AR 82-2

Aeronautical Report 82-2

SUPERSONIC BURNING IN SEPARATED FLOW REGIONS

(NASA-CR-168914) SUPERSONIC BURNING IN SEPARATED FLOW REGIONS Final Report (Wichita State Univ.) 53 p HC A04/MF A01 CSCL 21B

N82-23283

Unclas

G3/25 09882

by

Glen W. Zumwalt
Aeronautical Engineering Department
Wichita State University
Wichita, Kansas 57208

Final Report

for

NASA Langley Research Center (NSG 1575)
NASA Technical Monitor: R. Clayton Rogers

April 1982

CONTENTS

Symbols

A_{Base}	Area of the rearward facing base wall
h	Back step height from the separation edge to the trough corner
$\dot{m}$	Mass flow rate
M	Mach number
P	Pressure
V	Velocity
δ^*	Displacement thickness of the boundary layer at the separation corner
ρ	Density

Subscripts

a	Air
b	Base
∞	Free stream upstream of the trough
H_2 or h	Hydrogen

1. INTRODUCTION

This is the final Technical Report on research work performed at Wichita State University under Grant NSG 1575 from the National Aeronautics and Space Administration Langley Research Center, Hampton, Virginia. The Grant covered the period from January 1979 through October 1981. The NASA Technical Monitor at the start was Mr. Paul Huber; due to his retirement he was replaced by Dr. R. Clayton Rogers of the Hypersonic Propulsion Technology Group at Langley. The Grant title was "Supersonic Burning in Separated Flow Regions," but the research generally concentrated on (1) ignition in a supersonic-combustion ramjet (SCRamjet), and (2) using the vortex trough phenomenon to accomplish the stable burning and ignition.

The work reported herein is not a completed exploration of combustion applications of the vortex trough; further possibilities continue to come to mind. Neither is it an optimized design study for the SCRamjet ignitor; in several ways the tests fail to simulate the true SCRamjet conditions. This is, however, a demonstration of the feasibility of stable supersonic burning and ignition using the peculiar geometry of the vortex trough and has been developed sufficiently to give design guidance for installation in a hypersonic vehicle's SCRamjet.

2. TECHNICAL BACKGROUND

A review will first be presented of the work which preceded the research done under this Grant to define the "trough vortex" phenomenon and to provide a context for the SCRamjet-related study.

2.1 Genesis: The Intersecting Plume Problem

The work which led to the "trough vortex" began with attempts to solve the base recirculation for multi-jet rockets, such as those shown in Figure 1. In the case of the four-nozzle rocket, the plumes meet at planes of symmetry which form four corners. If the planes of symmetry act as though they were solid walls, a solution may be sought in the manner illustrated in Figure 2. The inviscid method of characteristics will provide the plume shape for any selected base pressure for flow from a given nozzle, having a known Mach number and exit pressure. One can then trace a typical streamline until it intersects the wall and is forced to turn into the plane of the wall.

A published work (Ref. 1) was found in which the author assumed that the turn into the wall would be accomplished isentropically. This was believed to be a reasonable assumption because the presence of viscosity on the jet plume surface would actually cushion the turn and make a rather smooth turn. He therefore attempted to preserve the velocity of the intersecting vector except for the component normal to the wall. In attempting to use his method, we discovered that he had made an error in the component actually deleted. After this was called to the authors' attention and they acknowledged the error, we corrected the method and tried to apply it to some selected example problems.

Working from the nearest wall location at mid-wall toward the corner in small angular intervals, it was found that a point was reached in each case at about half-way to the corner, where no solution existed mathematically. It seemed likely that the reversible-turn assumption was at fault, and so the method was changed to effect the vector's turn into the wall plane by a locally-oblique shock. Computations were again attempted, but the same failure of solution was encountered at about the same location. It was later discovered that Brewer at NASA/MSFC (Ref. 2) had also devised the oblique shock turn method for this configuration and also failed to find solutions. He decided that this indicated that the shock was detached beyond this point and that reverse flow would take place in the corner region from the jet back into the near wake.

Since analytical modeling seemed difficult and the geometry was quite simple, an exploratory experiment seemed in order. Axial-flow nozzles of about 6.5 cm^2 (1 in^2) exit area with Mach numbers about 1.5 and 2.0 were available. Channels having square cross-sections were fitted axially with the exhaust of the nozzles, as shown in Figure 3. On one of the walls, a matrix of flush pressure taps were placed for about one nozzle diameter downstream. When level lines were drawn for the pressures, the jet sheet was found to intersect the wall in a parabolic shape paralleling the inviscid jet plume intersection for the measured pressure ratio, as expected, near the center of the wall. However, about two-thirds of the way to the corner, the level lines wandered off in baffling patterns. A typical example is shown in Figure 4. It was also noted that the base pressure was behaving in a strange manner. Subsequently a series of tests were run which are summarized in Figure 5. The circular nozzles were exhausted through channels having cross-sections which were circular, square, triangular, and hexagonal, as indicated by the symbol shape. Each of these were made in three sizes of flow area. Base pressure (ratioed to supply pressure) is plotted against the area ratio of the sudden expansion at the end of the nozzle. For any one geometry, the results were predictable: base pressure decreased as area increased. But comparisons of results for different geometries led to another mystery.

2.2 The Rubber Tube Paradox

On Figure 5, three points are marked A, B and C. These correspond to the three expansion channels pictured in Figure 6. Here, the circular channel A is shown with a square channel B circumscribed around A tangential to the circle. Similarly, the triangular channel C is tangentially circumscribed over A. Although B and C are markedly larger in area than A, Figure 5 indicates that the base pressure is slightly higher. Thus, if we imagine that the channel A were elastic and capable of being stretched to increase its cross-section, then the effect of stretching it would depend on <u>how</u> it was stretched as well as <u>how much</u>. If it were stretched symmetrically, remaining circular, the base pressure would decrease. If it were stretched unsymmetrically by pulling it into three or four corners, the base pressure would rise. Unsymmetrical area increase had the same effect as area decrease, that is, the increase of the base pressure.

2.3 The Trough Experiment (Ref. 3)

A larger flow model was needed if this phenomenon were to be studied. A model of the corner was made, in the form of a 90° trough having walls of 10.67 cm, giving a step height into the corner of 7.62 cm. A block was fitted into the lower half of the W.S.U. 9" x 9" (23 x 23 cm) wind tunnel, and the trough was placed at the end of the block as shown in Figure 7. The trough could be tilted a few degrees. It was provided with static pressure taps on its walls and a total pressure probe could survey the field inside the trough. Flow visualization was accomplished by the use of lampblack-and-kerosene painted on the trough inner surfaces either as a solid coat or in a matrix of dots. The resulting flow pattern is illustrated in Figure 8. As the shear layer emanating from the step moves down into the corner it is squeezed by the converging walls and its edges roll under forming a vortex at either side. In the corner these somehow join to form a vortex pair lying in the corner and extending, not only downstream, but also spiraling upstream into the near wake. The oil streaks reveal the vortices reaching almost to the back step. Example photographs are shown in Figure 9 and 10, where the two trough vortices lie in the corner symmetrically. The more common occurrence is shown in Figure 11; one of the vortices slips under the other and they lie asymmetrically in the corner. Figure 11a shows vortex on the right hand side. Lowering the left edge 3 mm produced the left-side vortex domination of Figure 11b.

The "Rubber Tube Paradox" now seemed to be solved. When non-symmetrical wall expansion occurred forming corners, a vortex pair formed in each corner. The vortices occupied space which was denied to the main flow, actually reducing the available flow area and having the same effect as a physical reduction in cross-section. Figure 12 shows the total pressure isobars four step-heights downstream of the step. The effective removal of flow area is also seen in Figure 13, a longitudinal plane section showing the shear layer floating high above the corner.

Trough tests were run at Mach numbers of about 2 and 3 and with tilt angles of 0°, 5° and 10°. Qualitative results were similar for all of these.

2.4 Missile-Fin Tests

The most obvious application of the trough vortex phenomenon was as a base drag reducing device. If fins were extended from the base of a projectile or missile, vortex pairs would form in the corners having the same effect as a "sting," namely, an increased base pressure and reduced drag. Figure 14 shows a cone model used to evaluate the base pressure-raising ability of the trough. Care was taken to avoid support-mount interference. Repeated attempts were made to realize the hoped-for base pressure rise, but for all trough angles and lengths, the base pressure was essentially the same as for a flat base.

A square-base experiment finally clarified the nature of the trough vortex. The square cross-sectioned body shown in Figure 15 was machined to the shape of the streamlines in a Mach 2.0 induction wind tunnel. Two sets of four-bladed fins were extended from the base. The first is shown in Figure 15; the fins are parallel to the body sides. This produced no change in base pressure. The second set had fins reaching across the diagonals of the base. For these, the base pressure was about 30% higher.

Figure 15 summarizes the tests so far described. When the shear layer is squeezed by walls normal to the layer, as in cases (c) and (d), no vortices form. But when the low velocity layers are shortened more (or faster) than the faster layers, as in cases (a), (b) and (e), the edges curl into vortices and a higher base pressure results. The smaller the angle between the wall and the shear layer, the stronger the resulting vortices. Thus, the triangular channel produced a greater pressure rise than the square, and the square a greater rise than the hexagonal. Unfortunately, few missiles have square cross-sections, and no feasible means can be imagined for a fin system which would improve the drag of a circular vehicle. One possibility which suggested itself is shown in Figure 15: The Pencil Missile. It was believed that drag might be further improved by burning in the base troughs in the manner described below. Test of the Pencil Missile will be described in a later section of this report.

2.5 Subsonic Flow Studies (Ref. 4)

Concurrently with the igniter work to be given below, tests were made to determine whether the trough vortices occur for subsonic flows as well as for the supersonic. A trough with 2.5 inch (6.35 cm) step height was placed in a low-velocity wind tunnel having velocity of about 3.3 m/sec. Smoke was either injected into the base, placed to flow adjacent to the plate surface just upstream of the trough, or introduced through a thin tube placed in the trough corner at many locations. All attempts to detect corner vortices failed.

Suspecting that the absence of trough corner vortices was due to the very low Reynolds number of the low velocity flow, the 6.35 cm trough was moved to a water channel. The 2 ft square water channel at the Boeing Military Airplane Company in Wichita was made available for this. The tests were similar to the low-velocity wind tunnel studies except that liquid dye replaced the smoke for visualization. Relative free-stream water speeds varied from 0.3 to 1.5 m/sec, and extensions of the upstream flat plate provided Reynolds number changes. Flow visualization obtained was excellent, but the corners seemed devoid of the expected linear vortices.

A third subsonic test was performed, using the same trough, in the W.S.U. Beech Memorial, 2.1 x 3.05 m Wind Tunnel at about 290 Km/hr. Smoke and oil surface visualizations again failed to detect any sign of the corner vortices which so dominate the supersonic cases. Figure 17 shows the subsonic combinations of Reynolds number and Mach number which were tested, as well as various supersonic tests for which the vortex trough phenomenon was present. The conditions required to produce the trough vortex are still not well defined, but supersonic flow may be a necessity. If so, this must be regarded as disappointing since it might preclude the use of the vortex trough as a burner for turbojet engine. Tests are planned to fill in the gaps of Figure 17 in the high subsonic Mach range.

3. DEVELOPMENT OF SUPERSONIC FLAME HOLDERS/IGNITERS

Since the trough vortex creates a high level of mixing and a long residence time in the near wake, it appeared to be useful as a flame holder. A 6.35 cm deep steel trough was used in the W.S.U. 23 x 23 cm supersonic tunnel to study the possibilities of burning in the separated region adjacent to the back step. A spark plug igniter was placed in the base wall and hydrogen was added through a tube at the vee corner one step downstream of the base. Ignition was easily achieved over a fairly wide range of hydrogen flows, and combustion was self-sustaining without the spark igniter. Reignition could be readily achieved after the flame was extinguished by stopping the hydrogen flow. The actual amounts of hydrogen burned was limited, however. As the rate of hydrogen flow increased, the flame moved from the triangular slow-flow region at the base to the shear surface above it, and then downstream in the vortex pair sitting in the corner of the trough. These tests were all done at Mach 2.0 for the adjacent flow, with air total temperature about 21°C and total pressure about 482 kN/m^2 (70 psia).

3.1 Hydrogen Supply

For all of the tests reported herein, gaseous hydrogen was supplied from a commercial high-pressure bottle at room temperature. Hydrogen mass flow rate was measured by using a set of calibrated in-line flow nozzles. Pressure drop across the nozzle was sensed by a differential pressure transducer.

3.2 Air Supply

The air for all tests was supplied from storage tanks having maximum pressure of 250 psig and total volume of 700 cu.ft. Difficulty in ignition was experienced due to high humidity of the flowing air, especially in warm weather, even though air passed through a chemical dessicant dryer. A water-cooled heat exchanger and centrifugal water separater were installed just downstream of the compressor and the humidity problem was helped considerably. Even so, air used in all the tests was quite wet compared to that to be expected in flight. On warm, humid days, dry bulb temperature of 72°F (22°C) and wet bulb temperature of 65°F (18°C) were typical. Stream pressures in the wind tunnel test were approximately 9 psia (62 kN/m^2).

Thus, ignition was being achieved under conditions much less favorable to combustion than those encountered in a SCRamJet: colder, wetter, and at lower pressure.

3.3 Small Trough Wind Tunnel Tests

Under this grant a program was undertaken to develop very small supersonic igniter for use in a supersonic combustion ramjet. The trough was reduced to a 1 inch step height and stable combustion again accomplished alongside a Mach 2 flow of air, with hydrogen again inserted in the corner and directed toward the base. Because of the smaller size, most of the burning took place on the shear layer surface and in the downstream corner vortices.

The step height was again reduced to 0.25 inches (6.35 mm), with the hydrogen injection and spark igniter as shown in Figure 18. The flame was present only downstream of the shear layer, generally starting about one step height from the base and extending downstream beyond the end of the trough. The height was reduced to 0.20 inches (5 mm), then to 0.15 inches (3.8 mm). For these, the plate upstream of the step was kept very short to limit the size of the boundary layer at the step. This length was varied to find the effect of boundary layer height; Figure 19. For steps smaller than about 0.25 inches (6.4 mm), ignition became more difficult and the boundary layer had to be kept very small to permit stable burning. For air flows with higher temperatures, it would be expected that ignition would be easier.

A thermocouple was placed in the flame region for the combustion tests using step heights less than one inch. Platinum/platinum-rhodium thermocouples were used, with 3 mil wires forming the joint and 10 mil "posts." Alumina was baked on the junction and adjacent wires to prevent hydrogen combination with the platinum and to insulate electrically from free ions in the combustion gases. It was found that ignition was much more difficult to achieve when the thermocouple was removed. To determine whether or not this was simply a glow-plug effect, stainless steel, copper and ceramic protrusions of the same size were substituted. Only the copper gave the same ease of burning.

The ceramic was basically alumina, so apparently enough platinum is exposed on the thermocouple to act as a catalyst. The copper also serves well as a catalyst for the burning, but melts away quickly.

For troughs smaller than one inch in step height, Teflon material was used at first. This provided an easy method of producing the ignition spark. A wire was inserted into the trough corner about 3.5 step heights downstream of the base. The spark jumped from the wire to the aluminum base wall, so the incoming hydrogen was forced to flow past the spark. The Teflon surface gradually melted in the presence of the flame, requiring frequent replacement. Aluminum troughs with igniter wires placed as shown in Figure 24 replaced these early Teflon models. It was noted, however, that ignition was a bit easier with the Teflon version, due (we believe) not to the change in igniter location as much as to the heat sink provided by the aluminum which tended to cool the burning gases.

All of the above tests used troughs with a 90° angle and the corner was streamwise and untilted.

3.4 Open Jet Tests

For further tests with small trough models, the decision was made to design a new flow facility rather than continue testing in the 23 x 23 cm supersonic tunnel. Run times in the 23 x 23 cm blow-down-type tunnel were less than 10 seconds of steady flow. Also, access was difficult and vision obstructed. For very small troughs this large supersonic stream was not needed.

The first "open jet" tests used axisymmetric nozzles, with a small 90° vee-trough placed at the exit having the trough edges aligned with the nozzle wall, as shown in Figure 20a. This gave a slightly curved separation corner, but had the advantage of easily varying the boundary layer by adding cylindrical extension tubes to the nozzle, as seen in Figure 20b. Nozzles with design Mach numbers of 1.4 and 2.0 were available. (Mach numbers were lower at the end of the extension tubes due to the boundary layer build-up.)

The effect of boundary layer at the separation step is shown in Figure 21. It became increasingly difficult to maintain combustion as the turbulent boundary layer displacement thickness approached one-fourth the step height of the trough. Displacement thickness was computed by subtracting the effective exit area from the physical area. Effective area was defined as the area which would produce the measured Mach number for isentropic flow. This assumed zero boundary layer at the nozzle throat and adiabatic flow.

With the longer run time (over one minute) it was possible to experiment with re-ignition capabilities. With air flow established so that exit plane pressure was atmospheric, (1) the igniter spark was switched on, (2) hydrogen flow begun, (3) ignition achieved and (4) igniter switched off. Thus, stable burning was demonstrated. The hydrogen flow could then be stopped, and the four steps repeated.

An alternate starting method was to light the flowing hydrogen first, then start air flow. As the air supply pressure increased the flame was crowded down into the trough. At full $M=2$ air flow, the flame was a tiny glowing line hugging the trough corner, brightening where the thermocouple disturbed the flow, and (at high hydrogen mass flow rates) extending beyond the trough as a glowing plume.

The second set of "open jet" tests replaced the axisymmetric nozzles by a two-dimensional (2-D) nozzle producing plane flow. This is shown in Figure 23 with one side wall removed. To add versatility, a ½-inch thick center block could be inserted and the nozzle blocks spread apart. The exit dimensions without the center block were 1.0 x 1.5 inches. The igniter trough was attached to the 1.5 inch side (Figure 22) with trough edges flush with the nozzle wall. Hydrogen injection was moved to the base wall to better represent the most likely situation in a SCRamjet. The geometry is shown in Figure 24 in centimeters.

Pressure at the exit plane of the nozzle was monitored and supply pressure regulated to keep exit pressure equal to ambient. Thus, a parallel jet of air at $M=2$ was directed over the trough. Again, ignition was easily achieved, the flame was self-sustaining and reignition easily accomplished.

The trough experiments were repeated using the plane nozzle. A static pressure port was placed in the base just above the trough corner. Temperature and pressure measurements are shown in Figure 25.

3.5 Flush-Wall Tilt Trough

The igniters described to this point would be suitable for use on a strut or wall termination. A version of the vee-trough igniter suitable for use in a flat wall is shown in Figure 26. This is essentially a tilted trough cut off at the top edges to be flush with the wall. The shear layer is almost parallel to the wall so the base pressure is nearly equal to the stream pressure. This gives a nearly dragless igniter. Test results are shown in Figure 27. It can be seen that only about one-third as much fuel can be burned in the tilt trough as in the non-tilted one. The length to height ratio shown is 6:1, resulting from a tilt angle of 9°, and this appears to be the shortest practical design for $M = 2$ flow. An 8:1 (7° tilt) design is suggested as optimum, and should permit more hydrogen to be burned.

3.6 Upstream Fuel Injection

Further tests were made to find how the trough might be used to ignite a large flow of hydrogen. One method explored is shown in Figure 28. No hydrogen is supplied directly into the trough, but a hole having diameter equal to the step height was placed two step heights upstream. The hydrogen was injected normal to the wall and ignition could be achieved for a wide range of hydrogen mass flow rates.

While the flame appeared to be large, there was a question about the fraction of hydrogen which was actually burned. It was deemed unwise to continue tests of this sort until some method was devised to measure the combustion efficiency. This will be discussed in Section 4.

3.7 Yawed Wall Tests

One proposed use of the igniter is to be mounted on a yawed strut. The geometry shown in the lower sketch of Figure 19 was tested at Mach number of 2. Results were nearly indistinguishible from those for the non-yawed trough wall.

3.8 Back Step Comparison

As a comparative test, a simple back step with width of 3.81 cm and step height of 0.635 cm was placed at the end of the 2-D nozzle, as shown in Figure 29. Hydrogen was again injected from the base and two spark ignition locations were provided. Ignition could be achieved only with difficulty and over a narrow range of hydrogen flows. Further data will be taken to establish the hydrogen mass flow range as a function of "aspect ratio" of the base.

3.9 Base Burning on a Missile

A related thesis project, not directly supported by this grant, involved vee-trough burning in 60° troughs with 2.54 cm step height. This was done in a feasibility test for external base burning on a missile having a hexagonal cross-section, referred to in Section 2.4 as the Pencil Missile; see Figure 30.

Figures 31 and 32 show the test set up. A front support for the model was extended through the throat of the 23 x 23 cm $M = 2$ wind tunnel. All supply tubes and wiring were introduced from the plenum chamber giving an unobstructed base. An internal strain gage model balance (Figure 33) provided for relative movement in the streamwise direction between the support and the model.

The measured thrust was converted to specific impulse and plotted in Figure 34. The abscissa is the same as for Figures 25 and 27, mass flux of hydrogen in the triangular base divided by mass flux of air in the free stream. A single spark at the base center served to ignite all six channel flows without difficulty. The hydrogen was injected at the base wall, about in the center of each hexagonal segment, and directed inward toward the spark location.

4. CALORIMETER STUDY

The capability of the trough igniter for igniting a large hydrogen flow can only be evaluated by determining the burning efficiency. This could be done by chemical analysis of the outflow, but to do so requires much special equipment and effort. An alternate method which promises reasonable ease and accuracy is flow calorimetry.

Figure 35 illustrates the device constructed for this purpose. The rectangular nozzle and small trough igniter are placed in a plastic tube 20.3 cm (8 inches) in diameter. An attempt was made to diffuse and mix the flow to achieve uniform velocity in the exit plane at the top of the tube. A total pressure rake was rotated in the exit plane to determine the uniformity of the exit velocity. Conical diffusers and a normal-shock-producing wire grid were used to achieve uniform outflow.

When suitably uniform exit velocity was achieved, a grid of iron-constantin thermocouples was placed in the exit plane. The 29 thermocouples were cross-coupled to integrate the temperature readings electrically. At the time of this report writing, calorimetric data are beginning to be taken. Comparison of energy increase from air and hydrogen inflow to gas outflow with energy release for stoichiometric burning of the hydrogen supplied will give the the burning efficiency. Care must obviously be taken to achieve steady state conditions before recording data.

This project goes beyond the scope of the subject Grant, and was not supported monetarily except for using some models made under the Grant. This will be included in the Ph.D. Dissertation work of Robert A. Friedberg.

5. CONCLUSIONS

It has been demonstrated that the trough vortex phenomenon can be used for combustion of hydrogen in a supersonic air stream. This has been done in small sizes suitable for igniters in supersonic combustion ramjets so long as the boundary layer displacement thickness is less than 25% of the trough step height. A simple electric spark, properly positioned, ignites the hydrogen in the trough corner. The resulting flame is self-sustaining and re-ignitable.

Hydrogen can be injected at the base wall or immediately upstream of the trough. Care needs to be taken to introduce the hydrogen at low velocity to permit it to be drawn into the corner vortex system and thus experience a long residence time in the combustion region.

The igniters can be placed on a skewed back step for angles at least up to 30° sweep without affecting the igniter performance significantly. Certain metals (platinum, copper) were found to act catalytically to improve ignition. Comparison tests showed that the trough igniters burned with more ease and stability than plane back steps.

While most vee troughs tested had 90° corners, a related test showed that 60° corners performed combustion at least as well.

The tests were all performed under conditions less conducive to burning than are expected in a SCRamjet. The total temperature of the air was low (near 15°C) and the pressure was standard atmospheric or less (down to one-third).

REFERENCES

1. Lamb, J.P., Abbud, K.A., and Lenzo, C.S., "A Theory for Base Pressures on Multinozzle Rocket Configurations," AIAA Paper No. 69-570, 1969.

2. Brewer, E.B., and Craven, C.E., Experimental Investigation of Base Flow Field at High Altitude for a Four-Engine Clustered Nozzle Configuration," NASA TN-D-5164, May 1966.

3. Barr, Gary L., "Experimental Investigation of the Sudden Expansion of Supersonic Flow into a 90° Vee-Shaped Channel," M.S. Thesis, Aeronautical Engineering Dept., Wichita State University, 1972.

APPENDIX

Publications Resulting from this Grant

Publications reporting work which was partially supported by this Grant were as follows:

1. Zumwalt, G.W., "Experiments on Three-Dimensional Separating and Reattaching Flow," AIAA Paper 81-0259. Presented at the 19th Aerospace Sciences Meeting, St. Louis, MO, January 12-15, 1981.

2. Friedberg, Robert A. and Anwar Ahmed, "3-D Supersonic Combustion Experiments with Hydrogen in V-Trough," AIAA Paper 82-0417. Presented at the 20th Aerospace Sciences Meeting, Orlando, FL, January 11-14, 1982.

3. Friedberg, Robert A., "Burning Hydrogen in 90 Degree "V" Troughs Submerged in Mach 2 Air Flows," M.S. Thesis, Aeronautical Engineering Dept., Wichita State University, Wichita, KS, April 1981.

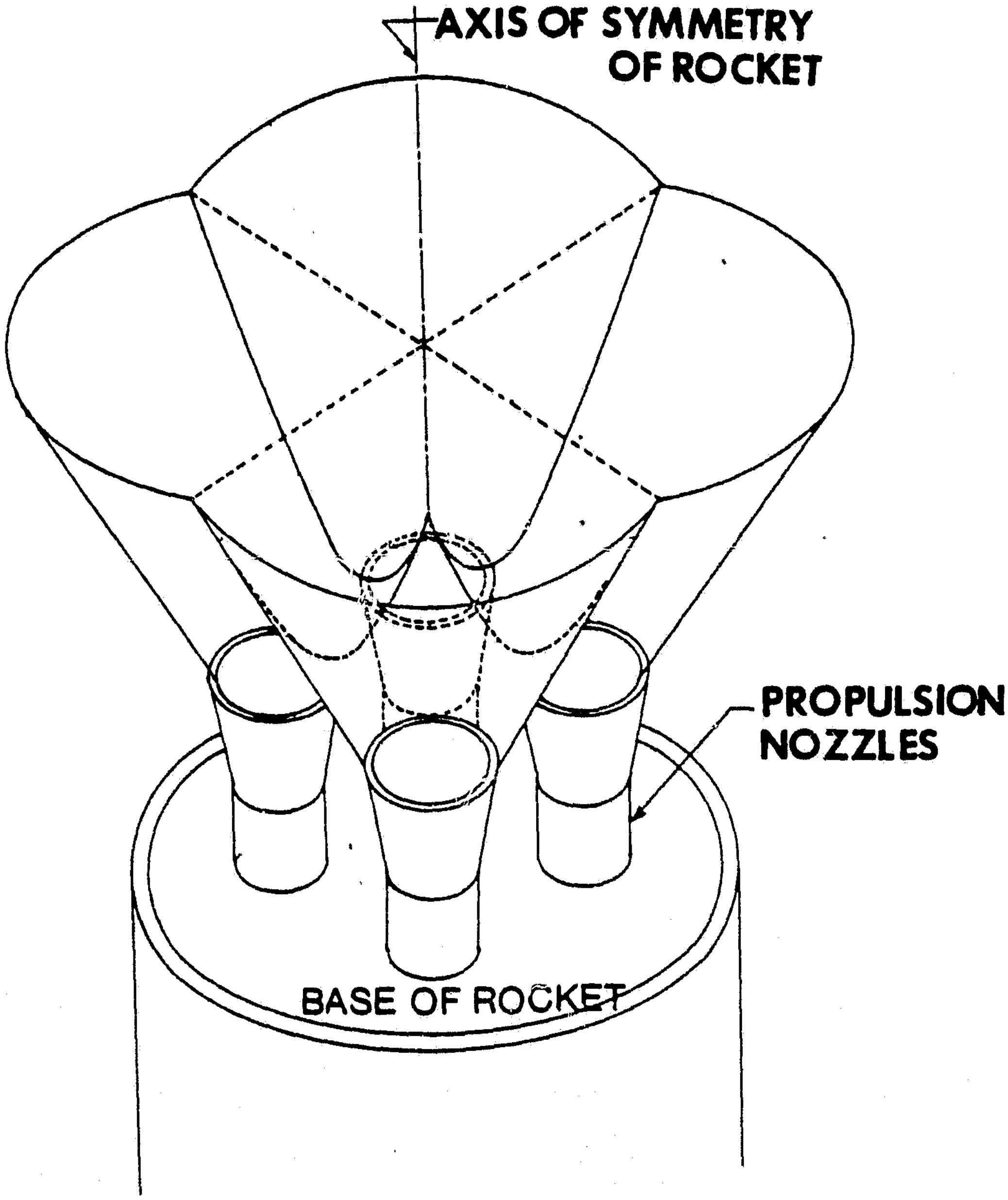


FIGURE 1. JET PLANE INTERSECTION FOR A MULTI-JET ROCKET

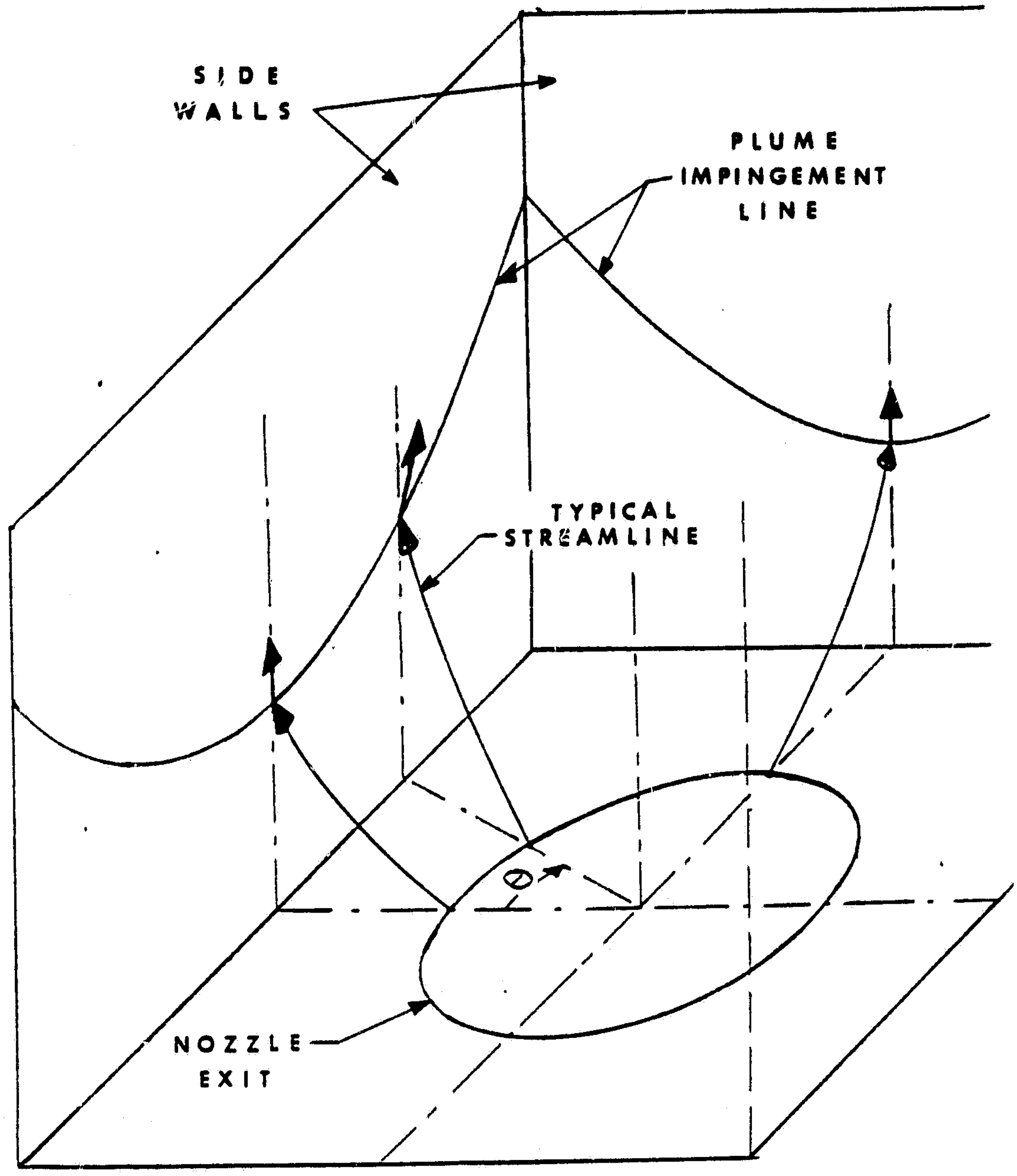


FIGURE 2. INVISCID JET INTERSECTION WITH A CORNER

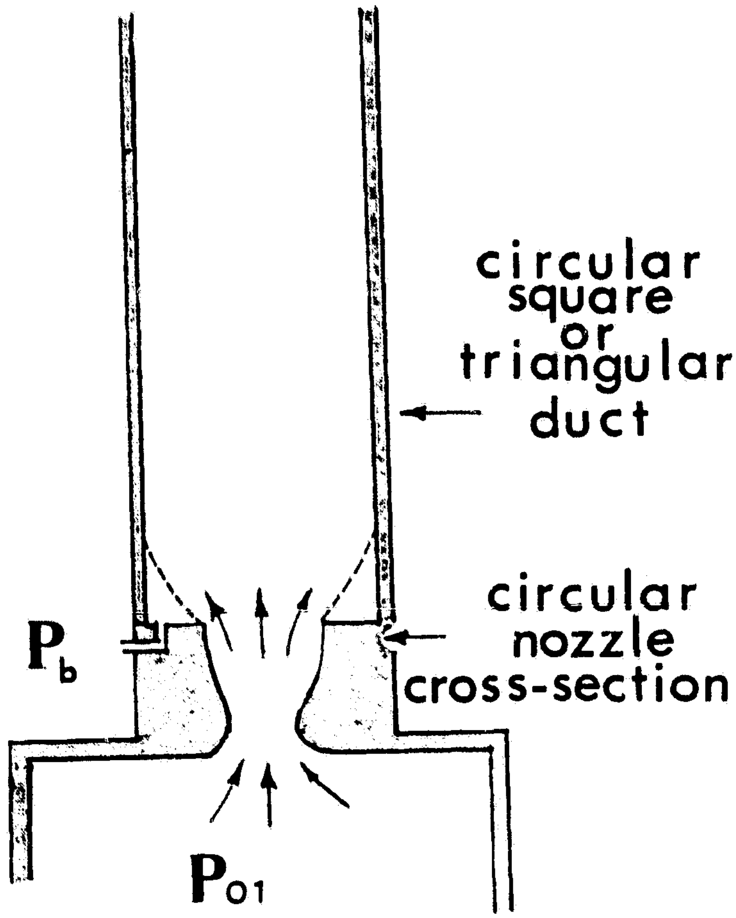


FIGURE 3. INTERNAL JET PLUME INTERSECTION EXPERIMENT

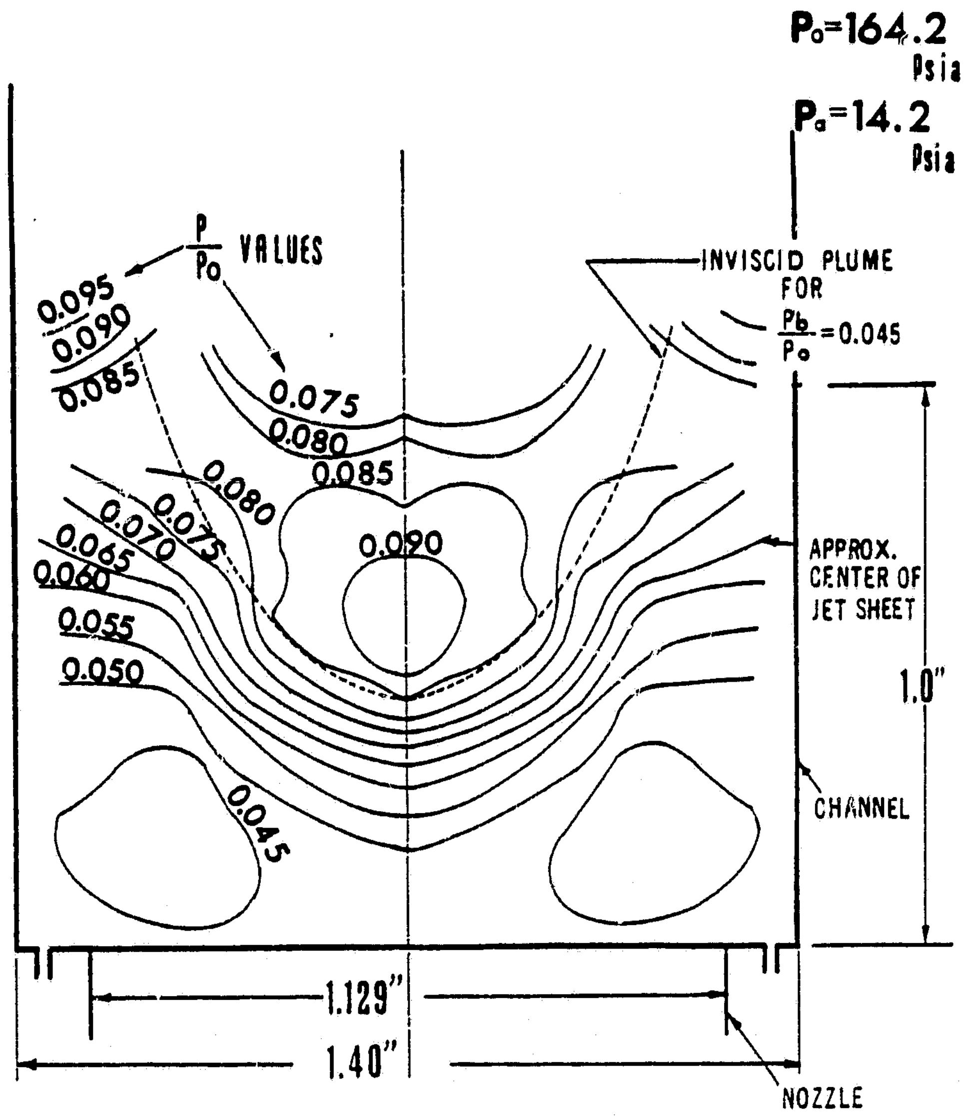


FIGURE 4. WALL ISOBARS FOR FIGURE 3 WITH A SQUARE CHANNEL

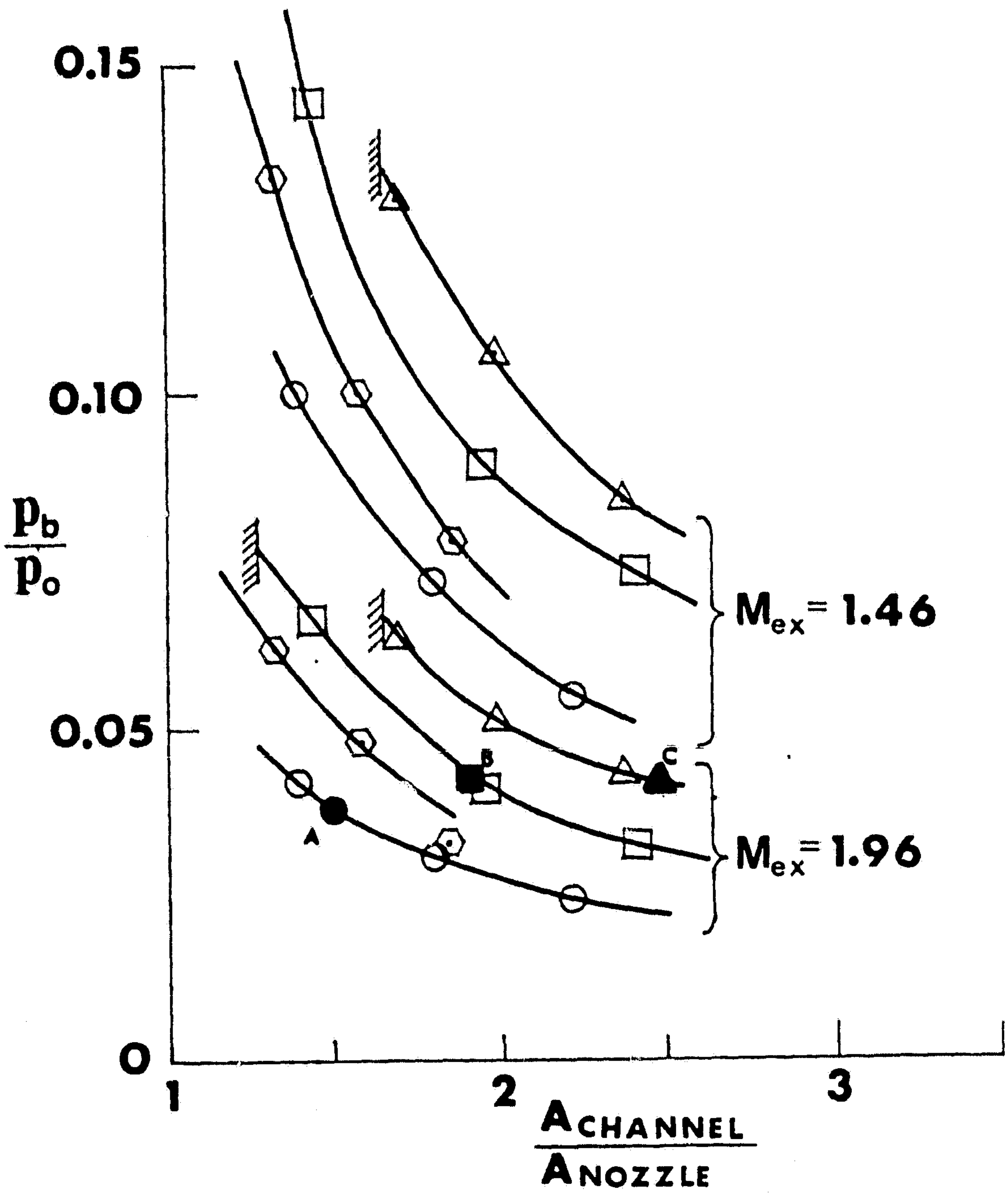


FIGURE 5. BASE PRESSURE FOR MODELS IN FIGURE 3

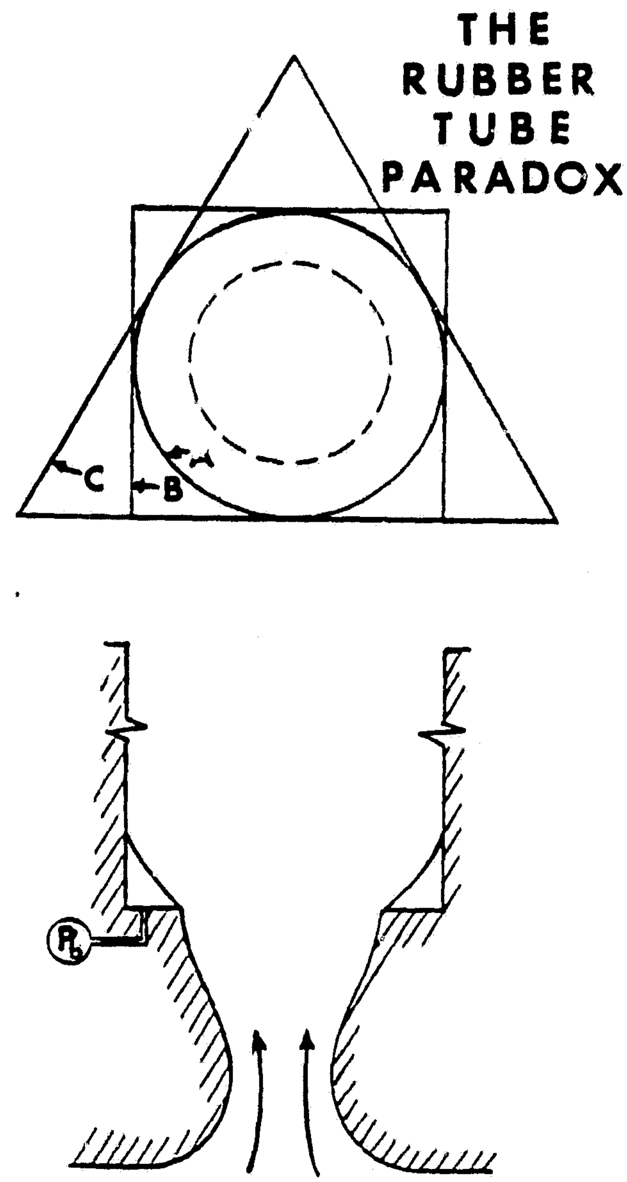


FIGURE 6. THE RUBBER TUBE PARADOX

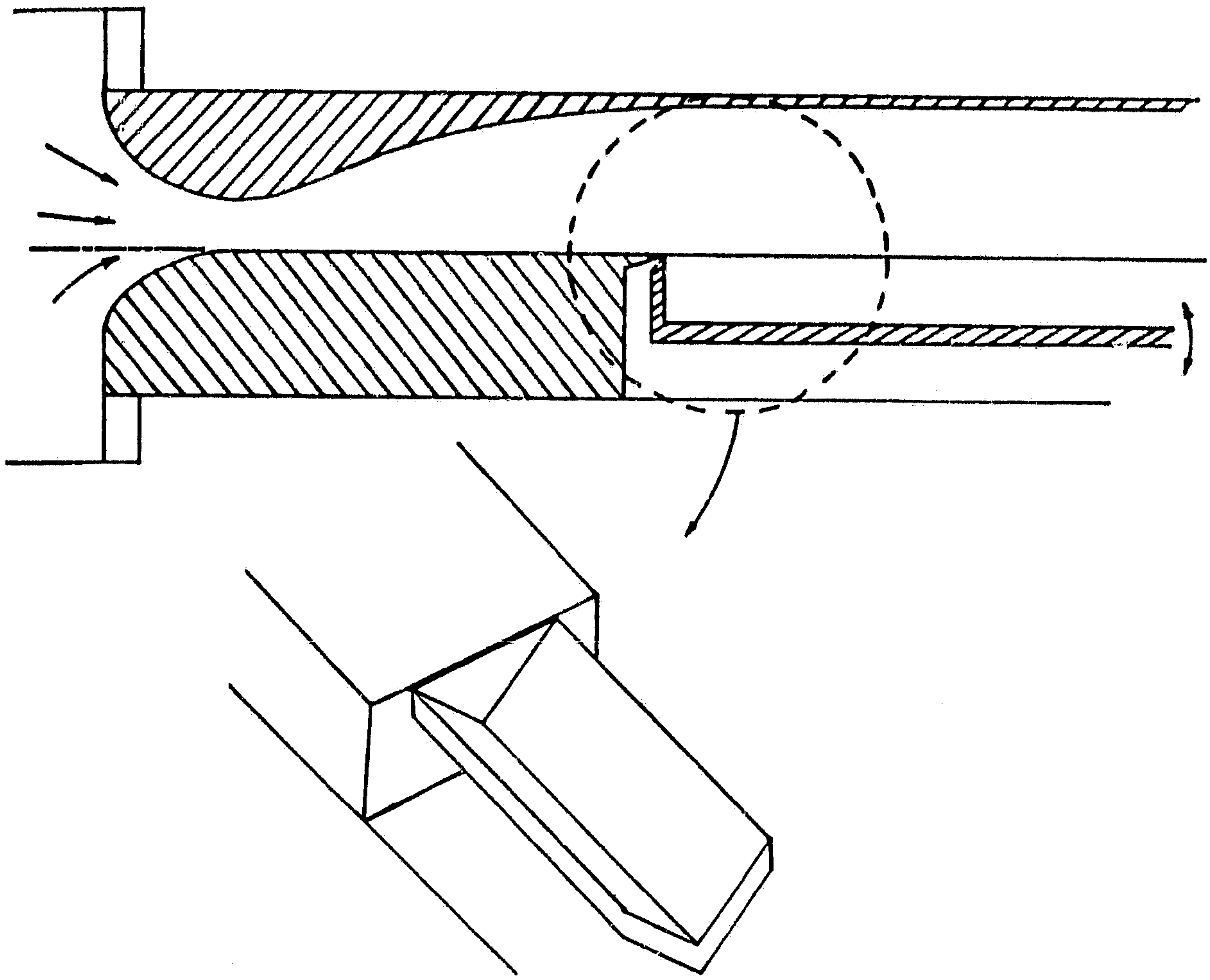


FIGURE 7. TROUGH MODEL IN THE SUPERSONIC WIND TUNNEL

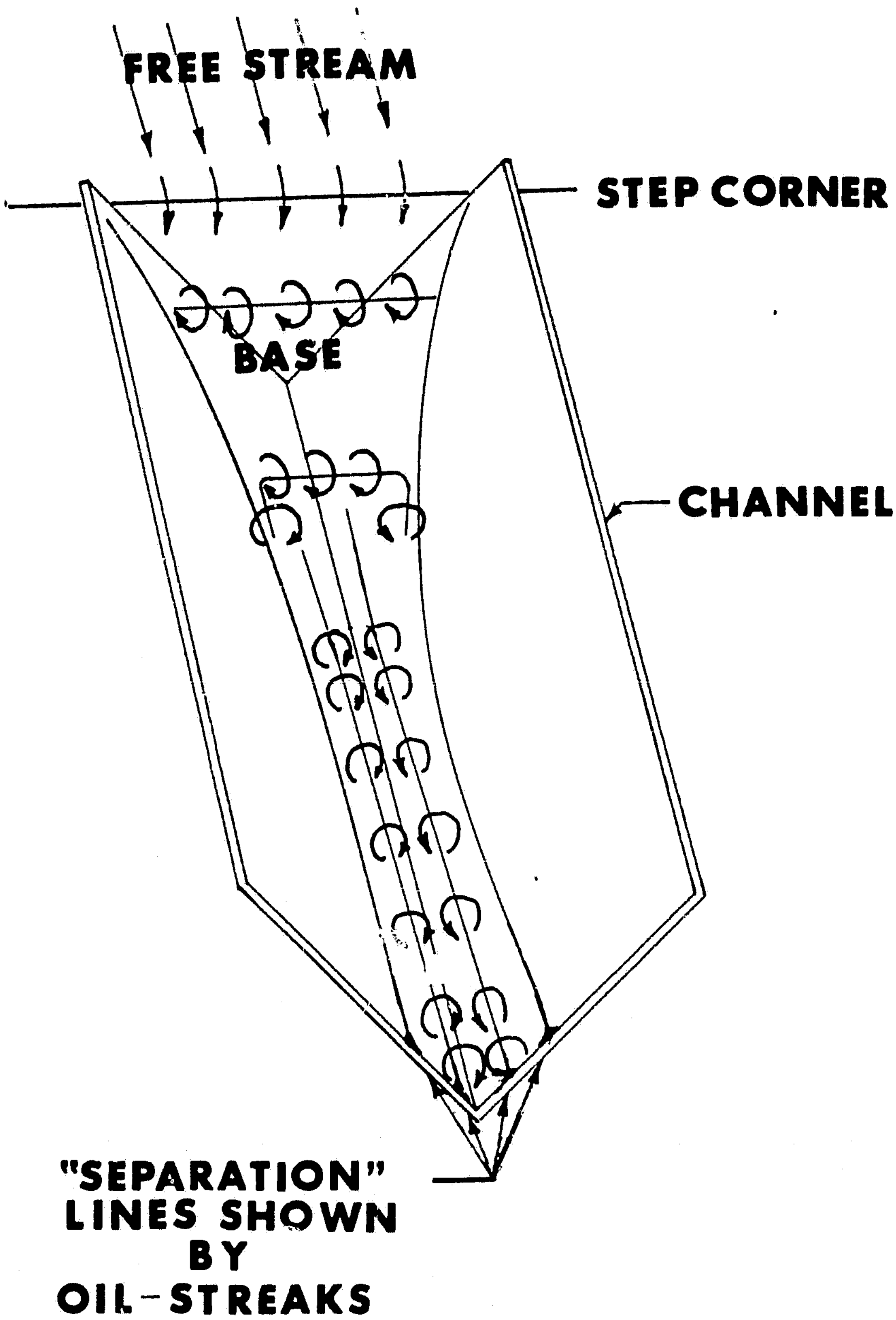


FIGURE 8. FLOW PATTERN INDICATED BY OIL STREAKS

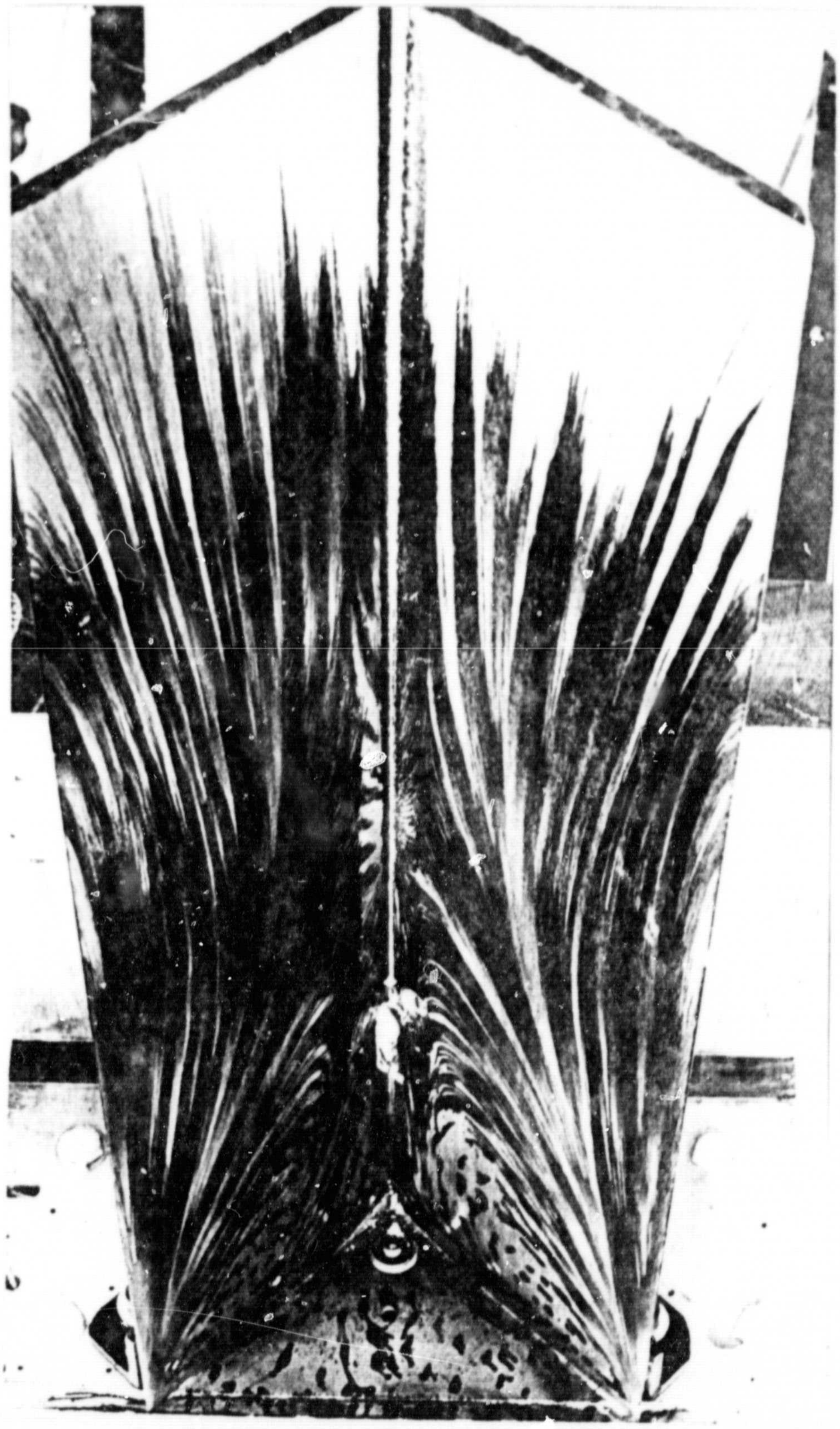
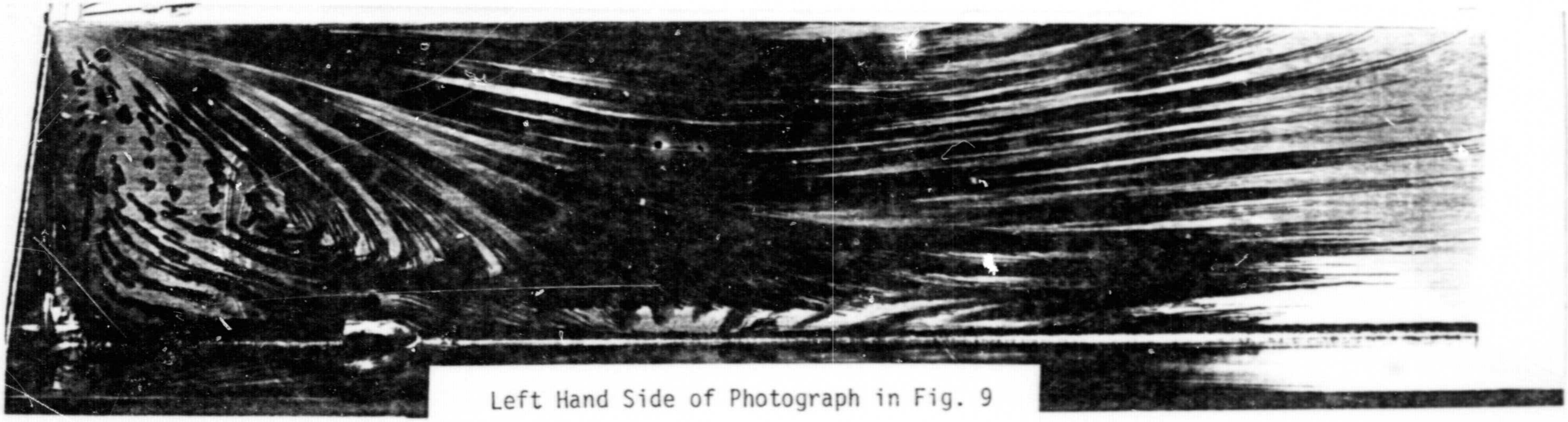
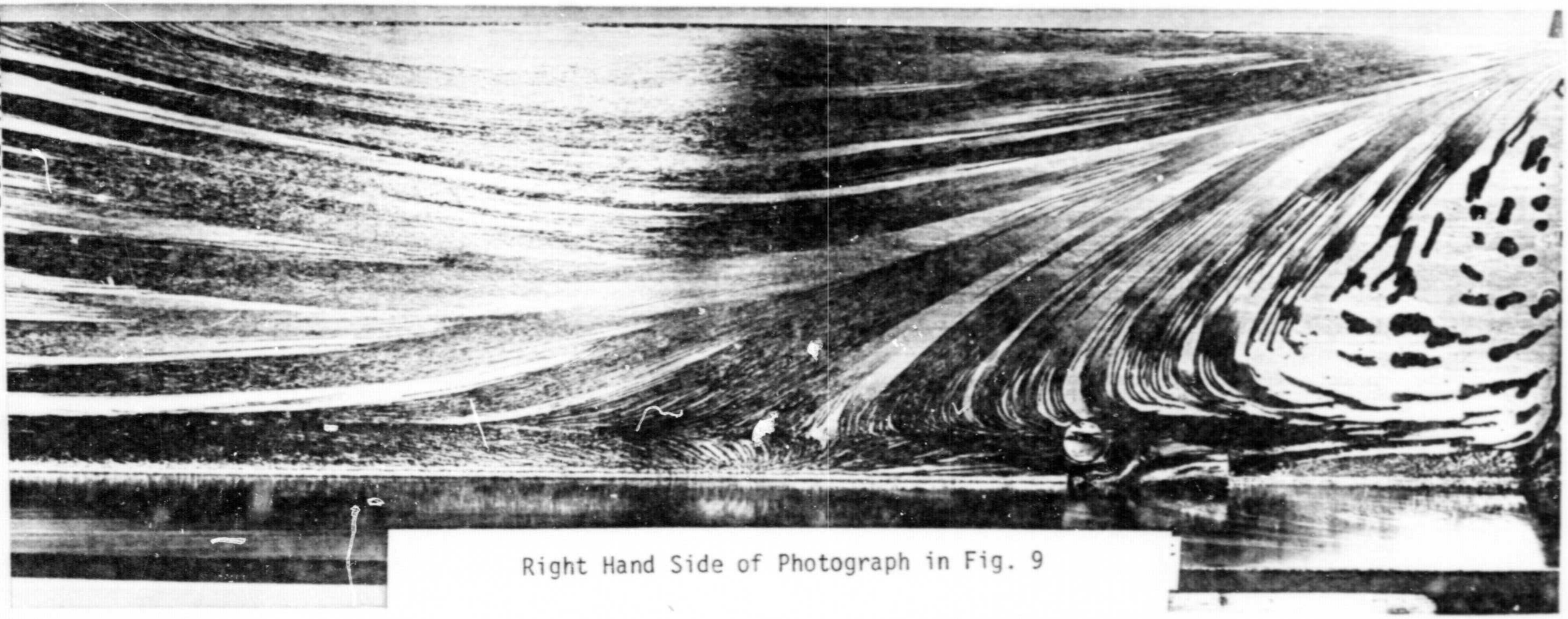


FIGURE 9. OIL STREAKS FOR SYMMETRICAL VORTICES



Left Hand Side of Photograph in Fig. 9



Right Hand Side of Photograph in Fig. 9

FIGURE 10. SIDE WALL PHOTOGRAPHS FOR FIGURE 9

Right Side Vortex Caused to Drop Into the Corner by a 1^{o} Leftward Misalignment

Asymmetry Reversed Due to Lowering Left Edge 3 mm.

FIGURE 11. NON-SYMMETRICAL TROUGH VORTICES

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

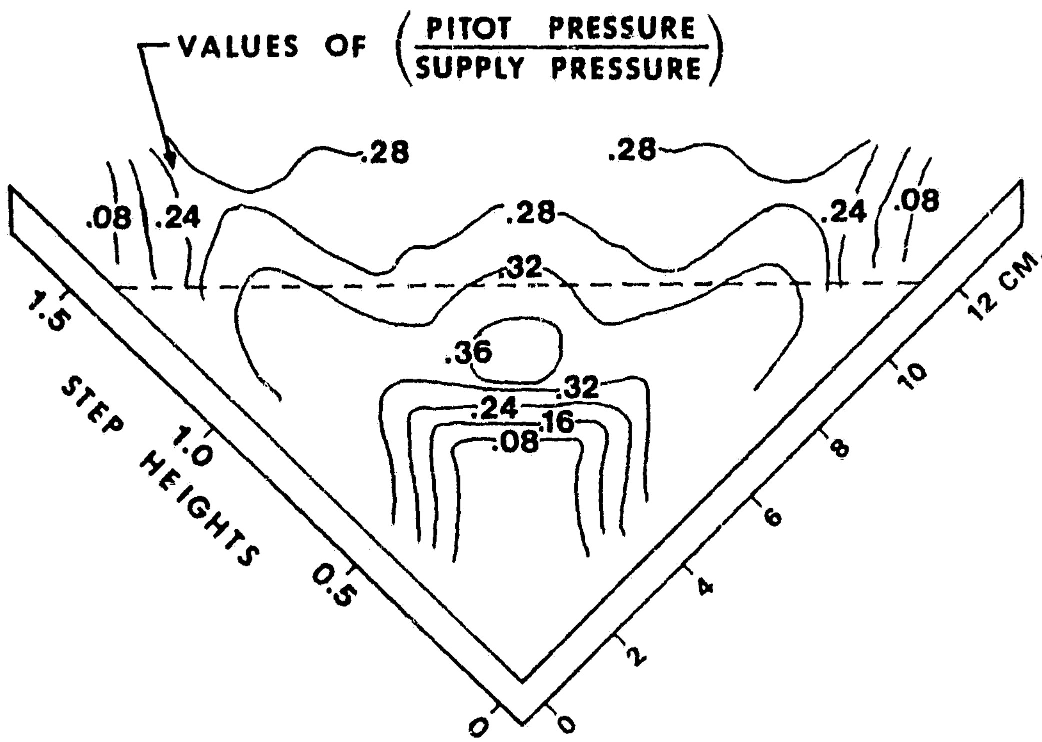


FIGURE 12. TOTAL PRESSURE ISOBARS ON A TRANSVERSE PLANE FOUR STEP HEIGHTS DOWNSTREAM FROM THE BASE

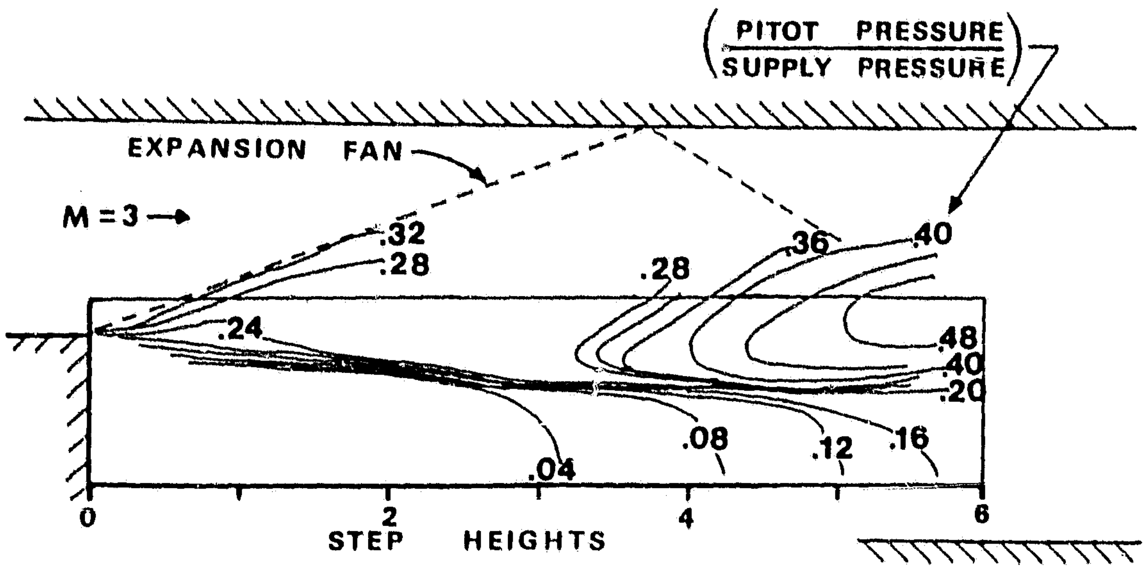


FIGURE 13. TOTAL PRESSURE ISOBARS ON THE LONGITUDINAL CENTER PLANE

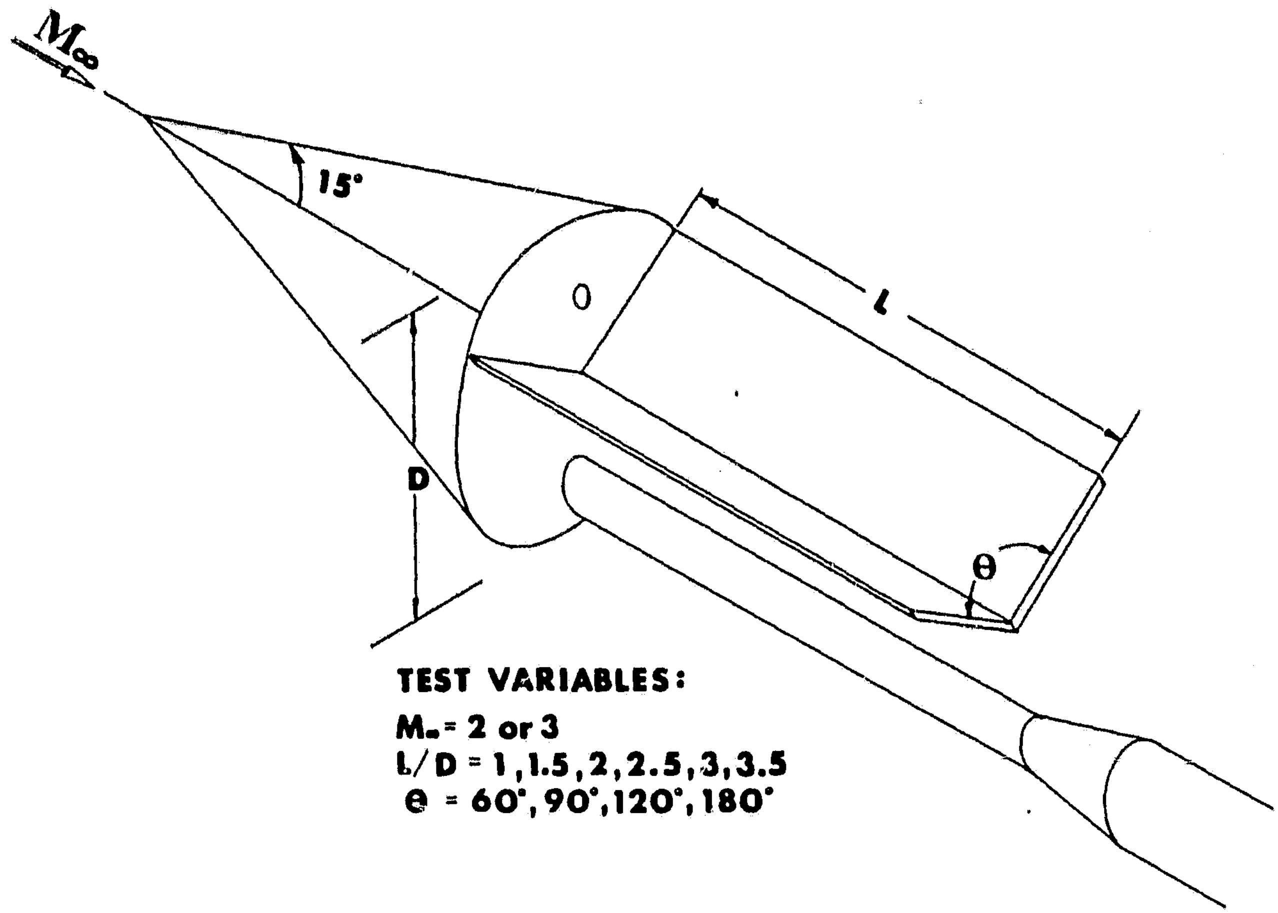
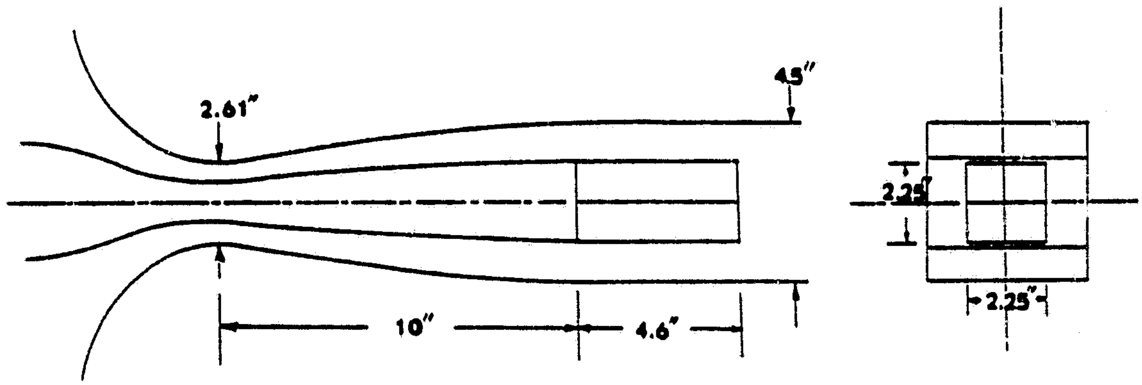


FIGURE 14. CONE MODEL WITH CENTERED FINS



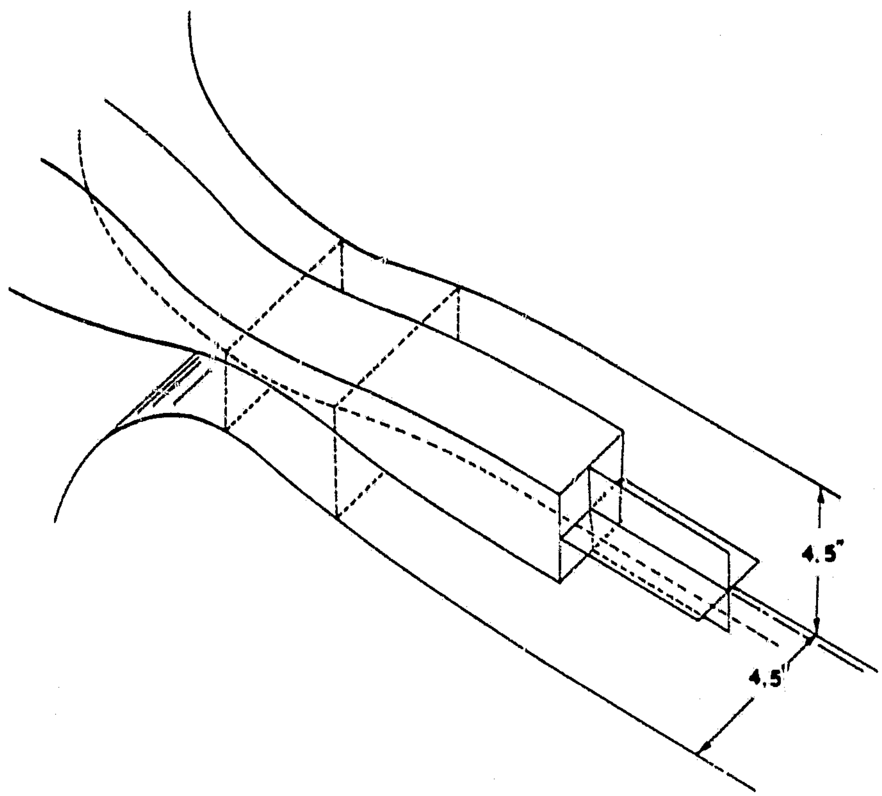


FIGURE 15. SQUARE BODY MODEL WITH BASE FINS

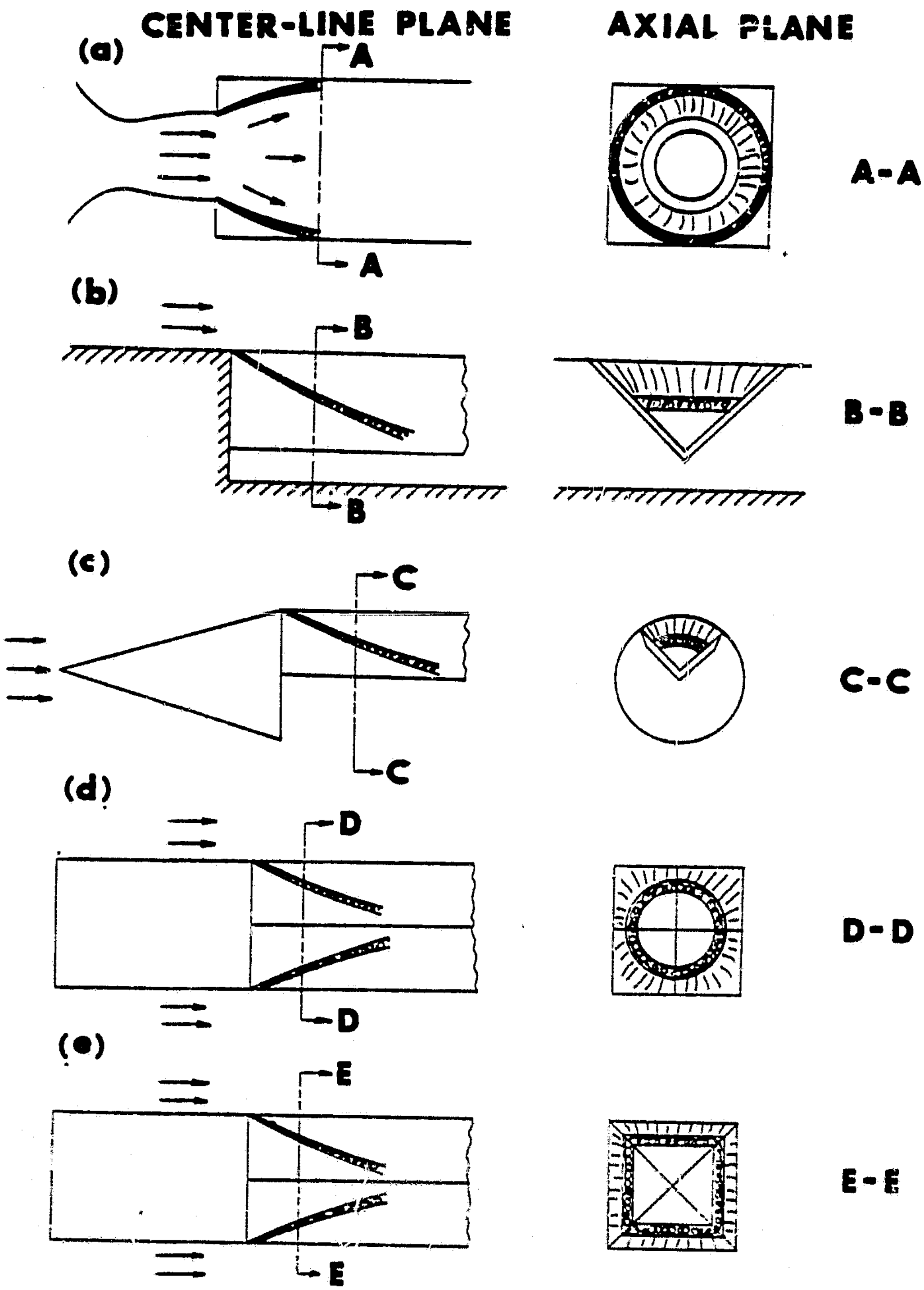


FIGURE 16. SUMMARY OF BASE TROUGH TESTS

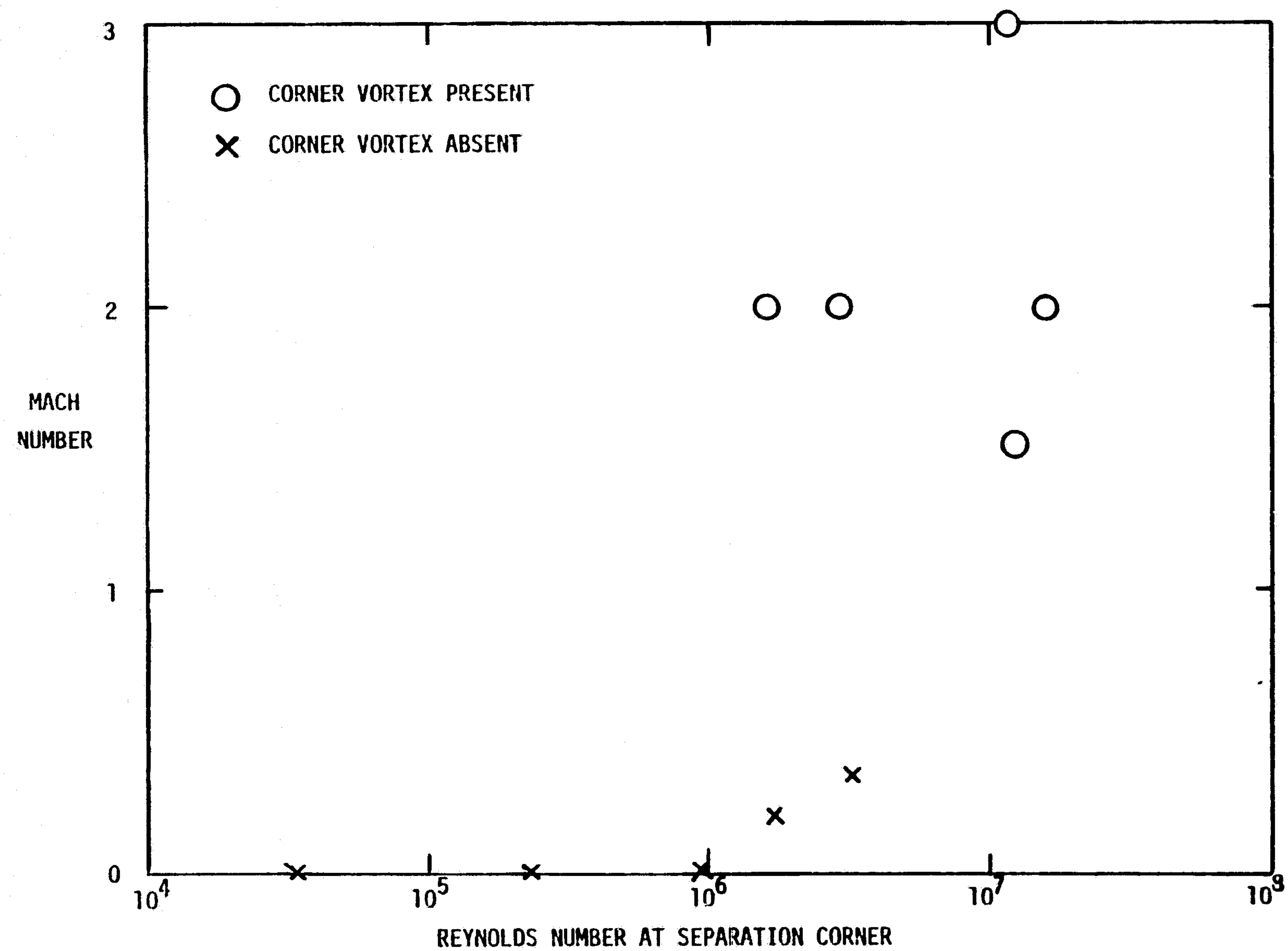


FIGURE 17. SUMMARY OF CONDITIONS OF TESTS

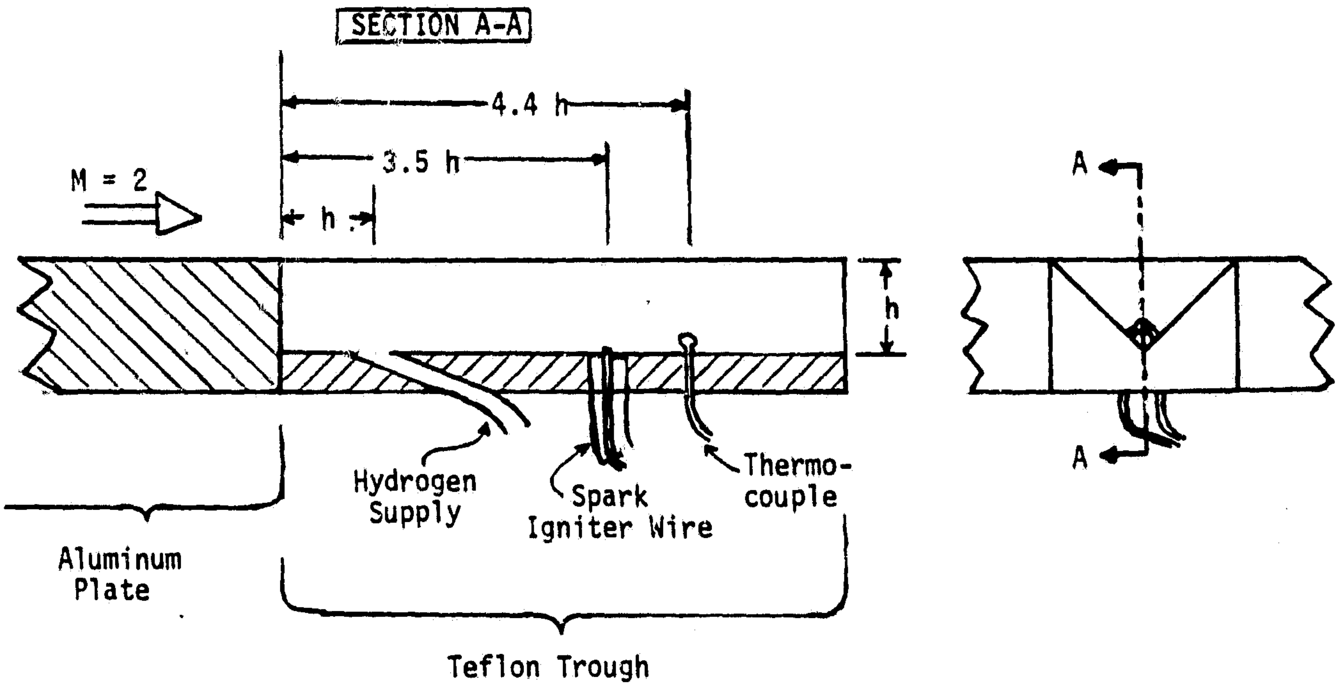


FIGURE 18. DESIGN FOR EARLY SMALL IGNITER TESTS

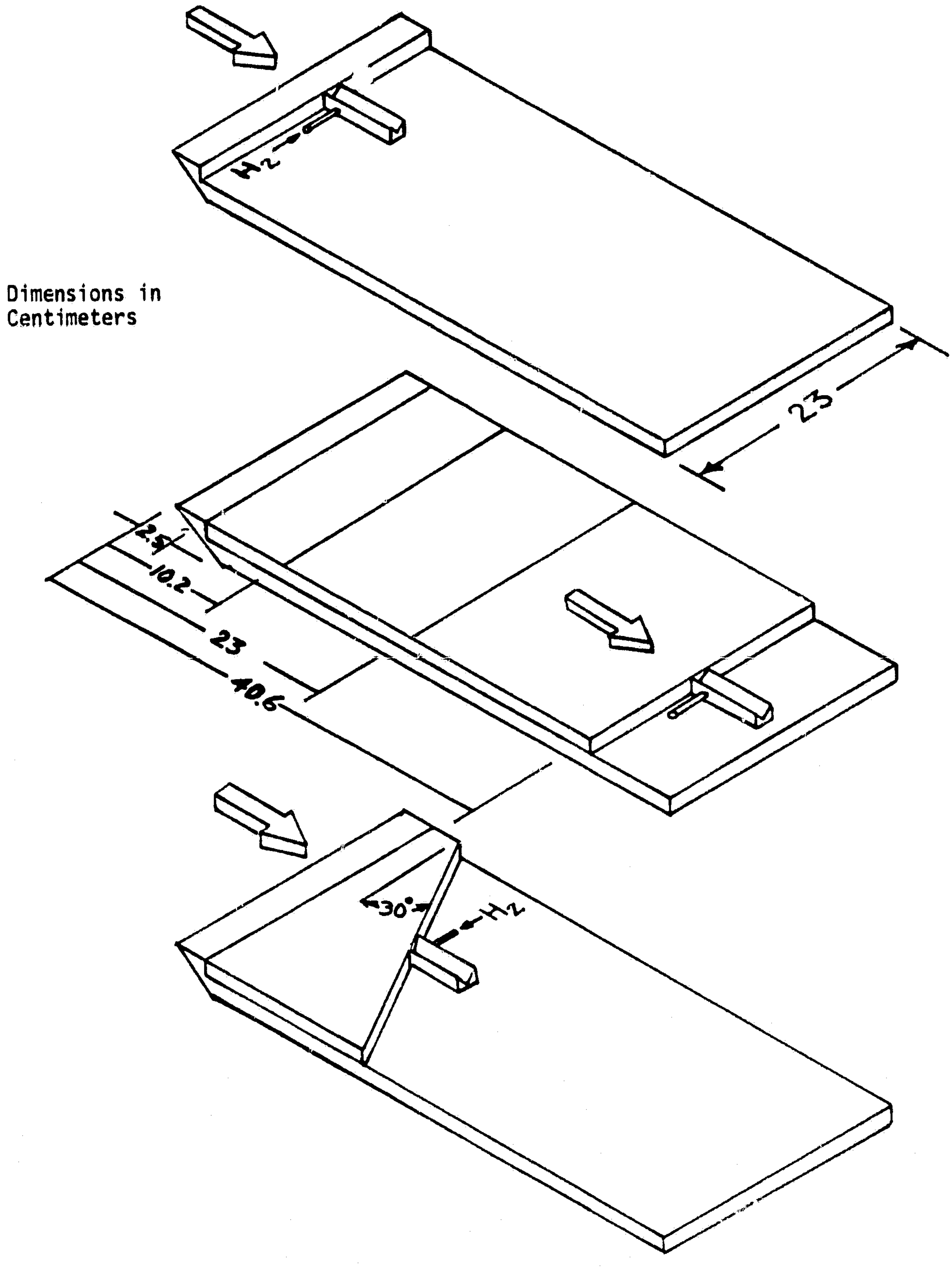
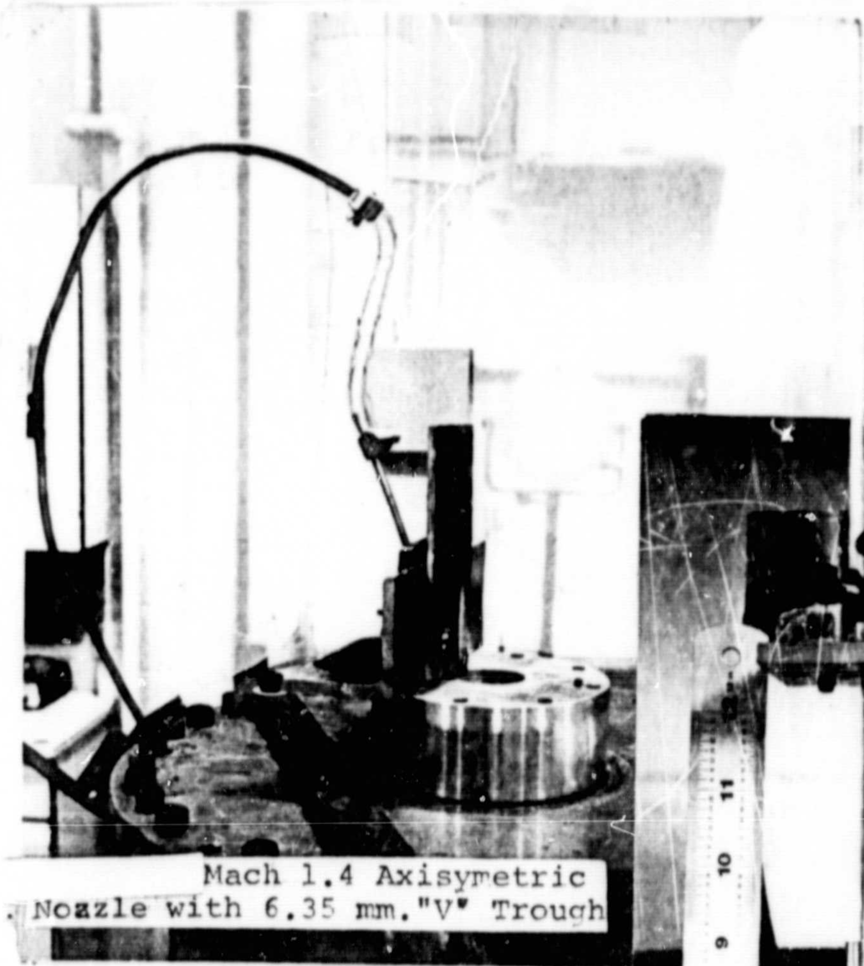
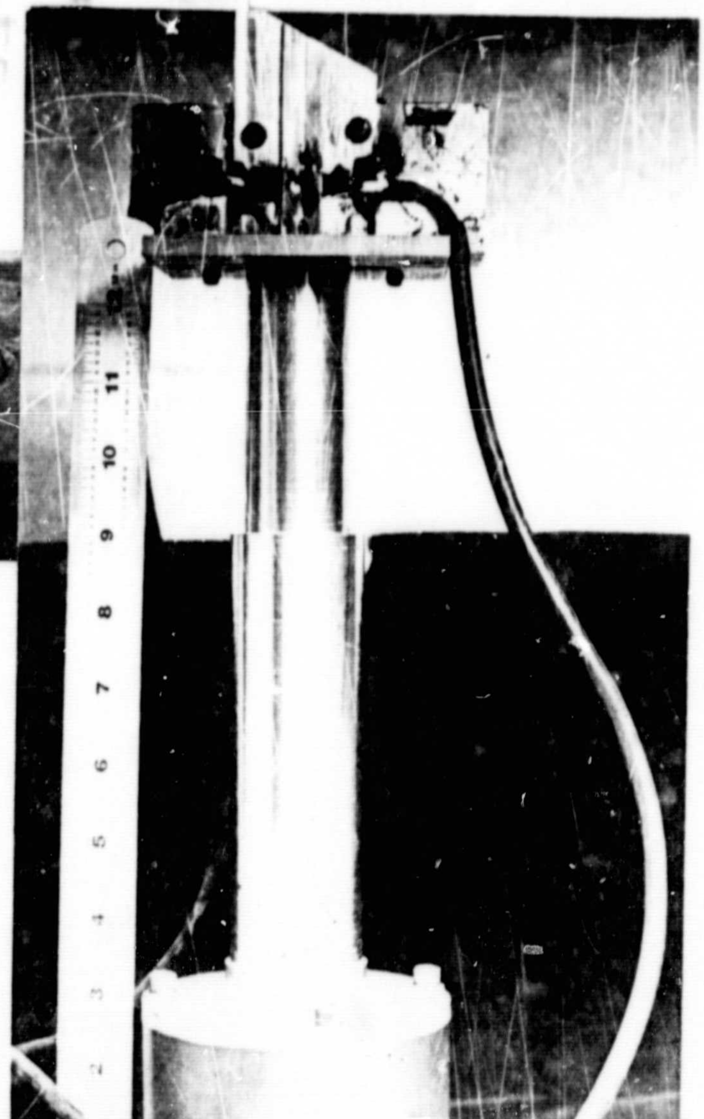


FIGURE 19. TEST FIXTURE FOR SMALL TROUGHS IN THE LARGE SUPERSONIC TUNNEL



Mach 1.4 Axisymetric
Nozzle with 6.35 mm."V" Trough



Mach 1.9 Axisymetric Nozzle
Extension to increase Boundary
Layer thickness and 6.35mm "V"

FIGURE 20. SMALL TROUGH MODELS WITH CIRCULAR NOZZLES

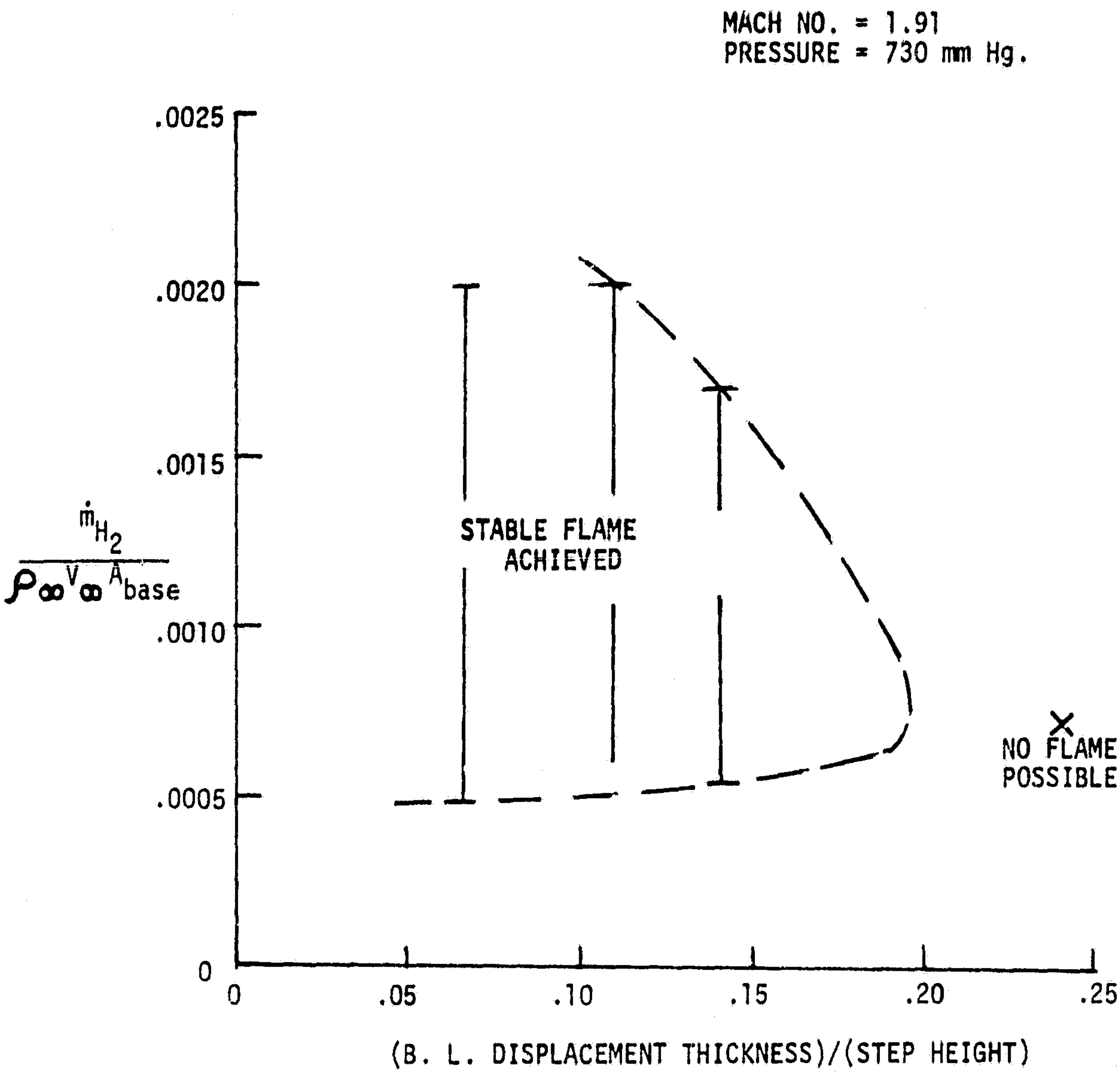


FIGURE 21. FLAMABILITY LIMITS FOR HYDROGEN BURNING IN 90° VEE TROUGH

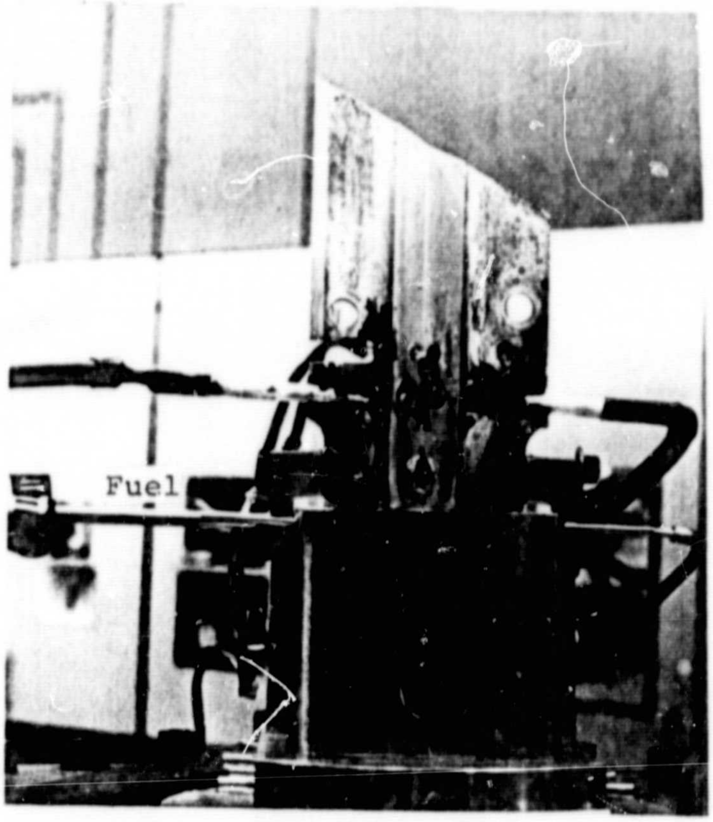
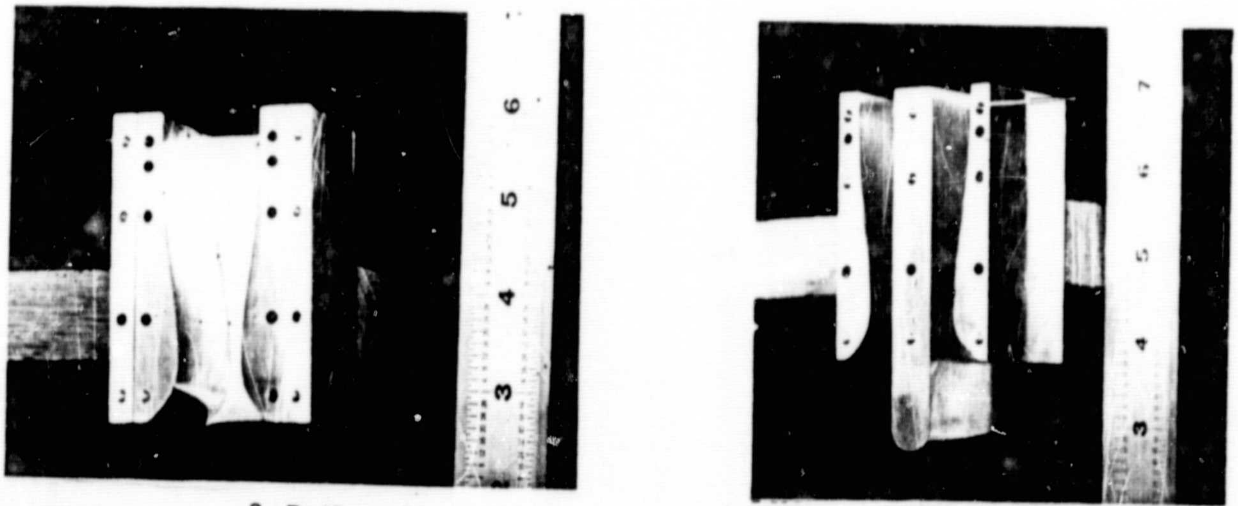


FIGURE 22. SMALL TROUGH MODEL ON RECTANGULAR NOZZLE



2-D Nozzle w/o Centerbody

2-D Nozzle w/ 12.5mm Centerbody

FIGURE 23. RECTANGULAR NOZZLES WITH SIDEWALL REMOVED

(dimensions are in centimeters)

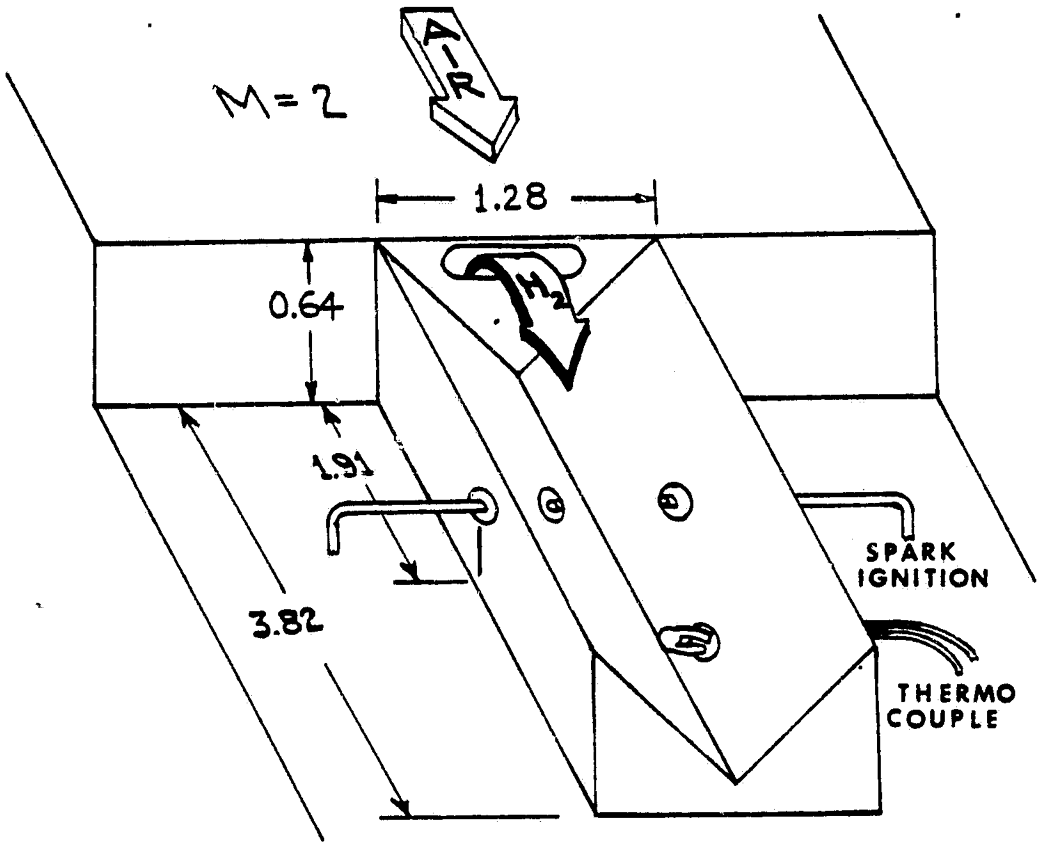


FIGURE 24. IGNITER TROUGH DESIGN

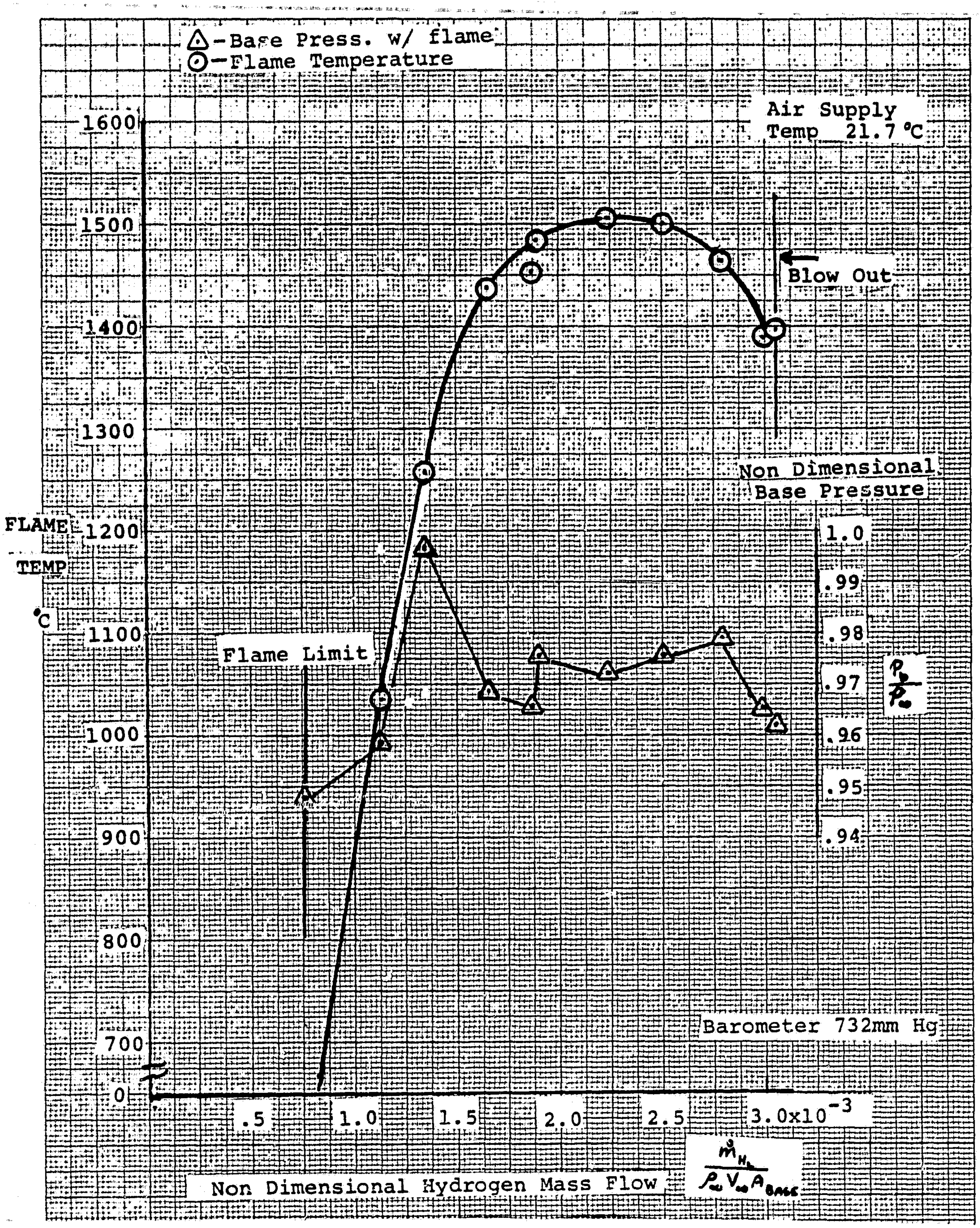


FIGURE 25. TEST RESULTS FOR A 90° TROUGH, 6.35 mm. STEP HEIGHT, AT $M = 1.92$. ($\delta^*/h = .024$)

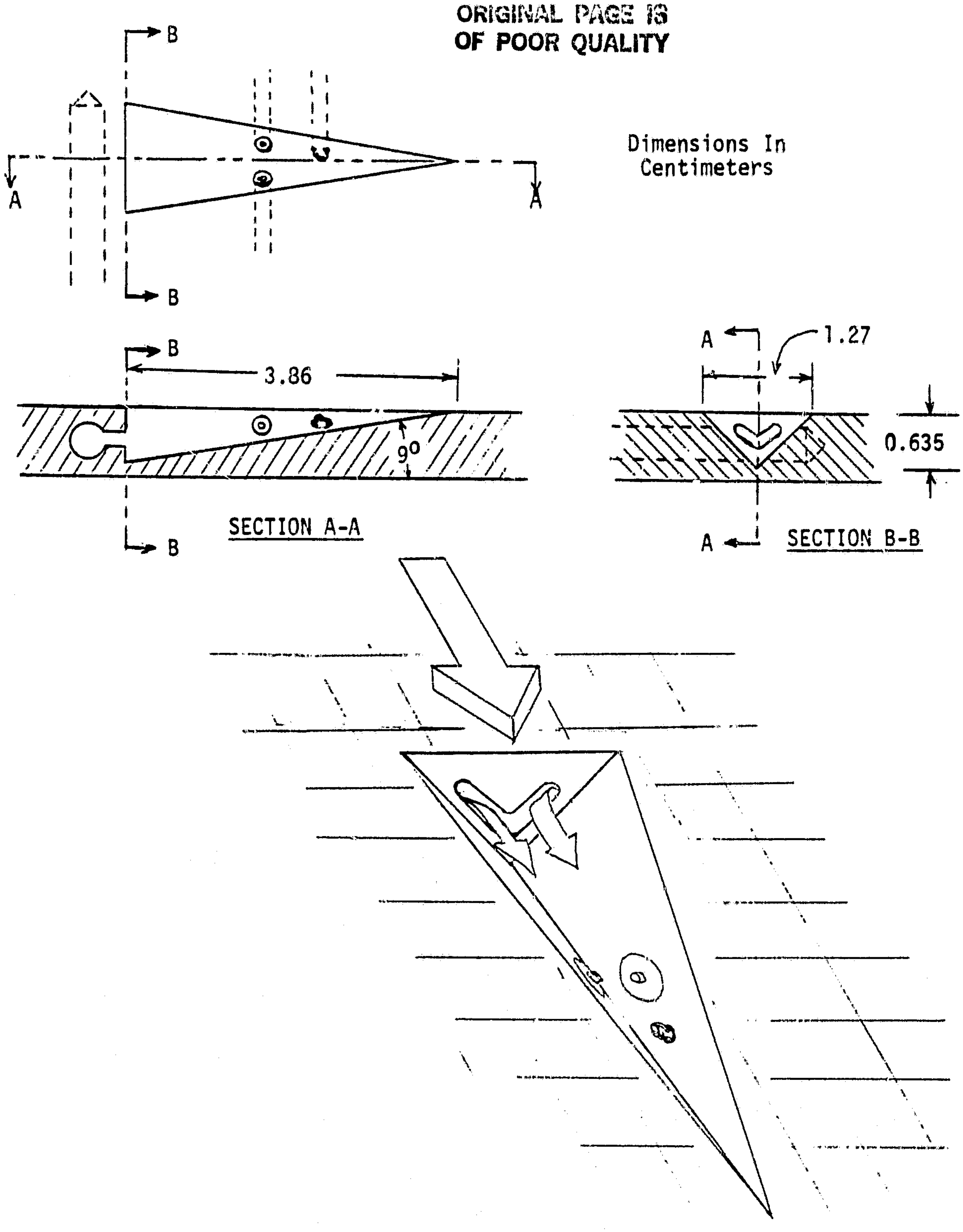


FIGURE 26. DESIGN OF TILTED TROUGH IGNITER

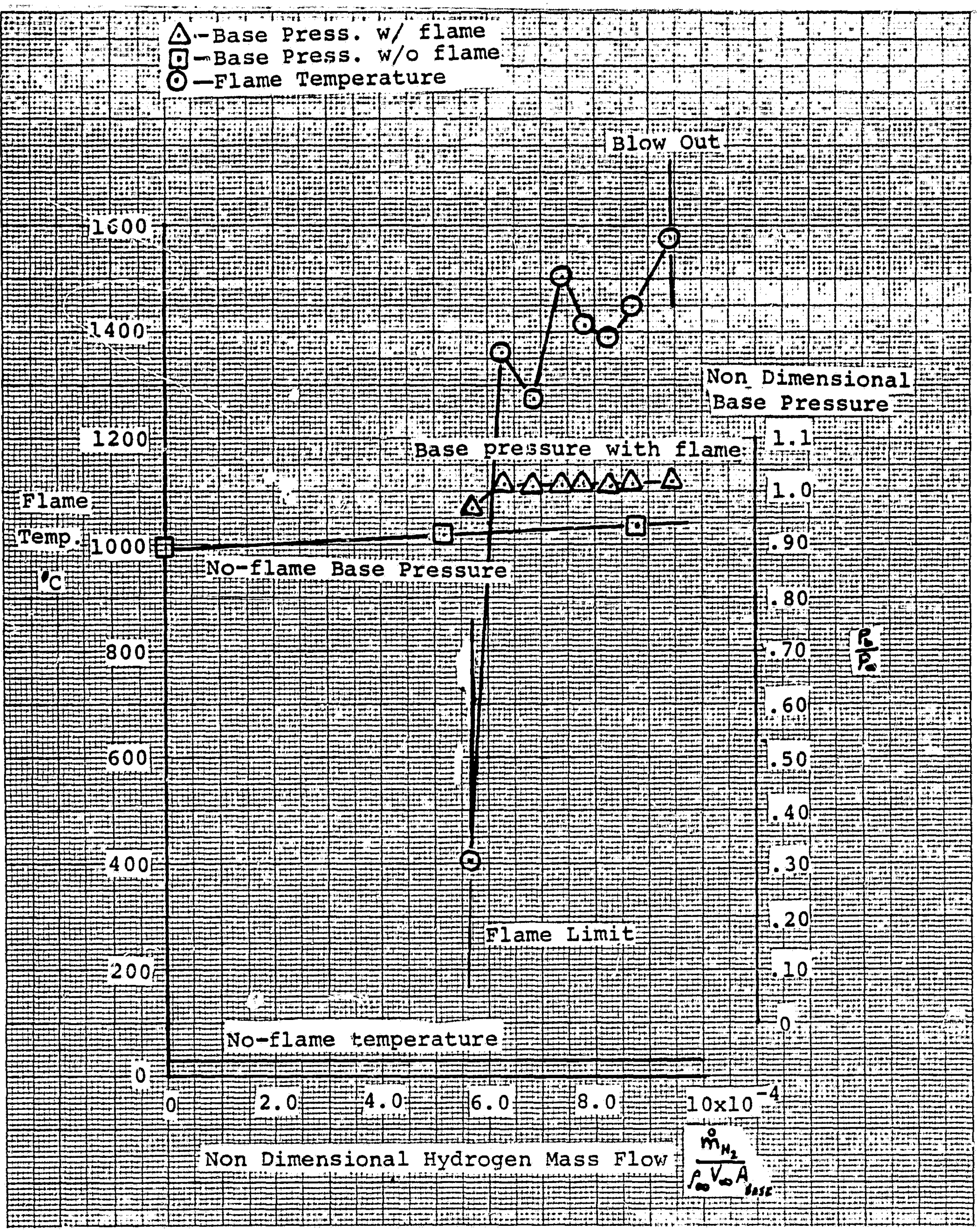


FIGURE 27. TEST RESULTS FOR A 90° TROUGH, 6.35 mm STEP HEIGHT, AT $M = 1.91$, WITH TROUGH TILTED UPWARD 9° ($\delta^*/h = .024$)

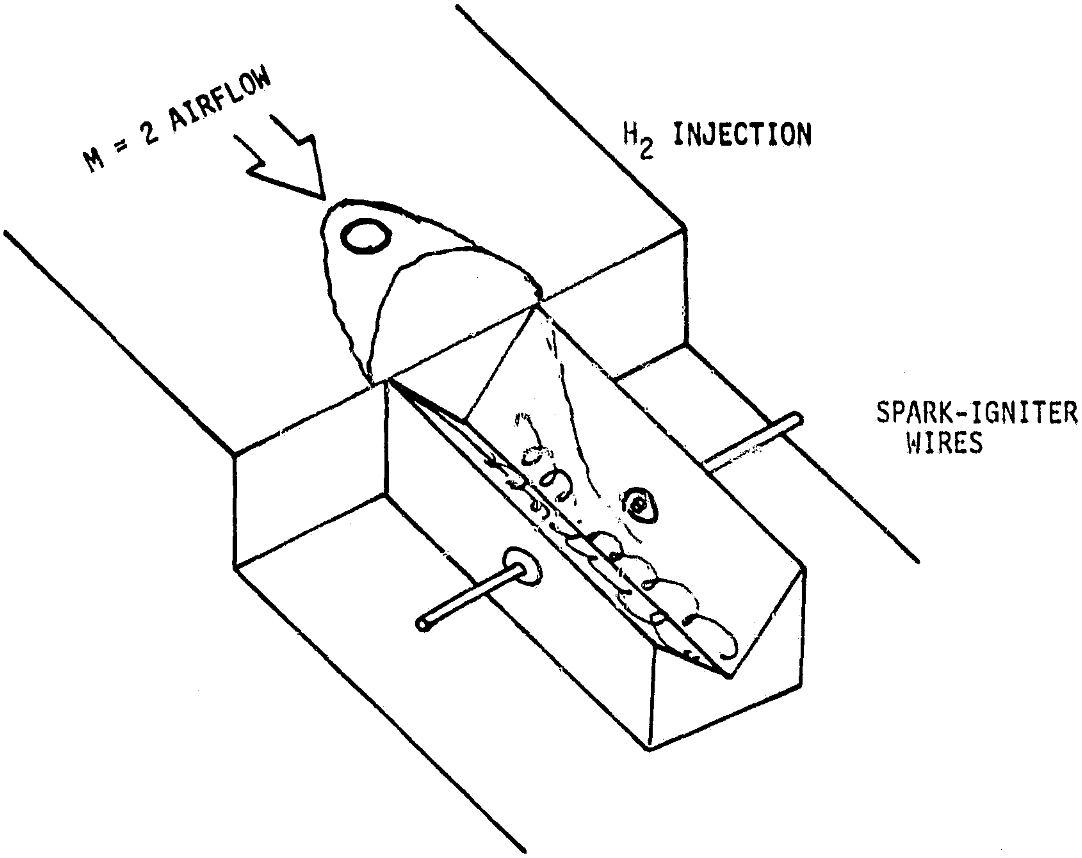


FIGURE 28. TEST SET-UP FOR A VORTEX TROUGH USED AS AN IGNITER FOR AN UPSTREAM COMBUSTIBLE FLOW

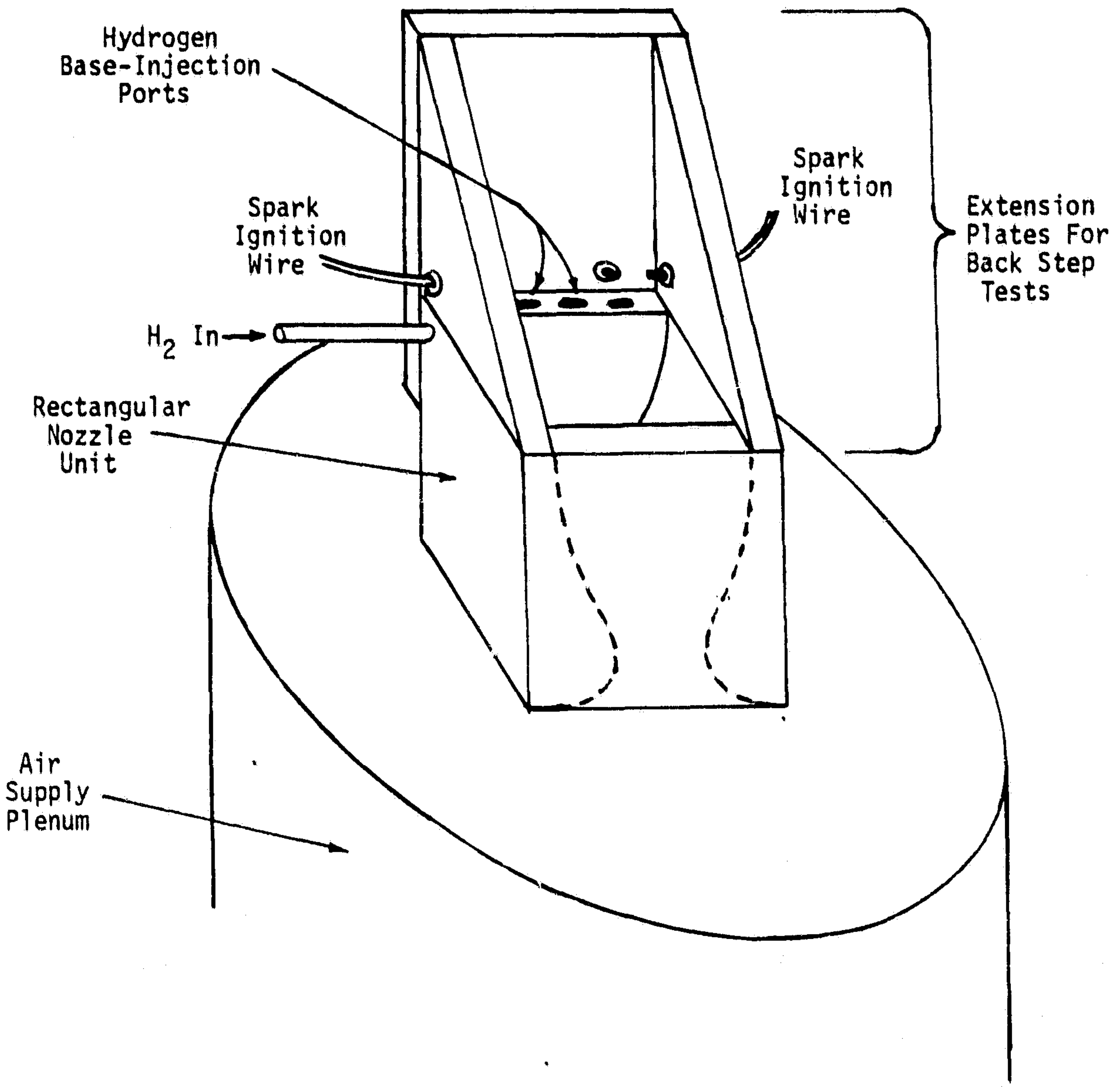


FIGURE 29. SCHEMATIC OF TWO-DIMENSIONAL BACK STEP BURNING TEST

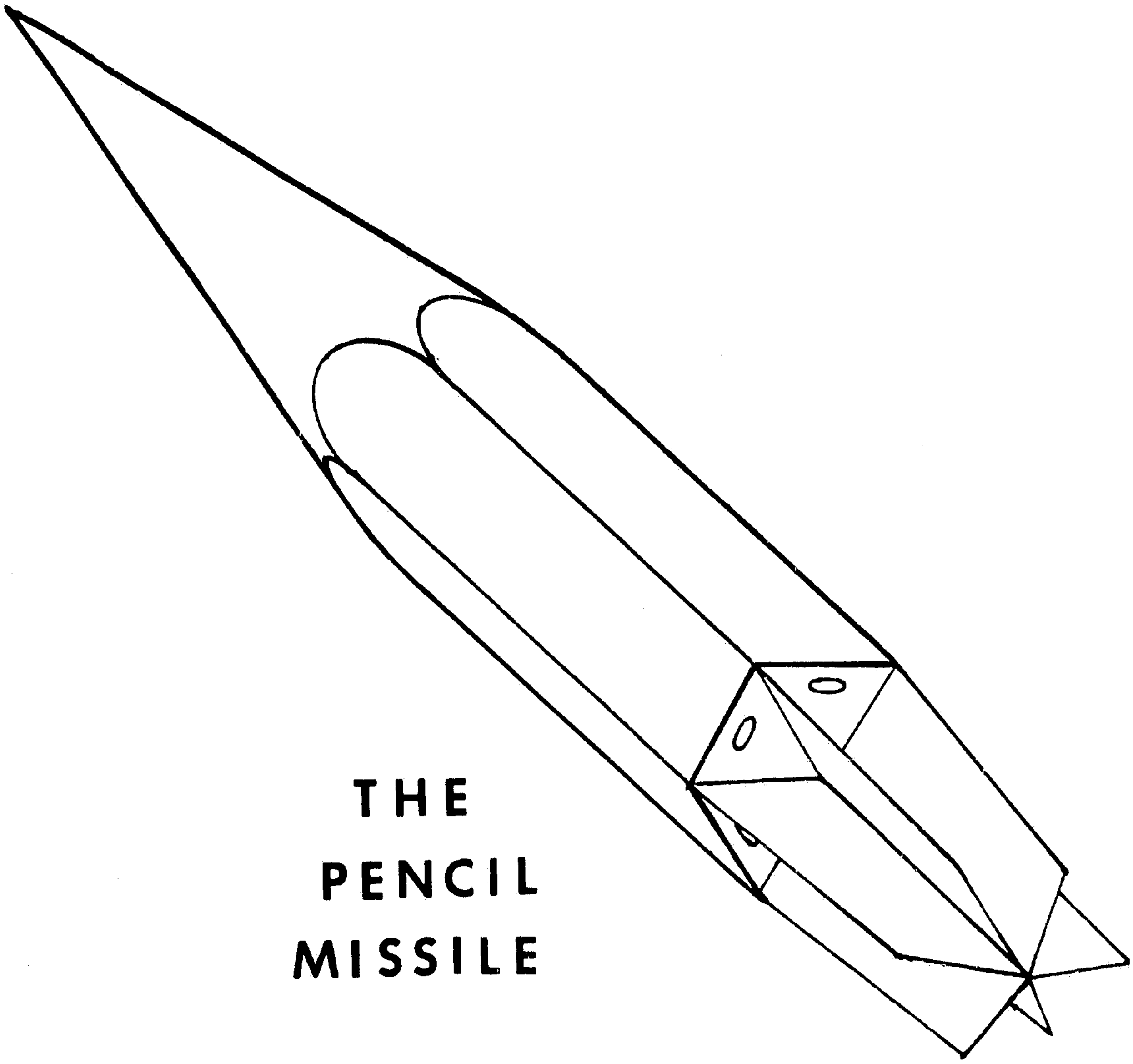


FIGURE 30. PROPOSED MISSLE WITH BASE BURNING BY TROUGH VORTICES

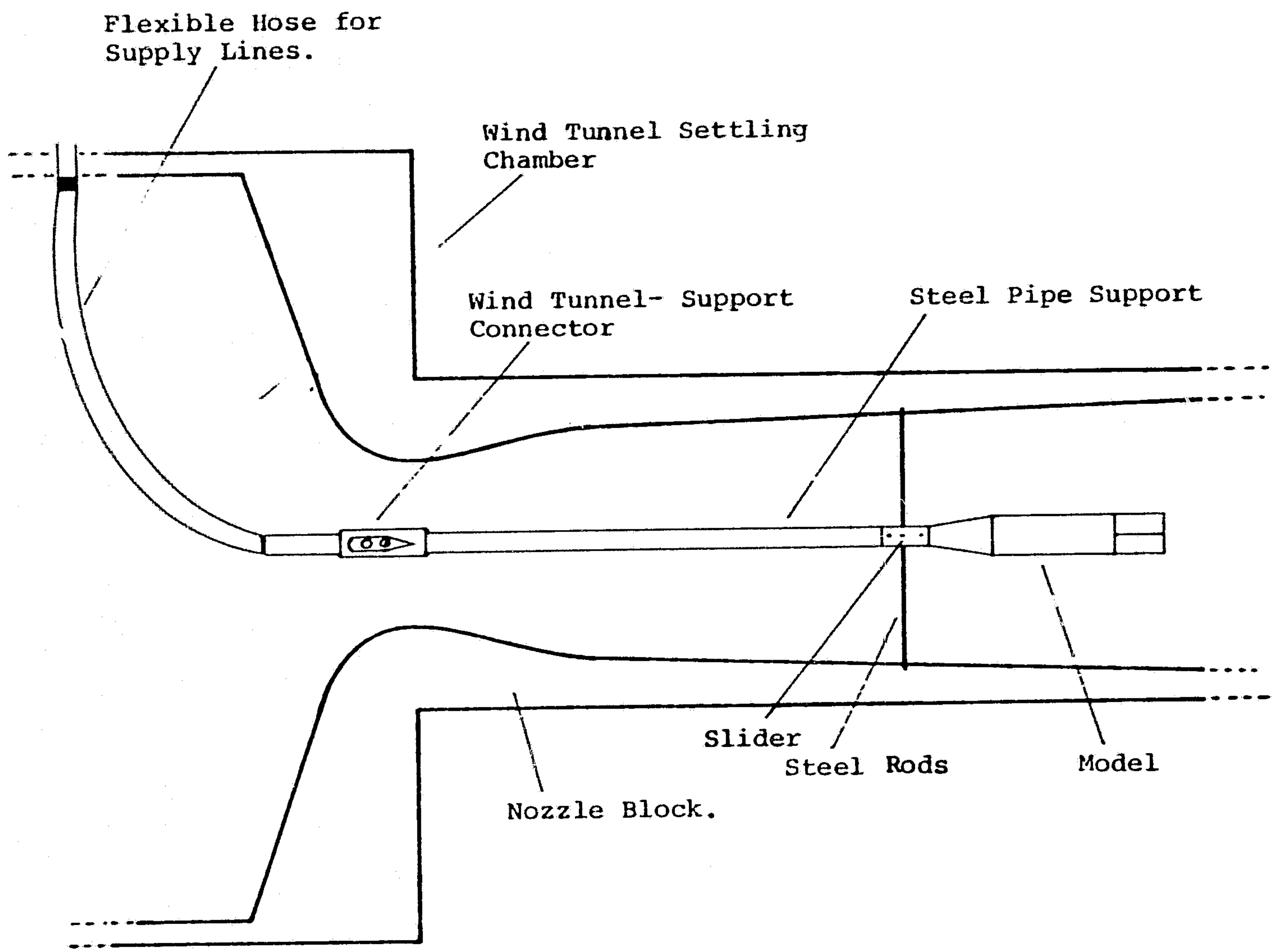


FIGURE 31. METHOD OF MOUNTING THE PENCIL MISSLE MODEL IN THE 23 x 23 cm WING TUNNEL

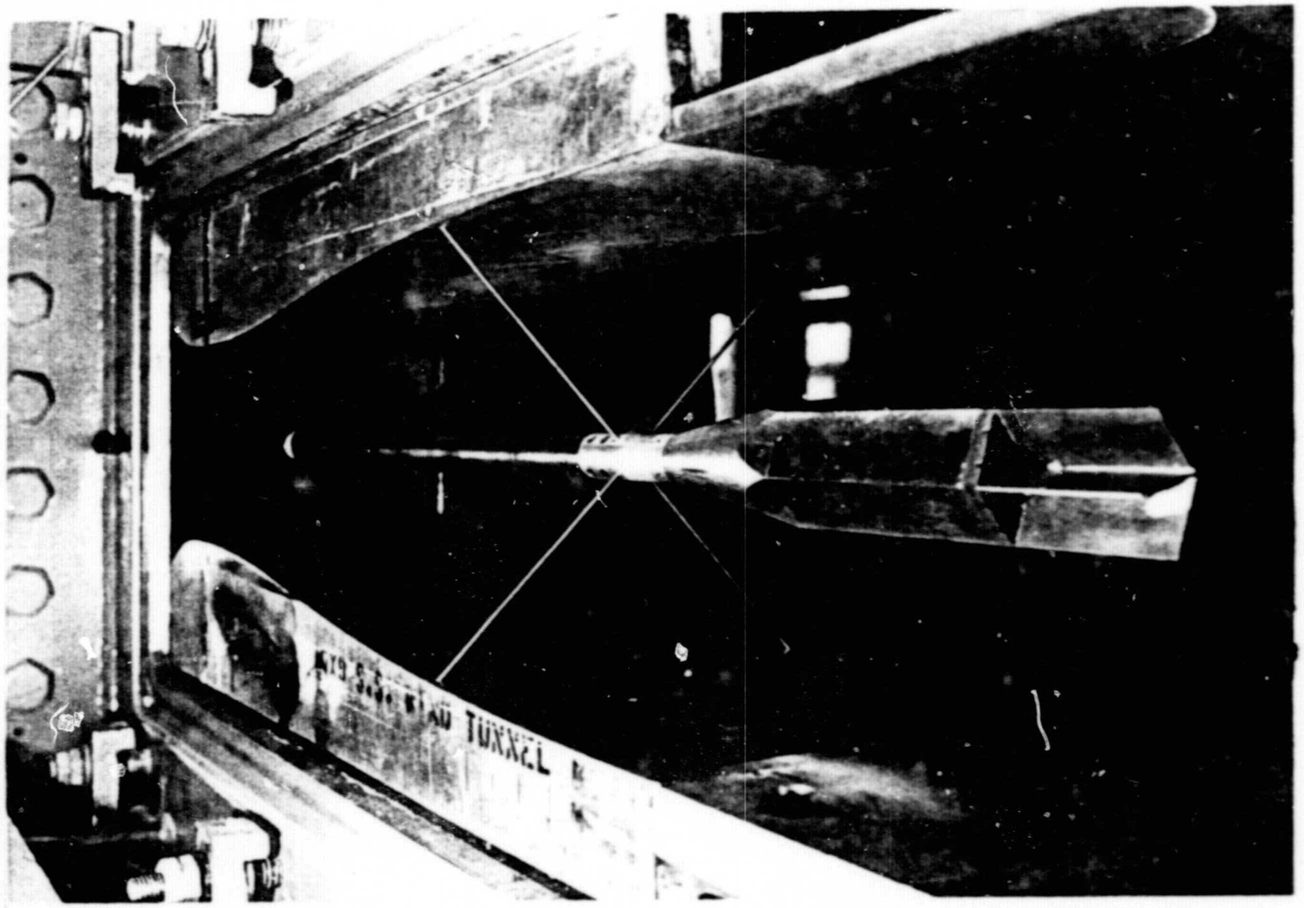


FIGURE 32. Rear End View of Model Installed in Tunnel showing Main Support, Tunnel-Support Interface and Flutter Damping and Model alligning Rods.

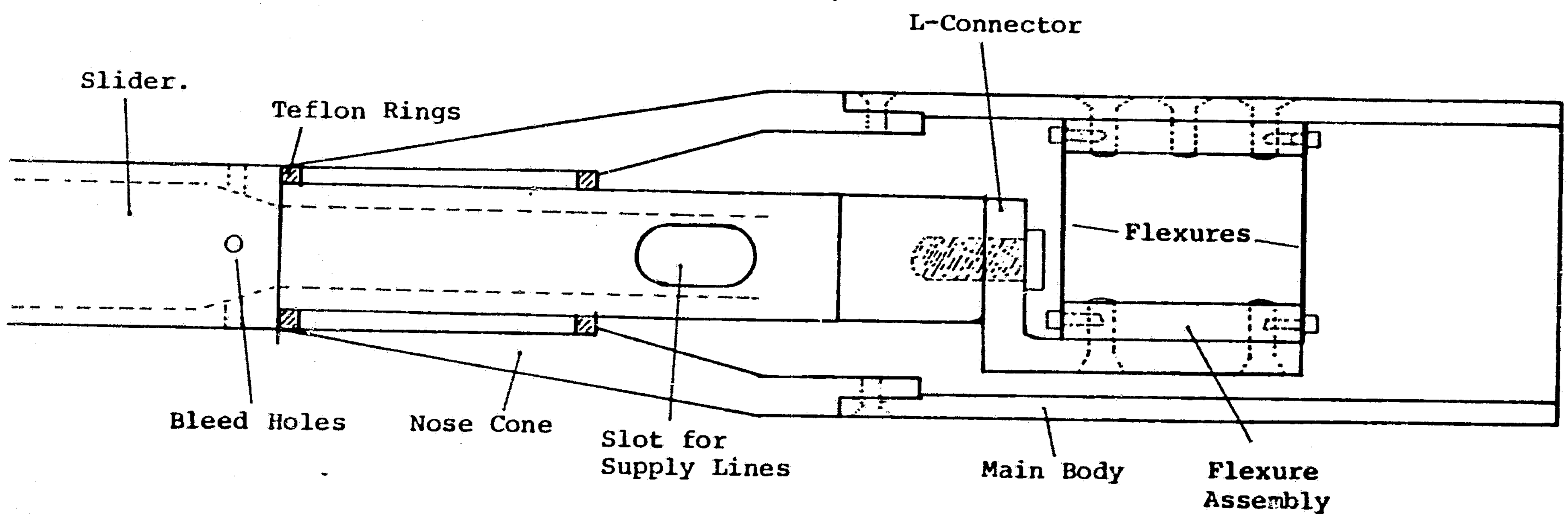


FIGURE 33. INTERNAL MODEL BALANCE FOR THE PENCIL MISSLE

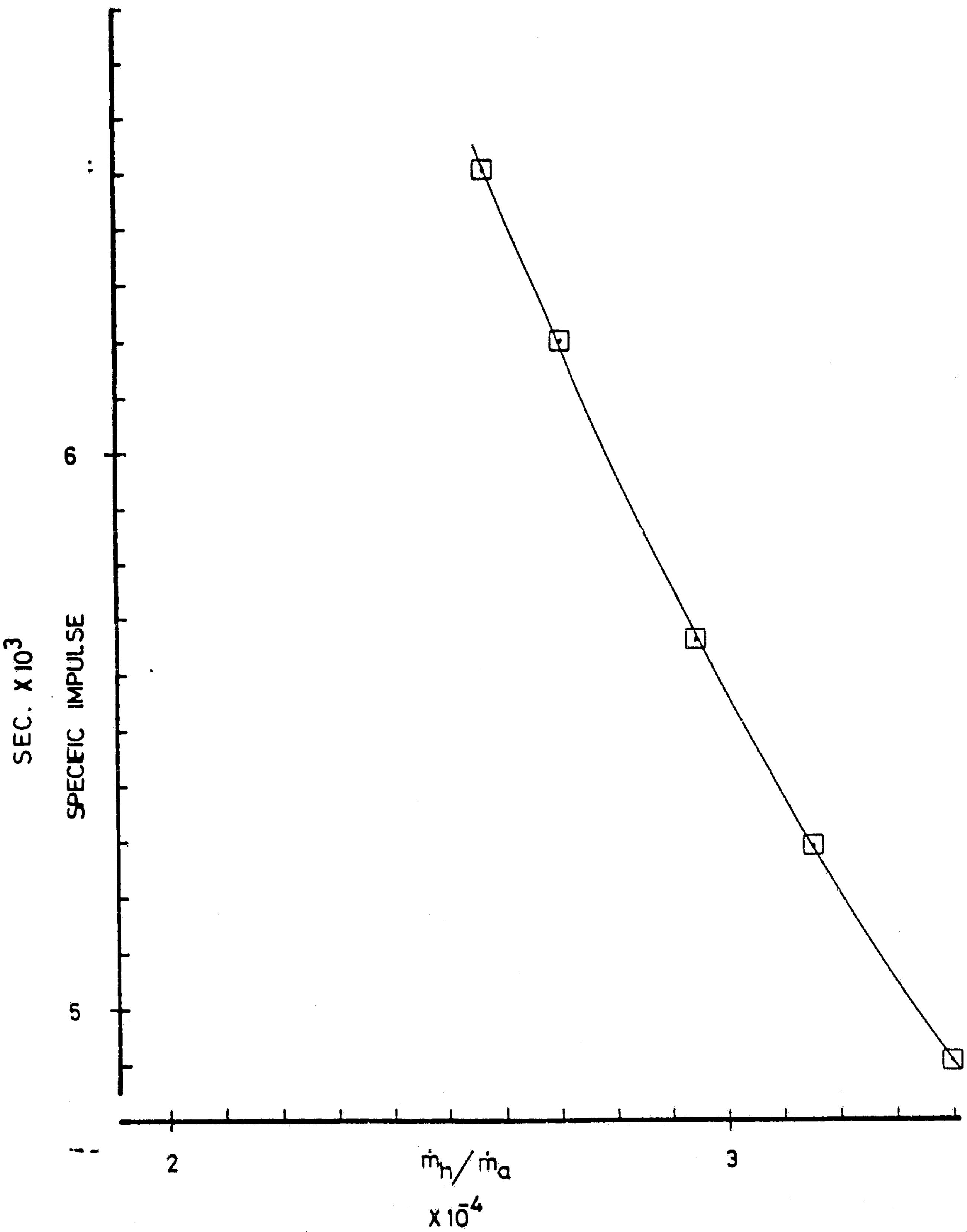


FIGURE 34. SPECIFIC IMPULSE FOR THE PENCIL MISSILE BASE BURNING TESTS

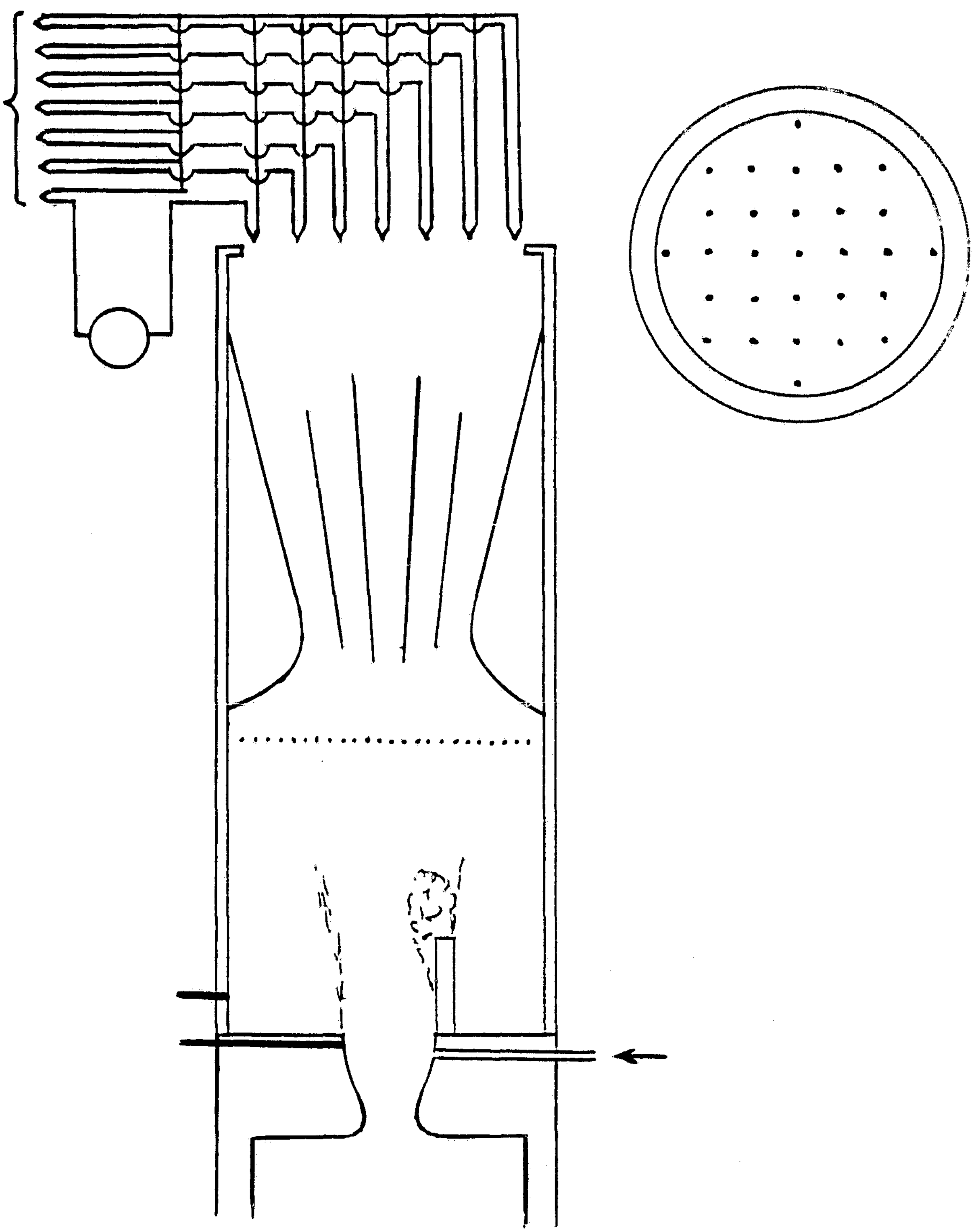


FIGURE 35. FLOW CALORIMETER SET-UP