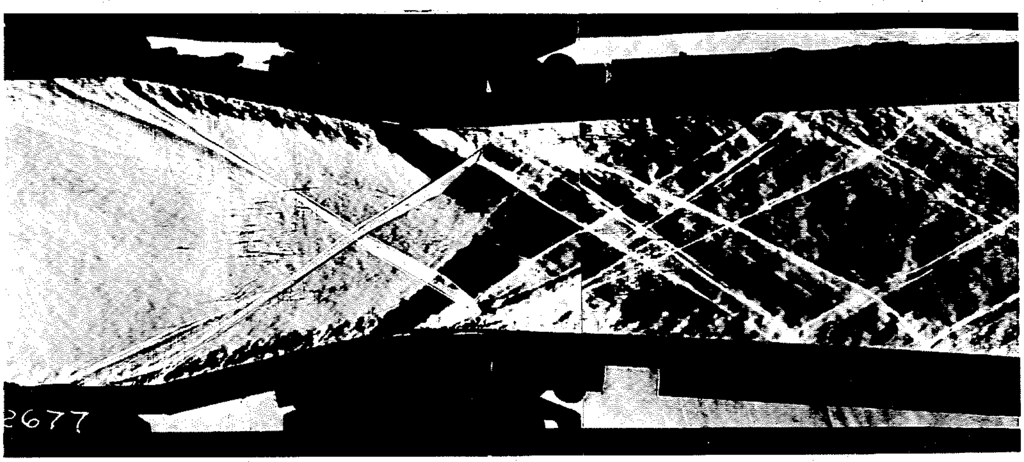


Second Throat Height 1.450

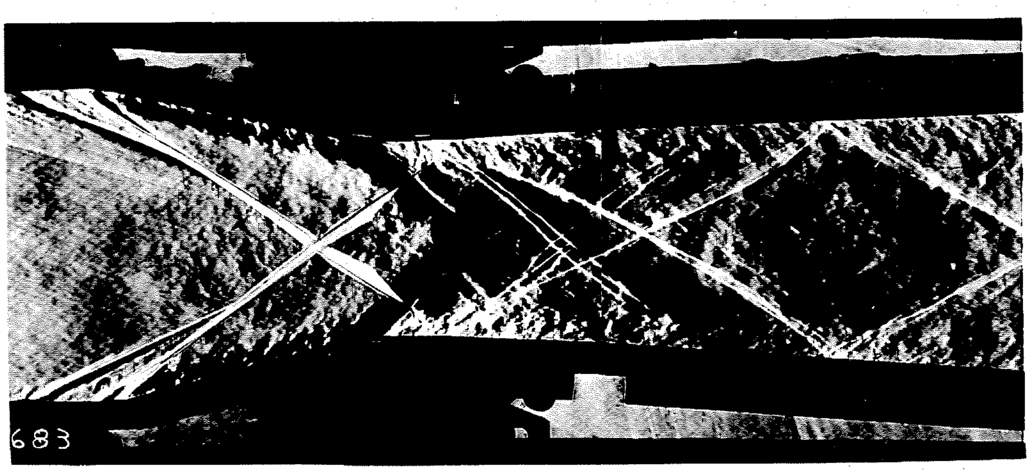
Figure 8-a $M = 2.60$, $\lambda = 3.41$



Second Throat Height 1.400

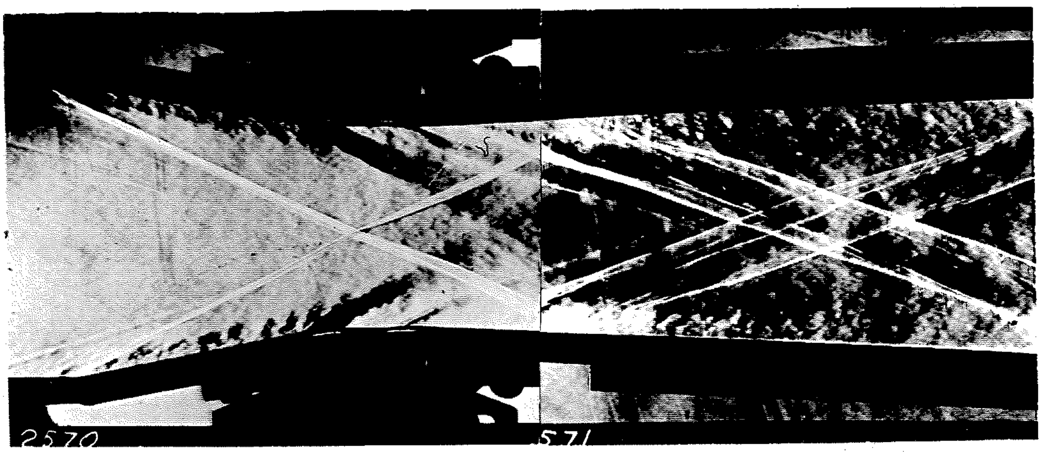


Second Throat Height 1.350

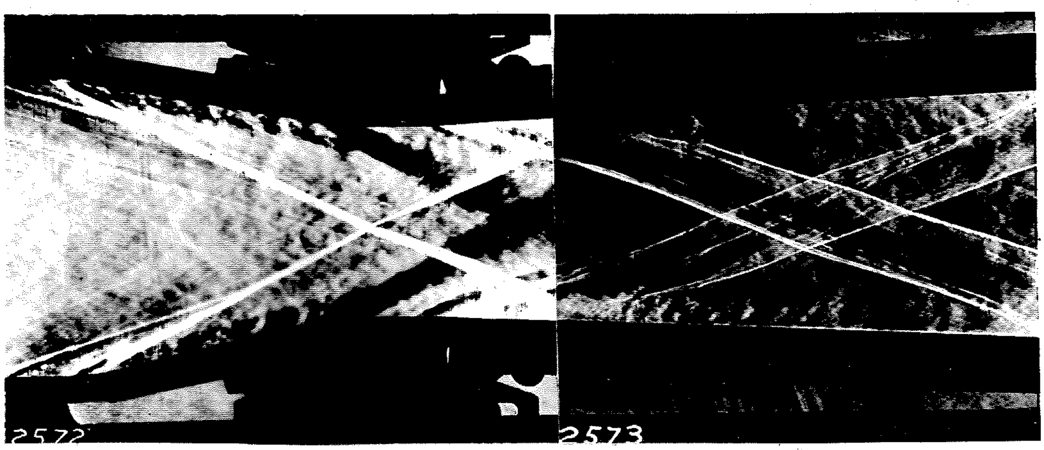


Second Throat Height 1.300
Flow Broke at 1.290

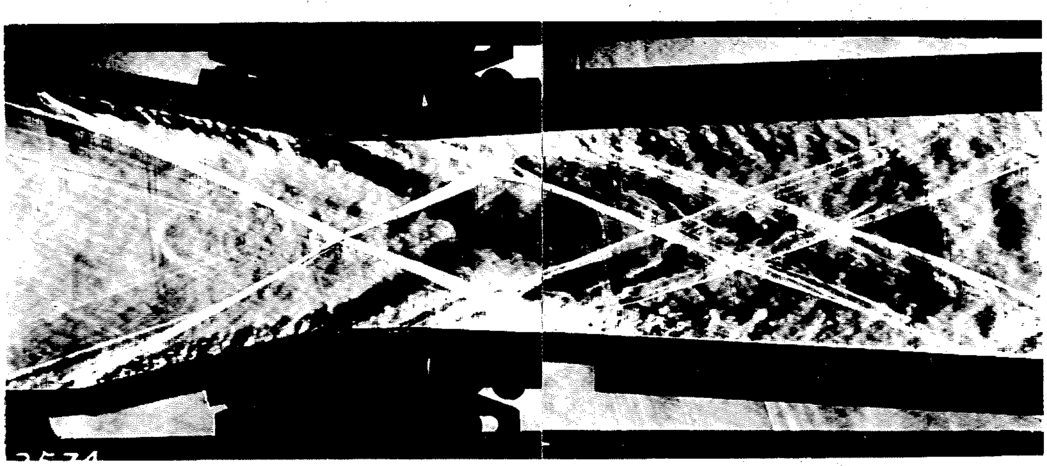
Figure 8-b



Second Throat Height 1.410

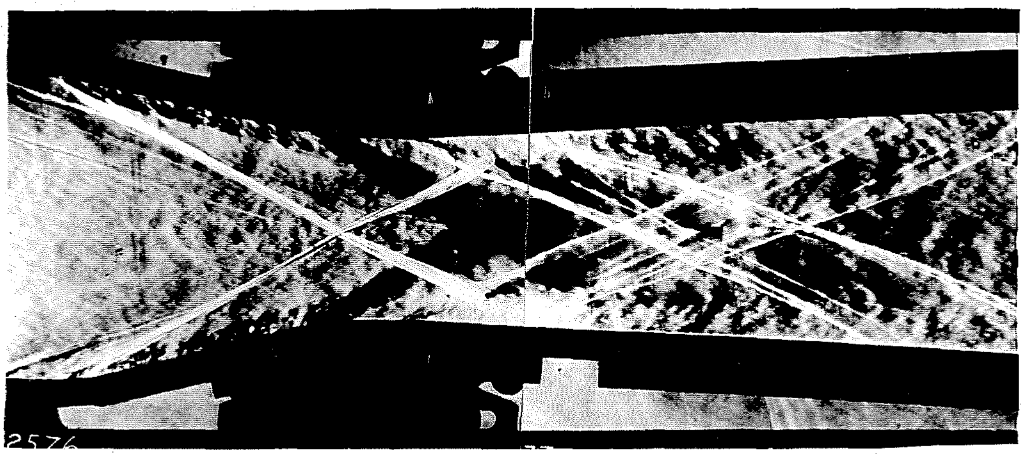


Second Throat Height 1.350

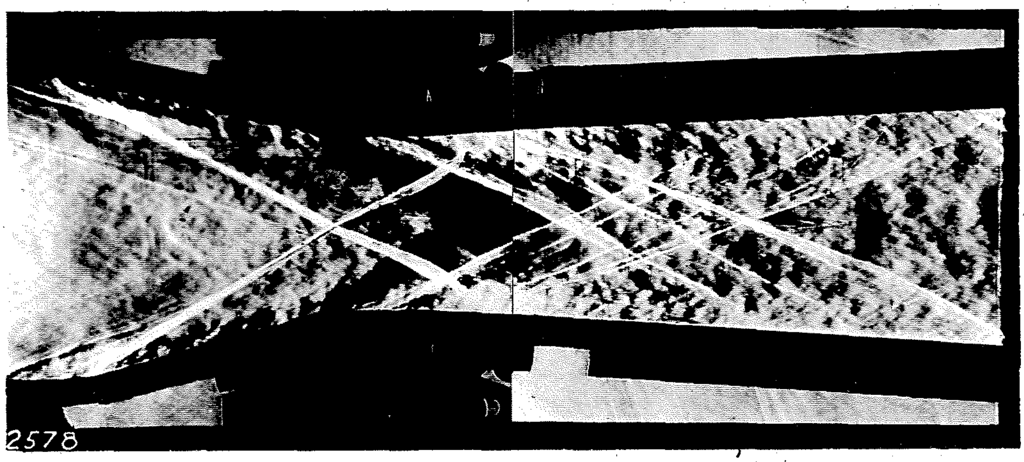


Second Throat Height 1.300

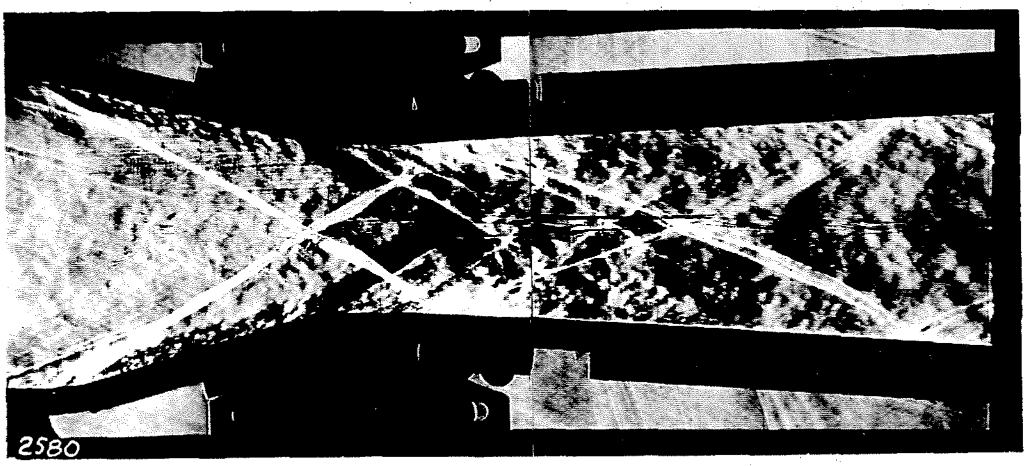
Figure 9-a M = 3.33, λ = 7.18



Second Throat Height 1.250



Second Throat Height 1.200



Second Throat Height 1.150
Flow Broke at 1.132

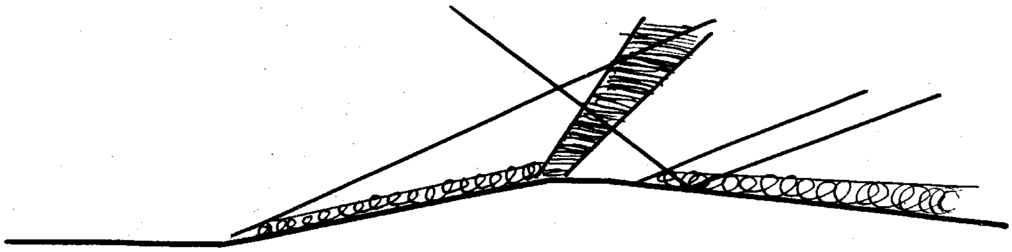
Figure 9-b

choking and boundary layer displacement thickness reduced the second throat area to a value smaller than that necessary for the mass flow. This would have been responsible for a simple choking off of the flow. To investigate this possibility, measurements of the boundary layer thinkness as appearing in the Schlieren photographs were taken, and liberal allowances for displacement thickness were made for all four walls of the tunnel. In addition, the entropy gain in the oblique shock wave system was computed, and allowance made for the increased area necessary because of the reduction of total head behind these waves. The area required for the mass flow after all of these computations was still only about one half that existing when the flow broke down experimentally. Hence it was felt that explanation of the flow failure as a result of choking due to simple boundary layer thickness and entropy gain was not sufficient.

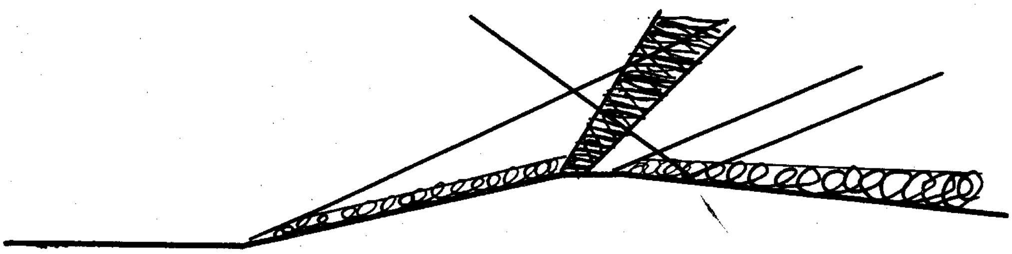
The flow breakdown due to choking was then observed carefully through the Schlieren apparatus several times, and the true nature of the failure became apparent. A pair of shock waves arising from either wall just beyond the second throat, but just upstream of the point where the initial oblique waves strike the opposite wall and reflect, may be seen in the first photographs of each series. Tracing this wave through the series of photographs, it is seen to move upstream until in the last picture of each series it is at the downstream edge of the expansion zone at the second throat entrance. When viewed in the Schlieren apparatus, further choking is seen to move this wave further upstream cancelling out an increasing portion of the expansion zone. This results in an incomplete turning of the flow at this corner and hence an effective separation which then reduces the effective area of the second throat and chokes off the flow. The series of events described here and depicted in the Schlieren photographs is diagrammed in Figure 10 for clarity. Such characteristics of the series of the events as are not clear from the photographs appear in this figure. The figure shows the shock wave and boundary layer separation moving upstream, decreasing the breadth of the expansion zone, and finally resulting in separation in the second throat. Once the shock wave pair had begun to enter the expansion zone, the second throat could be reduced only a few more thousandths of an inch. The expansion zone began to oscillate up and down stream, being very unstable, until finally the breakdown occurred. It was because of this instability and suddenness of the breakdown that no Schlieren photographs of the breakdown condition could be obtained.

This shock wave pair, which finally is responsible for breaking down the flow, is believed to be induced by the boundary layer. The thick boundary layer, so visible on the contracting sides of the channel, is greatly reduced by passage through the expansion zone as can be seen in the photographs. The boundary layer can also be seen to be quite thick again a short distance downstream. The point at which the boundary layer rethickens gives rise to the pair of shock waves in question. Now as the second throat narrows, the shock waves move upstream. Finally shock wave, boundary layer, and expansion zone interfere resulting in the separation and the effective choking off previously noted.

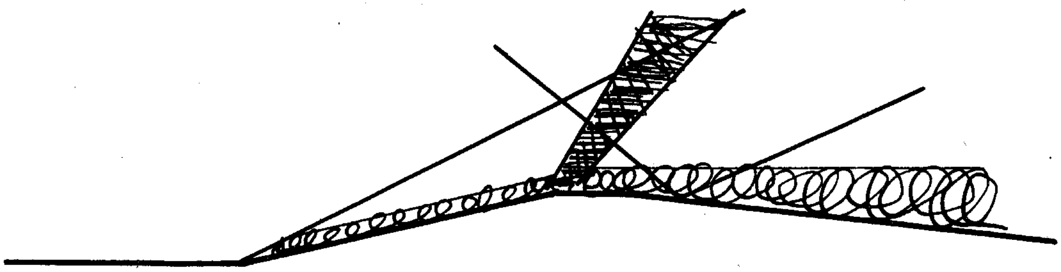
Thus it is certain that complications, not directly associated with boundary layer thickness, but nevertheless arising as a result of the boundary layer, preclude choking the second throat of a supersonic wind tunnel to a



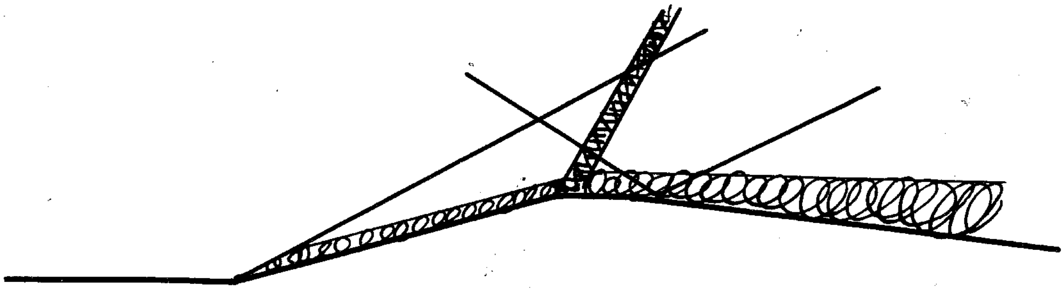
(a)



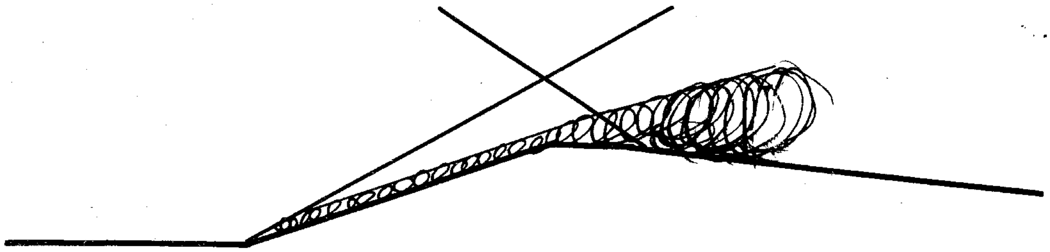
(b)



(c)



(d)



(e)

Figure 10

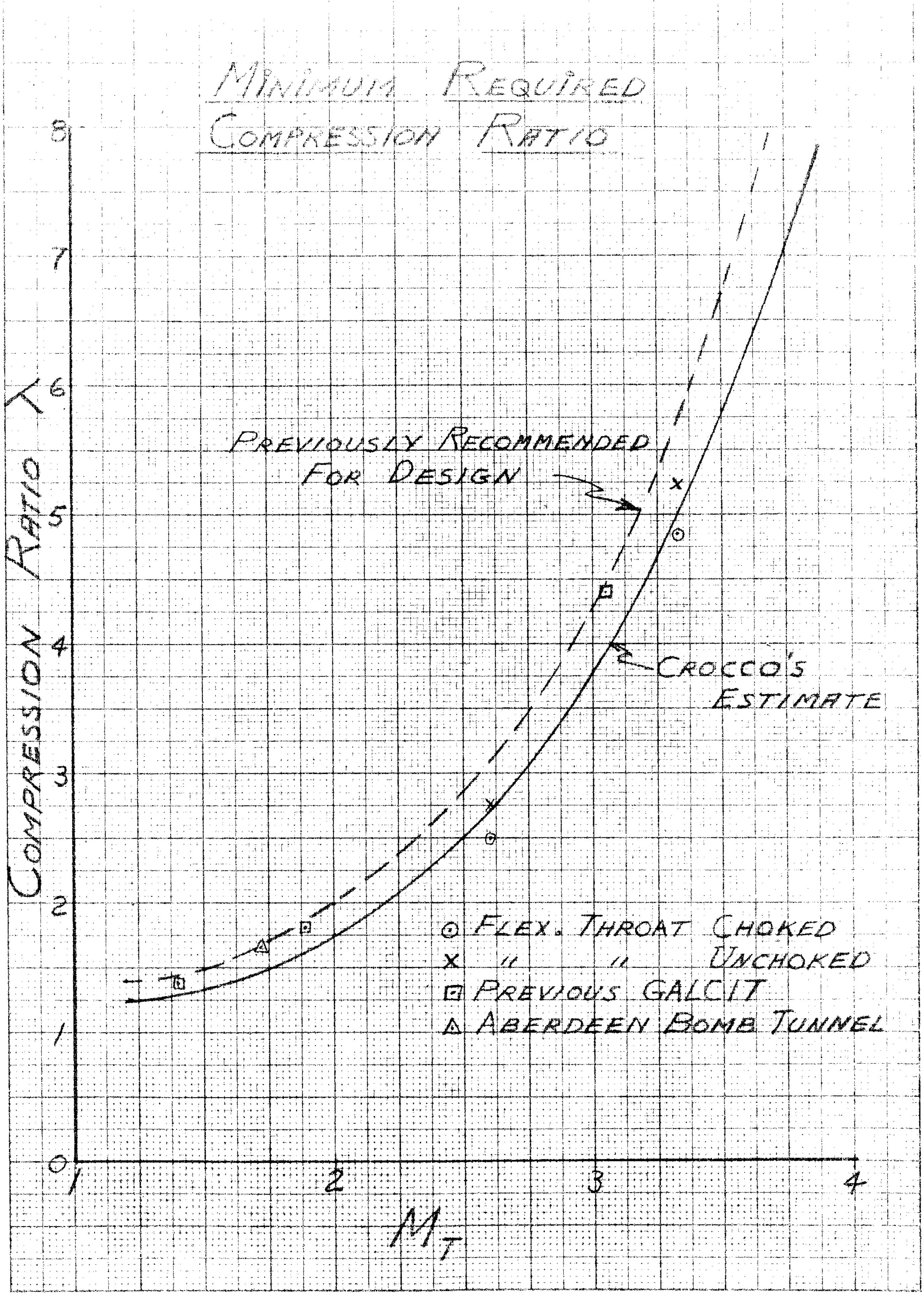
value at all approaching that theoretically possible. As expected, some energy gain is obtained as a result of choking; but the important benefits to be derived by reducing the second throat Mach number to unity are not experienced.

3. Minimum compression ratios: Following the procedure of section V, the minimum compression ratios at both Mach numbers were determined for the conditions: a) second throat at theoretical starting minimum, and b) second throat choked the optimum amount. An optimum amount of choking, somewhat less than the maximum attainable, was found to give the lowest overall compression ratios. In both cases, the λ_{min} in the choked condition was determined for a choking 0.050 inches less than that causing breakdown of the flow.

The conditions outlined in (a) and (b) above correspond to the cases of minimum compression ratio for operating 1) the fixed second throat tunnel, and 2) the variable second throat tunnel of this configuration. A summary of the values of λ_{min} obtained appears in Table 3. These minimum compression ratio values are compared in Figure 11 to those predicted by

M_T	Min Fixed	Min Choked	% Decrease
2.60	2.75	2.50	9.1
3.33	5.24	4.85	7.5

Table 3



MINIMUM REQUIRED
COMPRESSION RATIO
COMPRESSION RATIO λ
PREVIOUSLY RECOMMENDED
FOR DESIGN
CROCCO'S
ESTIMATE
⊙ FLEX. THROAT CHOKED
X " " UNCHOKED
⊡ PREVIOUS GALCIT
△ ABERDEEN BOMB TUNNEL
0
1
2
3
4
5
6
7
8
M_T

FIGURE 11

Crocco[8] and those recommended for design on the basis of previous experience in the 2-1/2 inch GALCIT tunnel. It appears that with the optimum choking, Crocco's estimate may be realized and bettered slightly by means of the flexible second throat. These compression ratios are lower than any previously measured in the GALCIT tunnel. The fact that the percentage decrease in λ_{min} with choking for the low Mach number is greater than that for the high M is attributed to the instability of the transition region at $M = 2.60$. Here it was not possible to maintain the transition as closely downstream of the second throat before choking as afterward, and hence the decrease of λ_{min} reflects not only the gain due to choking, but also the gain from increased stability of the transition region. A brief discussion of the stability of transition appears in a following part.

Figures 12 and 15 present Schlieren photographs of the length of the nozzle and flexible diffuser at the two Mach numbers, 2.60 and 3.33, and with excess compression ratio. Observe here that with excess compression ratio the flow remains supersonic throughout the diffuser. Transition occurred in the downstream tunnel section. These photographs indicate the smooth supersonic test section flow, and the general nature of the wave pattern at the diffuser entrance. Several determinations of the oblique shock wave angles indicated that they were, within the accuracy of measurement, the angles required by one-dimensional theory to turn the flow parallel to the sloping walls.

Schlieren photographs of the flow at minimum compression ratio for the unchoked and choked conditions appear in Figures 13, 14, 16 and 17 for the

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[8] Crocco, "Gallerie aerodinanische per alter velocita."

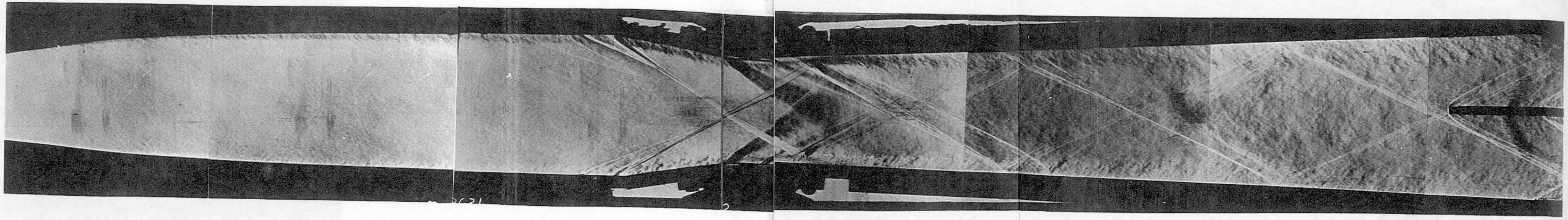

Figure 12 - $M = 2.60$, $\lambda = 3.41$, Second throat 1.550

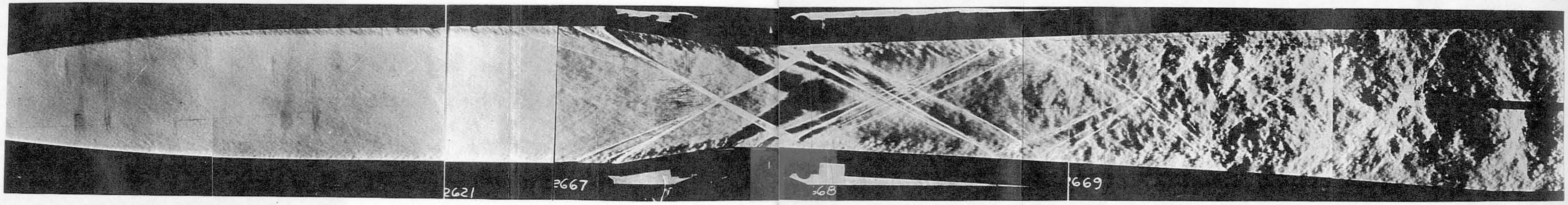

Figure 13 - $M = 2.60$, $\lambda = 2.75$, Second throat 1.550

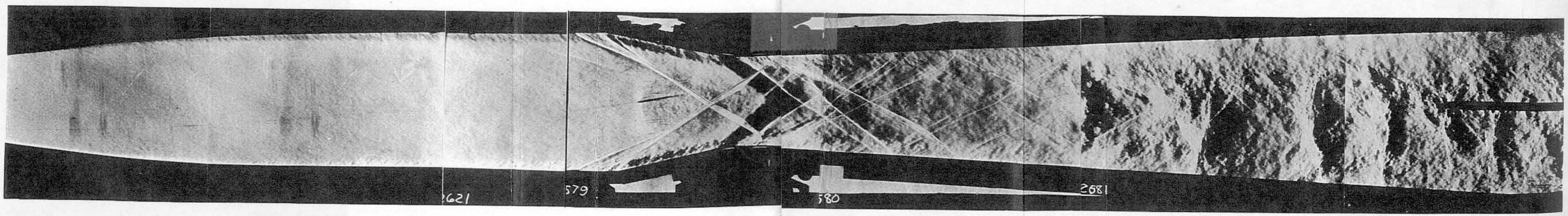

Figure 14 - $M = 2.60$, $\lambda = 2.50$, Second throat 1.350

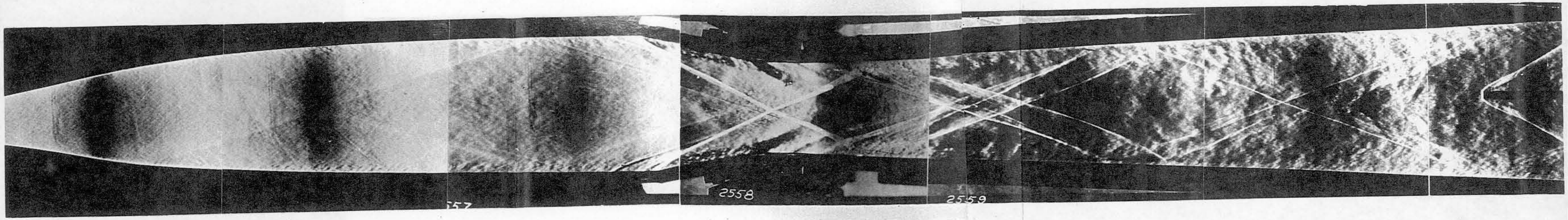

Figure 15 - M = 3.33, λ = 7.18, Second Throat 1.410

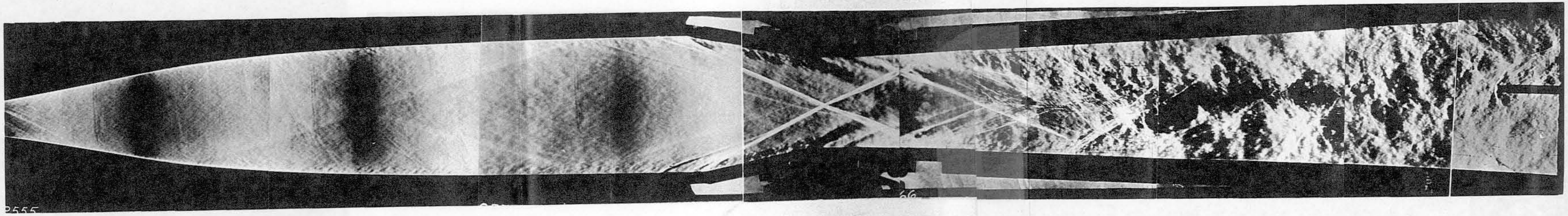

Figure 16 - M = 3.33, λ = 5.24, Second throat 1.410

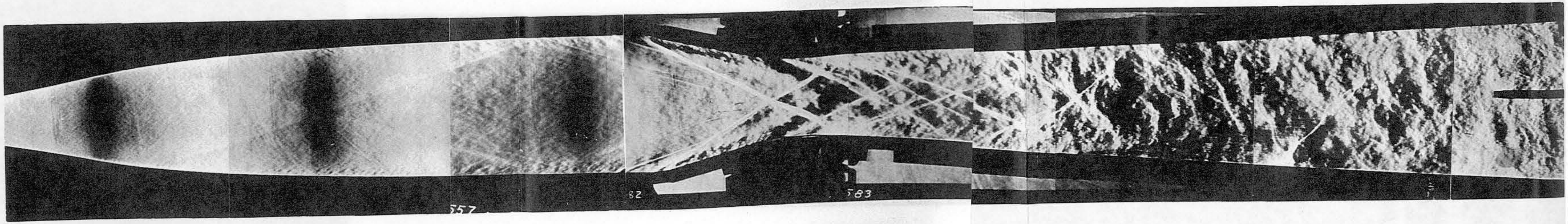

Figure 17 - M = 3.33, λ = 4.85, Second Throat 1.200

two Mach numbers. The most important features to be seen here are the nature of the transition itself, the proximity of the transition to the second throat, and the high turbulence accompanying the flow in the diffuser.

As noted in section II the flow transition is by means of a family of oblique shock waves. This was to be expected as soon as it was found that the second throat height could not be reduced to that necessary to give a a Mach number near unity. Measurements of the angle of the initial oblique shock waves and estimation of the area change to the point where transition occurs indicated that transition in the case of the M = 3.33 nozzle was still occurring at a Mach number of the order of three, whereas we had already noted that normal shock waves were not to be expected above a Mach number of 1.5 to 2. In the M = 2.60 nozzle, transition appeared to be taking place at a Mach number of about 2. The transition occurs three to four inches downstream of the second throat, and is followed by a region of high subsonic turbulence.

The existence of an optimum amount of choking for best overall compression ratio is linked to the causes of flow breakdown discussed in part 2 of this section. For when the second throat is reduced to a height very near the minimum, the interaction of boundary layer and expansion zone has already begun. Then if an unstable transition zone exists just downstream and transmits disturbances upstream through the boundary layer, enough fluctuation is induced at the second throat to cause breakdown. If the choking is a little less than minimum, flow at the second throat is little effected by the transition zone.

4. <u>Stability of the transition zone</u>: The positional stability of the

transition zone was observed to depend on rate of change of channel area. In the $M = 2.60$ unchoked condition, the diffuser angle was the smallest for any of the tests, and it was precisely here that the most difficulty was experienced in maintaining the flow near λ_{min}. The transition moved rapidly back and forth in the diffuser a distance of over one foot. In tests where choking had increased the diffuser angle to about 8 degrees the transition was quite stable about three inches downstream of the second throat. Hence it is certain that larger diffuser angles might be incorporated in this apparatus with possible further reduction of compression ratio.

5. Effect of surface roughness on test section flow: The existing set of wood nozzle blocks for Mach number 2.68 had the shellac surface which had previously been used exclusively in the GALCIT supersonic tunnel. When these blocks were first used, a distinct expansion wave pattern was observed originating at the first throat. This pattern, though decreasing in strength, nevertheless remained evident throughout the test section. Figure 18 indicates the nature of this flow. Upon examination after the first run, the blocks were found to be extremely rough, the shellac humping and buckling up from the wood surface. In the preliminary trials it was necessary to sand these blocks smooth between each run.

As a result, before the $M = 3.47$ blocks were put into the tunnel, the shellac was removed, and they were given a number of thin coats of Du Pont automotive laquer, sanded well between coats. Flow almost as smooth as with the usual steel nozzles was obtained with these blocks, and no care whatsoever

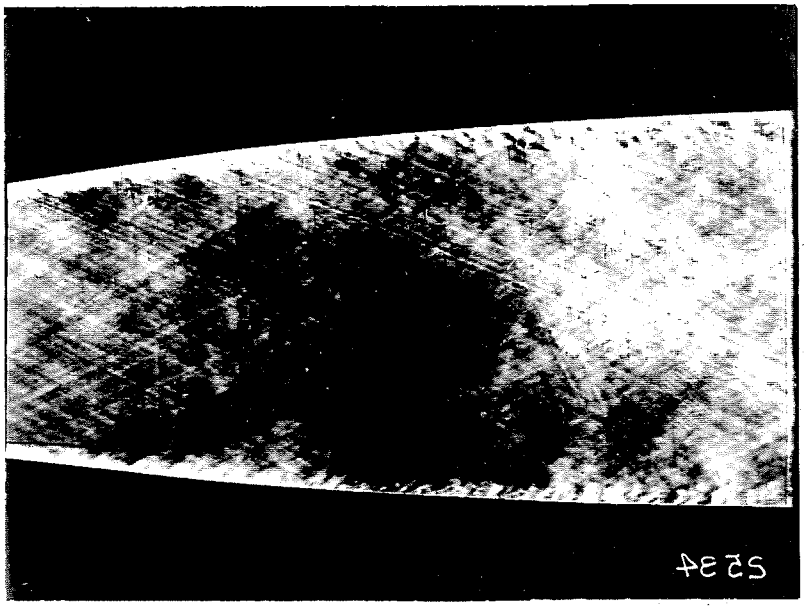

Figure 18 M = 2.60 Shellac

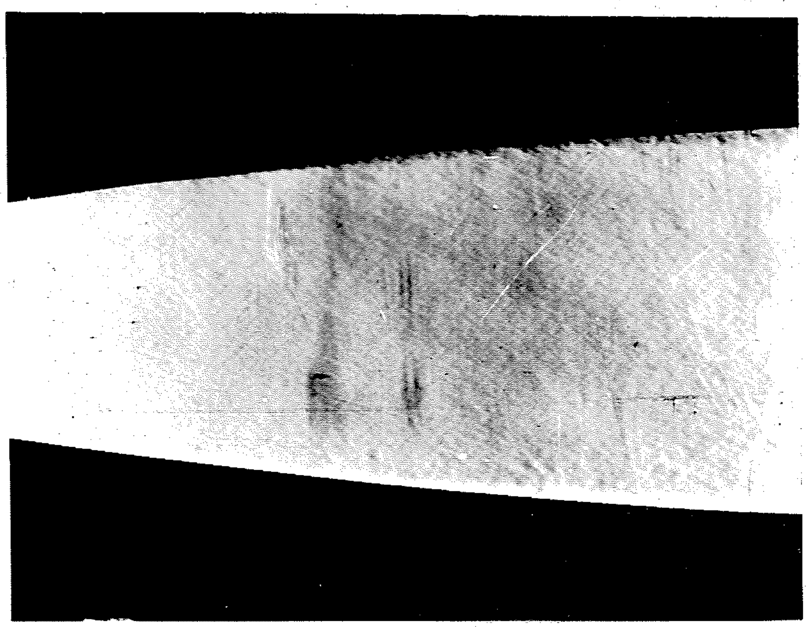

Figure 19 M ≈ 2.60 Laquer

was required for them throughout the tests. Subsequently the M = 2.68 blocks were also refinished, and the flow is illustrated for comparison in Figure 19 . The effect of surface roughness is clearly visible from Figures 18 and 19 . All the Schlieren photographs appearing in this report were taken with blocks having this smooth and durable surface. These results indicate the advantages, both in permanence of the surface and smoothness of flow, to be had from careful finishing of wood channel blocks.

VII CONCLUSIONS

1. Theoretical Minimum Starting Area: Experimental values of the minimum second throat area for starting at the two Mach numbers, 2.60 and 3.33, were found to check those predicted by theory within 3%. Starting was possible in both cases with a second throat slightly smaller than theoretically estimated. Thus the minimum starting area considerations of theory are felt to be reasonable design criteria.

2. Choking after Starting: When the second throat height was decreased after starting, it was found that only somewhat more than 1/3 of the theoretical design choking was realized at both Mach numbers. Further reduction of second throat area in both cases resulted in movement of the transition shock system upstream of the second throat with accompanying breakdown of flow in the test section.

3. Minimum Compression Ratios: The minimum compression ratios realized for the Mach numbers 2.60 and 3.33 were 2.50 and 4.85 respectively. These values, though lower than any previously measured in the GALCIT tunnel, were not as low as expected from the theory. Failure of these values to be reduced further is attributed to the inability to obtain design choking of the second throat and consequent reduction of the Mach number of transition.

4. Transition region: Rate of area change of the channel was observed to affect the positional stability of the transition shock system. In tests where the total diffusion angle was 8 degrees, the position of the transition could be held stable a few inches downstream of the second throat, whereas in tests where the diffusion angle was only 3 to 4 degrees, difficulty in establishing a stable transition near minimum compression ratio was experienced.

5. <u>Effect of surface roughness</u>: As observed in the Schlieren apparatus and presented in photographs herein, a striking effect of surface roughness on character of the supersonic test section flow was found.

6. For future work with flexible second throat supersonic diffusers, it is recommended that the diffusion angle be increased to the order of 15 degrees. It is felt as a result of this work that such a change would result in an even more stable position of the transition zone. It is further probable that additional pressure orifices throughout the diffuser would be of considerable value in analyzing the results.

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