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A Shock Tube for Transonic and Supersonic Aerodynamic Research

John Hopkins Atkinson

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TUBE

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A SHOCK TUBE FOR TRANSONIC AND SUPERSONIC
AERODYNAMIC RESEARCH

A Thesis
Presented to
the Faculty of the College of Engineering
The University of New Mexico

In Partial Fulfillment
of the Requirements for the Degree
Master of Science in Mechanical Engineering

by
John Hopkins Atkinson

June 1955

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A Thesis

Presented to

the Faculty of the College of Engineering

The University of Toronto

In partial fulfillment

of the requirements for the degree

Master of Science in Mechanical Engineering

By

John Joseph A. ...

June 1911

This thesis, directed and approved by the candidate's committee, has been accepted by the Graduate Committee of the University of New Mexico in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE

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5/27/1955
DATE

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CHAPTER I

INTRODUCTION TO THE PROBLEM

The shock tube as known today consists essentially of a constant cross section tube divided into two chambers by a breakable membrane. The chambers may be filled with like or different gases at various pressures. It was first used by Vielle¹ in France in 1899 about the time the first wind tunnels appeared. Vielle's shock tube consisted of a tube 22 mm. in diameter and 20 feet long which he filled with air to a pressure of about 400 psi. Vielle measured shock velocities up to about twice the velocity of sound. In spite of his success with these experiments the method was completely abandoned until about 1940.

Little was done toward the study of pressure waves released by a bursting diaphragm until Payman and Shepherd² conducted a series of experiments to study the air flow following the bursting of a diaphragm in a shock tube. They used copper-foil diaphragms to sustain high air pressures in the compression chamber of the tube.

¹W. Payman and W. C. F. Sheppard, "Explosion and Shock Waves, VI, The Disturbance Produced by Bursting Diaphragms with Compressed Air," Proceedings Royal Society, Series A, 186:294, 1946.

²Ibid., p. 293.

In 1943 Reynolds³ used the shock tube to produce shock waves of known strength for the calibration of piezo-electric pressure gages which were to be used in blast wave measurements.

In 1943 Smith⁴ made an extensive study of the reflection of shock waves, in which the shock waves were allowed to strike an inclined plate. His purpose was to study the conditions for the onset of Mach reflection.

Since 1950, both in the United States and Canada, interest has increased in the use of the shock tube as an aerodynamic wind tunnel.⁵

Bleakney⁶ and his co-workers are continuing the study of Mach reflection, using a shock tube equipped with a Mach-Zehender interferometer.

Previous studies have been made of the shock wave following the breaking of a diaphragm in a shock tube. This paper is concerned with an investigation of the transonic flow following the shock wave and contact surface rather than with the shock wave itself, with the

³J. Lukasiewicz, "Shock Tube Theory and Applications," National Aeronautical Establishment, Canada, Report 15, 1952, p. 7.

⁴Ibid., p. 7.

⁵Ibid., p. 7.

⁶Walker Bleakney, D. K. Weimer, and C. H. Fletcher, "Shock Tube: A Facility for Investigations of Fluid Dynamics," Review of Scientific Instruments, 20:807, 1949.

In 1943, Hertzberg and his group made the first attempt to produce

shock waves of much higher pressure than those of the explosion of
plexiglass or plastic, which was the first step in the production of
blast wave experiments.

In 1947, Smith made an extensive study of the
reflection of shock waves, in which the shock waves were
allowed to strike a flat surface. His purpose was
to study the conditions for the onset of Mach reflection.
Since 1950, work in the United States has continued

interest has increased in the use of the shock wave
as a tool for the study of high speed flows.

The study of the shock wave and its reflection
study of Mach reflection, which is the first step in the
with a Mach number of 1.5 or higher.

Previous work in this field has been done in the shock wave
following the discovery of the shock wave. It is known that
This paper is concerned with an investigation of the
transonic flow, which is the first step in the study of
flow rather than with the shock wave itself.

U. S. Bureau of Standards, Washington, D. C. 20535
National Bureau of Standards, Washington, D. C. 20535
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1955, p. 1.
1955, p. 1.

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1955, p. 1.

purpose in view that the shock tube might be used as an intermittent wind tunnel in the mechanical engineering laboratory.

A study of shock tube theory indicates that the shock tube as an intermittent wind tunnel has both advantages and disadvantages when compared to the standard wind tunnel. The principal advantage is the ease in producing various flow Mach numbers by setting the pressure ratio of the chambers for the desired Mach number flow within the range available. A second advantage is the fact that the flow at any particular station in the shock tube starts instantaneously. Thus, steady flow patterns may be studied in the process of formation. Principal disadvantages are the difficulty of measuring forces and torques on models, the lack of time for establishment of a desired flow in some cases, and the limit of the Mach number flow to a value less than 1.89 following the primary shock wave.

Purpose of the study. The purpose of building a shock tube was to give the University another research tool and demonstration instrument for aerodynamic studies since the present wind tunnel is not suitable for high velocity research. It was hoped that the shock tube might, when calibrated against theoretical flow data,

prove suitable as a test facility. To determine its usefulness as such a facility it was necessary to devise a method for measuring local velocities, investigate size effects of models in the flow and provide a check against standard wind tunnel data. The shock tube was to be built as simply and cheaply as possible.

Shock tube flow consists of a steady flow period following the primary shock wave developed at the bursting of the diaphragm, and a second steady flow period following passage of the discontinuity or contact surface between the high and low pressure gases. Preliminary investigation indicated that it might be possible to measure, by the use of wedges and airfoils, the flow in the cold gas following the contact surface. After the diaphragm is broken this region moves toward the low pressure end of the shock tube followed by a period of steady flow. The duration of steady flow is dependent on the shock tube length, the pressure ratio of the two gases, and the location of the test stations.

In this study the high pressure section of the shock tube is referred to as the chamber, and the low pressure section as the channel. Using air as the gas in both the chamber and the channel, and pressure ratios up to 60, it was considered likely that flow Mach numbers of

1.5 to 2.0 could be measured in the channel section.

Shock tube theory predicts a region of steady flow preceding the passage of the contact surface. This flow region follows the primary shock wave developed when the diaphragm is broken. A greater pressure ratio is necessary here for a given Mach number than in the flow following the contact surface, and there is a much shorter period of steady flow. The upper limit in Mach number is 1.89. Using air as the gas in both the chamber and channel, it appeared feasible to obtain flow Mach numbers up to 1.20 with a pressure ratio of 150.

After a workable shock tube is accomplished it is necessary to calibrate it for use as a test facility. Rather than go to the expense of an electronic method of calibration of the flow velocity between two points in the channel, which would only indicate the velocity of the passage of the primary shock wave, plans were made to measure the angle of the bow wave on a wedge from shadow-graph pictures. This meant that only supersonic flows could be measured. It was felt, however, that if these measured flows were consistent the curve could be extrapolated into the region below Mach 1.00.

It also appeared feasible to measure the Mach wavelet angle at striations on a wedge, if sufficient density were available, as a check against this bow wave

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It also appeared feasible to measure the Mach wavelet angle at stations on a wedge, if sufficient density were available, as a check against this bow wave

measurement. If the striation angles were visible it would be possible to measure local velocity on an airfoil or other object in the flow stream.

Wedges of different cross sectional areas might also be installed in the test section of the tube in order to observe the effect on the air flow.

Following the calibration of the shock tube with measured velocities plotted against the pressure ratio, a standard NACA airfoil for which wind tunnel information was available might then be installed in the shock tube and the critical Mach number determined.

measurement. If the object is not visible at some

be possible to measure the distance as an object or

other object in the line of sight.

Figure 1 shows the method of measuring the distance

also be measured in the line of sight of the object.

to observe the effect of the object.

Following the completion of the measurement the

measured distance is plotted against the pressure ratio.

a standard NACA airfoil for all of which the distance

was available and a curve is drawn through the points

and the critical pressure ratio is determined.

Figure 2 shows the results of the measurements for the

airfoil.

CHAPTER II

THE THEORY OF THE IDEAL SHOCK TUBE

Shock tube notation. Figure 1 illustrates schematically the shock tube after the diaphragm has been broken. It shows the tube of uniform cross section, closed at one end, this a gas-tight diaphragm and indicates the pressures, velocities and speed of sound expected in each part of the shock tube.

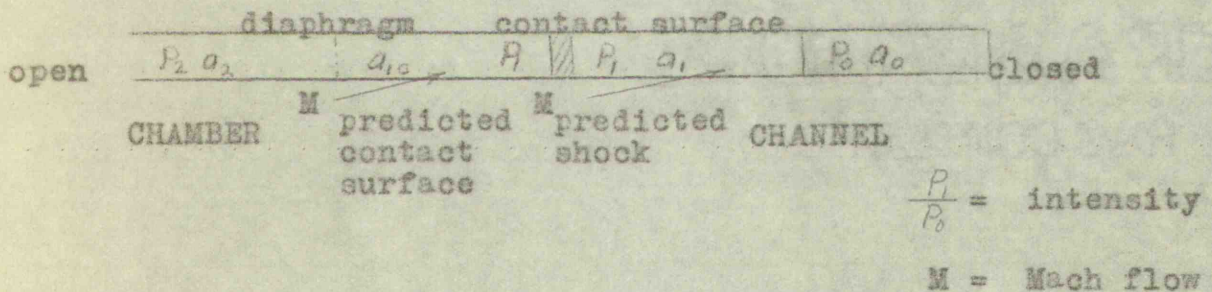


Figure 1. Shock Tube after the Diaphragm has been Ruptured

P_0 - gas pressure in the low pressure region of the shock tube

P_1 - gas pressure following the primary shock wave

P_2 - gas pressure in the chamber or high pressure region of the shock tube after rupture of the diaphragm (Atmospheric pressure for all tests in this paper)

a_0 - velocity of sound in the stationary gas

a_1 - velocity of sound in the gas following the primary shock wave

THE THEORY OF THE IDEAL GASES

Green's theorem. Let V be a volume bounded by a surface S .

Let ρ be the density of the gas, \mathbf{v} the velocity, p the pressure, \mathbf{f} the force per unit mass. Then the equations of motion are

$$\rho \frac{d\mathbf{v}}{dt} = -\nabla p + \rho \mathbf{f}$$

where $\frac{d}{dt} = \frac{\partial}{\partial t} + \mathbf{v} \cdot \nabla$ is the material derivative. The continuity equation is

$$\frac{d\rho}{dt} + \rho \nabla \cdot \mathbf{v} = 0$$

where $\nabla \cdot \mathbf{v}$ is the divergence of the velocity. The equation of state for an ideal gas is

$$p = \rho R T$$

where R is the gas constant and T is the temperature.

Let \mathbf{v} be the velocity of the gas, \mathbf{f} the force per unit mass, p the pressure, ρ the density. Then the equations of motion are

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Let \mathbf{v} be the velocity of the gas, \mathbf{f} the force per unit mass, p the pressure, ρ the density. Then the equations of motion are

$$\rho \frac{d\mathbf{v}}{dt} = -\nabla p + \rho \mathbf{f}$$

where $\frac{d}{dt} = \frac{\partial}{\partial t} + \mathbf{v} \cdot \nabla$ is the material derivative. The continuity equation is

$$\frac{d\rho}{dt} + \rho \nabla \cdot \mathbf{v} = 0$$

where $\nabla \cdot \mathbf{v}$ is the divergence of the velocity. The equation of state for an ideal gas is

$$p = \rho R T$$

where R is the gas constant and T is the temperature.

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$$p = \rho R T$$

where R is the gas constant and T is the temperature.

a_{1c} - velocity of sound in the gas following the contact surface

a_2 - velocity of sound in the high pressure air before the diaphragm is ruptured

U - velocity of propagation of the primary shock wave

Contact surface - The contact surface is the region of mixing gases at the diaphragm that moves down the channel following rupture of the diaphragm. The density and consequently the temperature is different although the pressure remains the same in this region. The gases following the passage of this contact surface are cooled.

The process following the rupture of the diaphragm.

In analyzing the process following the rupture of the diaphragm, the assumptions are made that the diaphragm is a plane barrier which disappears instantaneously, and that the gases in the chamber and channel obey the ideal gas laws.

When the diaphragm is suddenly broken a shock wave is formed in the channel, or low pressure portion of the tube, advancing along the tube away from the diaphragm. The shock wave is eventually reflected from the closed end of the tube. The breaking of the diaphragm also causes a rarefaction wave to be formed in the high pressure section of the tube, and moves back in the high pressure gas until it is reflected at the end of the compression chamber. The sequence of events may be shown

0.1 - velocity of sound in the gas following the constant volume

0.2 - velocity of sound in the gas following the constant pressure

U - velocity of propagation of the shock wave

Contact surface - The contact surface is the surface of mixing between the two gases. The contact surface is the surface of the shock wave. The density and temperature of the gas are different on either side of the contact surface. The pressure is the same on both sides. The velocity of the contact surface is the same as the velocity of the shock wave.

The process follows the law of the shock wave

In analyzing the process following the shock wave, the diaphragm, the shock wave, and the gas are considered. In a plane parallel with the shock wave, the gas is at rest. The gas is at rest in the shock wave and on either side of the shock wave. The gas is at rest in the shock wave and on either side of the shock wave.

When the diaphragm is suddenly removed, a shock wave is formed in the gas. The shock wave moves along the tube, dividing the gas into two regions. The shock wave is a surface of discontinuity. The pressure and temperature are different on either side of the shock wave. The velocity of the shock wave is the same as the velocity of the contact surface. The shock wave moves along the tube, dividing the gas into two regions. The shock wave is a surface of discontinuity. The pressure and temperature are different on either side of the shock wave. The velocity of the shock wave is the same as the velocity of the contact surface.

in the following way:¹

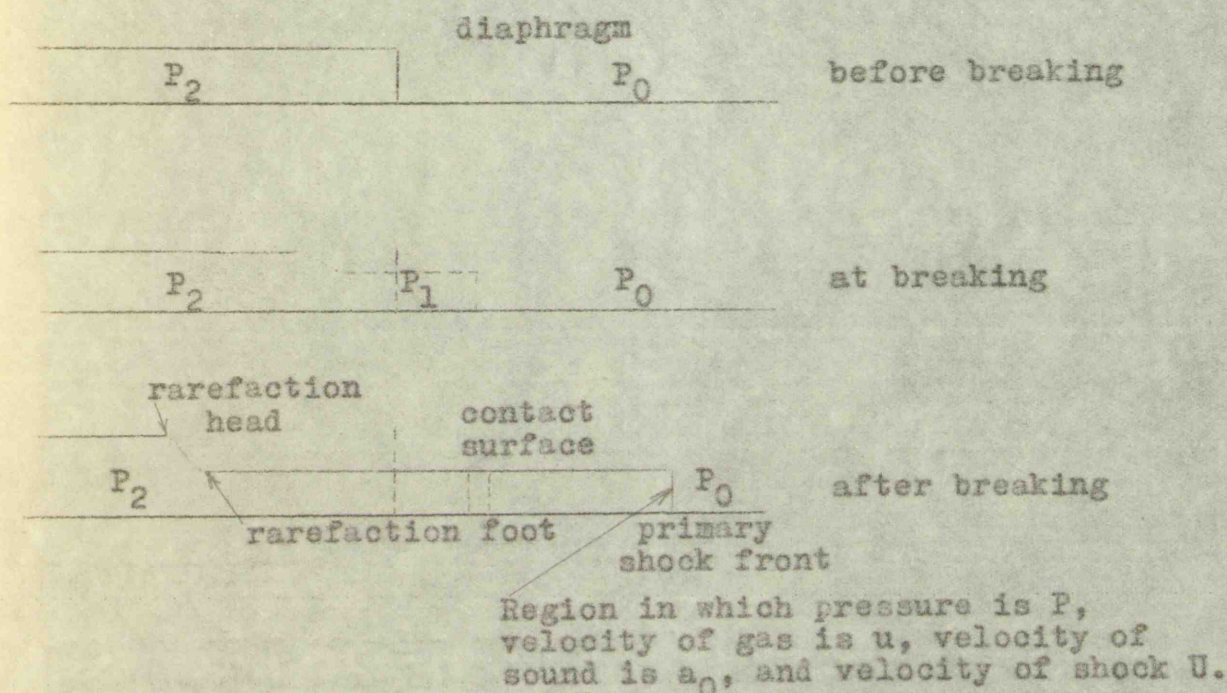


Figure 2. Development of Flows following the breaking of the Diaphragm

From the Rankine-Hugoniot equations we have the relationship between the particle velocity following the shock front and the pressure ratio $\frac{P_1}{P_0}$ at the shock front.

This is:

$$\frac{u^2}{a_0^2} = \frac{2}{\gamma} \frac{(\gamma - 1)^2}{\gamma - 1 + (\gamma + 1)\gamma} \quad (\text{II} - 1)$$

where $\gamma = \frac{P_1}{P_0}$, $a_0 = \sqrt{\gamma \frac{P_0}{\rho_0}}$ and u = particle velocity following the shock front.

¹C. W. Lampson, "Resume' of the Theory of Plane Shock and Adiabatic Waves with Applications to the Theory of the Shock Tube," Ballistics Research Laboratory, Technical Note, No. 139:26-29, Aberdeen, Maryland, 1950.

in the following way:

P_2	Before breaking	Diagram
P_1	at breaking	
P_2	After breaking	Diagram

Region in which pressure is P_1
 velocity of gas is u_1 , velocity of
 shock is u_s , and velocity of shock V

Figure 2. Development of flow following the
 breaking of the diaphragm

From the Rankine-Hugoniot equations we have the
 relationship between the pressure ratio following the
 shock front and the pressure ratio P_2/P_1 of the shock front.

This is:

$$\frac{P_2}{P_1} = \frac{2\gamma}{\gamma+1} \frac{V^2}{u_s^2} - \frac{\gamma-1}{\gamma+1}$$

where $\gamma = \frac{P}{\rho u^2}$ and V is the velocity following
 the shock front.

¹O. W. Lehmann, "Structure of the Shock of 70",
 Shock and Detonation Waves, 1950, p. 10.
 of the Shock Tube, J. Appl. Phys., 21, 1950, p. 10.
 cal Note, No. 100, 1950, p. 10.

The high pressure region in the chamber behind the diaphragm is propagating a rarefaction wave because of the relief of pressure by the bursting of the diaphragm. The velocity of the particles in the rarefaction wave where the pressure is P is given by

$$u = \pm \int \sqrt{\frac{dP}{d\rho}} \frac{d\rho}{\rho} \quad u = \pm \frac{2}{\gamma-1} a_2 \left[1 - \left(\frac{P_1}{P_2} \right)^{\frac{\gamma-1}{2\gamma}} \right]$$

where P_2 is the pressure of the region into which the wave advances and a_2 is the velocity of sound in this region.

Now $P = P_0 \frac{P_1}{P_0} = P_0 \gamma$

so the equation becomes

$$u = \pm \frac{2}{\gamma-1} a_2 \left[1 - \left(\gamma \frac{P_0}{P_2} \right)^{\frac{\gamma-1}{2\gamma}} \right] \quad (\text{II} - 2)$$

These two particle velocities must be equal at the diaphragm after breaking; otherwise a local region of vacuum or high pressure will develop in time. The area and particle velocity is the same on each side of the diaphragm but the density of the gases is different causing a discontinuity in both temperature and density but holding the same pressure. So, equating the two particle velocities we have

$$\frac{2a_0^2(\gamma-1)^2}{\gamma[\gamma-1+(\gamma+1)\gamma]} = \frac{4a_1}{(\gamma-1)^2} \left[1 - \left(\gamma \frac{P_0}{P_2} \right)^{\frac{\gamma-1}{2\gamma}} \right]^2$$

The high pressure region in the contact between the

diaphragm is propagating in the direction of the wave motion of the diaphragm. The velocity of the particles in the contact region is

where the sign is \pm in given by

$$u = \pm \sqrt{\frac{2\Delta p}{\rho}} \left[1 - \left(\frac{p}{p_0} \right)^{\frac{\gamma}{\gamma-1}} \right]$$

where p_0 is the pressure of the region free of wave advance and Δp is the velocity of sound in this region.

Now $p = p_0 \frac{\rho}{\rho_0} = p_0 \left(\frac{u}{c} \right)^{\frac{2\gamma}{\gamma-1}}$

so the equation becomes

$$u = \pm \sqrt{\frac{2\Delta p}{\rho}} \left[1 - \left(\frac{u}{c} \right)^{\frac{2\gamma}{\gamma-1}} \right]$$

These two particle velocities must be equal to the diaphragm after it has stopped oscillating. In the case of vacuum or high pressure will develop in time. The area and particle velocity is the same on each side of the diaphragm but the quantity of the wave is different causing a discontinuity in both temperature and density but holding the same pressure. So, setting the two

particle velocities as u we

$$\frac{2\Delta p}{\rho} \left(\frac{u}{c} \right)^{\frac{2\gamma}{\gamma-1}} = \frac{2\Delta p}{\rho} \left(\frac{u}{c} \right)^{\frac{2\gamma}{\gamma-1}}$$

Assume $a_0 = a_2$ which means that sufficient time must elapse after adjusting the chamber pressures for the temperature to equalize. Taking the square root of each side of the above equation we have

$$\frac{\sqrt{2}(\gamma-1)}{\sqrt{\gamma[\gamma-1+(\gamma+1)\gamma]}} = \frac{2}{\gamma-1} \left[1 - \left(\gamma \frac{P_0}{P_2} \right)^{\frac{\gamma-1}{2\gamma}} \right] \quad (\text{II} - 3)$$

Assume $\gamma = 1.4$, then

$$\frac{\gamma-1}{\sqrt{7(1+6\gamma)}} = 1 - \left(\gamma \frac{P_0}{P_2} \right)^{\frac{1}{7}} \quad (\text{II} - 4)$$

$$\text{or } \frac{P_2}{P_0} = \frac{\gamma}{\left[1 - \frac{\gamma-1}{\sqrt{7(1+6\gamma)}} \right]^7}$$

This equation then gives the ratio of pressures in the chamber and channel, before the diaphragm is broken, necessary to establish a shock wave of pressure traveling down the tube. This relationship is plotted in Figure 3. See Table I.

The pressure distribution in the tube following the breaking of the diaphragm is indicated in Figure 4. The rarefaction has a head pressure or leading point at the chamber pressure while the foot of the rarefaction is at the primary shock pressure of P . Figure 4 shows the rarefaction both as to pressure and position with R as the head, and F as the foot. The foot of the rarefaction moves into the chamber with the speed of sound a_2 .

Assume $\phi = 0$, which gives $\sin \phi = 0$ and $\cos \phi = 1$

elapse after adjusting the chamber pressure for the
pressure to equilibrium. Taking the initial value of ϕ as
of the above equation we have

$$\frac{\sqrt{2}(\gamma-1)}{\sqrt{\gamma-1+\gamma^2\phi^2}} = \frac{1}{\gamma-1} \left[1 - \left(\gamma - \frac{1}{\gamma} \right) \left(\frac{1}{\gamma^2} \right) \right]$$

Assume $\gamma = 1.4$, then

$$\frac{\gamma-1}{\sqrt{\gamma(1+\phi^2)}} = 1 - \left(\gamma - \frac{1}{\gamma} \right) \left(\frac{1}{\gamma^2} \right)$$

$$\text{or } \frac{p}{p_0} = \left[1 - \frac{\gamma-1}{\gamma} \left(\frac{1}{\gamma^2} \right) \right]^{\frac{\gamma}{\gamma-1}}$$

This equation gives the ratio of pressures
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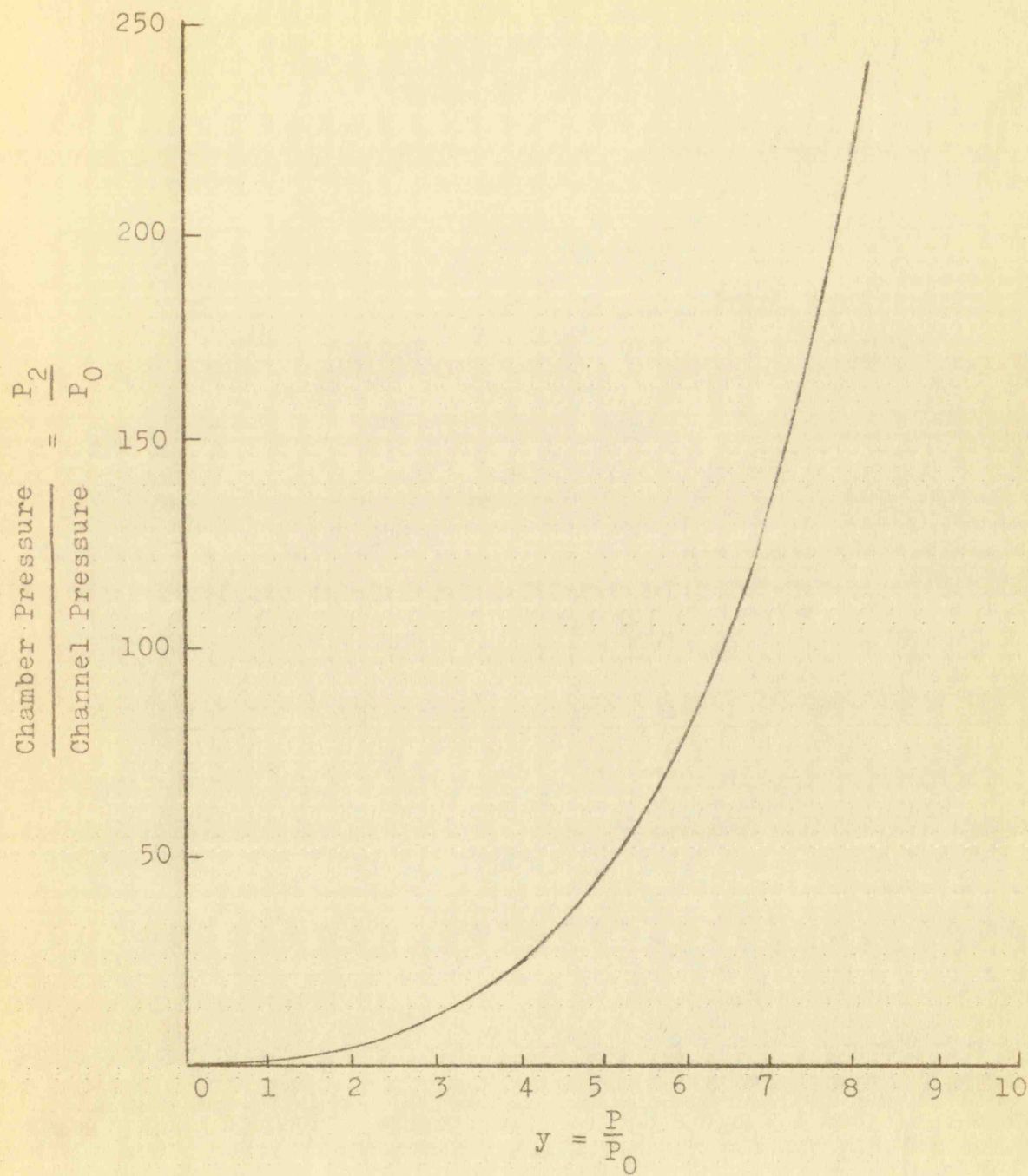
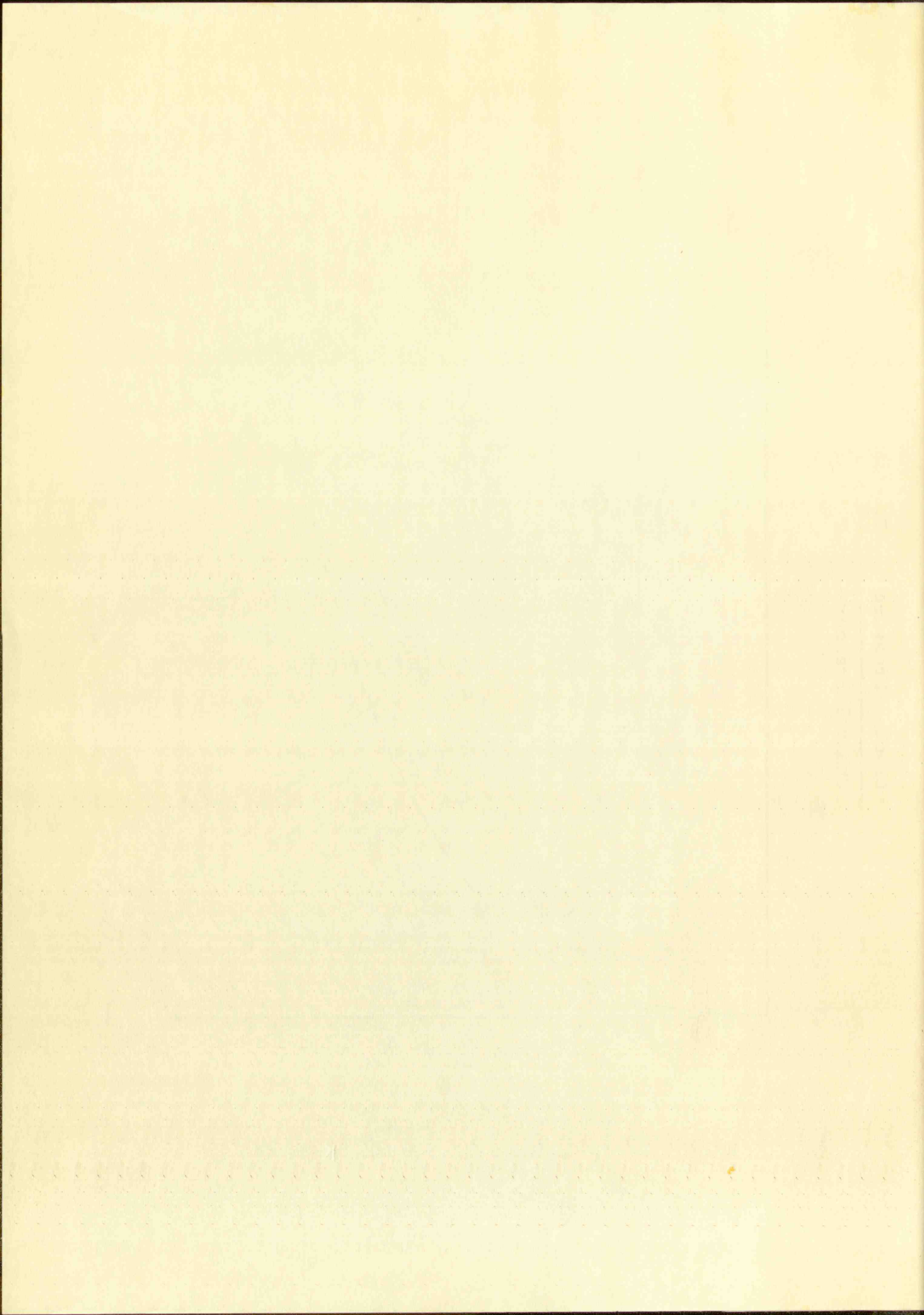


FIGURE 3

EFFECT OF PRESSURE RATIO ON SHOCK STRENGTH



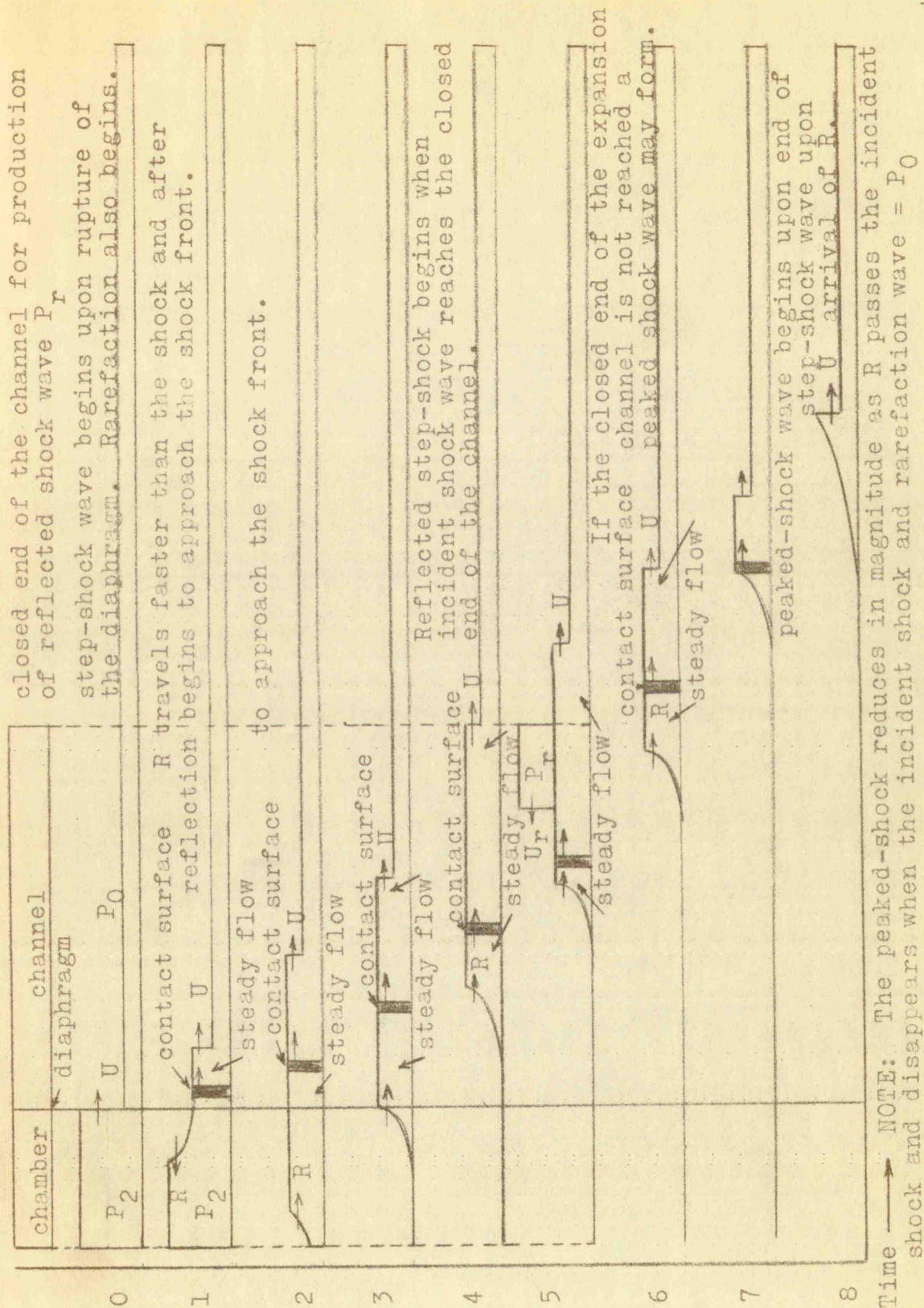


FIGURE 4
PRESSURE PHENOMENA IN THE SHOCK TUBE

The region between the foot of the rarefaction and the contact surface is also one of uniform flow.

The critical pressure of the gas determines whether the flow will be supersonic, sonic, or subsonic.² This is determined from

$$P_{CR} = P_2 \left(\frac{2}{\gamma+1} \right)^{\frac{2\gamma}{\gamma-1}} \quad (\text{II} - 5)$$

where $\gamma = 1.4$ for air $P_{CR} = P_2 \times \left(\frac{2}{2.4} \right)^7 = P_2 \times 0.279$

if $P_2 = 25 \text{ "Hg}$, then $P_{CR} = 6.98 \text{ "Hg}$.

From this then the foot of the rarefaction may move upstream, remain stationary, or move downstream, depending upon whether the pressure is greater than, equal to, or less than the critical pressure.

Figure 5a shows the pressure distribution when the pressure is less than the critical and Figure 5b when the pressure is greater than the critical pressure. In Figures 5c and 5d, the movements of the rarefaction, contact surface, and shock wave are shown.

²F. W. Geiger and C. W. Mautz, "The Shock Tube as an Instrument for the Investigation of Transonic and Supersonic Flow Patterns," Engineering Research Institute University of Michigan, Ann Arbor, June, 1949, pp. 15-17.

The region between the foot of the transition and the

contact surface is also one of uniform flow.

The critical pressure of the gas is given by

the flow will be supersonic, sonic, or subsonic. This

is determined from

$$p_{cr} = p_2 \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma}{\gamma - 1}}$$

(11-2)

where $\gamma = 1.4$ for air $p_{cr} = p_2 \left(\frac{2}{2.4} \right)^{\frac{1.4}{1.4 - 1}} = p_2 \times 0.528$

if $p_2 = 25$ "Hg then $p_{cr} = 13.2$ "Hg

From this then the foot of the transition may move up

stress, remain stationary, or move downstream, depending

upon whether the pressure is greater than, equal to, or

less than the critical pressure.

Figure 5a shows the pressure distribution when

the pressure is less than the critical and Figure 5b when

the pressure is greater than the critical pressure. In

Figures 5c and 5d, the movement of the transition, the

last surface, and shock wave are shown.

²E. W. Galt and G. W. Moore, "The Shock Tube as an Instrument for the Investigation of Transonic and Supersonic Flow Patterns," Engineering Research Institute University of Michigan, Ann Arbor, June, 1948, pp. 1-11.

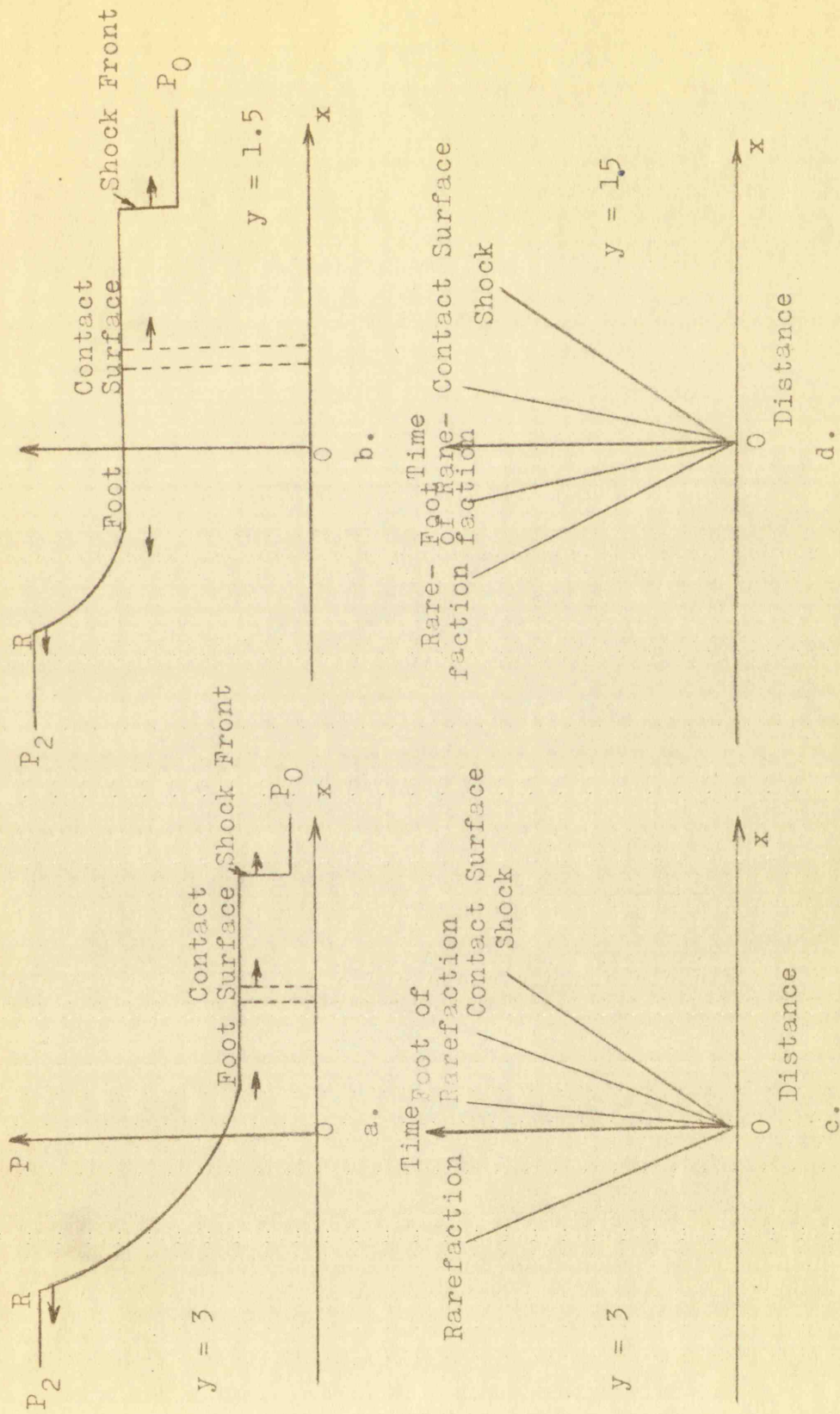


FIGURE 5
FLOW PRODUCED BY BURSTING DIAPHRAGM

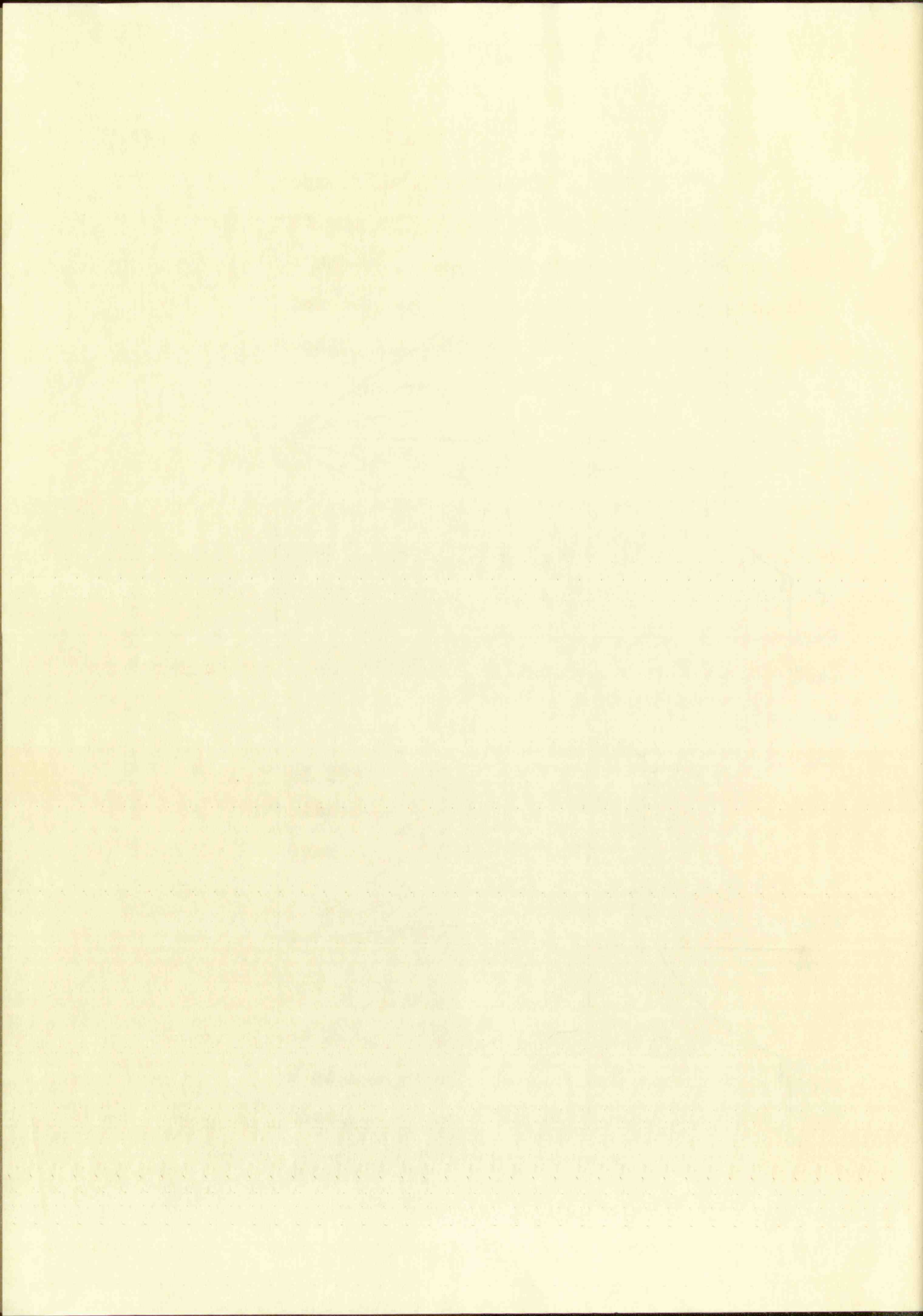


Figure 6 shows, through a double exposure, the complete flow in the shock tube. The air flows over a 3-degree steel wedge and the primary shock wave may be seen about one third of the way down the wedge from the leading edge. A detached bow wave may also be seen about one-eighth inch in front of the leading edge of the wedge. The detached wave is caused by the flow Mach number being just under the speed of sound 0.98. From the second exposure an attached shock wave may be seen at the leading edge of the wedge. The measured Mach flow was 1.3. The second exposure was taken in the flow region following the contact surface. The wake and expansion waves at the corners are due to be flow following the contact surface also.

The flow field produced by a plane shock. The pressures, P_0 and P_1 , and the corresponding flow speeds, V_0 and V_1 , on either side of the shock wave are related by the Rankine-Hugoniot relation

$$\frac{V_0}{V_1} = \frac{\frac{\gamma+1}{\gamma-1} + \frac{P_0}{P_1}}{1 + \left(\frac{\gamma+1}{\gamma-1}\right)\left(\frac{P_0}{P_1}\right)} \quad (\text{II} - 6)$$

where V_0 and V_1 are measured with respect to the shock front, and γ is the ratio of the specific heats of the gas. In the case of the shock wave proceeding into a stationary gas a transformation of the equation gives

Figure 6 shows, through a double exposure, the complete flow in the shock tube. The air flows over a 3-degree steel wedge and the primary shock wave may be seen about one third of the way down the wedge from the leading edge. A detached bow wave may also be seen about one-eighth inch in front of the leading edge of the wedge. The detached wave is caused by the flow Mach number being just under the speed of sound 0.98. From the second exposure an attached shock wave may be seen at the leading edge of the wedge. The measured Mach flow was 1.2. The second exposure was taken in the flow region following the contact surface. The wake and expansion waves at the corners are due to the flow following the contact surface also.

The flow field produced by a plane shock. The pressures, P and P_0 , and the corresponding flow speeds, V_0 and V_1 , on either side of the shock wave are related by the Rankine-Hugoniot relation

$$\frac{V_0}{V_1} = \frac{1 + \frac{\gamma + 1}{\gamma - 1} \left(\frac{P}{P_0} \right)}{\frac{\gamma + 1}{\gamma - 1} + \frac{P}{P_0}} \quad (\text{II} - 6)$$

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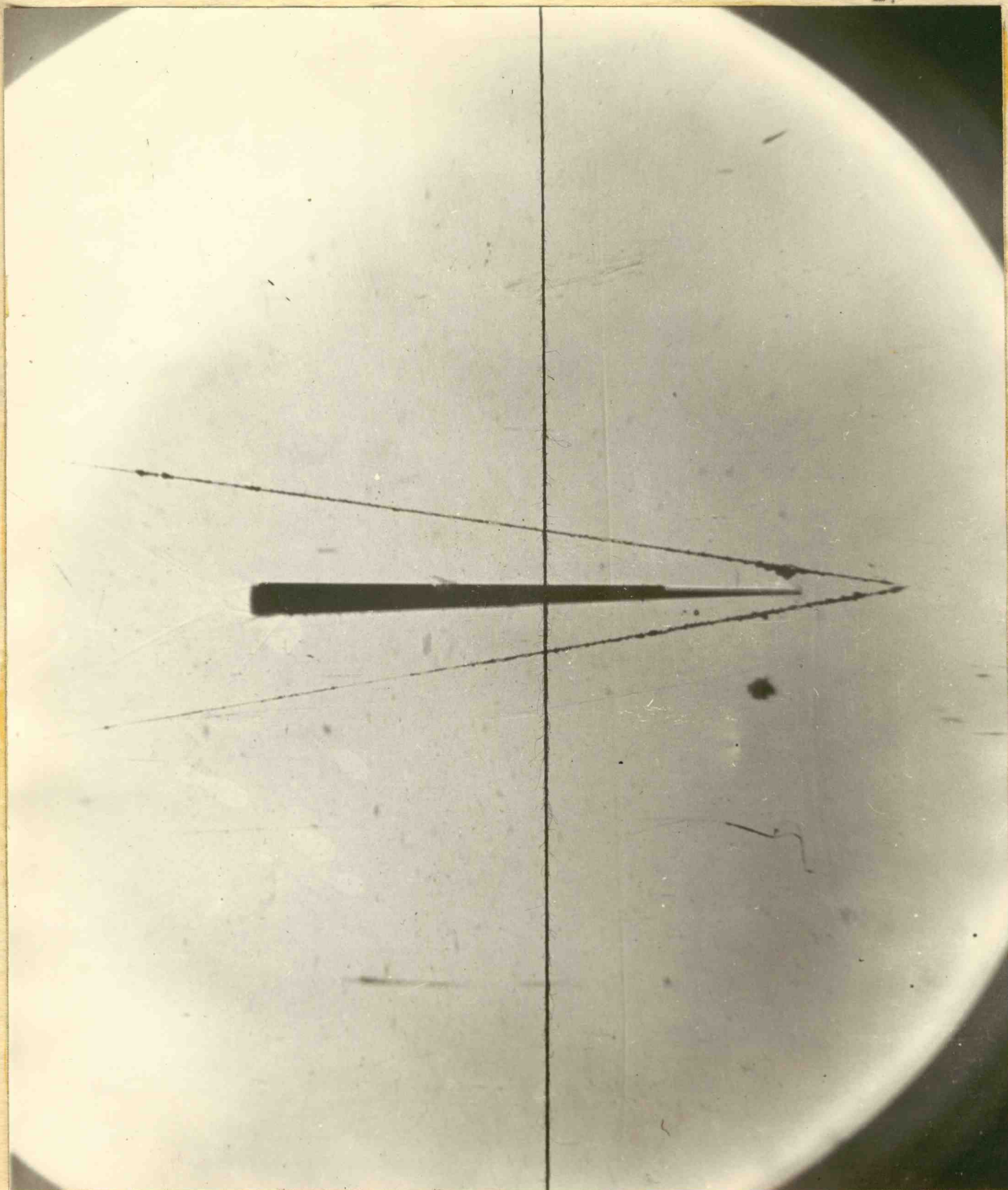


FIGURE 6

DOUBLE EXPOSURE M_{PS} 0.98, M_{PC} 1.3

led

FIGURE 6

DOUBLE EXPOSURE M_{PS} 0.38, M_{PO} 1.2

of the equation gives

$$\frac{U}{U-u} = \frac{\frac{\gamma+1}{\gamma-1} + \frac{P_0}{P_1}}{1 + \left(\frac{\gamma+1}{\gamma-1}\right)\left(\frac{P_0}{P_1}\right)} \quad (\text{II} - 7)$$

where P_0 is the pressure in the stationary medium, and P_1 is the pressure behind the shock wave. Adopting the notation

$$\gamma = \frac{P_1}{P_0} \quad 0 < \gamma < \infty; \quad \nu = \frac{\gamma+1}{\gamma-1};$$

the equation becomes

$$\frac{U}{U-u} = \frac{\nu + \frac{1}{\gamma}}{1 + \frac{\nu}{\gamma}} \quad (\text{II} - 8)$$

The relation between the shock speed, U , and the pressure ratio γ , is (Appendix A, III - 5)

$$U = a_0 \sqrt{\frac{6\gamma+1}{7}} \quad (\text{II} - 9)$$

The calculated data is shown in Table II and a plot of data may be seen in Figure 7.

Equations (II - 8) and II - 9) and the expression for the ratio of the sound speeds a_0 and a_1 are combined in Appendix IV to give

of the equation gives

$$\frac{U}{U-u} = \frac{\frac{1}{2} + \frac{1}{2} \frac{u}{U}}{1 + \frac{1}{2} \frac{u}{U}}$$

(II-5)

where P_0 is the pressure in the stationary medium, and

P is the pressure behind the shock wave. u is the

notation

$$\gamma = \frac{P}{P_0} = \frac{P}{P_0} > 1 > 0$$

the equation becomes

$$\frac{U}{U-u} = \frac{1 + \frac{1}{2} \frac{u}{U}}{1 + \frac{1}{2} \frac{u}{U}}$$

(II-6)

The relation between the shock speed u and the pressure

ratio γ is (see also (II-5))

$$U = P_0 \sqrt{\frac{2(\gamma+1)}{\gamma}}$$

(II-7)

The calculated data is shown in Table II as a function of

data may be seen in Figure 1.

Equation (II-5) and (II-7) can be expressed

for the ratio of the sound speeds u and U as

in Appendix II to give

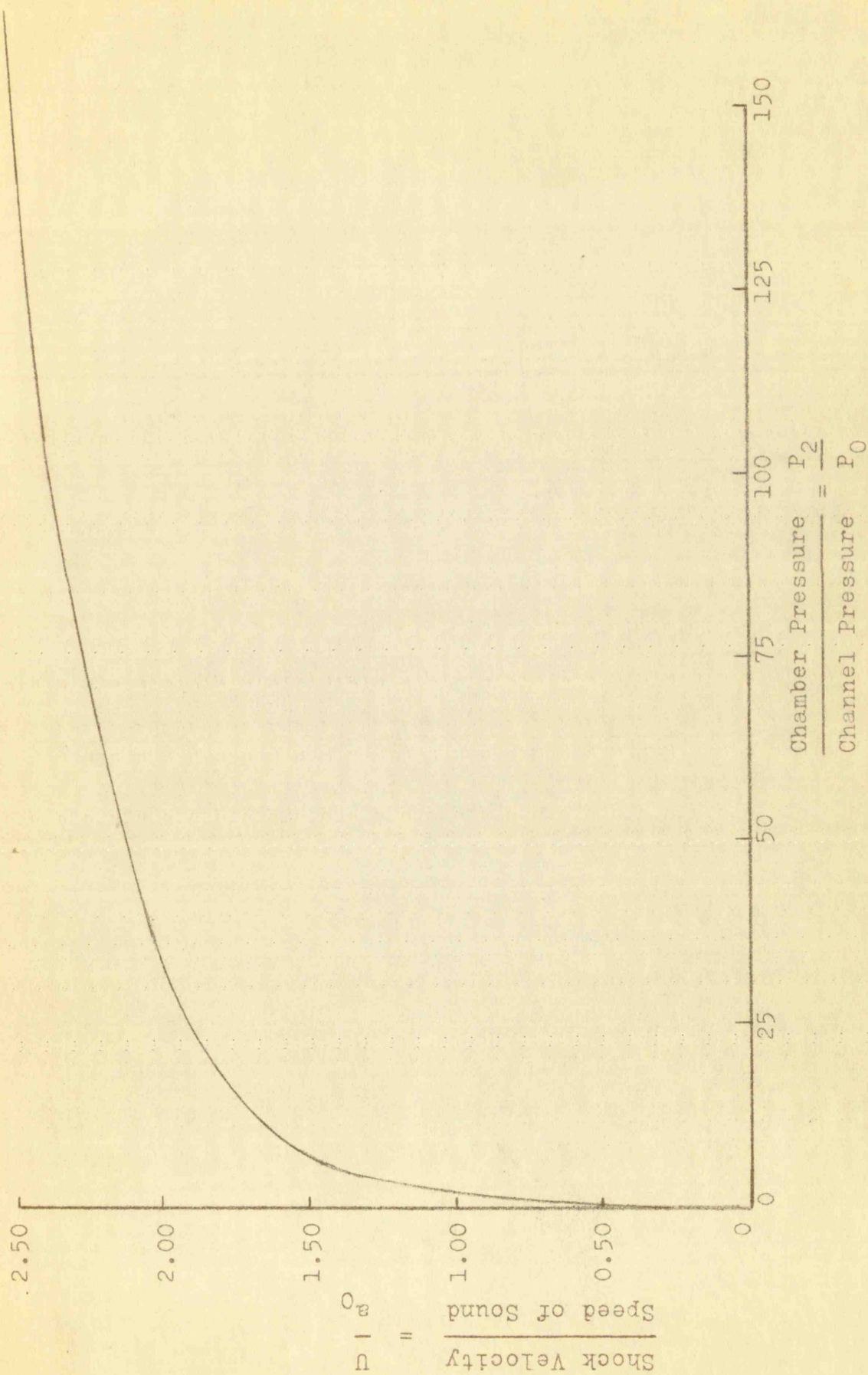


FIGURE 7

COMPARISON OF PRIMARY SHOCK VELOCITY TO
SHOCK TUBE PRESSURE RATIO

$$M_{PS} = \frac{U}{a_0} = \frac{2(\gamma-1)}{\sqrt{2\gamma[\gamma(\gamma-1)+\gamma+1]}} \quad (\text{II} - 10)$$

where M_{PS} is the Mach number of the flow. If the ratio of the specific heats is taken as 1.4 which is very nearly true for air, equation (II - 10) becomes

$$M_{PS} = \frac{5(\gamma-1)}{\sqrt{7\gamma(\gamma+6)}} \quad (\text{II} - 11)$$

The data has been calculated for Table III and a plot of the data may be seen in Figure 8.

The flow following the contact surface has a Mach number controlled by the speed of sound in the cooled gas. The derivation may be found in Appendix V.

$$M_{PC} = \frac{U}{a_{1c}} = \frac{\frac{2}{\gamma-1}(\gamma-1)}{\sqrt{\frac{2\gamma}{\gamma-1}[\gamma(\gamma+1)+1]-(\gamma-1)}} \quad (\text{II} - 12)$$

where M_{PC} is the Mach number of the flow. If the ratio of the specific heats is taken as 1.4 for air (II - 12) becomes

$$M_{PC} = \frac{5(\gamma-1)}{\sqrt{7(6\gamma+1)-(\gamma-1)}} \quad (\text{II} - 13)$$

The data has been calculated for Table IV and a plot of

$$M_{ps} = \frac{U}{C_0} = \frac{2(X-1)}{\sqrt{2(X-1)(X+1)}} \quad (11-10)$$

where M_{ps} is the Mach number of the flow. If the ratio of the specific heats is taken as 1.4 which is very nearly true for air, equation (11-10) becomes

$$M_{ps} = \frac{2(X-1)}{\sqrt{2(X+1)}} \quad (11-11)$$

The data has been calculated for M_{ps} and is shown of the data may be seen in Figure 8. The flow following the normal shock has a Mach number controlled by the speed of sound in the cooled gas. The derivation may be found in Appendix V.

$$M_{pc} = \frac{U}{C_0} = \frac{\frac{2(X-1)}{X-1}}{\sqrt{\frac{2(X-1)}{X-1} \left[\frac{X(X+1)}{X-1} + 1 \right] - 1}} \quad (11-12)$$

where M_{pc} is the Mach number of the flow. If the ratio of the specific heats is taken as 1.4 for air (11-12) becomes

$$M_{pc} = \frac{2(X-1)}{\sqrt{2(X+1)(X-1)}} \quad (11-13)$$

The data has been calculated for M_{pc} and is shown

CORRELATION OF MEASURED MACH FLOW TO PREDICTED MACH FLOW
FOLLOWING THE PRIMARY SHOCK FRONT

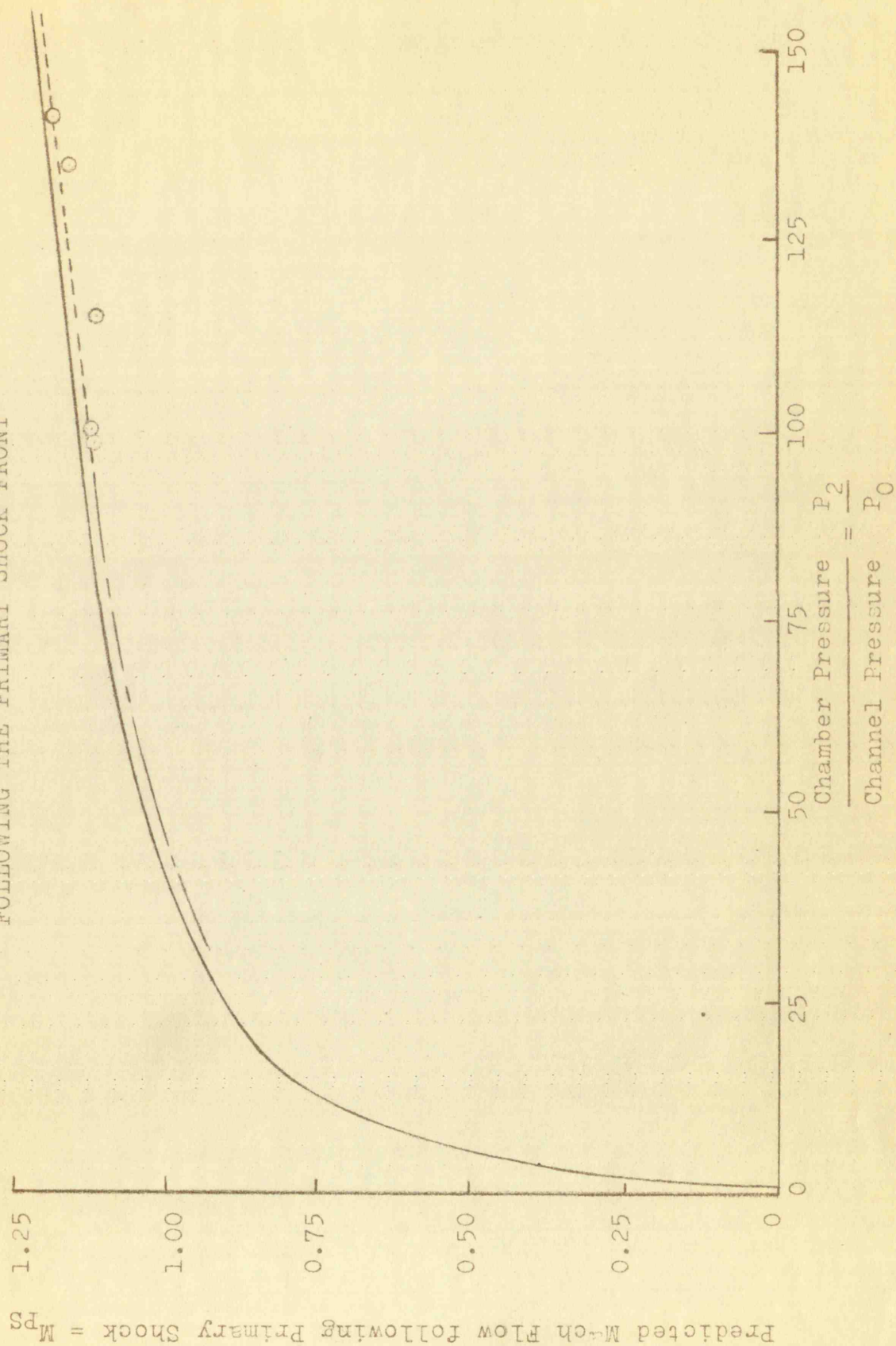
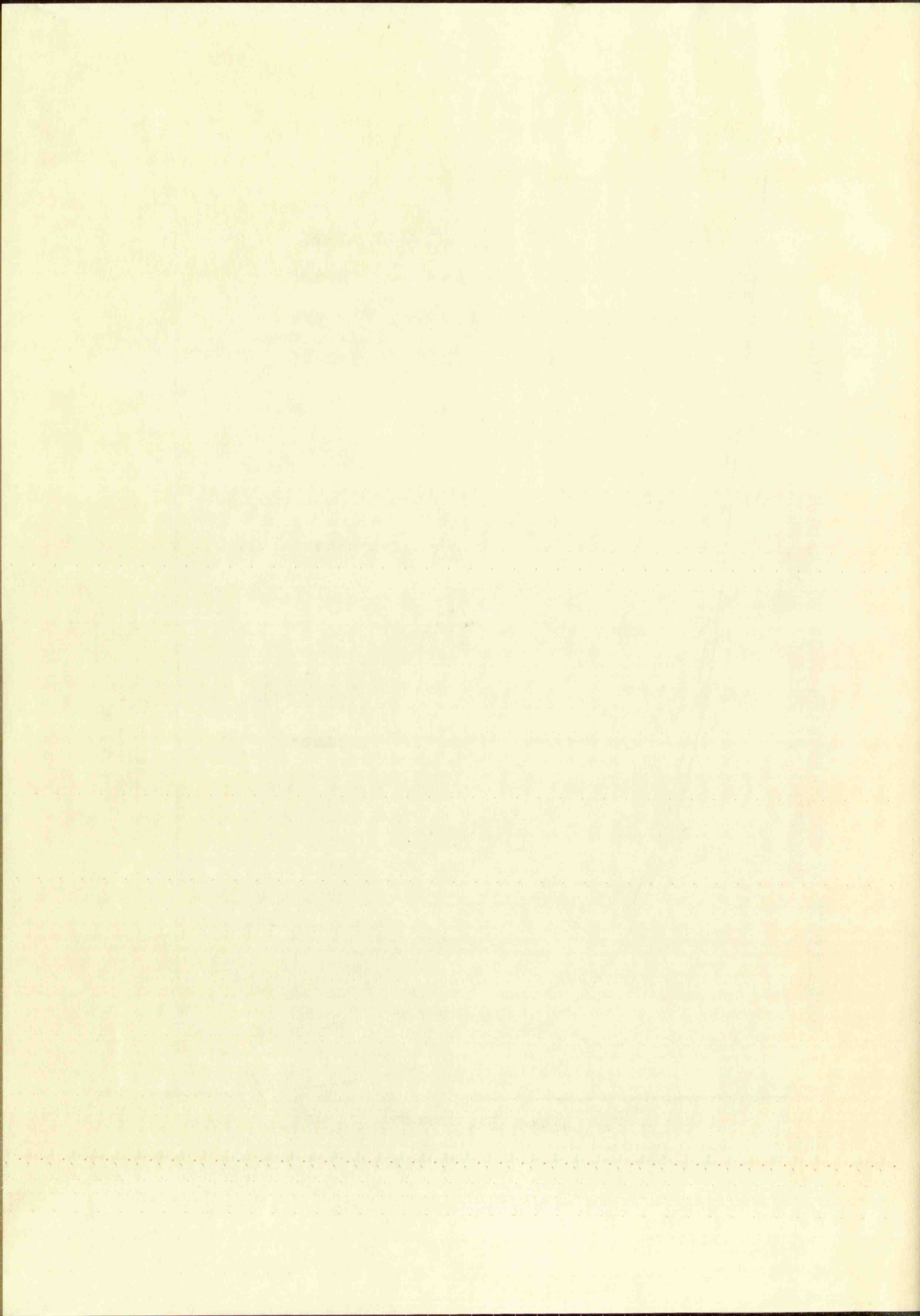


FIGURE 8



the data may be seen in Figure 9.

Reflections at the ends of the shock tube. C. W. Lampson³ shows that if a shock wave of pressure ratio γ is reflected from a rigid wall, the pressure ratio γ_r of the reflected shock is given by

$$\gamma_r = \frac{\gamma p + 2\gamma - 1}{\gamma + p} \quad (\text{II} - 14)$$

The velocity, U_r , of this shock wave with respect to the wall, in this case the end of the channel, is given by

$$U_r = a_0 \frac{(2\gamma + p - 1)}{\sqrt{(p+1)(p\gamma+1)}} \quad (\text{II} - 15)$$

where a_0 is the sound speed in the undisturbed gas. For air, with $p = 6$ equation (II - 15) becomes

$$U_r = a_0 \frac{2\gamma + 5}{\sqrt{7(6\gamma + 1)}} \quad (\text{II} - 16)$$

³Lampson, op. cit., pp. 24-26.

the data may be seen in Figure 2.

Reflections at the end of the pipe (11-12)

Laplace shows that if a shock wave of pressure ratio y is reflected from a rigid wall, the pressure ratio of the reflected shock is given by

$$(11-12) \quad \frac{y_r + y - 1}{y + 1} = \frac{y_r + y - 1}{y + 1}$$

The velocity, U_r , of this shock wave with respect to the wall, in this case the end of the pipe, is given by

$$(11-13) \quad U_r = a_0 \frac{(y_r + y - 1)}{(y + 1)}$$

where a_0 is the sound speed in the undisturbed gas, and y is the pressure ratio (11-12) between the air, with $y = \frac{p_r}{p_0}$ (11-12) between the air, with $y = \frac{p_r}{p_0}$

$$(11-14) \quad U_r = a_0 \frac{(y_r + y - 1)}{(y + 1)}$$

CORRELATION OF MEASURED MACH FLOW TO PREDICTED MACH FLOW
FOLLOWING THE CONTACT SURFACE

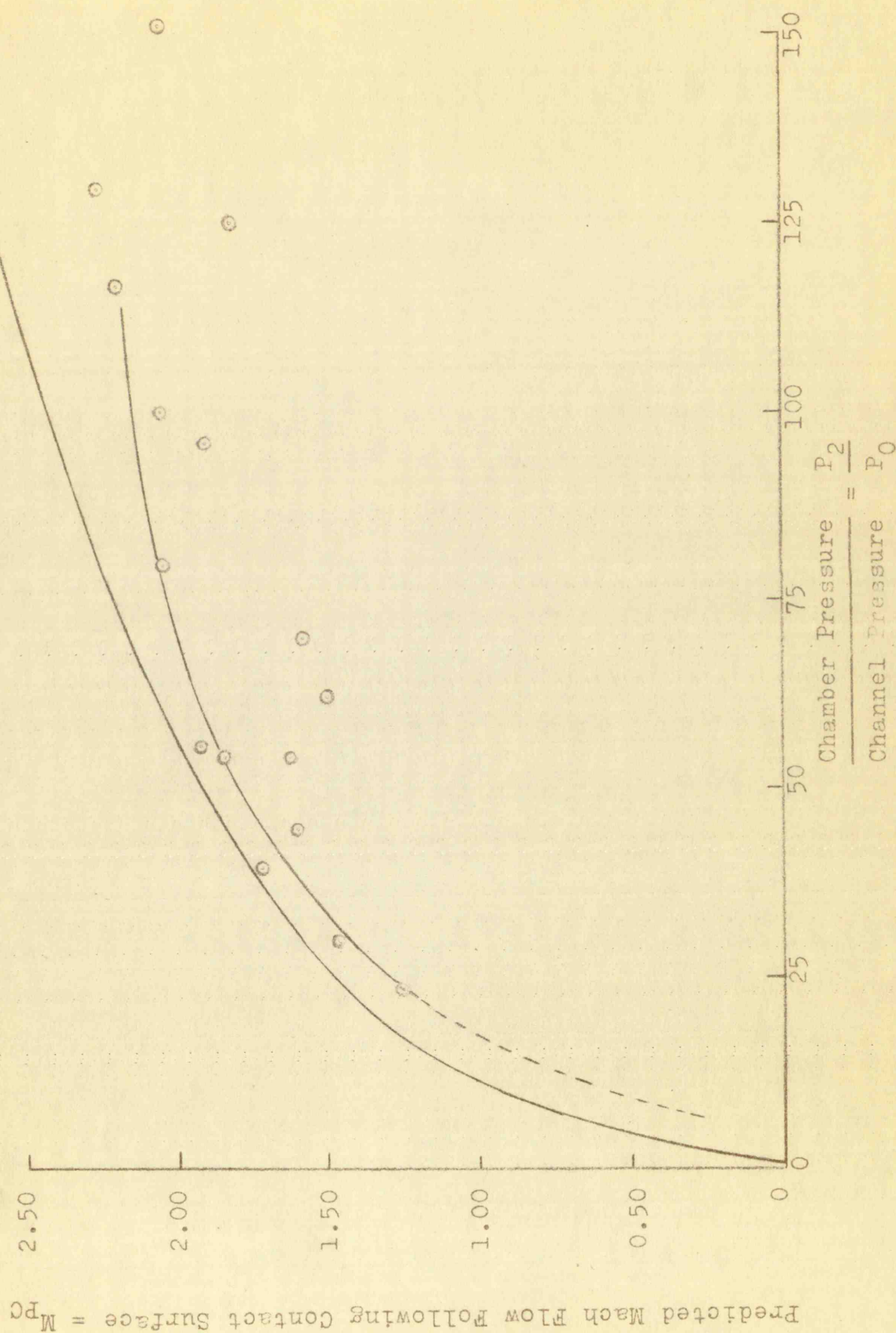
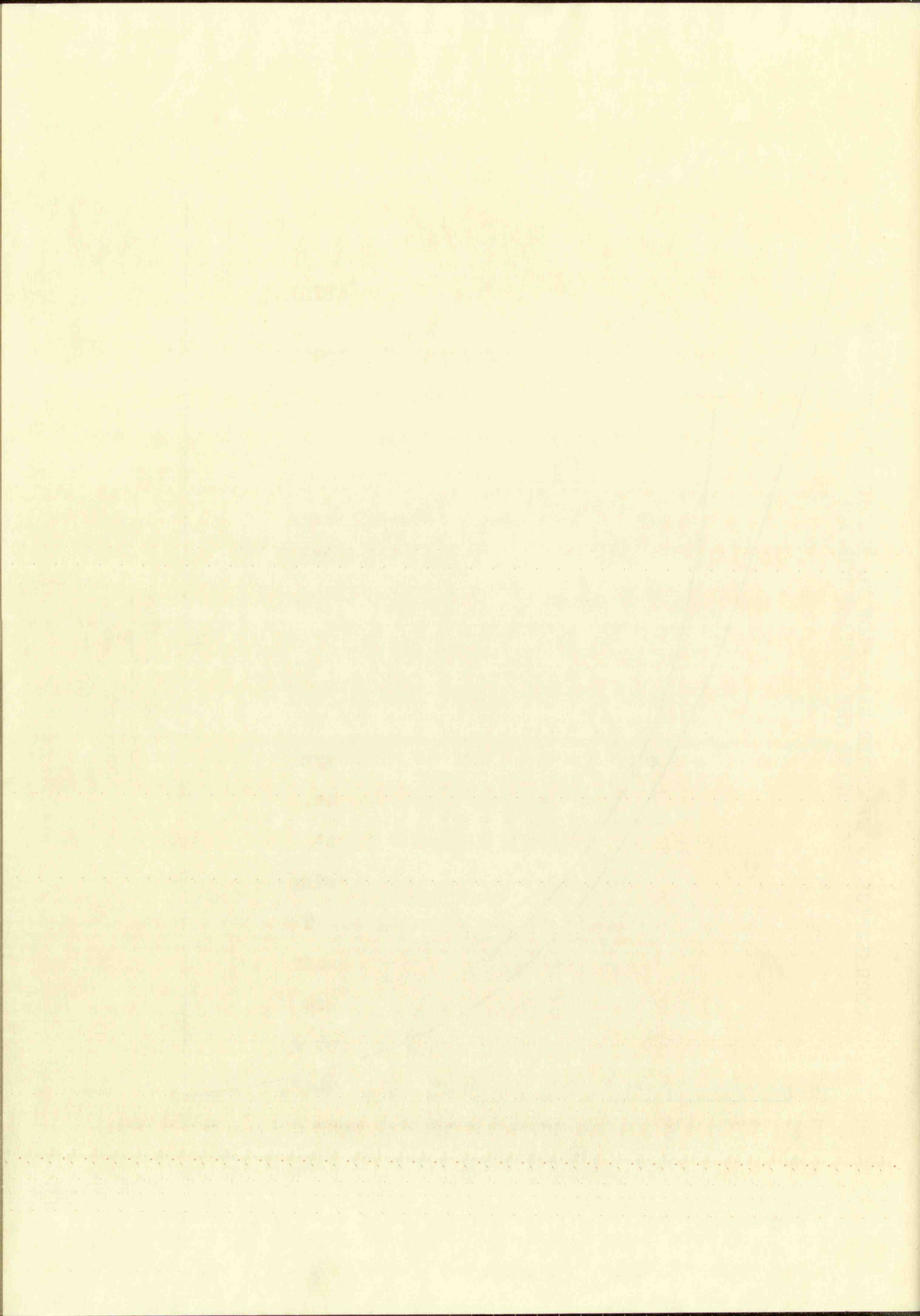


FIGURE 9



CHAPTER III

DESIGN AND CONSTRUCTION OF THE APPARATUS

The apparatus, see Figure 10, required for this study consisted essentially of (1) a tube divided into two parts consisting of a high pressure chamber and low pressure channel; (2) vacuum pumps for obtaining the necessary pressure differentials between these parts; (3) devices for measuring these pressure differences including a barometer, mercurial manometer, and a thermometer; (4) a diaphragm for maintaining these pressure differences; (5) a device for breaking the diaphragm to initiate the flow of gases into the channel; (6) the photographic apparatus for recording this flow; (7) the synchronizing mechanism for triggering the light source.

The tube. A wooden tube was first constructed out of one inch plywood, with 2 x 8 inch inside dimensions and adjustable up to 24 feet in length. This proved unsatisfactory since the leakage was excessive through the wood which resulted in inaccurate pressure readings.

A rectangular steel tube of 4 foot flanged sections was then made of 7 inch standard steel channel with a 1/4 inch steel plate welded over the open side. This tube

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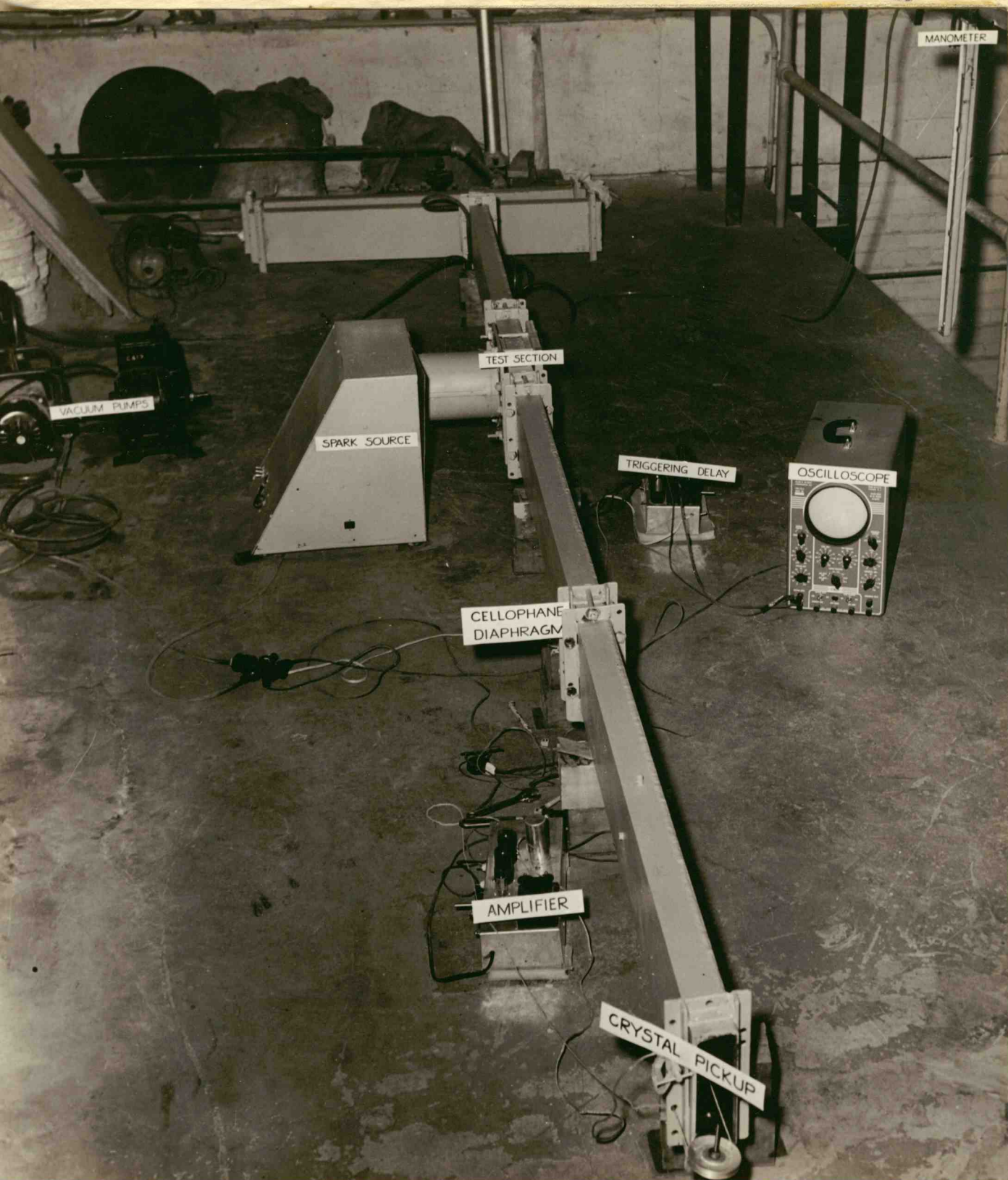


FIGURE 10
THE EXPERIMENTAL APPARATUS

THE EXHIBITION OF THE

PICTURE TO

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4-219

7-19 54

was adjustable in length up to 22 feet. The channel region of the tube included a two-foot test section at which point data was recorded. Two extra four-foot sections were available for use if needed. The vacuum seal at the section junctions was obtained by installing a 1/16 inch Victorite gasket packing greased with cup grease. The sections were fastened together with six 3/8 inch bolts, two on each side and one each on the top and bottom.

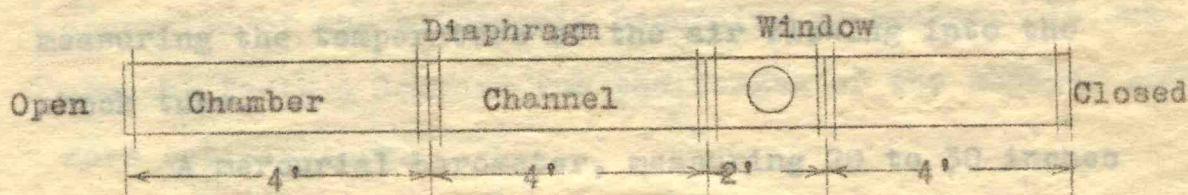


Figure 11. The Shock Tube

The vacuum system. Connections to the tube from the vacuum pump and manometer were made through the end plate of the channel to preserve the interchangeability of the sections of the tube. Non-collapsing 1/4 inch neoprene was connected with Schrader fittings to two holes in the end plate. One tube was connected to the mercurial manometer and one to the vacuum pumps. Pinch cocks on the neoprene tubing were used to seal off the shock tube. No screens were used over the openings in the end plate although a double filter filled with spun glass was installed to prevent particles of cellophane from clogging the vacuum pumps and entering the manometer.

Several hours.

was adjustable in length up to 12 inches. The channel
region of the tube included a two-foot long section
which pointed out and tapered. Two extra four-foot sections
were available for use if needed. The vacuum seal of the
section junction was obtained by installing a Jiffy Seal
Victrolite gasket packing pressed with oil pressure. The
sections were fastened together with six 1/4 inch bolts
two on each side and one each on the top and bottom.

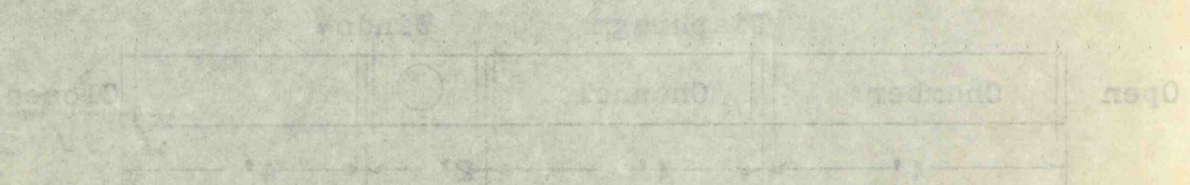


Figure 11: The Vacuum Tube

The vacuum system - Construction details

The vacuum pump and manometer were mounted on the end
plate of the chamber to provide the vacuum seal. The
of the section of the tube. The coil being 1/2 inch
reservoir was connected with the vacuum pump to the
holes in the end plate. The tube was connected to the
material manometer and one to the vacuum pump. The
coils on the end plate. The coils were 1/2 inch
about 1/2 inch. The coils were 1/2 inch
the end plate. The coils were 1/2 inch
glass was used. The coils were 1/2 inch
from the coils. The coils were 1/2 inch

Two vacuum pumps were used, a USAF rotary pump oil-sealed and with an air-oil separator for low pressure rapid evacuation, and a Cenco Hi-Vac two stage oil-sealed pump. The first drew a vacuum down to about 2 inches of mercury absolute. The second drew a vacuum down to about 0.1 inches of mercury absolute.

Devices for measuring temperatures and pressures

A standard Taylor mercurial thermometer was used for measuring the temperature of the air flowing into the shock tube.

A mercurial barometer, measuring 20 to 30 inches of mercury, absolute, was read for pressure measurements on the chamber.

A mercurial manometer, measuring 30 inches of mercury, was connected to the end plate of the channel section of the shock tube to measure the difference between the atmospheric pressure in the chamber and the pressure in the channel.

The diaphragm. The diaphragm consisted of 4 thicknesses of 0.001 inch cellophane wound on an aluminum frame and inserted between the gasketed ends of the chamber and channel. It was found by experiment that 4 thicknesses of the cellophane would withstand any pressure used even if allowed to maintain high pressure differentials for several hours.

Device for breaking the diaphragm. The device for breaking the diaphragm consisted of an "Exacto" blade mounted on an arm of 1/8 inch welding rod 3 1/2 inches in length. This arm pivoted on a stand mounted inside the shock tube chamber and moved through an arc to pierce the diaphragm when actuated. During early tests the diaphragm-breaking device was actuated by a small solenoid but it was found that the voltage surges interfered with the operation of the electronic recording apparatus. In subsequent tests a lanyard run the length of the chamber proved more satisfactory.

The photographic apparatus for recording the flow. Since the air flow in the shock tube takes place in less than 0.1 seconds, recording of the flow data by mechanical or electronic means is necessary. Thus a photographic device was needed to study the flow phenomena. The shadowgraph was chosen as the simplest and cheapest type of photographic apparatus. The flow in the tube was directed over wedges and other air foil models mounted in the test section in line with the shadowgraph light source and film. Figure 12 shows a block diagram of the triggering system.

The schematic diagram of the light source can be seen in Figure 13. The circuit consisted essentially of

Device for breaking the diaphragm. The device

for breaking the diaphragm consisted of an "E" shaped blade mounted on an arm of 1/8 inch diameter and 2 1/2 inches in length. This arm pivoted on a stand about 1 inch from the shock tube chamber and moved through it to break the diaphragm when actuated. During very high speed tests the breaking device was actuated by a small solenoid but it was found that the voltage surges interfered with the operation of the electronic recording system. In subsequent tests a lever for the breaking of the chamber proved more satisfactory.

The photographic apparatus for recording the flow

Since the air flow in the shock tube takes place in less than 0.1 second, recording of the flow data by means of an electronic system is necessary. This is accomplished by a device which is used to study the flow phenomenon. The photograph was placed at the stagnation and supersonic region of the flow. The flow in the tube was indicated by various and other air flow models mounted in the section in line with the flow. The light source and film, Figure 1, shows a black screen of the recording system.

The schematic diagram of the flow system can be

seen in Figure 1. The system consisted essentially of

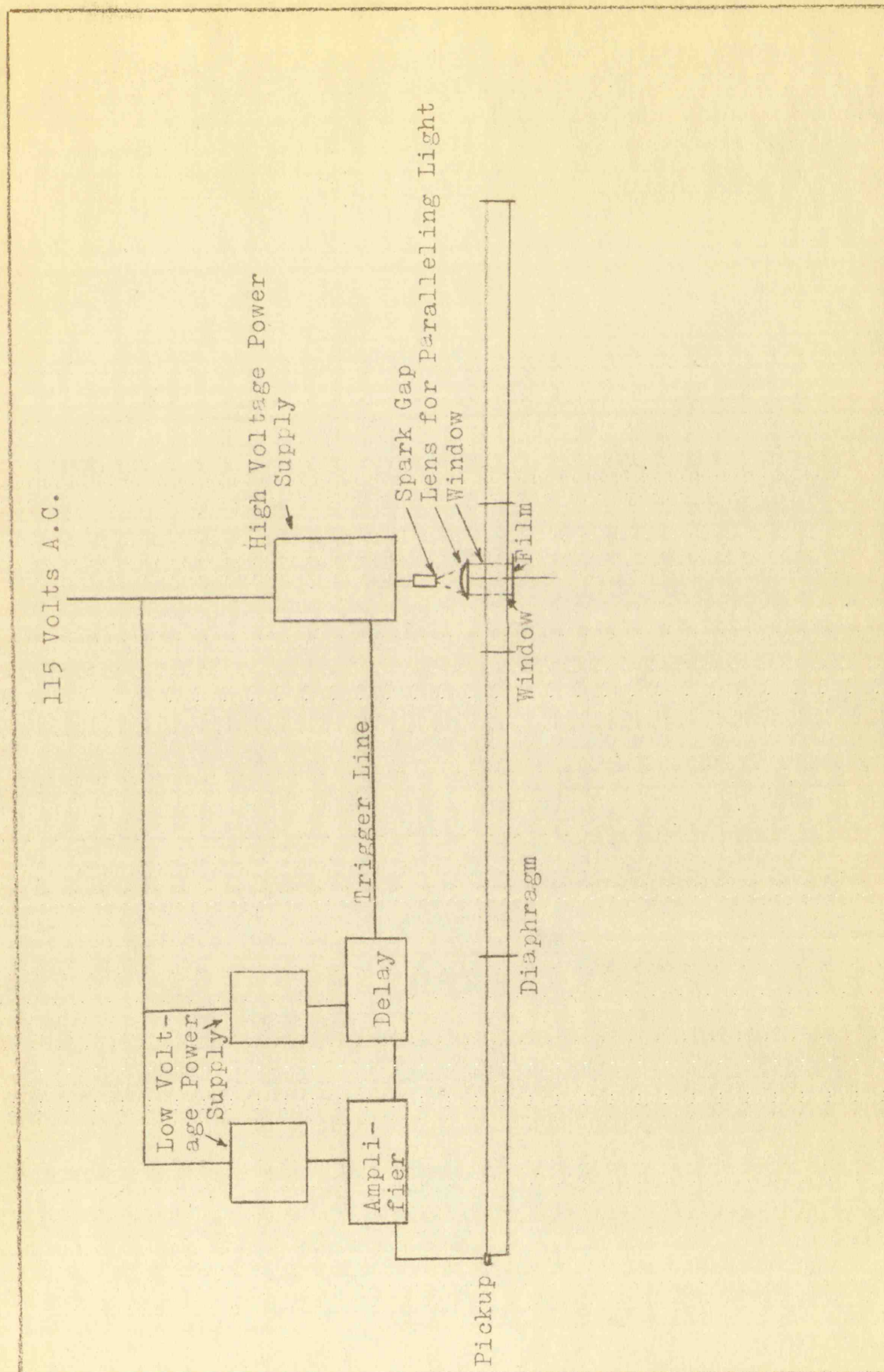
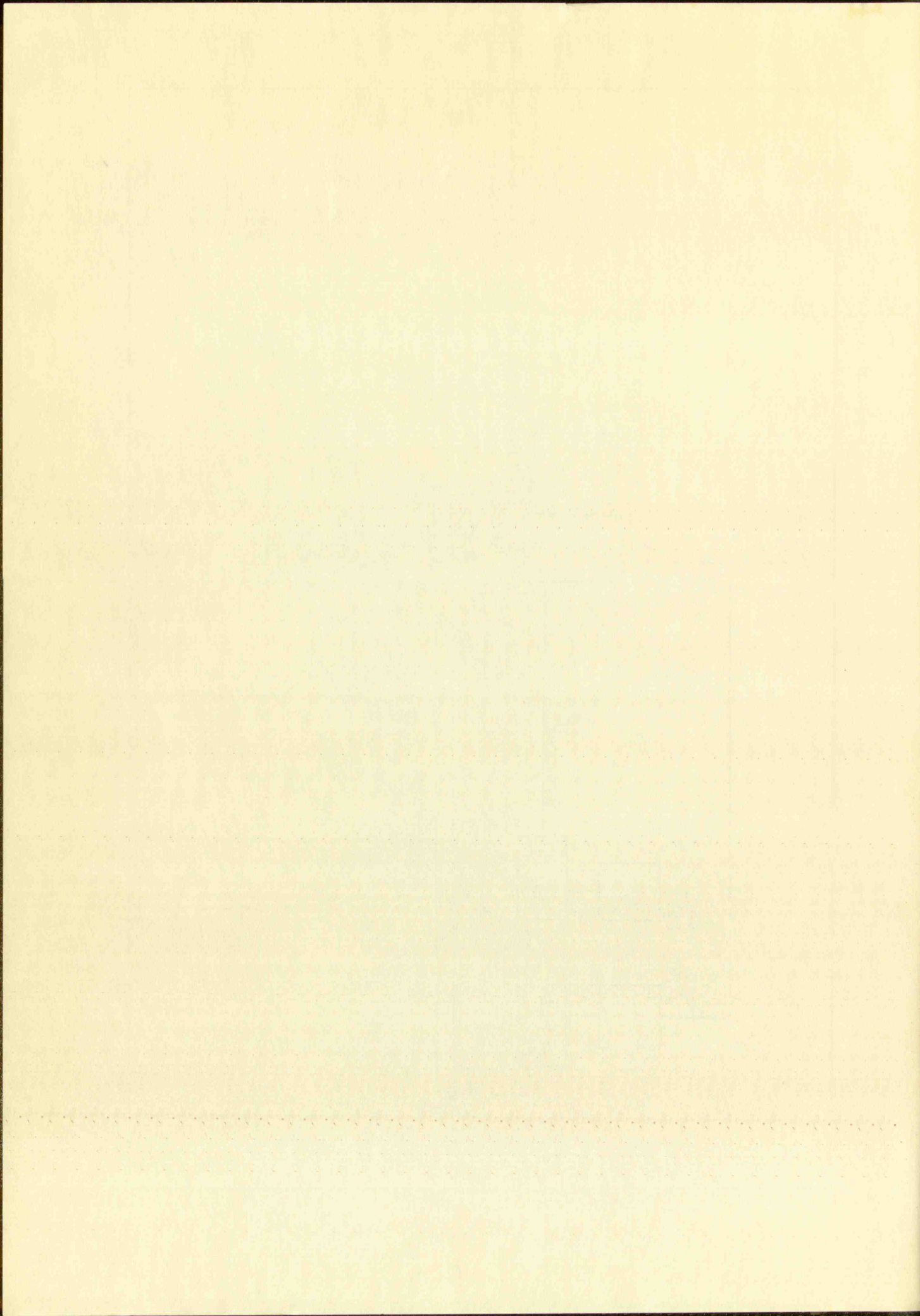


FIGURE 12

SCHEMATIC FOR APPARATUS FOR SHADOWGRAPH PHOTOGRAPHY



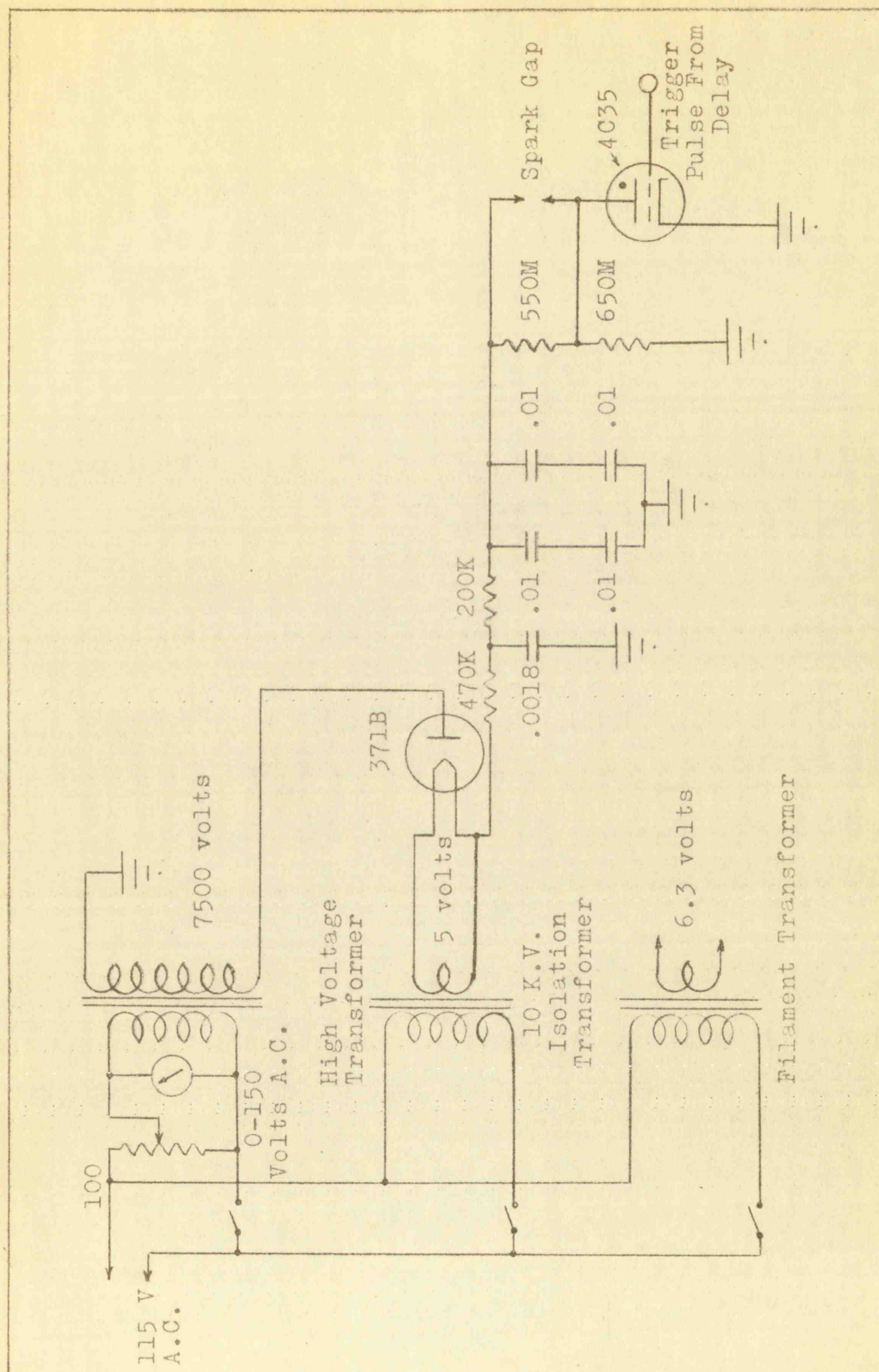
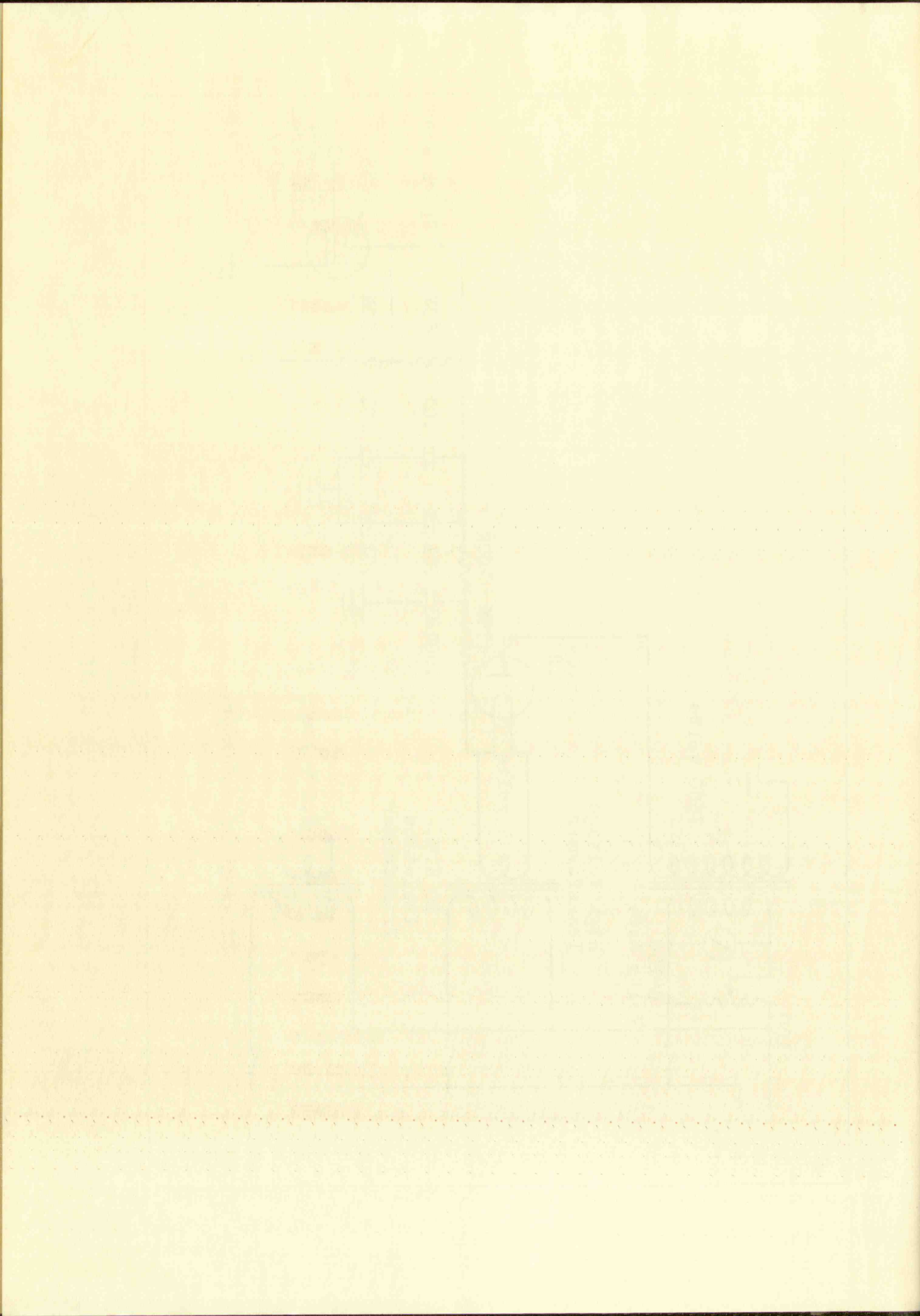


FIGURE 13

THE SPARK CIRCUIT



a high voltage power supply, a condenser to hold the electrical charge, and a discharge vacuum tube for triggering the spark.

The power supply used a 371 B rectifier. The bank of capacitors equal in capacity to 0.01 microfarads was available for 6 kilovolts. As a safety precaution resistors were installed across the spark gap and trigger tube to bleed off the system when not in use.

The trigger tube, a 4035 hydrogen thyratron, is designed to carry a current flow of 90 amperes. Although the capacitor bank probably discharges several times that current through the tube, the flow has a duration of about 0.1 microseconds and no tube damage has resulted from these brief overload currents. The duration of the spark has been checked by a rotating mirror as on the order of 0.1 microseconds.

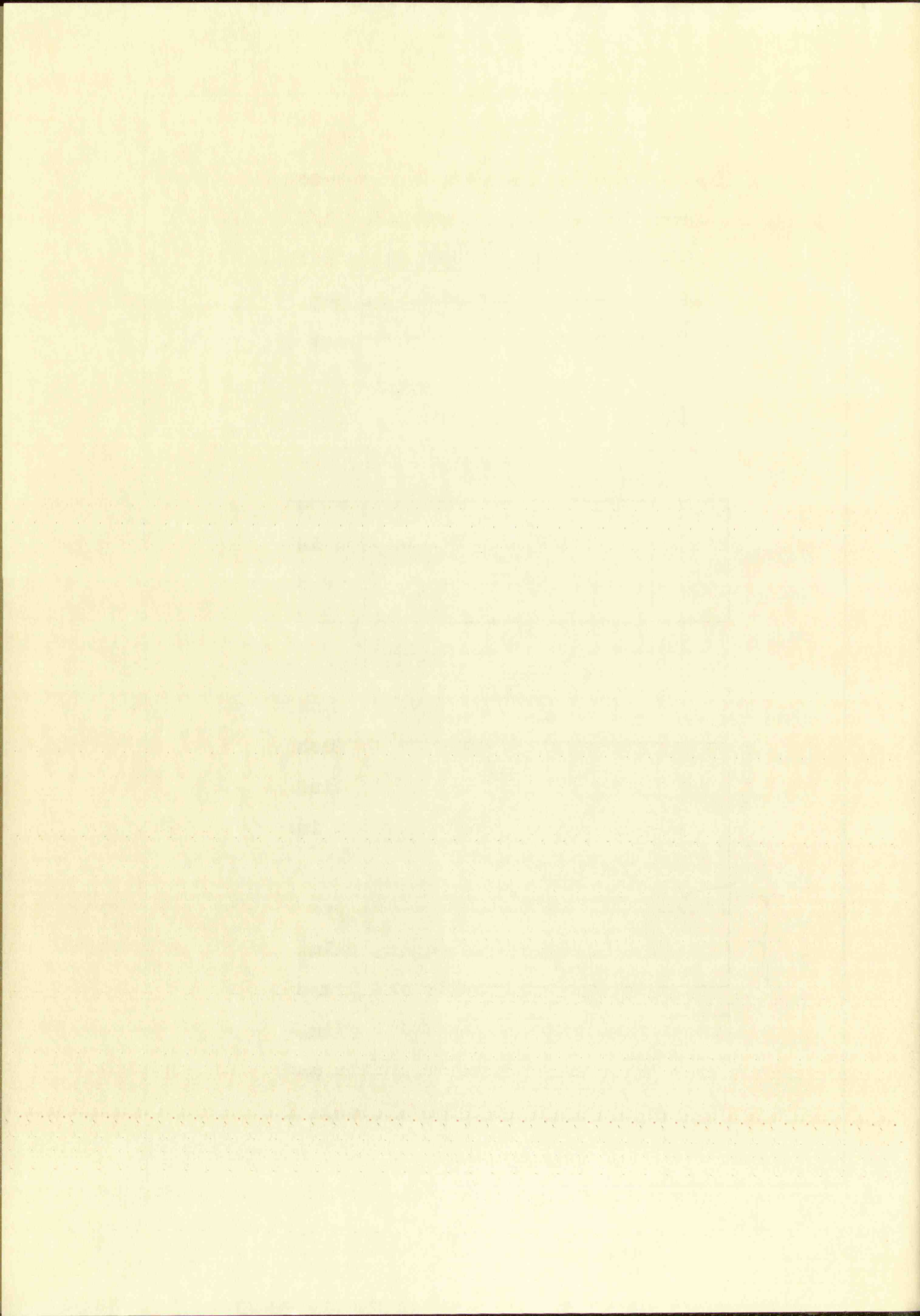
The spark head is a "Mycalex" block machined for mounting and for the installation of tungsten tips to carry the discharging ampereage from the capacitor bank. Figure 14 shows the dimensions of the spark head as used for this investigation. The block is mounted sufficiently far away from the chassis and components to prevent arc-over. The voltage could be varied from zero to about 6000 volts. The operating voltage, however, was about 5000 volts for most pictures in this investigation.

a high voltage power supply, a condenser to hold the electric charge, and a discharge switch for starting the spark.

The power supply used a 270 V resistor. The bank of capacitors equal in capacity to 0.01 microfarads was available for 6 kilovolts. As a safety precaution resistors were installed across the spark gap and a trigger tube to bleed off the system when not in use.

The trigger tube, a 600 V electron tube, was designed to carry a current flow of 20 amperes. Although the capacitor bank, which is electrically isolated from the current through the tube, and the tube is a function of about 0.1 microseconds and the tube has a resistance from these brief overhead currents. The duration of the spark has been checked by a rotating mirror on the tube at 0.1 microseconds.

The spark head is a 1/2 inch diameter spark head for mounting and for the installation of the spark head to carry the discharging voltage from the capacitor bank. Figure 14 shows the dimensions of the spark head and for this investigation. The spark is mounted and tested far away from the channel and subjected to pressure over. The voltage could be held from 5000 to 6000 volts. The spark is held at 5000 volts for about 0.1 microseconds.



The paralleling lens was a plano-convex 6-inch diameter lens with a focal length of 9 1/2 inches. The lens was mounted in a 6-inch stovepipe with a focal point at the forward electrode of the spark gap.

The windows of the shock tube test section were of 1/2 inch plate glass, 6 7/8 inches by 6 15/32 inches, mounted flush with the inside walls. The windows were supported by the top and bottom of the shock tube, and the test model which was mounted between them. To provide a vacuum seal the glass was fitted into a recessed area and sealed with "Permatex" No. 2 in a semi-plastic state, and a rubberized fabric gasket.

The test section of the shock tube was 2 feet long with an inside width of 1.840 inches. The wedge or airfoil model, machined to $1.840 \begin{smallmatrix} +.000 \\ -.003 \end{smallmatrix}$ inches in width, was held by the pressure of the glass windows after being set in place with clear cellophane tape inserted between the glass and the model.

The entire photographic system was light-tight from the light source to the film. Sheet film, 4 x 5, was held in a standard holder set outside the glass opposite the light source. The spark exposed the film. Eastman Super-Panchro Press Type B and Royal Pan film were used, followed by normal development in D-19 developer. Figure 15 shows a diagram of the light path for shadowgraph photography.

The parallel lens was a plane-convex 4-inch diameter lens with a focal length of 2 1/2 inches. The lens was mounted in a 6-inch aluminum plate with a hole at the forward extremity of the plate.

The window of the shock tube test section was of 1/2 inch plate glass, 6 1/2 inches by 6 1/2 inches, mounted flush with the inside wall. The window was supported by the top and bottom of the shock tube, and the test model which was mounted between them. To provide a vacuum seal, the glass was fitted into a recessed area and sealed with "Vaseline" oil. A rubber gasket, and a rubberized fabric gasket.

The test section of the shock tube was 2 feet long with an inside width of 1.640 inches. The window of the foil model, machined to 1.640 inches in width, was held by the pressure of the glass window in the shock tube in place with clear cyanoacrylate glue. The model, the glass and the model.

The entire photographic system was light-tight from the light source to the film. Light from a 150-watt lamp in a standard holder set outside the glass of the shock tube light source. The spark exposed the film. Eastman Kodak Panathro Press Type B and Royal Pan film were used, followed by normal development in 5-10 minutes. A diagram of the light path for the photographic system.

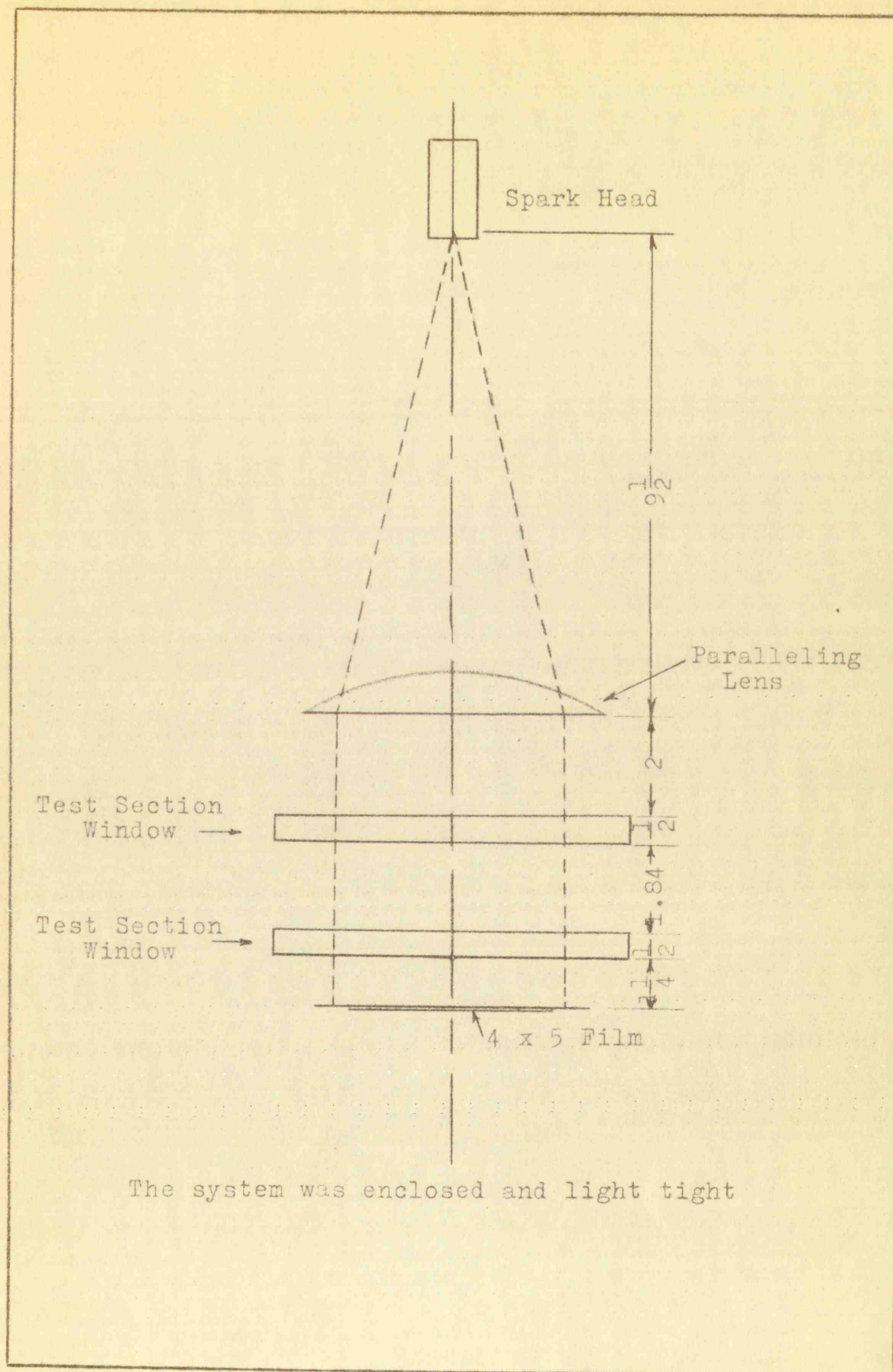
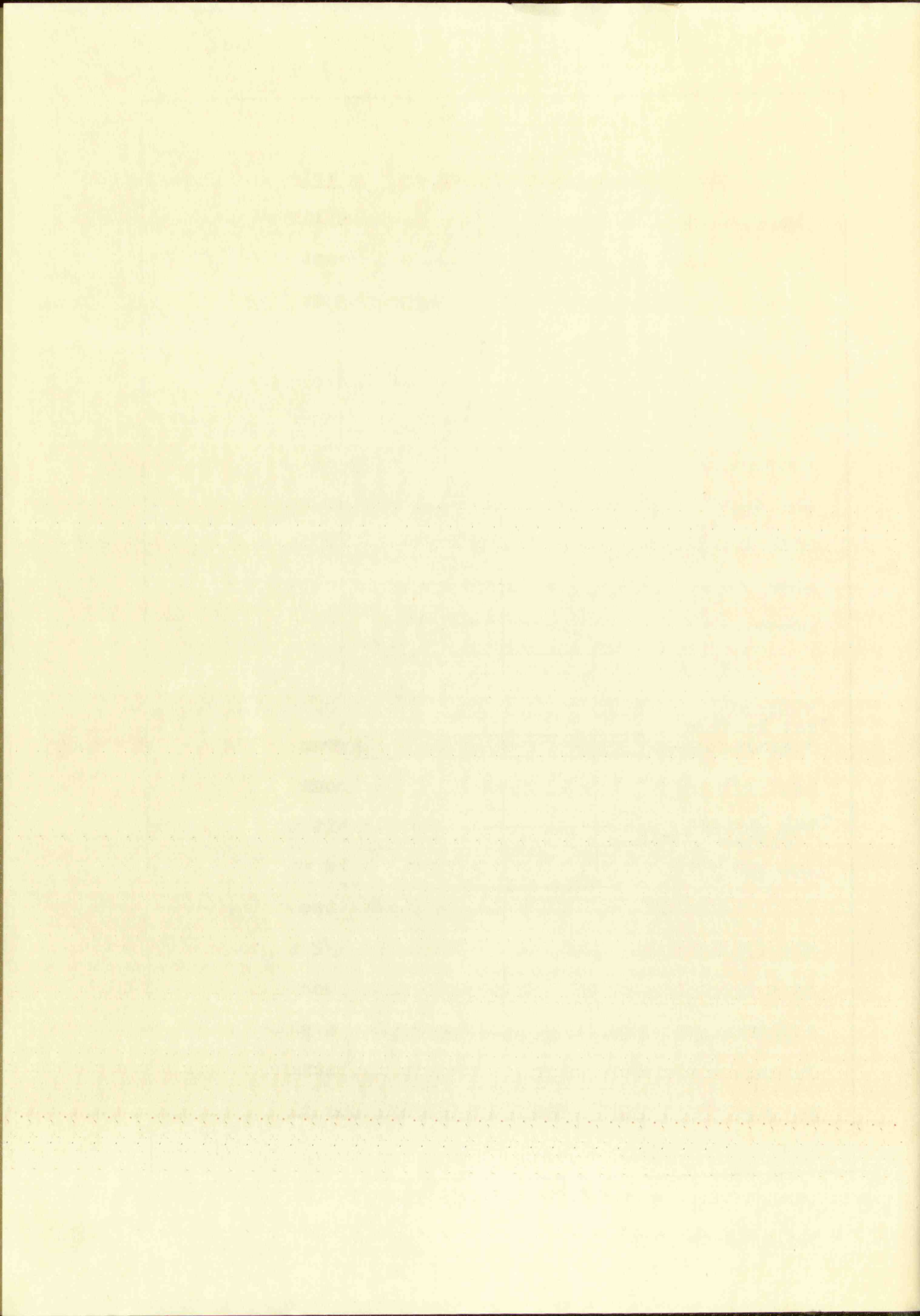


FIGURE 15

DIAGRAM OF LIGHT PATH FOR SHADOWGRAPH PHOTOGRAPHY



The synchronizing mechanism for triggering the light source. A crystal pickup with amplifier and delay system proved to be the simplest and least expensive synchronizing mechanism for the triggering of the shadow-graph light with the time of flow.

Figure 12 shows the place of the pickup in the system. In first trials an earphone pickup was used but because of its erratic behavior it was discarded. A small loudspeaker was then used with no better results. For this study a barium-titinate crystal $1/2$ inch in diameter and $1/4$ inch thick was used. The crystal was silvered on both flat surfaces and was fastened to the open end of the chamber with glue.

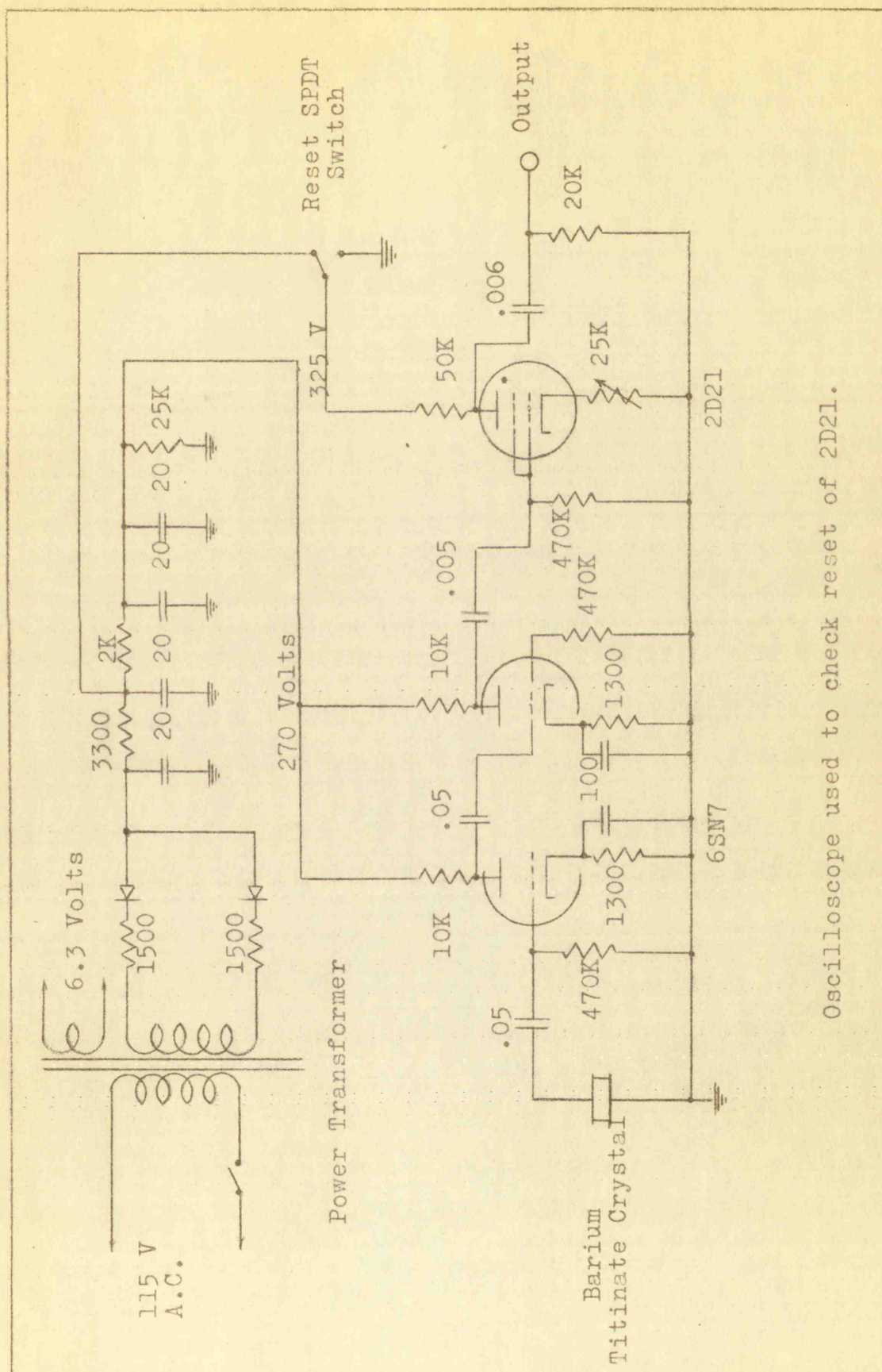
The amplifier is shown in the block diagram Figure 12. Figure 16 shows the schematic diagram of the resistance-coupled amplifier used to amplify the sound pulse from the barium-titinate crystal. The circuit was simple in that it used two stages of amplification with a single 6SN7. The 2D21 thyatron was used for two reasons. First, a sharp negative pulse of at least $22\frac{1}{2}$ volts was needed to trigger the delay. This pulse was possible with the action of the thyatron which provided a sharp voltage rise when the flow began. Second, isolation was needed for the delay unit. When triggered, the thyatron continued to conduct until reset, therefore, a single pulse

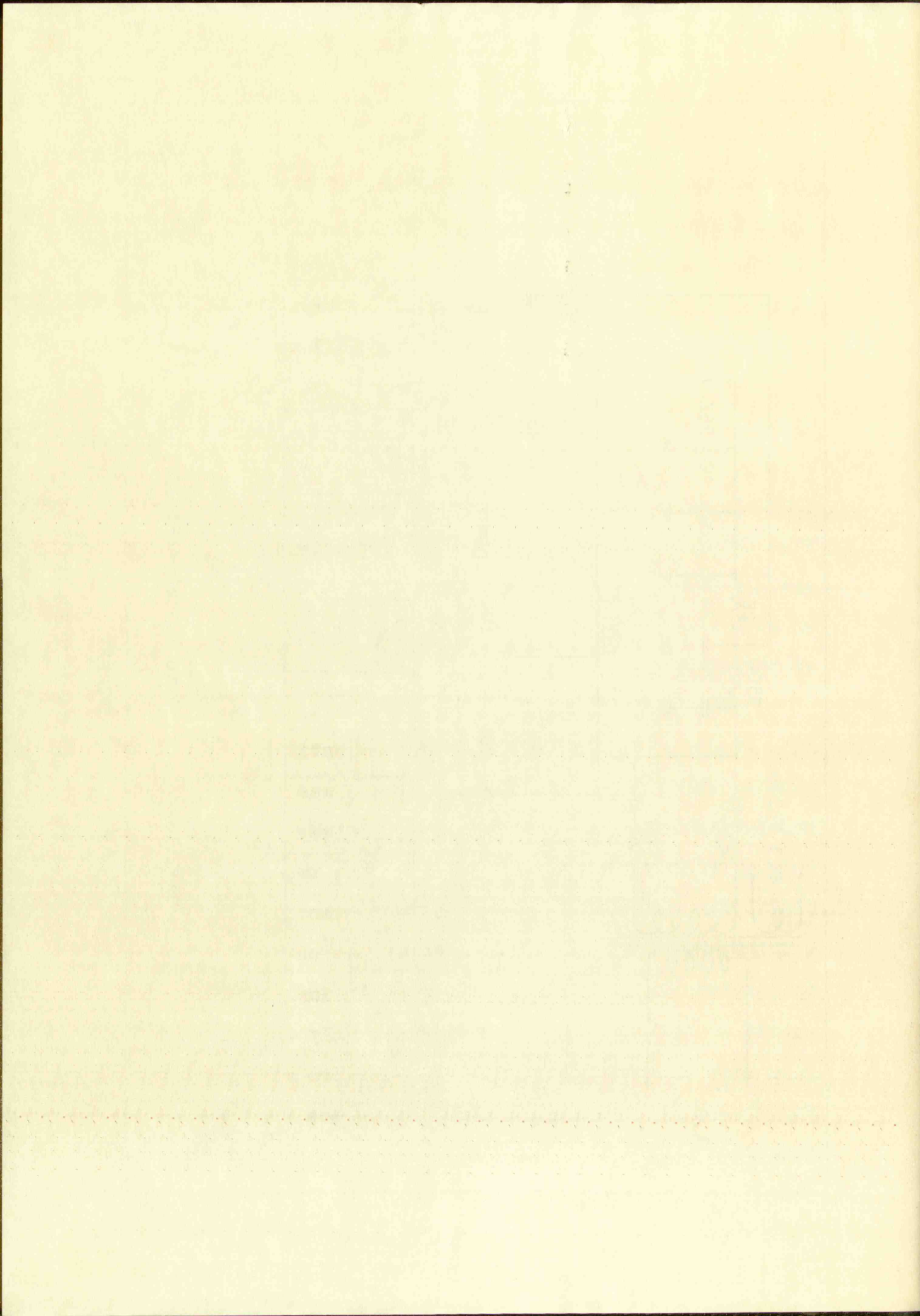
The synchronous nature of the light source

A crystal pickup with a light source and a crystal pickup proved to be the simplest and most accurate system for the triggering of the chamber with the light source.

Figure 12 shows the photo of the pickup in the system. In this photo an external pickup was used and because of its erratic behavior it was discarded. A small loudspeaker was then used with the pickup. For this study a barium-fluoride crystal 1/2 inch in diameter and 1/4 inch thick was used. The crystal was mounted on both flat surfaces and was mounted in the center of the chamber with glue.

The amplifier is shown in the photo of Figure 13. Figure 14 shows the approximate circuit of the vacuum coupled amplifier used to amplify the output of the barium-fluoride crystal. The circuit was made in that it used two stages of amplification with a single 6BN7. The 2D21 detector was used for the first stage and a sharp negative pulse of at least 100 volts was used to trigger the delay. This pulse was possible with the action of the thyatron which provided a sharp voltage rise when the flow began. Because of the delay for the delay unit, the thyatron was triggered to conduct only after the delay unit.





resulted from the amplifier regardless of the number of pulses received from the crystal pickup.

The schematic diagram of the delay circuit is shown in Figure 17. The pulse from the thyatron was received on the cathode of 1/2 of the 6SN7 used as a diode. The voltage was adjusted to a fixed level by the 25K ohm helipot. The adjustment of this voltage was critical in that it determined the delay time. The voltage was adjusted from 36 to 228 volts for full coverage of the delay. Flow travels in one direction only through the diode, therefore, all positive pulses are rejected. The negative pulse caused the 6SA7 to cutoff and at the same time cut off the other half of the 6SN7. The delay RC network of the 3920 *mpfd* capacitor and 1.8 megohm resistor determined the time of voltage buildup until the 6SA7 conducted. The plate voltage of this tube was determined by the helipot setting. The delayed pulse was taken from the cathode of the 6SA7 and amplified by the two-stage amplifier of the 12AU7. Two stages were necessary in order to give a positive pulse on the grid of the output thyatron. The 2D21 thyatron was equipped with an indicator to show when it was conducting. It triggered only once but kept conducting until reset. The voltages were critical and the power supply for the delay circuit was regulated only

resulted from the amplifier regardless of the number of
pulses received from the crystal oscillator.
The schematic diagram of the delay circuit is
shown in Figure 17. The signal from the crystal oscillator
received on the cathode of V1 of the 62AT tube is
diode. The voltage was adjusted to a fixed level by the
25K ohm potentiometer. The adjustment of this voltage was
critical in that it determined the delay time. The voltage
was adjusted from 50 to 550 volts for full coverage
of the delay. Flow travels in one direction only through
the diode, therefore, all positive pulses are rejected.
The negative pulse caused the 62AT to conduct and at the
same time cut off the other half of the 62AT. The delay
RC network of the 62AT was adjusted and 1.8 ohm resistor
determined the time of voltage oscillation until the 62AT con-
ducted. The pulse voltage of this tube was determined by
the helipot setting. The delay pulse was then transferred
cathode of the 62AT and amplified by the 62AT tube.
The 62AT tube was necessary in order to
give a positive pulse on the grid of the 62AT tube.
The 62AT tube was equipped with an internal resistor in order
when it was conducting. It conducted only once but kept
conducting until reset. The voltage was critical and
the power supply for the delay circuit was regulated only

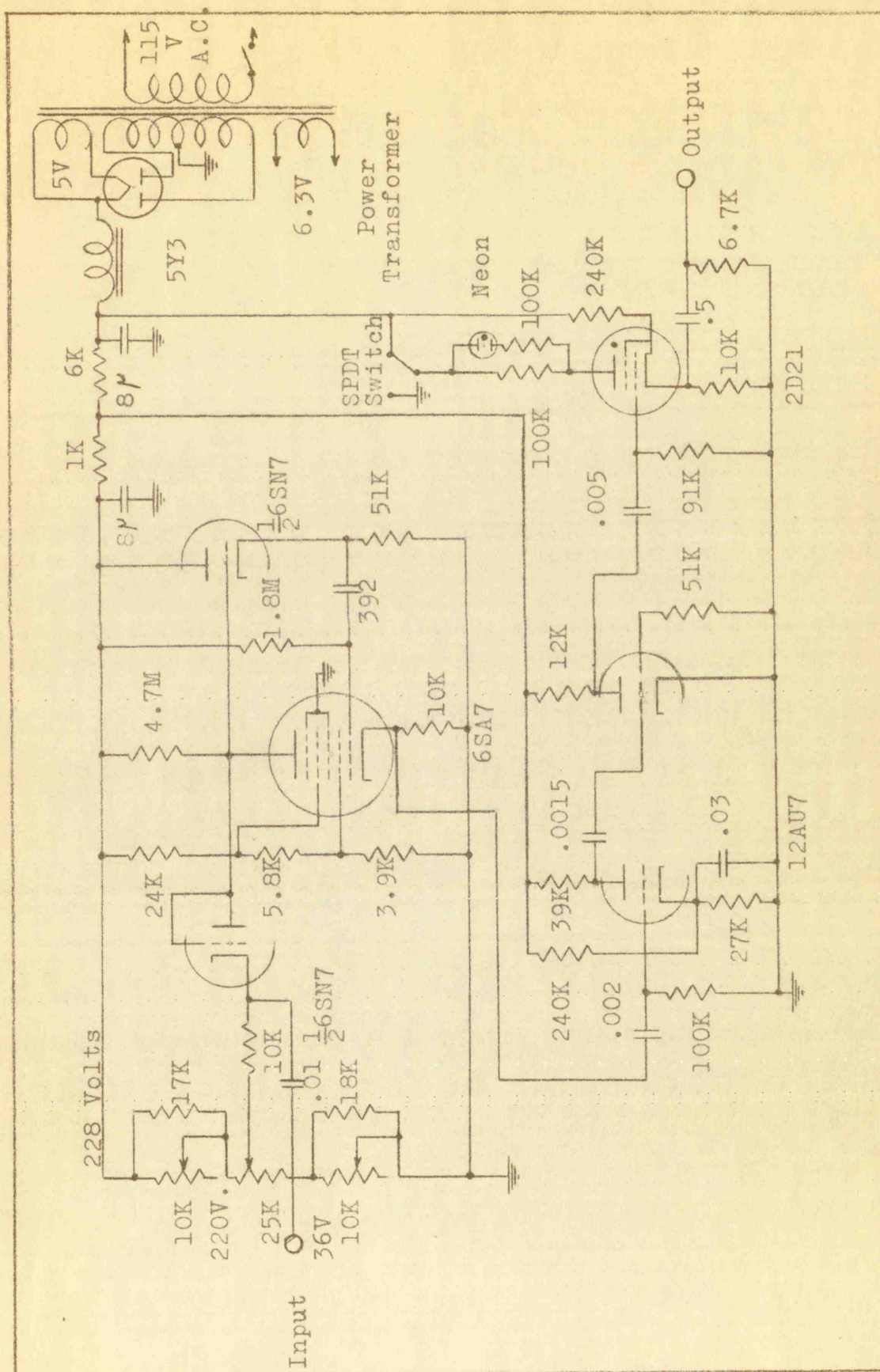
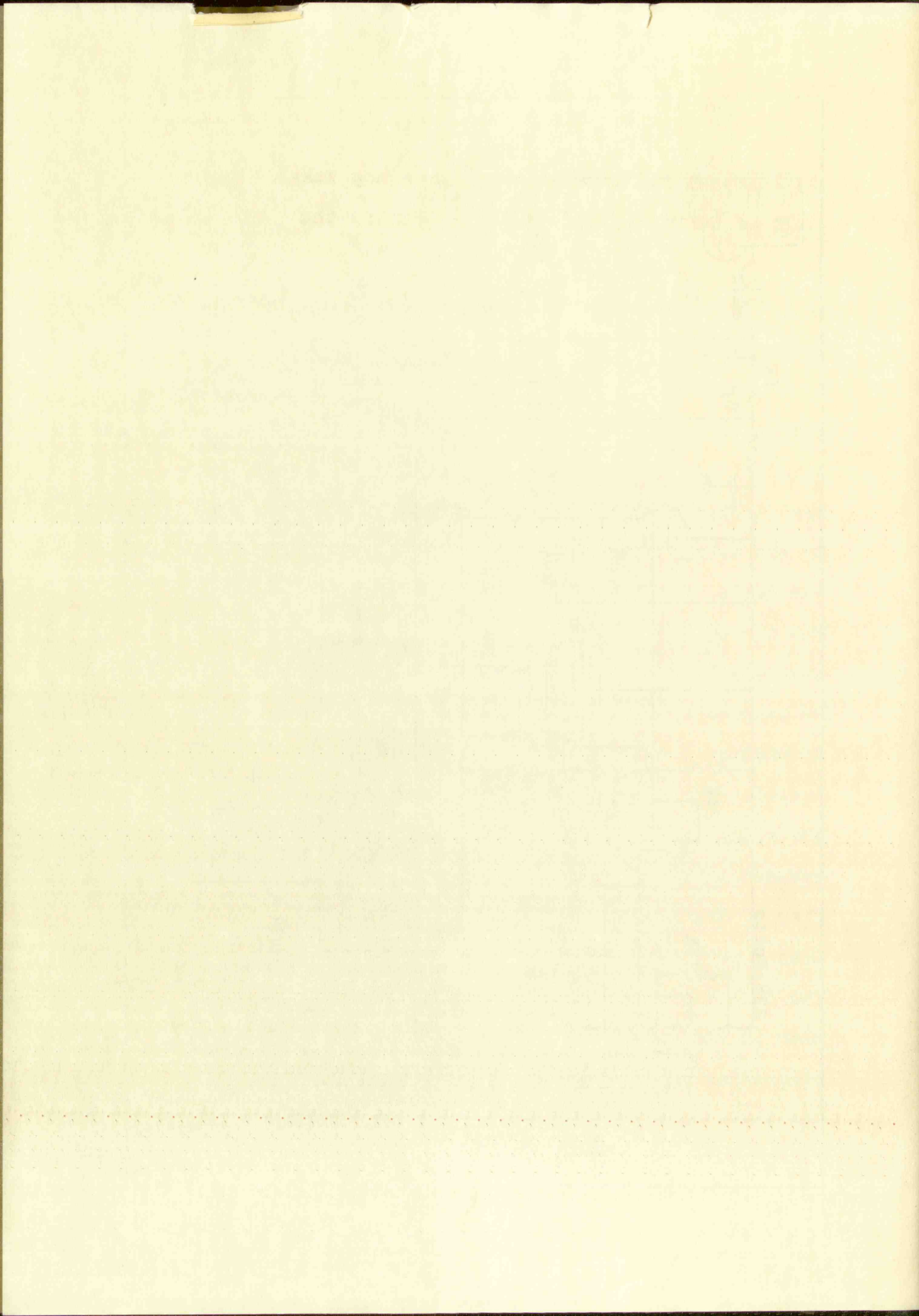


FIGURE 17

THE DELAY CIRCUIT



by load. A positive voltage pulse was taken off to trigger the 4C35 hydrogen thyatron in the light source.

by load. A positive voltage pulse was taken off to
trigger the 4035 hydrogen amplifier in the light source.

CHAPTER IV

TEST PROCEDURE

Assembly of the shock tube. The sections of the shock tube were lined up and bolted together. The test section was checked with a square with respect to both the floor and the spark light source.

Assembly of the cellophane diaphragm. The cellophane was cut and wound around the aluminum frame to give 4 thicknesses, and the frame inserted between the chamber and channel of the shock tube. The top and bottom bolts were snugged down and the alignment of the two sections observed. The side bolts were then drawn down and tightened followed by the final tightening of all bolts. Improper adjustment of the gaskets or bolts caused the cellophane to be strained and to break.

Calculations made for data desired. The barometer was read and corrections made for the temperature. From this corrected reading a value was calculated for the pressure needed in the channel and the delay setting needed in order to obtain the desired results.

Vacuum drawn. The low-vacuum pump was started and vacuum was drawn on the channel section until about 22

CHAPTER IV

TEST PROCEDURE

Assembly of the shock tube.

The shock tube was lined up and bolted together. The test section was checked with a square with respect to both the floor and the spark light source.

Assembly of the calibration apparatus.

The phane was cut and wound around the aluminum frame to give a thickness, and the two tapered sections of the shock tube and channel of the shock tube. The tapered sections were snugged down and the alignment of the two sections observed. The side holes were then covered with tape. The final tightening of the bolts was followed by the final tightening of the bolts. Proper adjustment of the phane of bolts around the calibration phane to be obtained and to be.

Calibration of the shock tube.

was read and corrections made for the temperature. This corrected reading a value was calculated for the pressure needed in the channel and the value needed in order to obtain the desired results.

Vacuum system.

Vacuum was drawn on the system and the system was

inches of mercury was shown on the manometer. If further vacuum was needed for the pressure ratio desired the small Hi-Vac pump was put into operation. The two pumps were not operated at the same time. When the pressure was within about 1/2 inch of the desired pressure in the channel, the amplifier, delay unit, and spark light source were turned on. When the desired pressure had been reached in the channel the vacuum pump was switched off and the high voltage applied to the spark gap to ready it for triggering.

Triggering operation. A test triggering pulse was applied to the amplifier to make certain that it would operate satisfactorily. The amplifier and delay were then reset ready for the trigger pulse. The manometer was sealed off from the shock tube after a final reading. The film was installed in the holder, and the cover slide on the film pack raised. The lanyard operating the diaphragm-breaking device was then pulled to break the cellophane. The sound from the breaking operation traveled down the steel tube to the barium-titanate crystal pickup, was amplified, delayed the amount calculated, and the spark discharged exposing the film. The slide was immediately closed over the film. The time involved between rounds varied from 20 to 30 minutes. If during the triggering

cycle an unexpected voltage surge took place the delay system might be unreliable and that round rejected.

Maintenance. Between test runs the tubing to the filter for the vacuum pumps had to be blown clear of shattered cellophane. After from between 20 to 25 rounds the back plate of the tube had to be taken off and the packed cellophane removed. Also, the test section had to be dismantled and the plate glass windows cleaned. The entire tube was not purged of cellophane particles except at this time and frequently bits of cellophane may be seen in the flow stream in the shadowgraph pictures.

After approximately 100 rounds had been fired the spark head had to be disassembled and cleaned. A 5/64 inch drill was run down the passage in order to clean it of tungsten particles.

cycle an unexpected voltage surge took place the relay system might be unreliable and the record rejected.

Maintenance. Between each run the tubing to the

filter for the vacuum pump had to be blown clean of shattered cellophane. After each between 10 to 15 rounds the back plate of the tube had to be taken off and the soaked cellophane removed. Also, the seal between had to be dismantled and the plate glass window cleaned. The entire tube was not changed or cellophane replaced except at this time and frequently bits of cellophane may be seen in the flow stream in the photograph chamber. After approximately 100 rounds had been fired the spark head had to be disassembled and cleaned. A 1/2 inch drill was run down the passage in order to clean it of hardened particles.

CHAPTER V

CALIBRATION OF THE APPARATUS

Delay unit calibration. The delay unit was calibrated by the use of a Textronic 514 D Oscilloscope, with a Krohn-Hite push button oscillator Model 440-A used to calibrate the oscilloscope sweep. The delay unit and the oscilloscope were triggered with a 22 1/2 volt battery. Figure 18 shows a plot of the calibration curve with the voltages used at the high and low ends of the helipot. See Figure 17 for the schematic diagram on the delay unit.

Velocity measurements in the shock tube. Since the shock tube was designed to be used for aerodynamic investigations, it was of first importance that the flow become stabilized at any desired Mach number. The usefulness of the tube was determined by the duration of constant flow.

Shock tube theory predicts a constant flow following the primary shock wave of 800 microseconds at Mach 1.15 with a pressure ratio of 100. The theory also predicts a constant flow for approximately 2900 microseconds following the contact surface. Steady flow following the

CHAPTER 7

CALIBRATION OF THE APPARATUS

Delay unit calibration

Delayed by the use of a Tektronix 114 D Oscilloscope, a Krohn-Hite push button oscilloscope Model 440-A used to calibrate the oscilloscope sweep. The delay unit and the oscilloscope were triggered with a 25,000 volt battery. Figure 18 shows a plot of the calibration curve with the voltages used at the high and low ends of the battery. See Figure 17 for the schematic diagram of the delay unit.

Velocity measurements in the shock tube

The shock tube was designed to be used for velocity measurements, it was of that importance that the flow become stabilized at any desired Mach number. The uniformity of the tube was determined by the duration of constant flow.

Shock tube theory predicts a constant flow follow-

ing the primary shock wave of 500 microns and Mach 1.15 with a pressure ratio of 100. The shock wave is a constant flow for approximately 100 microns. Following the constant surface, shock wave reflecting

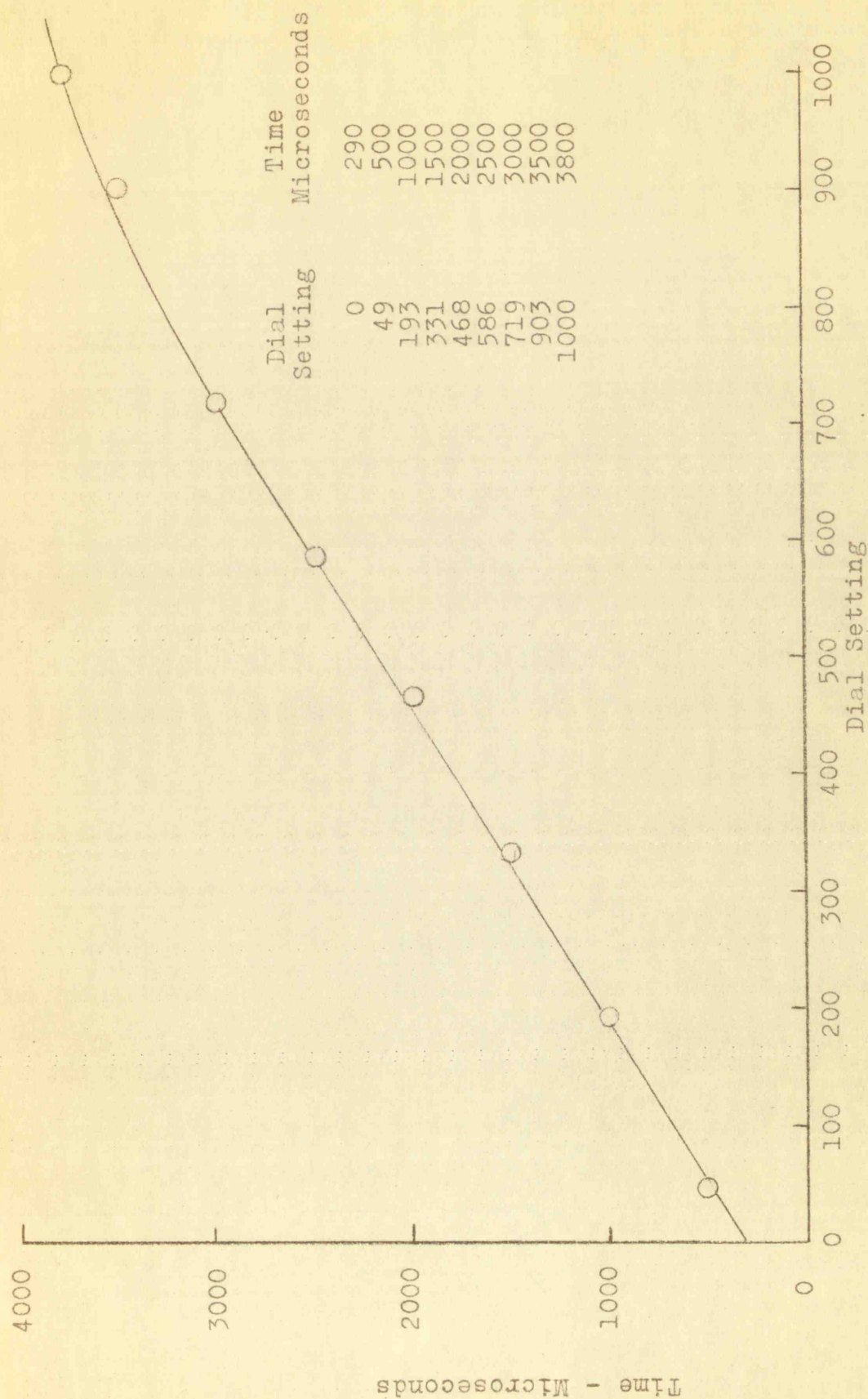
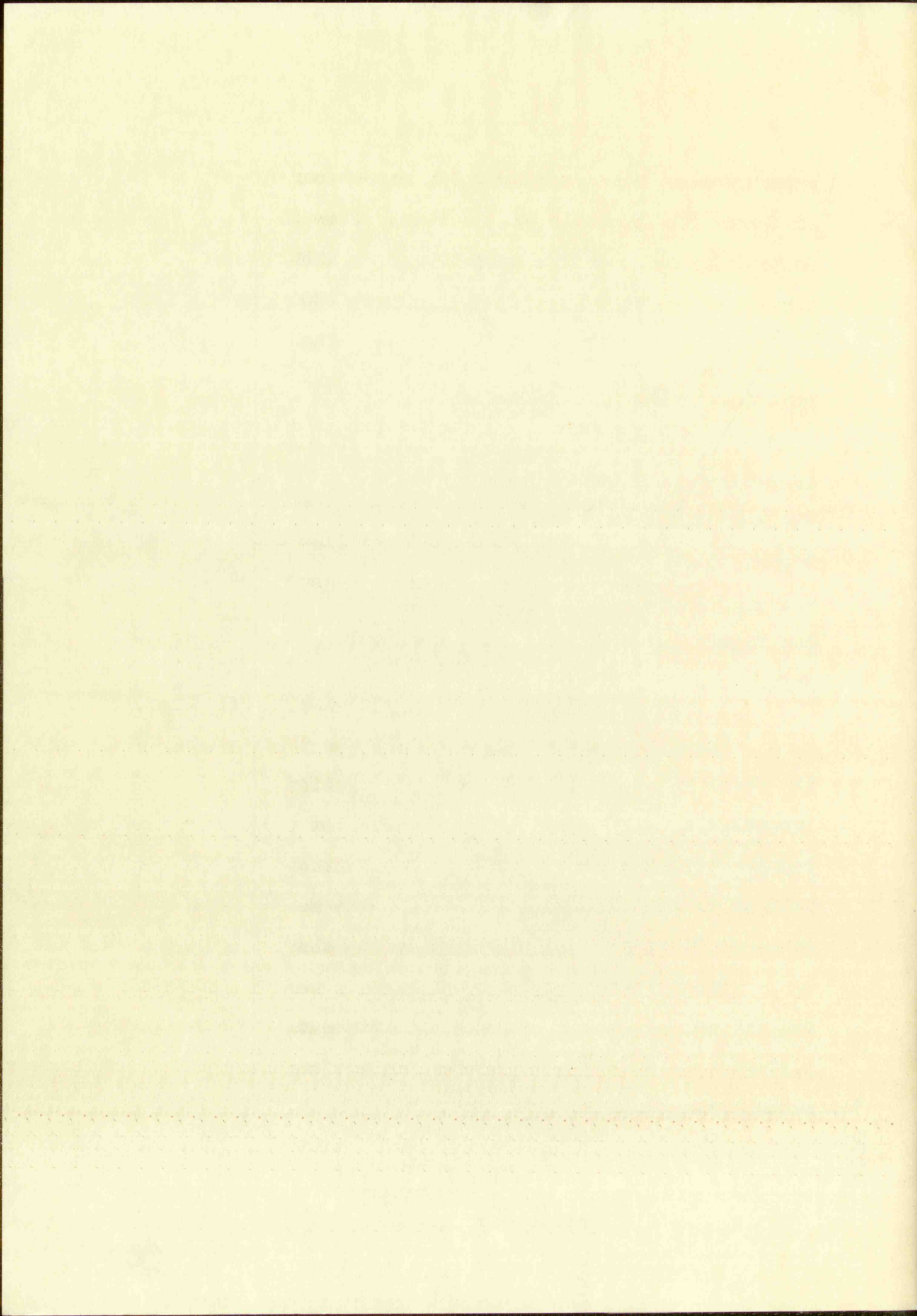


FIGURE 18

CALIBRATION OF THE DELAY CIRCUIT



primary shock wave was about 300 microseconds as shown in Figure 19 and Table VI. A steady flow following the contact surface did not actually occur although velocity measurements could be made up to about 2000 microseconds after passage of the contact surface. The delay unit was designed to provide for a maximum delay time of 3800 microseconds, and about 1900 microseconds were required from the pickup trigger until the contact surface had passed the test window. After the contact surface had passed the test section, the measured flow velocities dropped off as the delay time was increased. At this pressure ratio either the flow velocity decreased with longer time delay or the flow could not stabilize because the test section was too close to the diaphragm.

According to the ideal shock tube theory, steady flow of about 2100 microseconds was predicted following the primary shock wave, and a steady flow of 3300 microseconds was predicted following the contact surface at a pressure ratio of about 40. The flow Mach following the primary shock wave was found to be about 1.00. The flow Mach following the contact surface was found to be about 1.75 and it did not seem to attenuate with time as had the flow Mach following the contact surface at a pressure ratio of 100.

primary shock wave was about 100 microseconds in flow
in Figure 19 and Table VI. A steady flow following the
contact surface did not actually occur although velocity
measurements could be made up to about 2000 microseconds
after passage of the contact surface. The delay until
was designed to provide for a maximum delay time of 200
microseconds, and about 1000 microseconds were required
from the pickup trigger until the contact surface had
passed the test window. After the contact surface had
passed the test section, the measured flow velocity
dropped off as the delay time was increased. At this
pressure ratio either the flow velocity decreased with
longer time delay or the flow could not be obtained because
the test section was too distant from the pickup.
According to the test shock tube theory, the
flow of about 2100 microseconds was predicted following
the primary shock wave, and a steady flow of 2100 micro-
seconds was predicted following the contact surface as
a pressure ratio of about 1.75. The flow was following
the primary shock wave was found to be about 1.75. The
flow Mach following the contact surface was found to be
about 1.75 and it did not seem to increase with time as
had the flow Mach following the contact surface at
pressure ratio of 1.75.

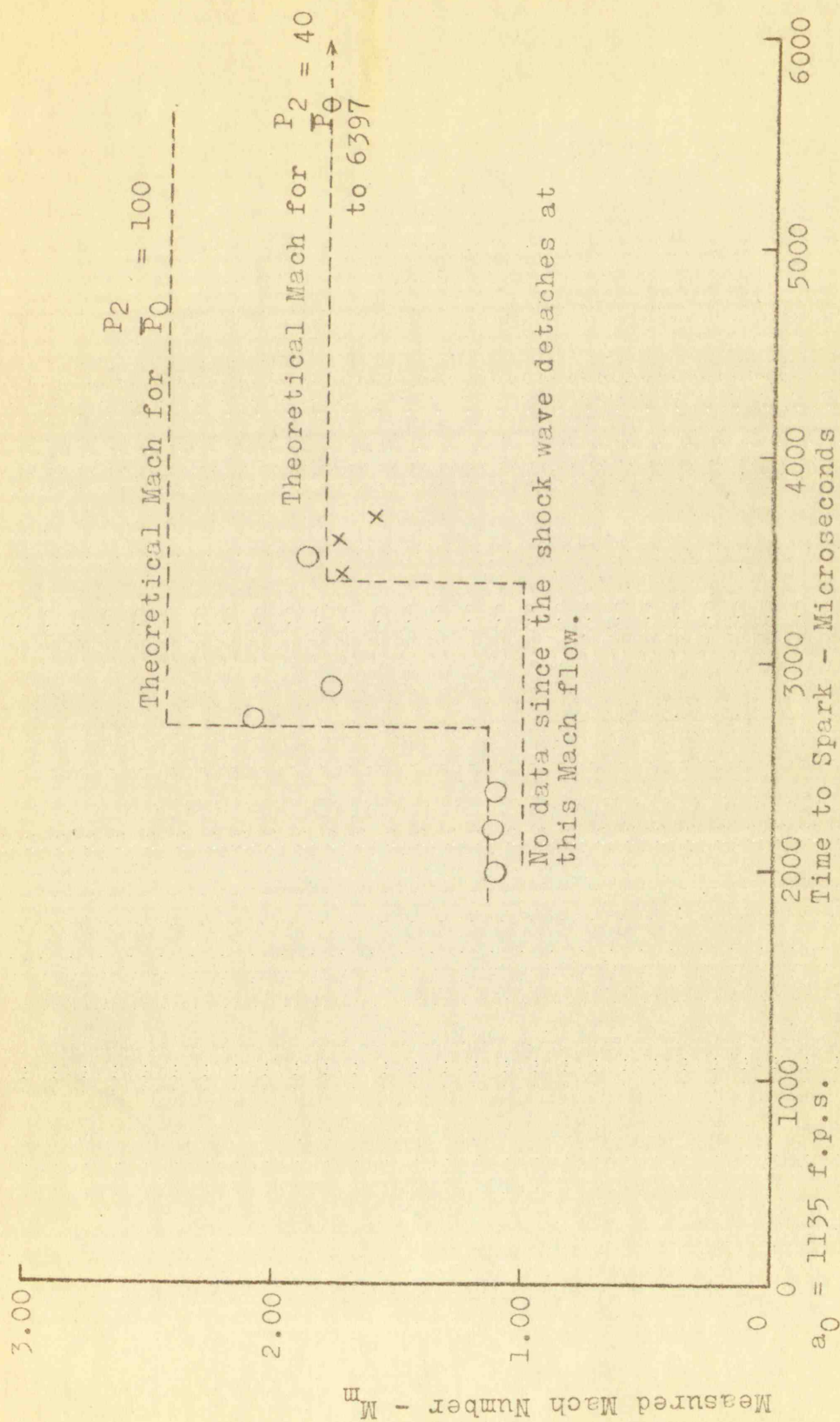


FIGURE 19

COMPARISON BETWEEN THEORETICAL MACH FLOW AND MEASURED MACH FLOW AT PRESSURE RATIOS OF 100 AND 40

The predicted Mach number was obtained from the original shock tube pressure ratio, and was calculated by use of the ideal shock tube theory.

The measured Mach number was calculated from the attached shock wave angle and in this paper is designated as M_m . Nonviscous flow was assumed in the calculation.

When the wedge of half-angle θ was placed in the flow stream, the measured Mach number for steady flow was obtained by¹

$$\frac{1}{M_m^2} = \sin^2 \beta - \frac{\gamma+1}{2} \frac{\sin \beta \sin \theta}{\cos(\beta-\theta)} \quad (V-1)$$

where β was the inclination of the bow-wave in the direction of the flow as indicated in Figure 20.

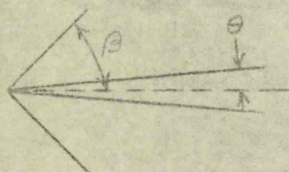


Figure 20. Shock Wave Attached to Wedge

Several wedges with different total angles were used in early experiments. A soft wood wedge became battered immediately. A hardwood wedge was used for some time but eventually became battered from cellophane particles. An aluminum wedge of 18 1/2 degrees included angle

¹H. W. Liepmann and A. E. Puckett, Introduction of Aerodynamics of a Compressible Fluid, (New York: John Wiley and Sons, Inc., 1947), pp. 51-58.

The predicted Mach number was obtained from the original shock tube pressure ratio, and was calculated by use of the ideal shock wave theory.

The measured Mach number was obtained from the attached shock wave angle and its ratio to the upstream flow was assumed in the calculation.

When the wedge of half-angle θ is placed in the flow stream, the measured Mach number for steady flow was obtained by¹

$$\frac{1}{M_{\infty}^2} = \frac{1}{\sin^2 \theta} \left(\frac{1 + \gamma}{2} - \frac{\gamma + 1}{2 \cos^2 \theta} \right) \quad (\gamma = 1.4)$$

where θ was the inclination of the shock wave in the direction of the flow as indicated in Figure 2C.



Figure 2C. Shock Wave Attached to Wedge

Several wedges with different shock angles were used in early experiments. A soft wood wedge became battered immediately. A hardened wedge was used for some time but eventually became battered from reflection of the shock. An aluminum wedge of 15.1 degrees thickness was used.

¹H. W. Liepmann and A. E. Driscoll, *Investigation of Aerodynamics of a Compressible Fluid Flow*, John Wiley and Sons, Inc., 1949, pp. 11-12.

was also used but the leading edge became rough and had to be resharpened several times. For final velocity measurements a hardened steel wedge was accurately ground to a total angle of 3 degrees. Using this wedge, and with the assumption that 1.400 for air, equation (V-1) can be written

$$M_m = \frac{1}{\sqrt{\sin^2 \beta - \frac{0.0314 \sin \beta}{\cos(\beta - 1.5)}}} \quad (V - 2)$$

This equation was plotted and the graph used to read M_m . Figure 21 shows this graph. This graph shows that the lower the measured angle the higher the Mach number. Note that the steepness of the curve decreases with decreasing Mach number until at 77 degrees the curve again swings upward. At that point the Mach number was 1.098. No measurement of the Mach number was possible below 1.098 since the bow wave detached at any lower Mach flow. After the leading edge of the wedge became slightly roughened from the battering of cellophane particles the shock detached at Mach 1.12 and measurement of the angle becomes difficult.

In order to obtain the shock wave angle as a function of time, it was necessary to take a series of pictures, using the same pressure ratio. The spark delay was varied for each picture. The resulting shadowgrams might be compared to a high-speed motion picture of a single shock wave.

was also used but the leading edge became rough and had to be resurfaced several times. For final velocity measurements a hardened steel wedge was used. Using this wedge, and with the assumption that $\mu = 1.40$ for air, equation (5-1) can be written

$$M = \frac{0.034 - 2.119}{\cos(9 - 1.1)} \sqrt{2.119 - 0.034} = 1.40$$

This equation was plotted and the graph used to read M . Figure 21 shows this graph. This graph shows that for lower the measured angle the higher the Mach number. Note that the steepness of the curve decreases with increasing Mach number until at 75 degrees the curve again turns upward. At that point the Mach number was 1.40. The measurement of the Mach number was possible below 1.00 since the bow wave detached at very lower Mach flow. After the leading edge of the wedge became slightly rougher from the battering of successive particles the shock detached at Mach 1.15 and measurement of the angle became difficult.

In order to obtain the exact wave angle as a function of time, it was necessary to take a series of photographs using the same pressure ratio. The shock delay was varied for each picture. The resulting measurements of the shock compared to a high-speed motion picture of a shock wave.

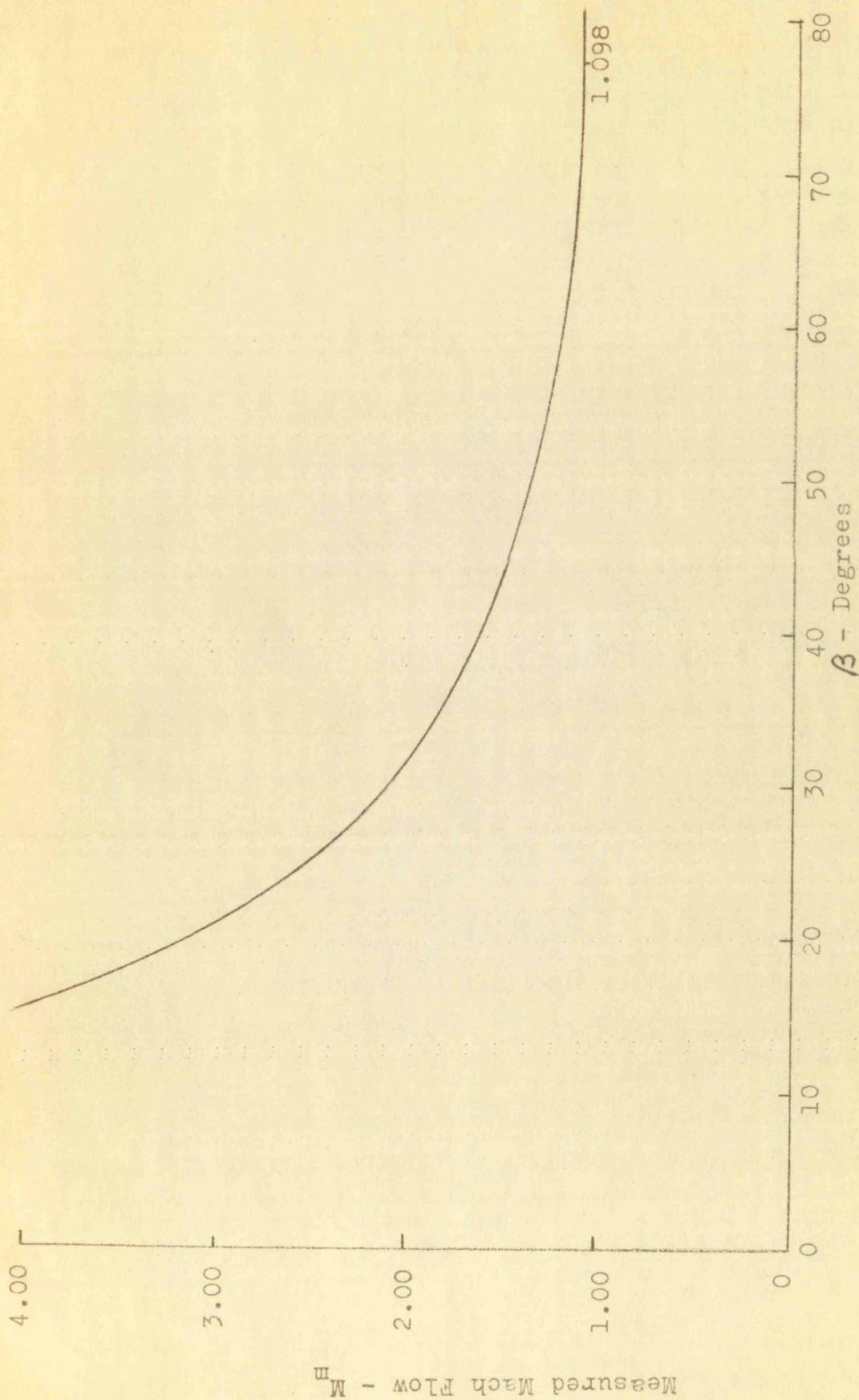


FIGURE 21

ONE-HALF BOW WAVE ANGLE PLOTTED AGAINST MEASURED MACH FLOW FOR A
THREE-DEGREE WEDGE

The variation between the measured Mach number of the flow following the primary shock wave and the predicted Mach flow may be seen in Figure 8. It was found that the Mach flow closely followed that predicted by the theory between 1.10 and 1.18. This region was of little use, however, since the duration of flow was only about 300 microseconds. Above Mach 1.18 the density of air was not sufficient for the use of shadowgraph pictures.

The variation between the measured Mach number of the flow following the contact surface and the predicted Mach flow may be seen in Figure 9. The flow can not be predicted accurately at pressure ratios greater than 40 or with Mach numbers greater than 1.75. Below a pressure ratio of 25, results are not obtainable because of the limitations of the present electronic delay system to 3800 microseconds.

Measurement of the shock wave angle. The angle formed by the attached shock wave, 2β , was measured directly from the photographic film. The method followed was to measure the largest included angle to nearest half degree. The error involved in one-half degree measurement of 2β varies with the Mach number as can be seen in Figure 21. The higher the Mach number the steeper the slope and the greater the error. The measurement

The variation between the measured Mach number
of the flow following the primary shock wave and the pre-
dicted Mach flow may be seen in Figure 3. It was found
that the Mach flow closely followed that predicted by
the theory between 1.10 and 1.12. This region was of
little use, however, since the variation of flow was only
about 300 microseconds. Above Mach 1.12 the density of
air was not sufficient for the use of shadowgraph technique.
The variation between the measured Mach number of
the flow following the secondary shock wave and the predicted
Mach flow may be seen in Figure 4. The flow curves were
predicted separately at pressure ratios of 1.10, 1.12,
or with Mach numbers greater than 1.12. The ratio of 1.10
ratio of 1.12, results are not within the accuracy of the
limitation of the present experiment. The Mach number is
3800 microseconds.

Measurement of the shock wave angle

formed by the attached shock wave. The angle was measured directly
from the photograph. The shock wave angle was measured
to measure the largest included angle to the shock wave.
The error involved in the angle measurement was
about 0.5 degrees. The shock wave angle was measured
in Figure 5. The shock wave angle was measured
the slope and the gradient of the shock wave.

was difficult both when the Mach number was small and when the shock tube and spark source were not perfectly aligned. When the Mach number was low the bow waves were ragged, curved, and sometimes double.

Figures 22 and 23 show the three-degree wedge with the attached shock wave at two different Mach numbers. Here the wake as well as the wavering shock lines are visible.

It may be concluded from the data taken at points between 1.10 and 1.18 Mach flow that the actual flow Mach number was slightly low due to energy lost in shattering the cellophane diaphragm² or from boundary losses³ due to the walls of the shock tube.

²F. W. Geiger, C. W. Mautz, The Shock Tube as an Instrument for Investigation of Transonic and Supersonic Flow Patterns, (Engineering Research Institute, University of Michigan, Ann Arbor, June 1949), pp. 86-88.

³R. J. Emrich, C. W. Curtis, "Dissipation of a Shock Traveling in a Tube," The Physical Review, 77:573, Feb., 1950.

was difficult both when the Mach number was small and when the shock tube and spark source were not perfectly aligned. When the Mach number was low the bow waves were ragged, curved, and sometimes double. Figures 22 and 23 show the three-degree wedge with the attached shock wave at two different Mach numbers. Here the wake as well as the trailing shock lines are visible.

It may be concluded from the data taken at points between 1.10 and 1.18 Mach flow that the actual flow Mach number was slightly low due to energy lost in chattering the cellophane diaphragm, or from boundary losses due to the walls of the shock tube. The ratio of the velocity of the shock wave to the velocity of the gas behind it is about 1.1. The distance of the shock wave from the diaphragm was about 1.5 cm.

Investigation of the shock wave

Formed by the attached shock wave, the shock wave is seen in the photograph. The shock wave is seen to be curved and ragged.

W. W. Gelfand, C. W. Mautz, The Shock Tube as an Instrument for Investigation of Transonic and Supersonic Flow Patterns, (Engineering Research Institute, University of Michigan, Ann Arbor, June 1949), pp. 86-88.

R. J. Emrich, G. W. Gurtis, "Dispersion of a Shock Traveling in a Tube," The Physical Review, 77:277, Feb., 1950.

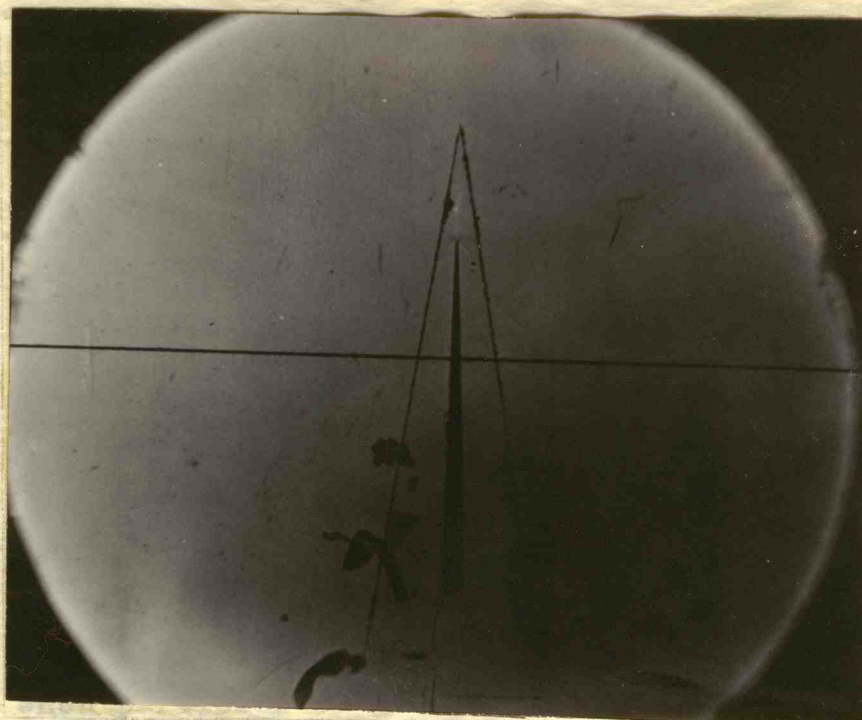


FIGURE 23
Round Number 245

ATTACHED BOW WAVE
 $M_{\infty} = 2.07$, 3° Wedge

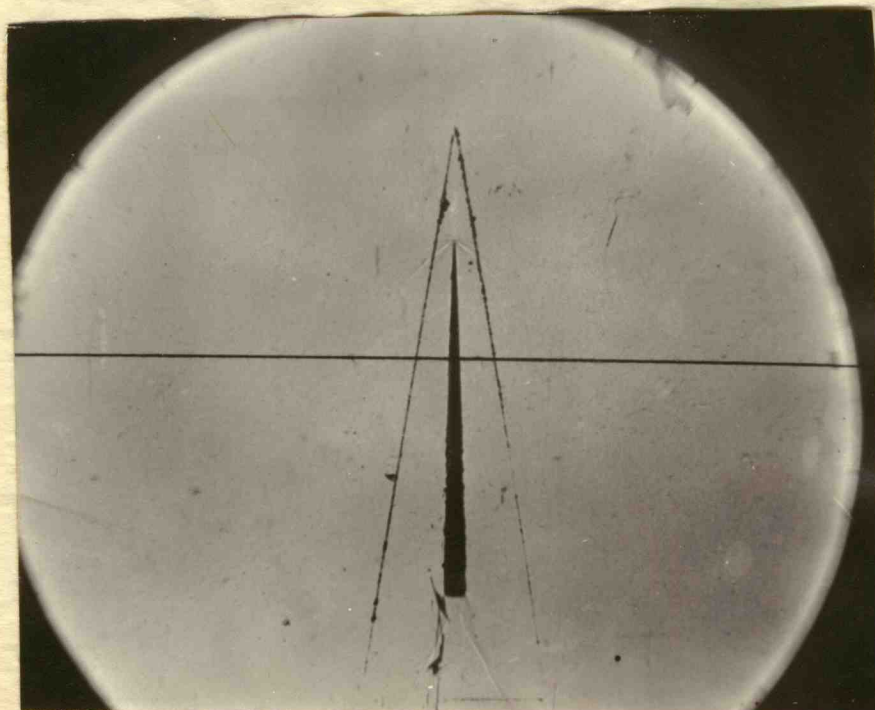


FIGURE 22
Round Number 247

ATTACHED BOW WAVE
 $M_{\infty} = 1.56$, 3° Wedge

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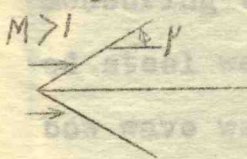
CHAPTER VI

AERODYNAMIC STUDIES

Velocity measurements on models from striation

Mach waves. During preliminary testing it was noted that the saw marks on one face of the wooden wedge caused Mach wavelets to be developed. It seemed feasible that velocity might be measured from these wavelet angles.

Dwinnell¹ points out that a Mach angle is formed when particles of gas at supersonic velocities strike surface roughnesses in passing over an object in the flow path. The Mach wavelet may be seen in some shadowgraph and schlieren pictures when the density is sufficient. Dwinnell used the following equation to show the relationship between Mach wavelet angle and Mach number:



$$\mu = \sin^{-1} \frac{1}{M_{local}} \quad (VI - 1)$$
$$M_{local} = \frac{1}{\sin \mu}$$

Sibert² says that one way of determining the Mach number is to measure the angle between a weak shock and the direction of motion along a body surface and then

¹James H. Dwinnell, Principles of Aerodynamics, (New York: McGraw-Hill Book Company, Inc., 1949), p. 185.

²Harold W. Sibert, High-Speed Aerodynamics, (New York: Prentice-Hall, Inc., 1948), p. 160.

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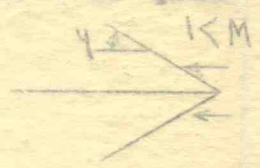
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¹James H. Dainelli, Principles of Aerodynamics, (New York: McGraw-Hill Book Company, Inc., 1942), p. 185.

²Harold W. Sibert, High-Speed Aerodynamics, (New York: Prentice-Hall, Inc., 1948), p. 100.

find the Mach number from the above equation. This necessitates laying off a tangent at the point of flow measurement except in the case of a wedge.

The chance striations caused by the saw cuts in preliminary experiments were too close together to form clearout wavelet angles. Table VII gives data taken from the four pictures in which this occurs. Two of these pictures are shown in Figures 24 and 25. They show correlation between the measured Mach flow and the measured striated angles. The other two pictures show a detached shock wave at the flow velocities measured. The distortion of the shock waves in all four pictures seems to indicate that they are near the detaching point.

A later series of pictures shows the possibilities of accurate local velocity measurement on a model by measuring the Mach wavelet angle. The three-degree hardened steel wedge used in the velocity calibration from the bow wave was used.

First the wedge was scored on one face to a depth of about 0.005 inches at a distance of $1/2$ inch from the leading edge and also to the same depth at a distance of 1 inch from the leading edge on the other face. No satisfactory data were obtained. The 1 inch striation gave some indication of Mach angle but the $1/2$ inch striation showed none. This was attributed to the rougher scoring

find the Mach number from the above equation. This necessitates laying off a tangent at the point of flow measurement except in the case of a wedge.

The chance variations caused by the saw cuts in preliminary experiments were too close together to form clear-cut wavelet angles. Table VII gives data taken from the four pictures in which this occurs. Two of these pictures are shown in Figures 24 and 25. They show correlation between the measured Mach flow and the measured attached angles. The other two pictures show a detached shock wave at the flow velocities measured. The distortion of the shock waves in all four pictures seems to indicate that they are near the detaching point.

A faster series of pictures shows the possibilities of accurate local velocity measurement on a model by measuring the Mach wavelet angle. The three-degree hardened steel wedge used in the velocity calibration from the flow wave was used.

First the wedge was scored on one face to a depth of about 0.005 inches at a distance of $1/8$ inch from the leading edge and also to the same depth at a distance of 1 inch from the leading edge on the other face. No satisfactory data were obtained. The 1 inch station gave some indication of Mach angle but the $1/8$ inch station showed none. This was attributed to the rougher scoring

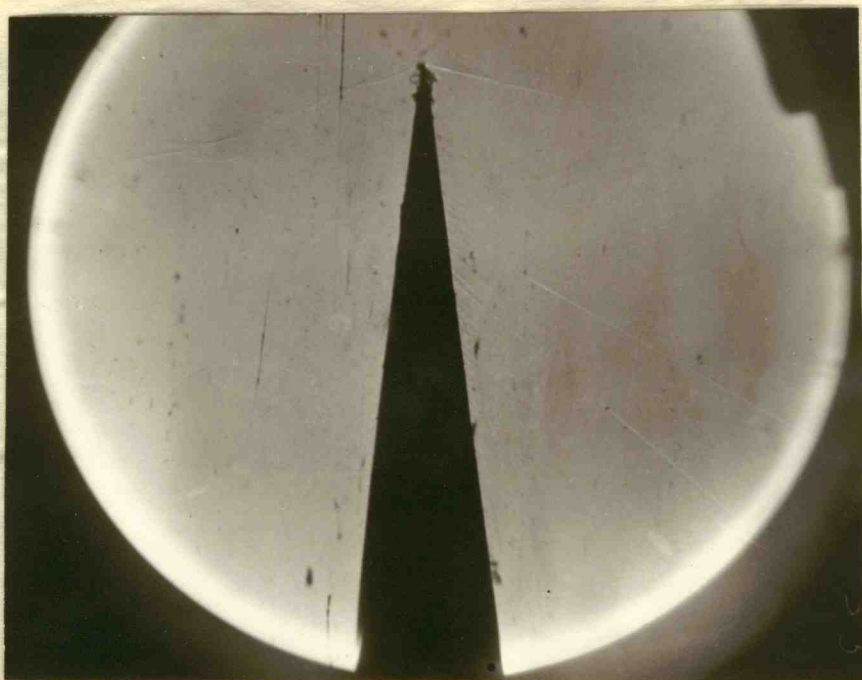


FIGURE 25

Bow Wave $M_m = 1.34$
 Striation $M_m = 1.44$ to 1.27



FIGURE 24

Bow Wave $M_m = 1.42$
 Striation $M_m = 1.39$ to 1.31

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at the 1 inch striation.

Following the unsatisfactory results from notching the wedge, a strip of 0.0015 inch cellophane tape was applied to the wedge as a striation to interrupt the flow. This proved satisfactory and excellent agreement resulted from measurements made on the striations and on the bow wave. The tape striation, while temporary, proved satisfactory for a series of 30 pictures. See Table IX for the tabulated results. Figures 26, 27, 28 and 29 show representative results.

From this series of pictures it appears that local velocities can be satisfactorily measured on models by a striation across the model at any point local velocity is desired in the supersonic range.

Size effect of models on shock tube flow. In wind tunnels as the flow velocity increases the tunnel finally reaches a choked condition where the boundary layers of air build up in thickness and obscure the observed flow, distorting force measurements on the models. The shock tube on the other hand has a slow build up of these boundary layers, and data may be obtained before choking occurs. Figures 30, 31, 32, and 33 show the build up to this choking condition. Sibert³ shows that the choking Mach flow

³Sibert, op. cit., pp. 85-86.

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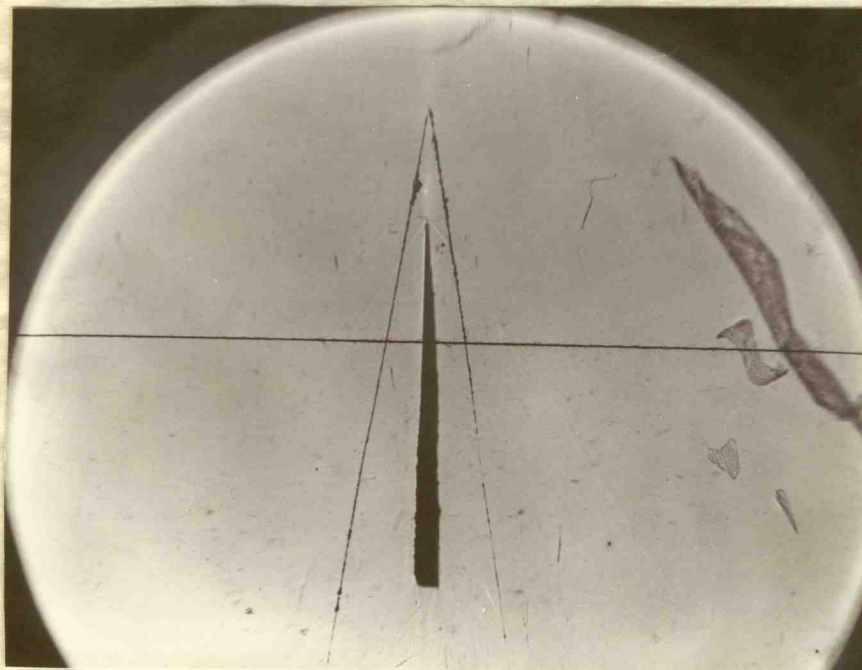


FIGURE 26

Round Number 257

Bow Wave $M_n = 2.28$

Mach Wave $M_n = 2.28$

Delay Time - 3850 Microseconds

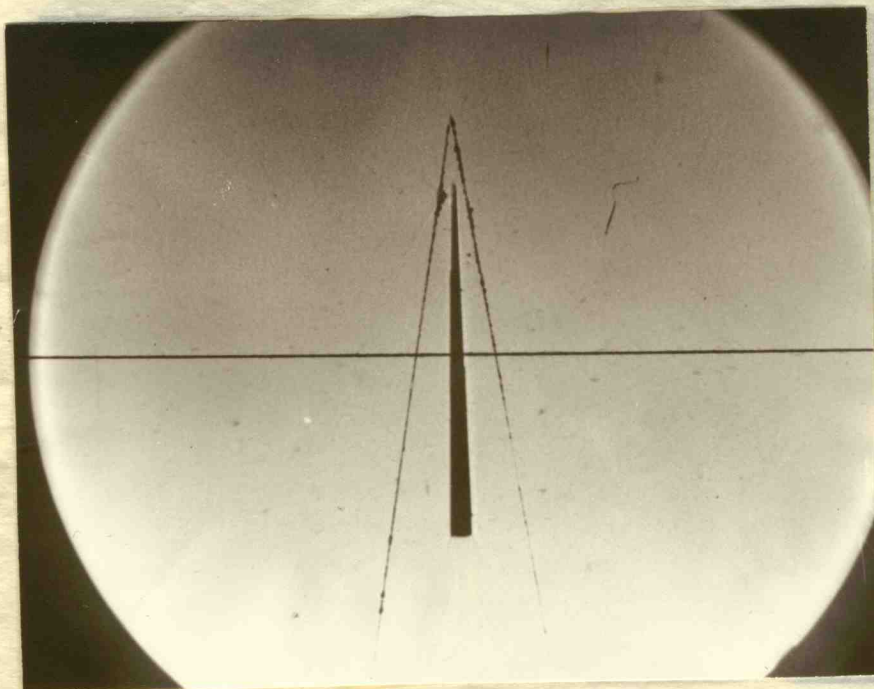


FIGURE 27

Round Number 262

Bow Wave $M_n = 2.03$

Mach Wave $M_n = 2.03$ and 2.00

Delay Time - 3300 Microseconds

Depth time - 2820 microns

Depth time - 2820 microns

Depth time - 2820 microns

Depth time - 2820 microns

Depth time - 2820 microns

Depth time - 2200 microns

Depth time - 2200 microns

Depth time - 2200 microns

Depth time - 2200 microns

Depth time - 2200 microns

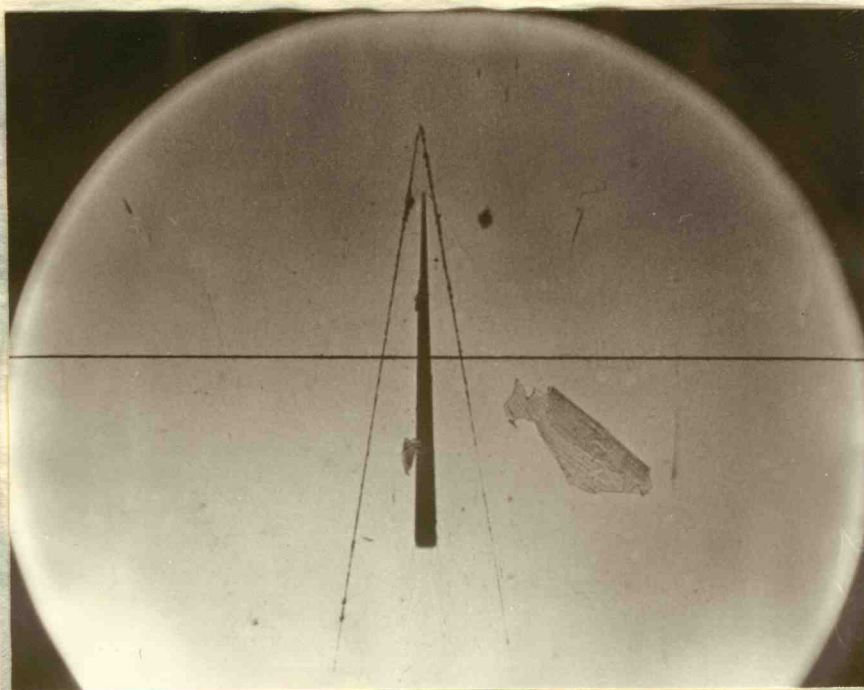


FIGURE 28
Round Number 282

Bow Wave $M_M = 2.10$
Mach Wave $M_M = 2.10$

Delay Time - 2640 Microseconds

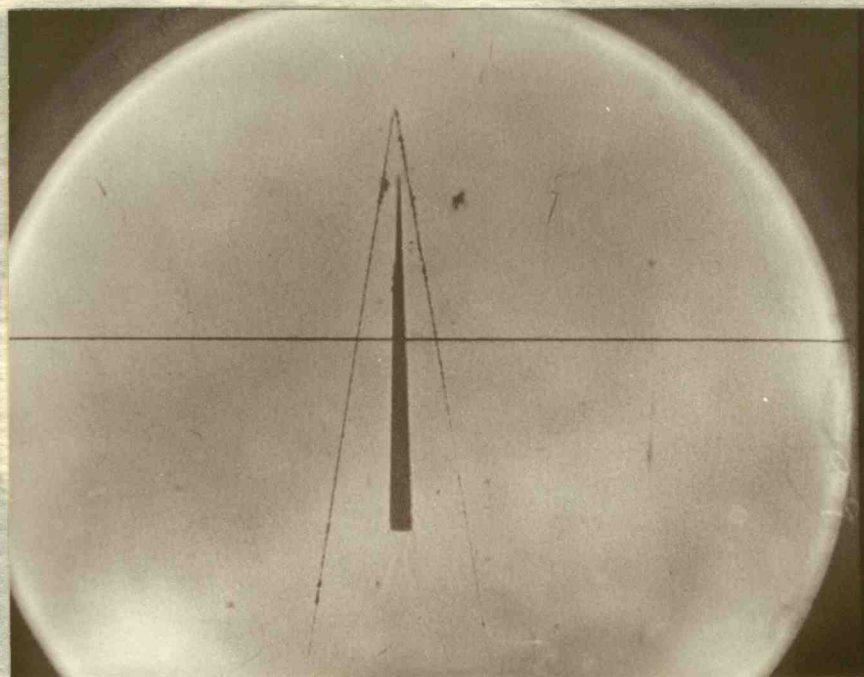


FIGURE 29
Round Number 289

Bow Wave $M_M = 1.60$
Mach Wave $M_M = 1.60$

Delay Time - 3675 Microseconds

12
DATE: June 24, 1950

FROM: Mr. J. H. ...

TO: Mr. J. H. ...

RE: ...

DATE: June 24, 1950

FROM: Mr. J. H. ...

TO: Mr. J. H. ...

RE: ...

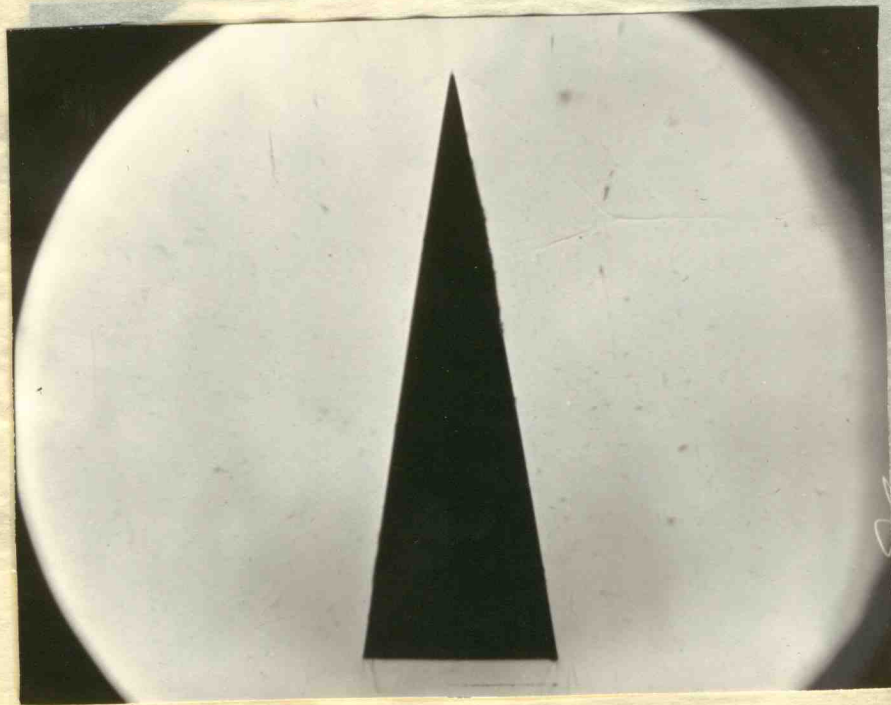


FIGURE 30

Round Number 33

Aluminum Wedge 18 1/2-degrees

MPS = 1.12

Time after Primary Shock

Wave Passage = 300 Microseconds

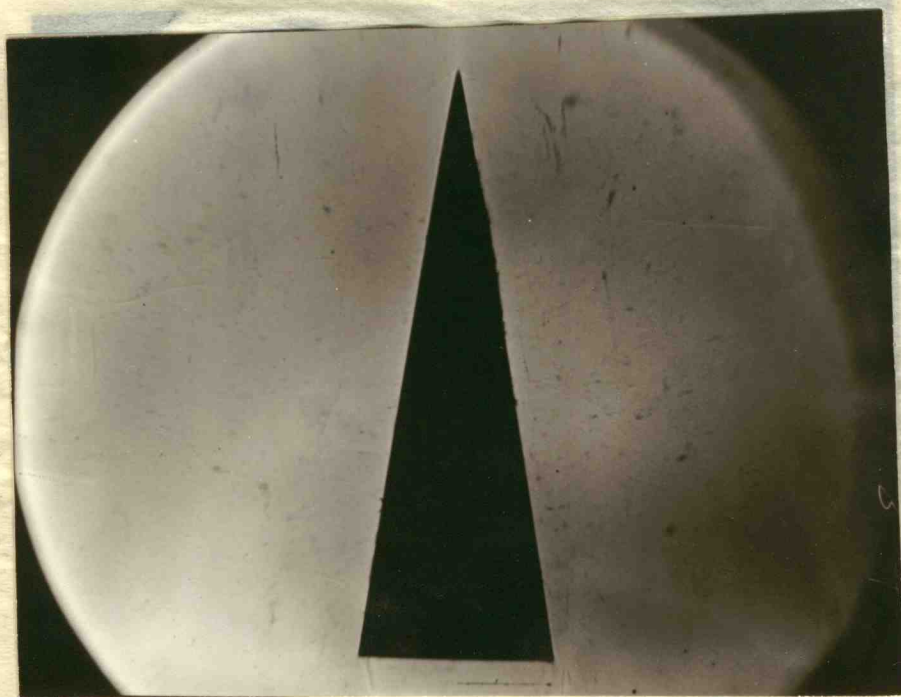


FIGURE 31

Round Number 34

Aluminum Wedge 18 1/2-degrees

MPS = 1.12

Time after Primary Shock

Wave Passage = 500 Microseconds

Анализ - 200 мг
время - 20 мин
температура - 200 °C
время - 20 мин

Время - 20 мин
температура - 200 °C

Анализ - 200 мг
время - 20 мин
температура - 200 °C
время - 20 мин

Время - 20 мин
температура - 200 °C

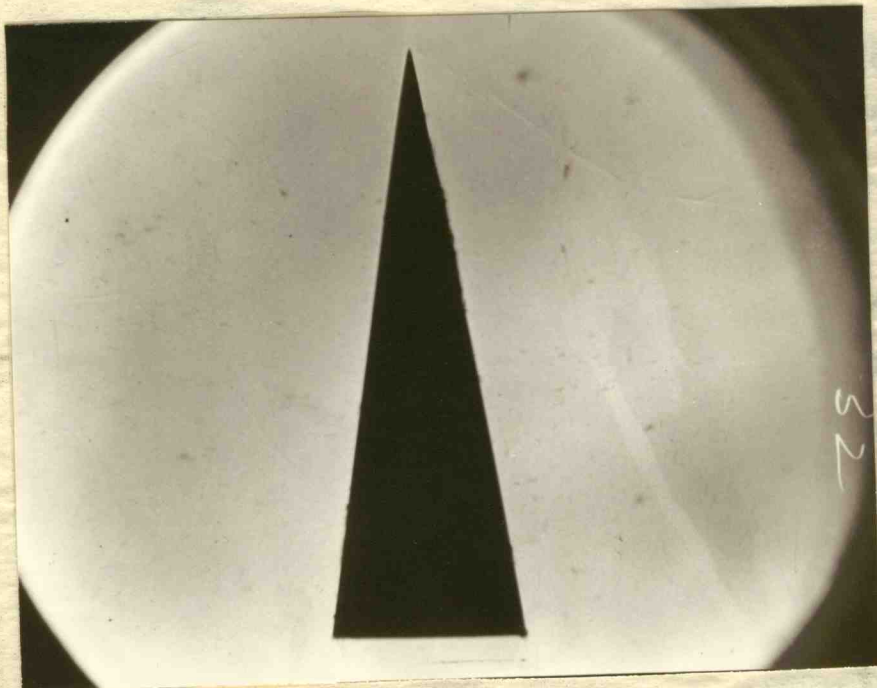


FIGURE 33
Round Number 32
Aluminum Wedge $19\frac{1}{2}$ -Degrees
MPS = 1.12
Time after Primary Shock
Wave Passage = 900 Microseconds

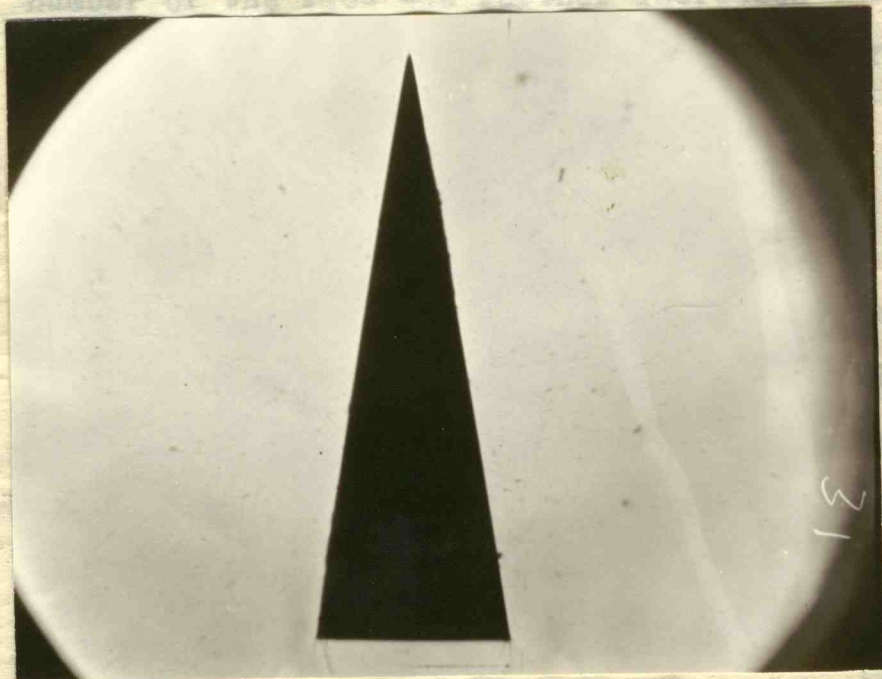


FIGURE 32
Round Number 31
Aluminum Wedge $18\frac{1}{2}$ -Degrees
MPS = 1.12
Time After Primary Shock
Wave Passage = 700 Microseconds

Мале Бвеведе = 100 Иголосесонга
Леме вдез блитиуа гроок

№12 = 1.1.15

Угитини дезде 18 1/3-дегесе

Нонну динрет 21

Мидне 25

Мале Бвеведе = 300 Иголосесонга
Леме вдез блитиуа гроок

№12 = 1.1.15

Угитини дезде 18 1/3-дегесе

Нонну динрет 25

Мидне 25

can be determined from the following equation if the area at the model location is known and if the presented area of the model is known.

$$\frac{A_{model}}{A_{throat}} = 1 - \frac{1.728 M_{CH}^3}{(1 + 0.2 M_{CH}^2)^3} \quad (VI - 2)$$

A derivation of the equation may be found in Appendix VI. The wedge used for these experiments had a presented area in cross section of 1.840 square inches. The shock tube cross sectional area was 11.270 square inches. The indicated Mach number at which choking occurred in this duct was 0.67.

When the critical speed, Mach 0.67 of the model, is exceeded a shock wave formed on the model. As the Mach number of the flow was further increased the shock extended from the model through the flow stream and finally reached the top and bottom walls of the tube. M_{CH} , the wind tunnel choking Mach number, is defined as the flow Mach number at which the shock wave extended completely across the tunnel at the model. When this occurred, essentially sonic speed existed at the model location. At velocities above the choking Mach number the tunnel acted like a nozzle rather than a wind tunnel and any aerodynamic data taken in this range were questionable.

Figure 30 shows a lambda shock wave attached to the model. Figure 31 shows the lambda shocks slightly more

can be determined from the following equation if the area at the model location is known and if the presented area of the model is known.

$$\frac{A_{\text{model}}}{A_{\text{throat}}} = 1 - \frac{1.728 M_{\text{th}}^2}{(1 + 0.2 M_{\text{th}}^2)} \quad (\text{VI} - 2)$$

A derivation of the equation may be found in Appendix VI. The wedge used for these experiments had a presented area in cross section of 1.340 square inches. The shock tube cross sectional area was 11.370 square inches. The indicated Mach number at which choking occurred in this duct was 0.67.

When the critical speed, Mach 0.67 of the model, is exceeded a shock wave formed on the model. As the Mach number of the flow was further increased the shock extended from the model through the flow stream and finally reached the top and bottom walls of the tube. M_{th} , the wind tunnel choking Mach number, is defined as the flow Mach number at which the shock wave extended completely across the tunnel at the model. When this occurred, essentially sonic speed existed at the model location. At velocities above the choking Mach number the tunnel acted like a nozzle rather than a wind tunnel and any aerodynamic data taken in this range were questionable.

Figure 30 shows a lambda shock wave attached to the model. Figure 31 shows the lambda shocks slightly more

developed. Figure 32 shows the lambda shocks fully developed, and with Figure 33 the shocks have widened out further to reach the top and bottom walls of the shock tube. The bow wave will probably detach from the leading edge of the wedge. If the glassed sides of the tube could be viewed from top to bottom this could easily be seen in its entirety. In all of these pictures the expansion region at the corners of the wedge can be seen.

Observation of rapid choking of the shock tube.

Several shadowgraph pictures were made of a large wedge with 90 degrees included angle. Figure 34 shows the primary shock wave just past the tip of the wedge and the reflection taking place from this blunt object. In Figure 35 the primary shock wave has passed and the reflected and detached wave is well along toward choking the shock tube. The wake trailing out from the rear of the wedge can also be seen. With this wedge the shock tube will choke at about Mach 0.35.

Aerodynamic experiments with the NACA 0009 airfoil.

An NACA 0009 airfoil model of aluminum 3 inches long was mounted in the test section. Rubber gaskets were used between the model and the glass. It was hoped that the critical flow Mach number might be found but due to the limitations of shadowgraph photography it could not be

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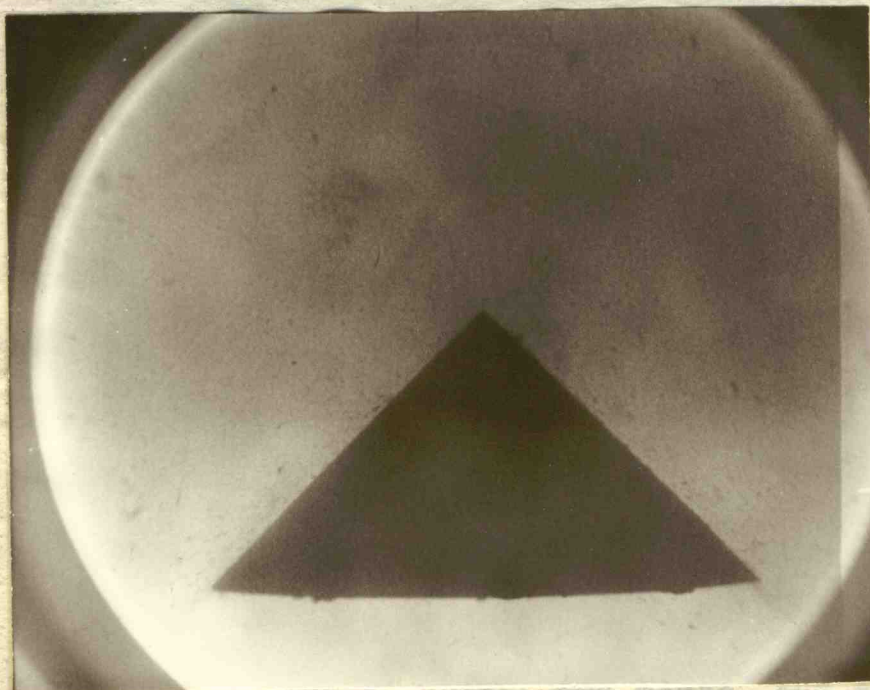


FIGURE 35
Round Number 4
 $M = 1.20$
100 Microseconds after the
Primary Shockwave

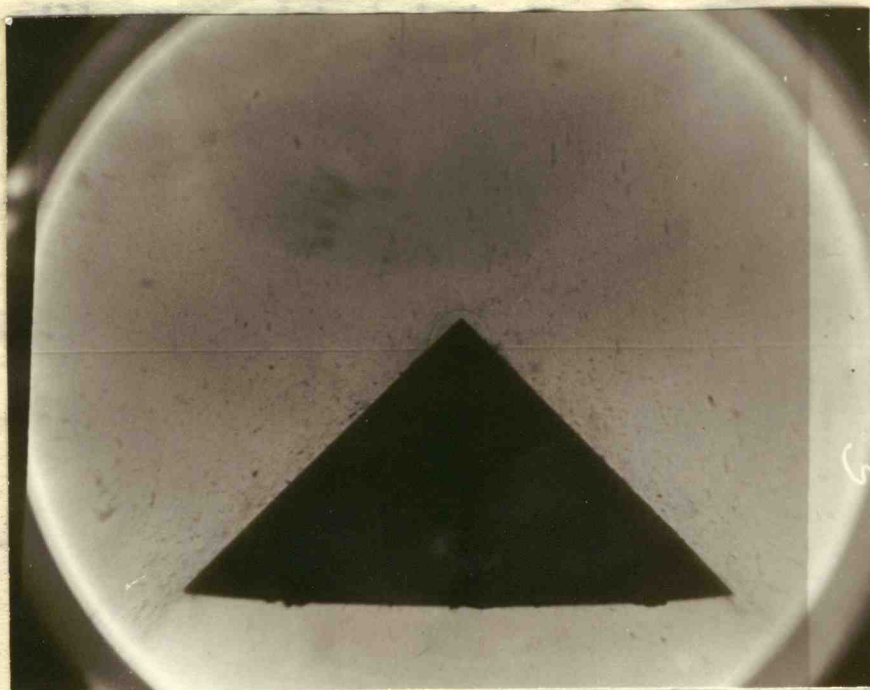


FIGURE 34
Round Number 3
 $M = 1.20$
Primary Shockwave

БЛАНКЕТЫ СПОСОБОВ
100 ПРОЦЕНТОВЫХ СПОСОБОВ
M = 1.50

КОМПЬЮТЕРЫ
ЛИСТЫ 12

БЛАНКЕТЫ СПОСОБОВ
M = 1.50

КОМПЬЮТЕРЫ
ЛИСТЫ 24

determined.

The continual changing of conditions due to flow disturbances in the shock tube make it necessary to make all angle measurements as close as possible to the model where the flow has stabilized. Figures 36 and 37 show the relatively fixed position of the detached shock wave and the changing position and shape of the shock wave further out from the model. Experimental tests were made at both subsonic and supersonic flow velocities to view effects on the model.

According to Geiger and Mautz⁴ the study of the detached shock wave can be put on a quantitative basis. For Mach numbers greater than unity a blunt-nosed model will cause a detached shock wave. The detached shock wave in the shock tube originates at the time when the primary shock wave reaches the nose of the model and proceeds against the flow reaching a stationary limiting position. This is true providing that the tube does not become choked. The distance from the most forward part of the wave to the model can be measured on the film and plotted as a function of time.

⁴F. W. Geiger and C. W. Mautz, The Shock Tube as An Instrument for the Investigation of Transonic and Supersonic Flow Patterns, (Engineering Research Institute, University of Michigan, Ann Arbor, Mich.) pp. 110-116.

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⁴F. W. Geiger and C. W. Maue, The Shock Tube as an Instrument for the Investigation of Transonic and Supersonic Flow Patterns, (Engineering Research Institute, University of Michigan, Ann Arbor, Mich.) pp. 110-116.

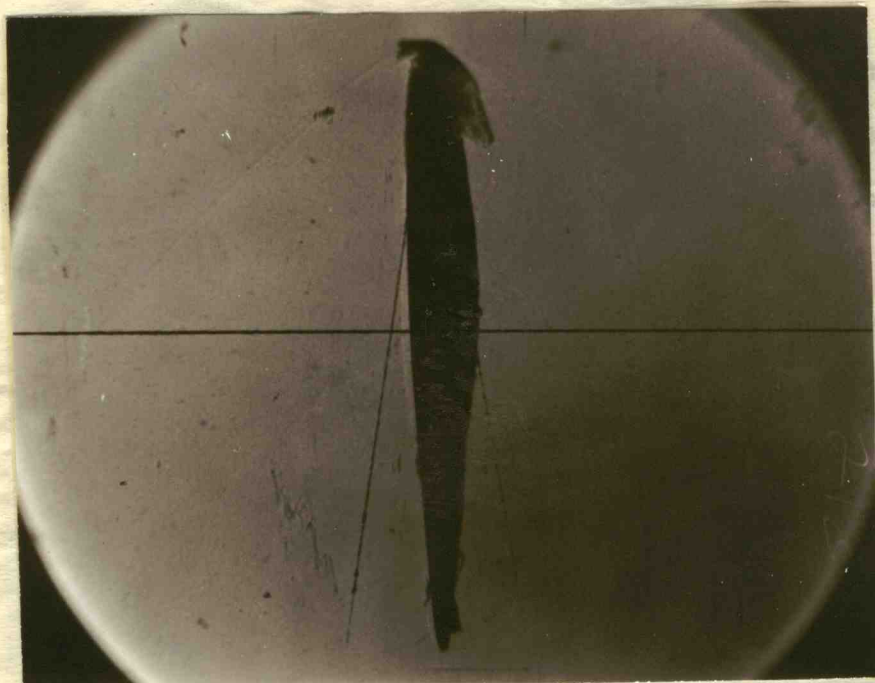


FIGURE 37
Round Number 213
 $M = 1.15$
Angle of Attack - 0°
Delay Time - 2540 Microseconds

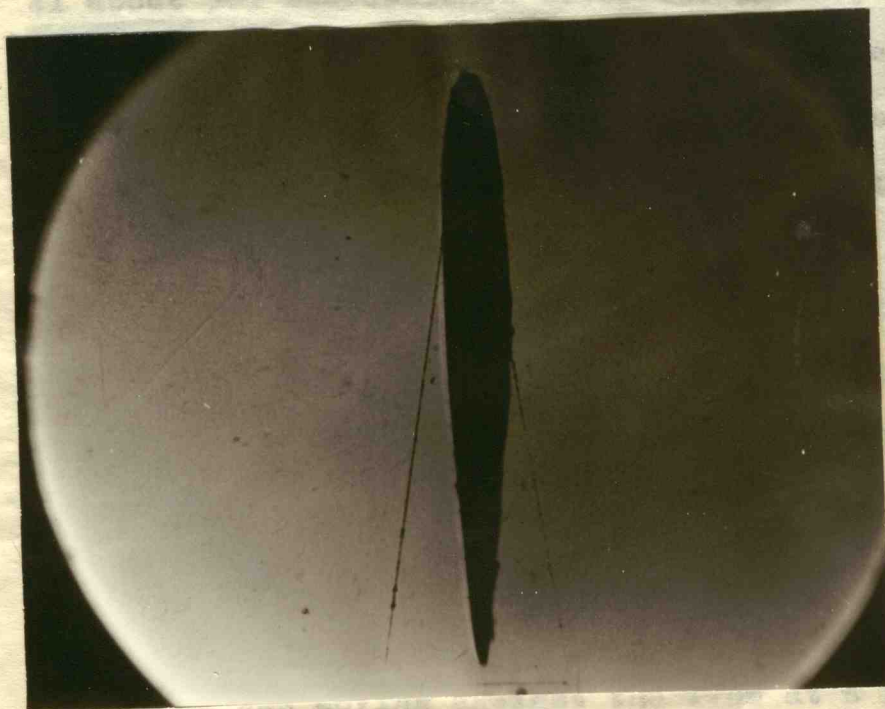


FIGURE 36
Round Number 207
 $M = 1.20$
Angle of Attack - 0°
Delay Time - 2175 Microseconds

Дата зняття - 22.12.1950
 Висота над поверхнею - 0.

М = 1.12
 Нормальний зріз
 Листок 21

Дата зняття - 22.12.1950
 Висота над поверхнею - 0.

М = 1.50
 Нормальний зріз
 Листок 22

A detached wave also occurs in the shock tube when flow is subsonic. Then it moves against the flow, becoming weaker as it moves. In this case no equilibrium position is reached.

Figures 38 and 39 show a detached shock wave in a supersonic flow. Figure 38 shows the detached shock wave at 2175 microseconds and Figure 39 shows the detached shock at 2540 microseconds. The pictures were taken about 365 microseconds apart. The detached shock wave is stable.

Figures 40 and 41 show the 0009 airfoil at zero degrees angle of attack. The primary shock wave has passed in both pictures. Figure 40 was taken about 100 microseconds after passage of the primary shock wave and Figure 41 about 365 microseconds after the primary shock wave had passed. Both shadowgraphs were taken at Mach 0.98. The wake can be seen as well as the detached shock waves. In all of the above figures considerable distortion may be seen around the airfoil due to the pressure forcing the rubber gasket to work out around the edges of the model. Cellophane also became lodged against the rubber gaskets and satisfactory measurements of the detached shock wave were not possible. This method of mounting the model was abandoned following this series of pictures. Figure 42 shows the 0009 airfoil with an angle of attack of 4 degrees. The shock was moving against the flow at a flow Mach of

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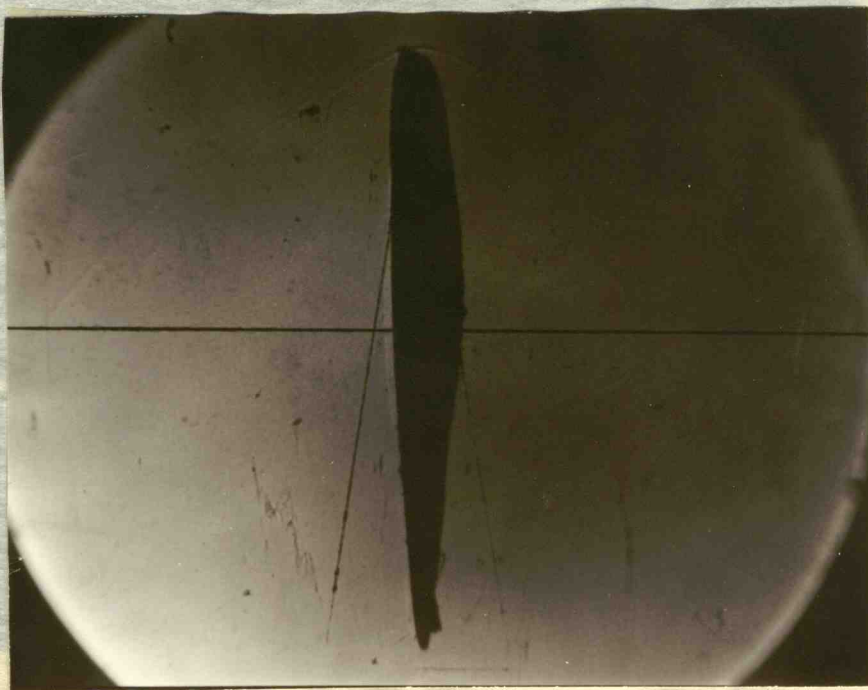


FIGURE 39
Round Number 214
 $M = 1.10$
Angle of Attack = 0°
Delay Time - 2540 Microseconds

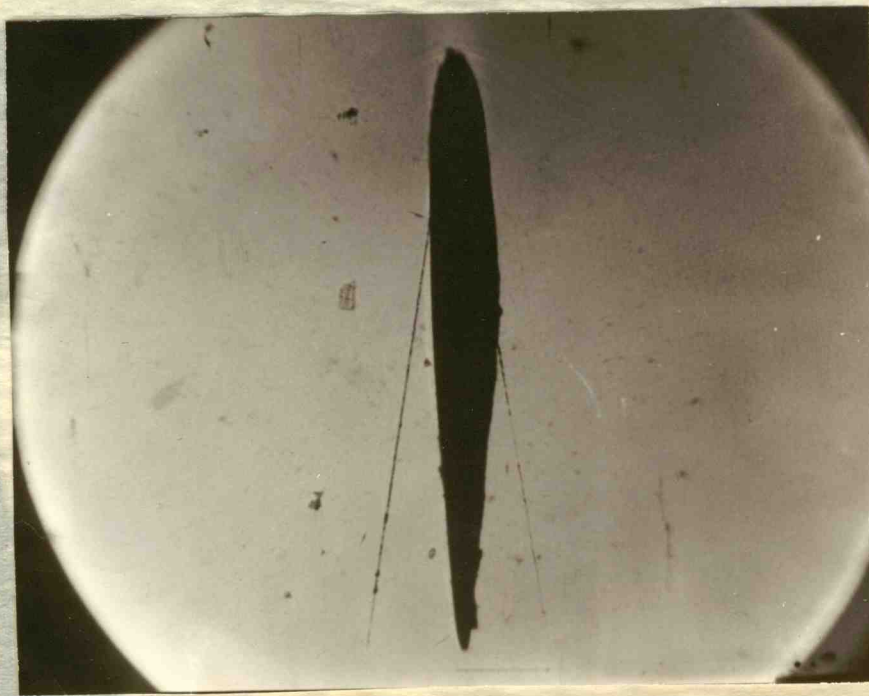


FIGURE 38
Round Number 210
 $M = 1.10$
Angle of Attack = 0°
Delay Time - 2175 Microseconds

Регла. лист - 3240 Нисловесонца
Улица од Варшав - 08
М - 1.10
Нолит Министр. СТО
БИСОНЕ 28

4539

Регла. лист - 3112 Нисловесонца
Улица од Варшав - 08
М - 1.10
Нолит Министр. СТО
БИСОНЕ 28

4538

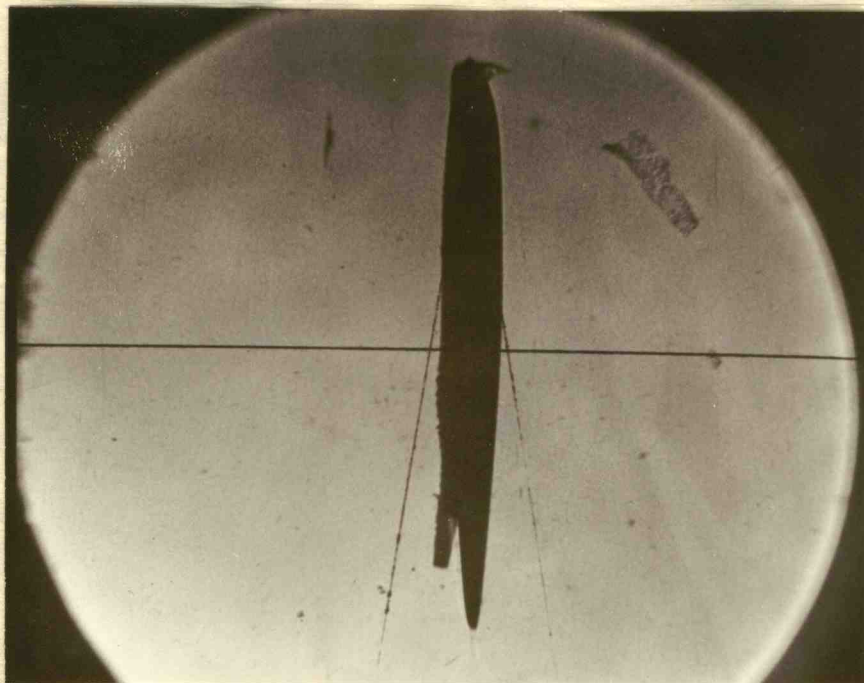


FIGURE 41
Round Number 224
 $M = 0.98$
Angle of Attack = 0°
Delay Time - 2540 Microseconds

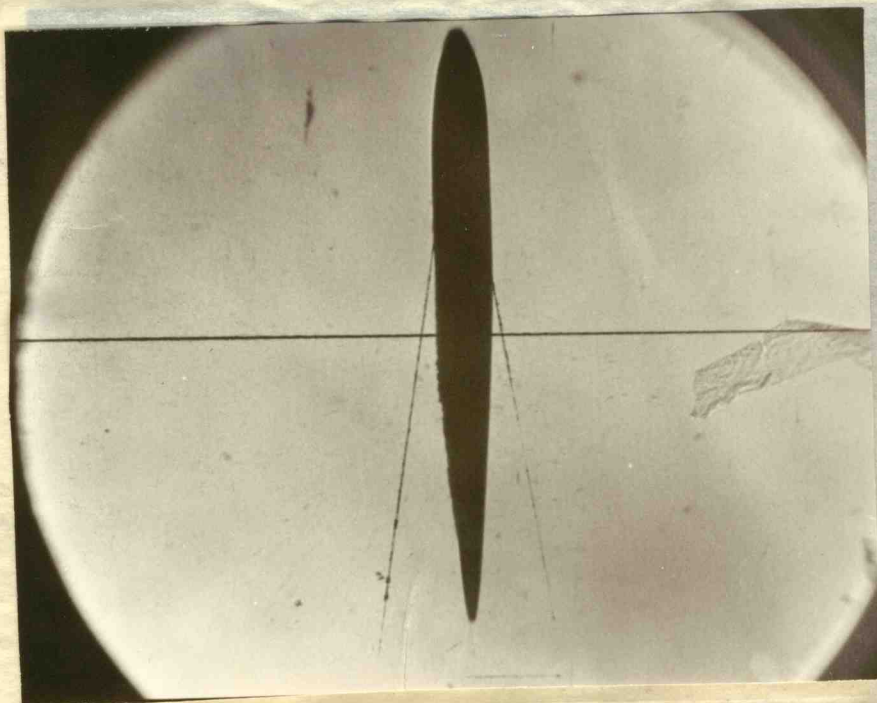


FIGURE 40
Round Number 219
 $M = 0.98$
Angle of Attack = 0°
Delay Time - 2175 Microseconds

DETA TYPE - 5210 MICROSCOPY
 VALUE OF VOLTAGE = 0.0
 30.0 C. M. = 0.00
 HONEY HUNTER 554
 EAGLE 40

DETA TYPE - 5210 MICROSCOPY
 VALUE OF VOLTAGE = 0.0
 30.0 C. M. = 0.00
 HONEY HUNTER 554
 EAGLE 40

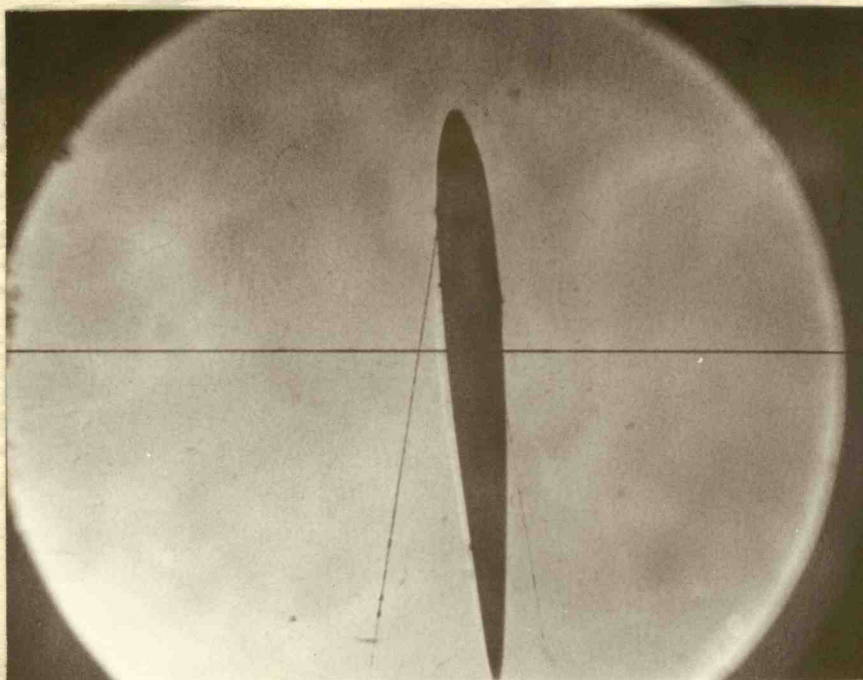


FIGURE 43
Round Number 227
 $M = 1.02$
Angle of Attack = 4°
Delay Time - 2540 Microseconds

.81 and there should have been some indication of a supersonic region on the upper surface of the airfoil since there was a 4 degree angle of attack. The density of the

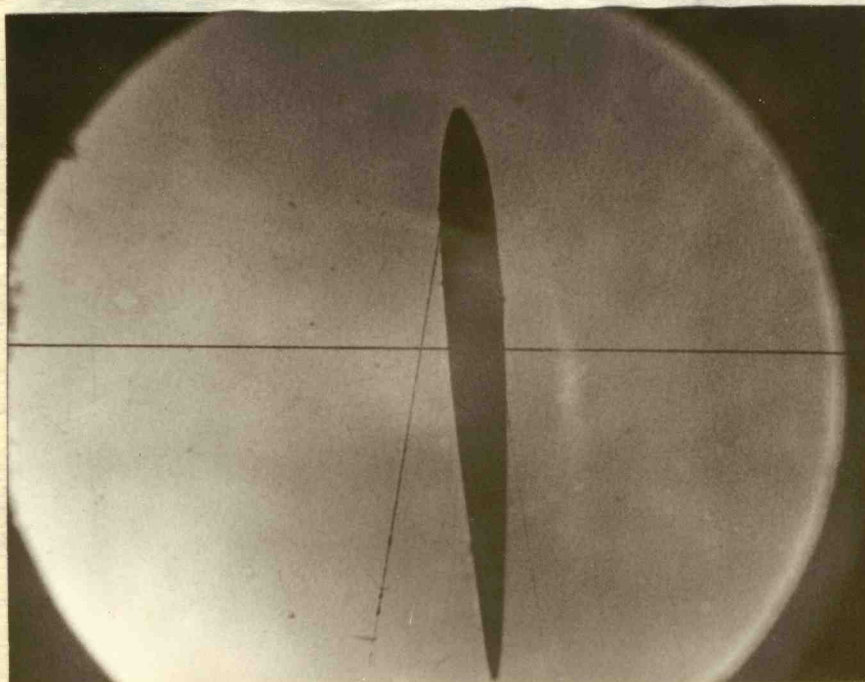


FIGURE 42
Round Number 229
 $M = 0.87$
Angle of Attack = 4°
Delay Time - 3100 Microseconds

DATA LINE - 2700 MICROSCOPIC
 VALUE OF VERTICAL = 4.

$\mu = 0.81$

LONG NUMBER 532

LONGER 45

DATA LINE - 5240 MICROSCOPIC
 VALUE OF VERTICAL = 4.

$\mu = 1.05$

LONG NUMBER 551

LONGER 42

0.87. Figure 43 shows the 0009 airfoil with an angle of attack of 4 degrees and a flow Mach of about 1.02. The detached shock wave became stabilized at its position after 500 microseconds flow.

Transonic flow around an airfoil.⁵ The transonic region of airflow is between the subsonic and supersonic regions. It begins when some point on the airfoil surface attains supersonic speed, and ends when the complete flow pattern is supersonic.

Figure 44 is a shadowgraph of flow at about Mach .81 and there should have been some indication of a supersonic region on the upper surface of the airfoil since there was a 4 degree angle of attack. The density of the air, however, may not have been sufficient to photograph satisfactorily.

Figure 45 shows the flow at about Mach 0.90 and although there is no indication on the upper surface, a shock can be seen extending from the lower surface near the rear of the airfoil. Probably most of the upper surface is subjected to supersonic flow.

Figure 46 shows the effect of supersonic flow. The flow Mach is 1.05 at zero degrees angle of attack and a

⁵Dwinnell, op. cit., pp. 190-195.

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²Datamall, op. cit., pp. 190-192.

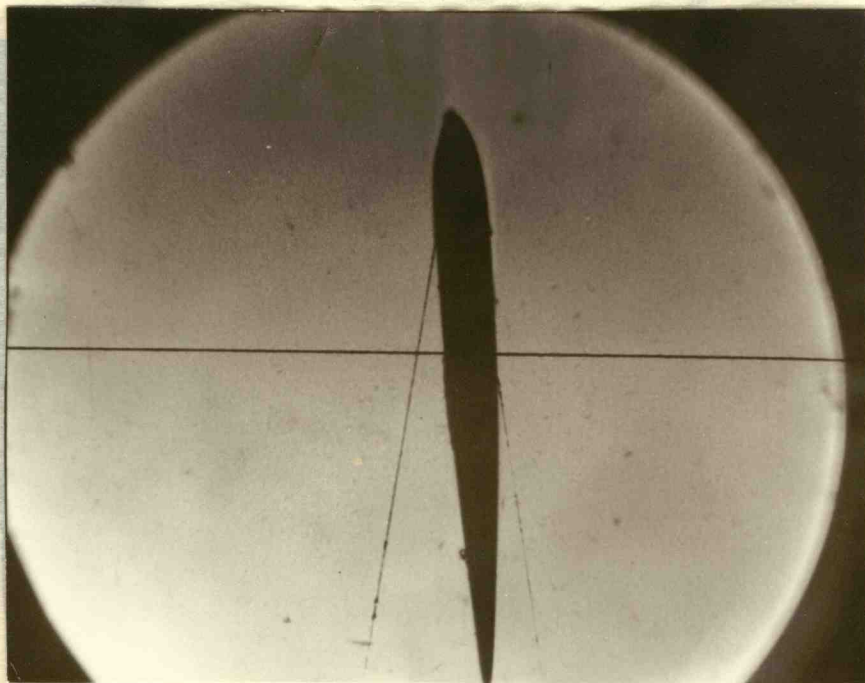


FIGURE 44 236
 Round Number
 Mach 0.81
 Angle of Attack = 4°
 Delay Time - 2740 Microseconds

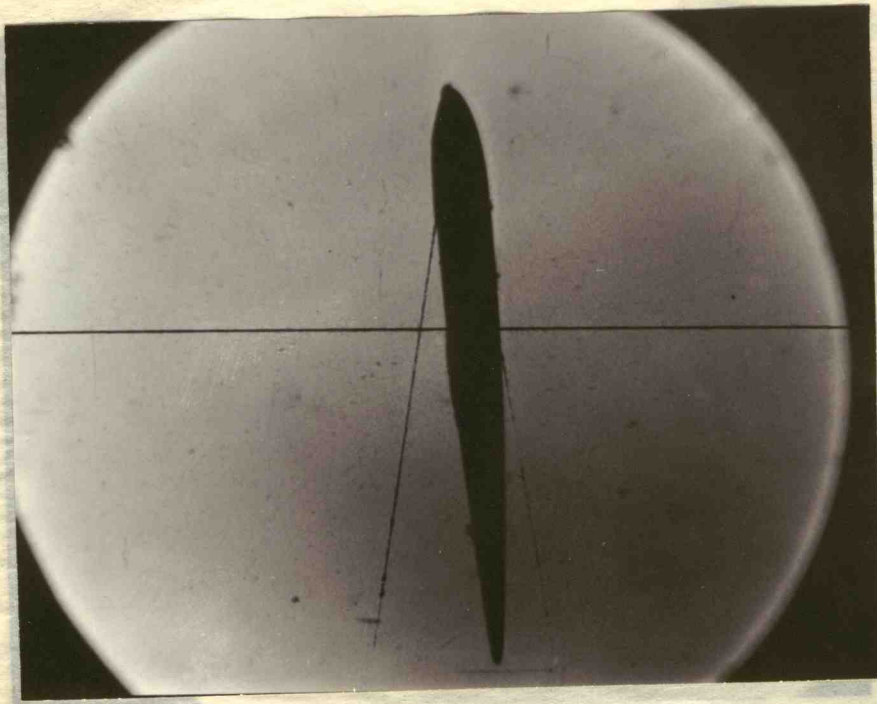


FIGURE 45 237
 Round Number
 Mach 0.90
 Angle of Attack = 4°
 Delay Time - 2920 Microseconds

Делта време - 3350 мкс
Угол отклонения = 4°
Время отсчета 0.30
Время отсчета 521
Время отсчета 42

Делта време - 3140 мкс
Угол отклонения = 4°
Время отсчета 0.31
Время отсчета 520
Время отсчета 44

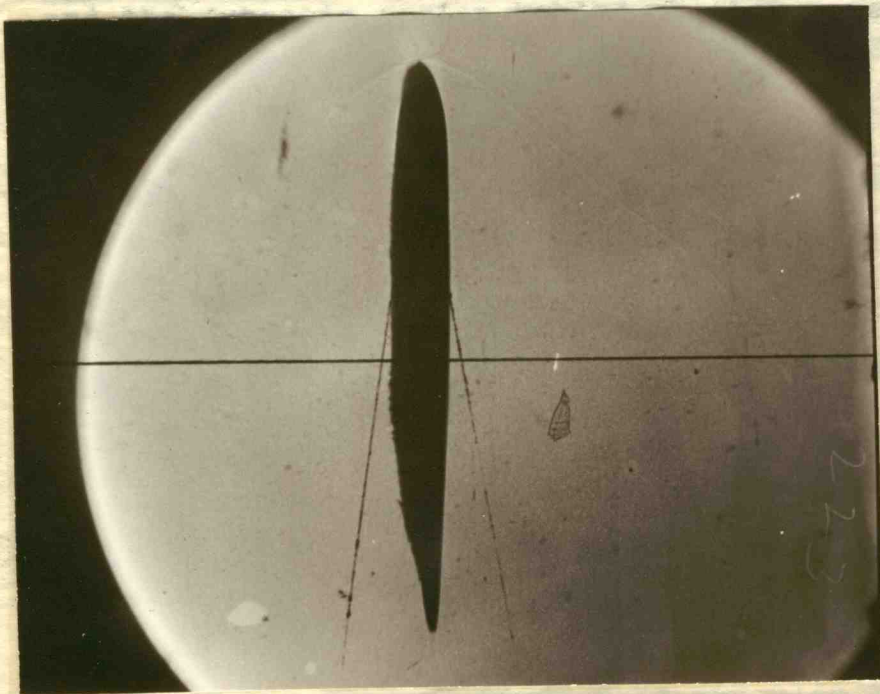


FIGURE 46
Round Number 222
Mach 1.02
Angle of Attack = 0°
Delay Time - 2540 Microseconds

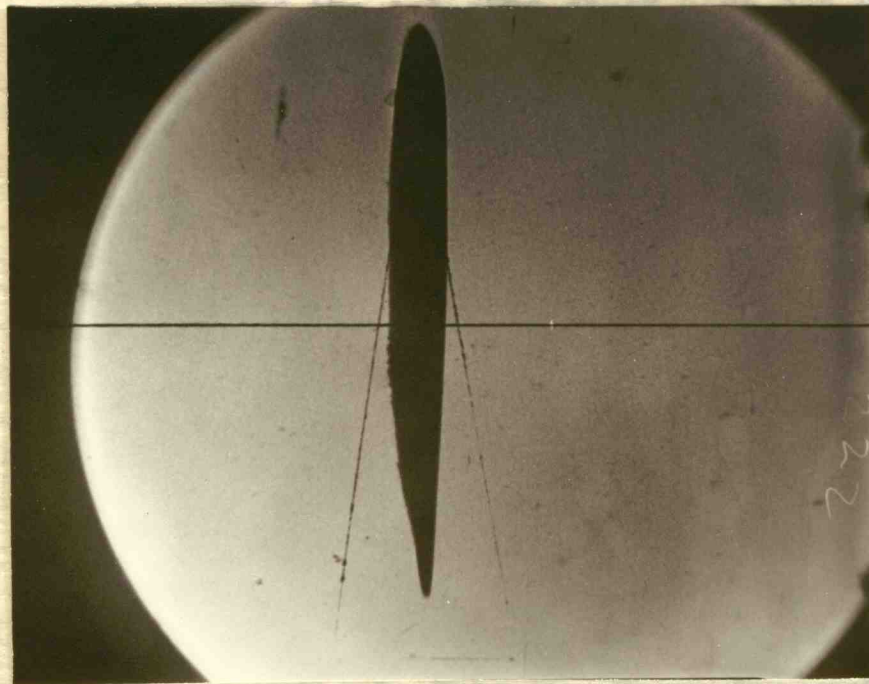


FIGURE 47
Round Number 223
Mach 1.14
Angle of Attack = 0°
Delay Time - 2540 Microseconds

Deferal time - 3240 MTCOLONOCORAS
value of VTCOR = 0.
WCF J'03
BONNY WAMPEN SSS
BICONE 40

Deferal time - 3240 MTCOLONOCORAS
value of VTCOR = 0.
WCF J'04
BONNY WAMPEN SSS
BICONE 41

41

47

mere trace of the detached shock wave can be seen while the shock fanning out from the trailing edge of the airfoil is clearly seen. There is a range behind the weak bow wave in which flow is subsonic. This range blends into a supersonic region over most of the airfoil to the trailing shock wave.

Figure 47 shows the flow at about Mach 1.14. At this Mach flow the bow wave approaches the leading edge of the airfoil, and the subsonic area following the bow wave is reduced. The position of the bow wave with respect to the leading edge and the extent of the subsonic region depend somewhat upon the shape of the leading edge. With a sharp leading edge, the bow wave may become attached and the subsonic region behind the bow wave may shrink to zero. The Mach number at which attachment occurs depends upon the wedge angle of the leading edge.

limited, however, being between Mach 1.10 and 1.15. Above the Mach 1.15 no shadowgraph pictures could be obtained, probably because the density of the air in the channel section of the shock tube was not sufficient to show up in a shadowgraph study. The duration of the flow in this range of Mach 1.10 to 1.15 was found to be only about 300 microseconds instead of the predicted 850 microseconds.

A velocity calibration was made in the flow region following the primary shock wave and also in the flow region following the contact surface. A 3 degree wedge

CHAPTER VII

SUMMARY OF RESULTS

In this investigation the flow in the cold gas behind the contact surface was found to follow the theory within 10 per cent up to Mach 2.00. Beginning at pressure ratios of 50 and above the flow became erratic. It appears possible that the test section was too close to the diaphragm for a full development of steady flow. No investigation of this phenomena was made.

The flow following the primary shock wave and before the contact surface had arrived was investigated. Over the range covered, the flow velocity was within 5 per cent of that predicted. This range was extremely limited, however, being between Mach 1.10 and 1.15. Above the Mach 1.15 no shadowgraph pictures could be obtained, probably because the density of the air in the channel section of the shock tube was not sufficient to show up in a shadowgraph study. The duration of the flow in this range of Mach 1.10 to 1.15 was found to be only about 300 microseconds instead of the predicted 800 microseconds.

A velocity calibration was made in the flow region following the primary shock wave and also in the flow region following the contact surface. A 3 degree wedge

CHAPTER VII

SUMMARY OF RESULTS

In this investigation the flow in the cold gas behind the contact surface was found to follow the theory within 10 per cent up to Mach 2.00. Beginning at pressures of 50 and above the flow became erratic. It is more probable that the best results are too close to the diagram for a full development of the theory. No investigation of this phenomenon is made.

The flow following the contact surface, above and below the contact surface, was investigated. Over the range covered, the flow velocity was within 10 per cent of that predicted. This range was extremely limited, however, being between Mach 1.10 and 1.15. Above the Mach 1.15 no shock wave was seen in the photograph probably because the density of the air in the channel section of the shock tube was not sufficient to show up in a schlieren image. The variation of the flow in this range of Mach 1.10 to 1.15 was found to be only about 100 microseconds behind of the predicted 50 microseconds.

A velocity calibration was made in the flow region following the primary shock wave and also in the flow region following the contact surface. A velocity range

made of hardened steel was used in both investigations. The difference between the predicted Mach flow and the actual measured flow in the region following the primary shock wave did not exceed 5 per cent. The variation was more pronounced in the flow region following the contact surface where the difference was as high as 10 per cent, but this region was usable only with flows up to Mach 1.75.

The measured Mach flow was always below the predicted Mach flow. This may have been caused by a failure of the diaphragm to burst instantaneously or by attenuation from the wall boundary layer of the shock tube. Either, or a combination of the two, could cause this reduction from the theoretical. No attempt was made to determine the cause. Following the contact surface the Mach flow was 16 per cent below the predicted Mach flow of 2.44, and seemed to attenuate rapidly as the delay time was increased. It is possible that this attenuation was due to lack of time for proper development of the flow at this pressure ratio of 100. Attenuation was slight at Mach 1.75 at a pressure ratio of 40. This will bear further investigation.

In tests made on the 3-degree steel wedge an agreement of 2 per cent was found in the measured bow wave angle and the Mach angle formed by a striation on the wedge. Local velocity is in agreement with the theory. A striation

made of aluminum sheet and was in both vertical and horizontal.
The difference between the vertical and horizontal flow and the
actual measured flow in the region following the primary
shock wave did not exceed 5 per cent. The vertical flow
more pronounced in the flow region following the shock
surface where the difference was as high as 10 per cent.
but this region was small and only with flow rate of 1.77
the measured flow was 1.77. Always below the measured
flow. This may have been caused by a failure of the
discharge to burst instantaneously or by a resistance in
the wall boundary layer of the shock tube. It is likely that
combination of the two could cause this deviation from
the theoretical. The theoretical flow rate is 1.77 and the
measured flow rate is 1.77. Following the shock wave the flow rate is 1.77
10 per cent below the theoretical flow rate of 1.77. It
seems to decrease rapidly in the delay time and is
It is possible that this phenomenon was due to lack of
time for proper development of the flow at this pressure
ratio of 100. Investigation of a shock at Mach 1.77
pressure ratio of 100. This will be done in the future.
Section.
In some cases in the 1-100 pressure ratio range
point of 1.77 and the flow rate was 1.77. The flow rate was 1.77
and the flow rate was 1.77. The flow rate was 1.77
local velocity at a distance of 1.77. A difference

formed by cellophane tape $1\frac{1}{2}$ thousandths thick was sufficient to give an excellent indication of local velocity.

Mach configurations over the NACA 0009 airfoil could be studied and its operation predicted. However, in the flow region following the primary shock wave there was apparently not enough density change of the air to make satisfactory measurements of the critical Mach number.

forest by cellophane tape 1 1/2 inches thick was sufficient to give an excellent indication of local velocity. Each configuration over the HACA 0009 at 1000 could be studied and its operation predicted. However, in the flow region following the primary shock wave there was apparently not enough density change of the air to make satisfactory measurements of the critical Mach number.

CHAPTER VIII

CONCLUSIONS AND RECOMMENDATIONS

The shadowgraph method of photography has definite limitations in high-speed velocity research. A change in density gradient causes the light rays to either converge or diverge, and hence produces either an increase or a decrease in light intensity on the screen. The method is not completely satisfactory unless the flow contains steep density gradients. Where weak shocks occur and density gradients are not steep enough no results may be observed. It was found that above Mach 1.18 in the flow following the primary shock wave, density was not sufficient to indicate any results.

Conclusions. The shock tube as presently available is a satisfactory test facility for limited experimentation. Flow velocities are available from sonic or Mach 1.00 to about Mach 1.75 with the shadowgraph method of observation in the flow region following the contact surface.

Triggering of the spark-delay system from the sound waves through the shock tube is satisfactory since the results were consistent and predictable.

CONCLUSIONS AND RECOMMENDATIONS

The shadowgraph method of photography has definite limitations in high-speed velocity tests. A change in density gradient causes the light rays to either converge or diverge, and hence produces either an increase or a decrease in light intensity on the screen. The method is not completely satisfactory unless the flow contains steep density gradients. Where such shocks occur and density gradients are not steep enough to register as observed. It was found that above Mach 1.18 in the flow following the primary shock wave, density was not sufficient to indicate any results.

Conclusions. The shock tube is presently available is a satisfactory test facility for limited experiments. High velocities are available from sonic or Mach 1.00 to about Mach 1.75 with the shadowgraph method of observation in the flow region following the constant surface.

Fracturing of the quartz-silica window from the sound waves through the shock tube is a serious problem. The results were consistent and predictable.

The spark source is satisfactory and should be usable for any shadowgraph work.

From the data taken it appears that local flow velocities may be measured on a model by use of an interrupting surface such as cellophane tape. This was used on a 3 degree wedge and satisfactory correlation resulted between readings of the bow wave and readings of the Mach wavelet angle.

Recommendations. The electronic delay, while satisfactory over the range it covered, does not have sufficient range for many of the lower pressure ratio studies and should probably be constructed for wider time coverage than from 300 to 3800 microseconds.

All other investigators studied used an optical method of triggering the delay circuits and it is possible that more consistent results might be obtained with an optical triggering system.

At all pressure ratios the mercurial manometer used for reading the pressure difference between the atmosphere and the channel was difficult to read accurately. Verniers might help but probably an oil manometer with its longer scale would be more satisfactory.

The first part of the paper is devoted to a discussion of the
possible for the theory of the
from the point of view of the
velocity may be considered as a function of the
rupting surface - the anisotropy of the
on a 3.5 mm scale and a 1.5 mm scale. The
between the two scales of the
waves.

Experimental results
The results of the experiment show that the
anistropy of the surface is not a function of the
velocity and should be considered as a function of the
coverage of the surface.

All other things being equal, the
method of measuring the surface anistropy is in the
able that there is a direct relation between the
an optical measuring method.

At all points along the surface the
used for measuring the surface anistropy is the same
monitors and the optical method is used for
verification of the results. The results show that
the optical method is a very reliable method.

TABLE I

PRESSURE RATIO NECESSARY FOR VARIOUS
SHOCK WAVE STRENGTHS

$$\frac{P_2}{P_0} = \frac{\gamma}{\left[1 - \frac{\gamma-1}{\sqrt{7(1+6\gamma)}}\right]^2} \quad \gamma=5 \quad \frac{P_2}{P_0} = \frac{5}{\left[1 - \frac{4}{\sqrt{7(1+30)}}\right]^2}$$

Shock Strength	Chamber Pressure Channel Pressure	Shock Strength	Chamber Pressure Channel Pressure
$\gamma = \frac{P_2}{P_0}$	$\frac{P_2}{P_0}$	$\gamma = \frac{P_2}{P_0}$	$\frac{P_2}{P_0}$
0.0	0.0	5.5	61.48
0.5	0.32	6.0	81.10
1.0	1.00	6.5	102.40
1.5	2.31	7.0	137.20
2.0	4.34	7.5	174.50
2.5	7.27	8.0	222.50
3.0	11.33	8.5	276.00
3.5	16.90	9.0	350.00
4.0	24.10	9.5	442.00
4.5	33.58	10.0	547.00
5.0	45.70		

TABLE 1

RELATIONSHIP BETWEEN PRESSURE RATIO AND SHOCK WAVE VELOCITY

SHOCK WAVE VELOCITY

$$\frac{P}{P_0} = \frac{1}{\gamma} \left[\frac{2}{\gamma + 1} + \frac{\gamma - 1}{\gamma + 1} \left(\frac{V}{V_0} \right)^2 \right]$$

$$\frac{V}{V_0} = \sqrt{\frac{2}{\gamma + 1} \left(\frac{P}{P_0} - \frac{1}{\gamma} \right)}$$

Shock Strength	Shock Pressure	Shock Velocity	Shock Wave Velocity
$\frac{P}{P_0}$	$\frac{P}{P_0}$	$\frac{V}{V_0}$	$\frac{V}{V_0}$
0.0	0.0	0.0	0.0
0.5	0.5	0.5	0.5
1.0	1.0	1.0	1.0
1.5	1.5	1.5	1.5
2.0	2.0	2.0	2.0
2.5	2.5	2.5	2.5
3.0	3.0	3.0	3.0
3.5	3.5	3.5	3.5
4.0	4.0	4.0	4.0
4.5	4.5	4.5	4.5
5.0	5.0	5.0	5.0

TABLE II

PRESSURE RATIO NECESSARY FOR VARIOUS RATIOS OF SHOCK
FRONT VELOCITIES TO THE SPEED OF SOUND IN THE GAS

$$\frac{U}{a_0} = \sqrt{\frac{6\gamma + 1}{7}}$$

$$\gamma = 5 \quad \frac{U}{a_0} = \sqrt{\frac{30 + 1}{7}} = 2.105$$

$$\frac{P_2}{P_0} = 45.7$$

Chamber Pressure Channel Pressure $\frac{P_2}{P_0}$	Primary Shock Velocity U Velocity of Sound a_0	Chamber Pressure Channel Pressure $\frac{P_2}{P_0}$	Primary Shock Velocity U Velocity of Sound a_0
0.0	0.0	61.48	2.21
0.32	0.76	81.10	2.30
1.00	1.00	102.40	2.39
2.31	1.19	137.20	2.48
4.34	1.36	174.50	2.57
7.27	1.51	222.50	2.65
11.33	1.65	276.00	2.73
16.90	1.78	350.00	2.80
24.10	1.89	442.00	2.88
33.58	1.20	547.00	2.96
45.70	2.105		

TABLE II

PRESSURE RATIO NECESSARY FOR VARIOUS RATIOS OF SHOCK
FRONT VELOCITIES TO THE SPEED OF SOUND IN THE GAS

$$\frac{U}{a_0} = \frac{\sqrt{\frac{2\gamma}{\gamma+1}}}{\gamma} \quad \frac{U}{a_0} = \frac{\sqrt{\frac{2\gamma}{\gamma+1}}}{\gamma} = 2.102$$

Channel Pressure $\frac{p_2}{p_1}$	Primary Shock Velocity $\frac{U}{a_0}$	Channel Pressure $\frac{p_2}{p_1}$	Primary Shock Velocity $\frac{U}{a_0}$
0.0	0.0	0.48	2.21
0.32	0.76	0.70	2.30
1.00	1.00	102.40	2.39
2.21	1.12	137.50	2.48
4.24	1.26	174.50	2.57
7.27	1.31	222.50	2.65
11.35	1.65	276.00	2.73
16.90	1.78	350.00	2.80
24.10	1.82	442.00	2.88
33.58	1.90	567.00	2.95
45.70	2.102		

TABLE III

PRESSURE RATIO NECESSARY FOR VARIOUS SHOCK TUBE STRENGTHS
AND PREDICTED MACH FLOWS FOLLOWING THE
PRIMARY SHOCK FRONT

$$M_{PS} = \frac{5(\gamma - 1)}{\sqrt{7\gamma(\gamma + 6)}}$$

$$\gamma = 5 \quad M_{PS} = \frac{5 \times 4}{\sqrt{35 \times 11}} = 1.02$$

Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow	Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow
$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{PS}	$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{PS}
0.0	0.0	---	5.5	61.48	1.07
0.5	0.32	---	6.0	81.10	1.11
1.0	1.00	0.0	6.5	102.4	1.15
1.5	2.31	0.31	7.0	137.20	1.19
2.0	4.34	0.47	7.5	174.50	1.22
2.5	7.27	0.62	8.0	222.50	1.25
3.0	11.33	0.73	8.5	276.00	1.28
3.5	16.90	0.82	9.0	350.00	1.30
4.0	24.10	0.90	9.5	442.00	1.33
4.5	33.58	0.96	10.0	547.00	1.35
5.00	45.70	1.02			

TABLE II

PRESSURE RATIO NECESSARY FOR VARIOUS RATIOS OF SHOCK FRONT VELOCITIES TO THE SPEED OF SOUND IN THE GAS

$$\frac{U}{a_0} = \sqrt{\frac{2}{\gamma+1} \left(\frac{p_2}{p_1} \right)}$$

$$\frac{U}{a_0} = \sqrt{\frac{2}{\gamma+1} \left(\frac{p_2}{p_1} \right)}$$

$$\frac{U}{a_0} = \sqrt{\frac{2}{\gamma+1} \left(\frac{p_2}{p_1} \right)}$$

Channel Pressure Ratio $\frac{p_2}{p_1}$	Primary Shock Velocity $\frac{U}{a_0}$	Channel Pressure Ratio $\frac{p_2}{p_1}$	Primary Shock Velocity $\frac{U}{a_0}$
0.0	0.0	0.0	0.0
0.32	0.32	0.32	0.32
1.00	1.00	1.00	1.00
2.71	1.71	2.71	1.71
4.74	1.76	4.74	1.76
7.57	1.81	7.57	1.81
11.33	1.85	11.33	1.85
16.90	1.88	16.90	1.88
24.10	1.90	24.10	1.90
33.29	1.92	33.29	1.92
45.70	1.94	45.70	1.94

TABLE III

PRESSURE RATIO NECESSARY FOR VARIOUS SHOCK TUBE STRENGTHS
AND PREDICTED MACH FLOWS FOLLOWING THE
PRIMARY SHOCK FRONT

$$M_{PS} = \frac{5(\gamma - 1)}{\sqrt{7\gamma(\gamma + 6)}}$$

$$\gamma = 5 \quad M_{PS} = \frac{5 \times 4}{\sqrt{35 \times 11}} = 1.02$$

Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow	Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow
$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{PS}	$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{PS}
0.0	0.0	---	5.5	61.48	1.07
0.5	0.32	---	6.0	81.10	1.11
1.0	1.00	0.0	6.5	102.4	1.15
1.5	2.31	0.31	7.0	137.20	1.19
2.0	4.34	0.47	7.5	174.50	1.22
2.5	7.27	0.62	8.0	222.50	1.25
3.0	11.33	0.73	8.5	276.00	1.28
3.5	16.90	0.82	9.0	350.00	1.30
4.0	24.10	0.90	9.5	442.00	1.33
4.5	33.58	0.96	10.0	547.00	1.35
5.00	45.70	1.02			

TABLE III

PRESSURE RATIO MEASURED BY THE TUBES UNDER TEST
AND PREDICTED FROM FOLLOWING THE
DATA OF CHOKI REPORT

$$M_p = \frac{P}{P_0} \sqrt{\frac{1 + \gamma}{1 - \gamma}}$$

$$M_p = \frac{P}{P_0} \sqrt{\frac{1 + \gamma}{1 - \gamma}}$$

Shock Strength	Pressure Ratio	Pressure Ratio	Pressure Ratio	Pressure Ratio	Pressure Ratio
$M = \frac{P}{P_0}$	$\frac{P}{P_0}$	$\frac{P}{P_0}$	$\frac{P}{P_0}$	$\frac{P}{P_0}$	$\frac{P}{P_0}$
0.0	0.0	0.0	0.0	0.0	0.0
0.5	0.5	0.5	0.5	0.5	0.5
1.0	1.0	1.0	1.0	1.0	1.0
1.5	1.5	1.5	1.5	1.5	1.5
2.0	2.0	2.0	2.0	2.0	2.0
2.5	2.5	2.5	2.5	2.5	2.5
3.0	3.0	3.0	3.0	3.0	3.0
3.5	3.5	3.5	3.5	3.5	3.5
4.0	4.0	4.0	4.0	4.0	4.0
4.5	4.5	4.5	4.5	4.5	4.5
5.0	5.0	5.0	5.0	5.0	5.0

TABLE IV

PRESSURE RATIO NECESSARY FOR VARIOUS SHOCK TUBE STRENGTHS
AND PREDICTED MACH FLOWS FOLLOWING THE
CONTACT SURFACE

$$M_{pc} = \frac{5(\gamma-1)}{\sqrt{7(6\gamma+1) - (\gamma-1)}}$$

$$\gamma = 5 \quad M_{pc} = \frac{5 \times 4}{\sqrt{7 \times 31 - 4}} = 1.86$$

$$\frac{P_2}{P_0} = 45.7$$

Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow	Shock Strength	Chamber Pressure Channel Pressure	Predicted Mach Flow
$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{pc}	$\gamma = \frac{P_1}{P_0}$	$\frac{P_2}{P_0}$	M_{pc}
0.0	0.0	—	5.5	61.48	2.04
0.5	0.32	—	6.0	81.10	2.25
1.0	1.00	0.0	6.5	102.40	2.45
1.5	2.31	0.28	7.0	137.20	2.65
2.0	4.34	0.59	7.5	174.50	2.84
2.5	7.27	0.75	8.0	222.50	3.04
3.0	11.33	1.05			
3.5	16.90	1.26			
4.0	24.10	1.47			
4.5	33.58	1.66			
5.0	45.70	1.86			

TABLE IV

PRESSURE RATIO NECESSARY FOR VARIOUS RATIO TESTS
AND LISTED EACH POINT WITH THE
CONTACT SURFACE

$$M_{pc} = \frac{A(X-D)}{\sqrt{V(X+D) - (V-D)}}$$

$$M_{pc} = \frac{A(X-D)}{\sqrt{V(X+D) - (V-D)}}$$

$$M_{pc} = \frac{A(X-D)}{\sqrt{V(X+D) - (V-D)}}$$

Shock Strength	Pressure Channel	Pressure Channel	Shock Channel	Pressure Channel	Shock Channel
0.0	0.0	0.0	0.0	0.0	0.0
0.5	0.5	0.5	0.5	0.5	0.5
1.0	1.00	1.00	1.00	1.00	1.00
1.5	1.31	1.31	1.31	1.31	1.31
2.0	1.74	1.74	1.74	1.74	1.74
2.5	2.21	2.21	2.21	2.21	2.21
3.0	2.72	2.72	2.72	2.72	2.72
3.5	3.26	3.26	3.26	3.26	3.26
4.0	3.83	3.83	3.83	3.83	3.83
4.5	4.42	4.42	4.42	4.42	4.42
5.0	5.03	5.03	5.03	5.03	5.03

TABLE V

CALCULATED DATA FOR MEASURED MACH FLOW AS TAKEN FROM
ONE-HALF BOW WAVE ANGLE MEASUREMENTS

$$M_m = \frac{1}{\sqrt{\sin^2 \beta - \frac{0.0314 \sin \beta}{\cos(\beta - 1.5)}}} \quad M_m = \frac{1}{\sqrt{(0.866)^2 - \frac{0.0314 \times 0.866}{0.5225}}} = 1.16$$

$\beta = 60^\circ$

One-half Bow Wave Angle β	Measured Mach Flow M_m	One-half Bow Wave Angle β	Measured Mach Flow M_m
10	6.33	50	1.35
15	4.11	55	1.26
20	3.07	60	1.19
25	2.46	65	1.16
30	2.07	70	1.11
35	1.80	75	1.10
40	1.60	80	1.10
45	1.46	85	1.17

NOTE: From 77 degrees on the shock wave will be detached and no measurements can be made below 1.098 Mach flow.

TABLE V

ONE-HALF FOR EACH MEASUREMENT
 CALCULATED DATA FOR EACH MEASUREMENT

$$M_m = \frac{1}{\sqrt{\sin^2 \theta - \cos^2 \theta}}$$

$$M_m = \frac{1}{\sqrt{\sin^2 \theta - \cos^2 \theta}}$$

One-half for each measurement	Calculated data for each measurement	One-half for each measurement	Calculated data for each measurement
10	2.11	10	2.11
15	4.11	15	4.11
20	7.07	20	7.07
25	9.14	25	9.14
30	11.0	30	11.0
35	12.6	35	12.6
40	14.1	40	14.1
45	15.5	45	15.5

NOTE: From 10 to 45 the data were left as calculated and no measurements were made before 10 or after 45.

TABLE VI
PRESSURE RATIO --- VELOCITY CALIBRATION FOLLOWING THE PRIMARY SHOCK

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Predicted Mach Number M _{ps}	Time After Triggering Microseconds	Bow Wave Angle Degrees	Measured Mach Number M _m	Remarks
196	95.5	39.9	1.00	2540	---	---	detached shock
201	95.0	39.8	1.00	3020	---	---	detached shock
202	95.0	39.8	1.00	2175	---	---	detached shock
206	88.5	40.0	1.00	3300	---	---	detached shock
239	89.0	135.0	1.18	2175	72	1.16	none
255	78.0	100.0	1.15	2175	70	1.13	none
270	68.0	142.0	1.19	2175	62.5	1.19	none
271	69.0	98.0	1.15	2350	69	1.13	none
280	69.0	102.0	1.17	1975	71	1.12	none

4800	11.0	555.0	1.11	2512	11	1.11	4500
4815	11.0	560.0	1.11	5120	21	1.11	4500
4830	11.0	565.0	1.11	5312	21	1.11	4500
4845	11.0	570.0	1.11	5512	21	1.11	4500
4860	11.0	575.0	1.11	5712	21	1.11	4500
4875	11.0	580.0	1.11	5912	21	1.11	4500
4890	11.0	585.0	1.11	6112	21	1.11	4500
4905	11.0	590.0	1.11	6312	21	1.11	4500
4920	11.0	595.0	1.11	6512	21	1.11	4500
4935	11.0	600.0	1.11	6712	21	1.11	4500
4950	11.0	605.0	1.11	6912	21	1.11	4500
4965	11.0	610.0	1.11	7112	21	1.11	4500
4980	11.0	615.0	1.11	7312	21	1.11	4500
4995	11.0	620.0	1.11	7512	21	1.11	4500

4800 11.0 555.0 1.11 2512 11 1.11 4500
 4815 11.0 560.0 1.11 5120 21 1.11 4500
 4830 11.0 565.0 1.11 5312 21 1.11 4500
 4845 11.0 570.0 1.11 5512 21 1.11 4500
 4860 11.0 575.0 1.11 5712 21 1.11 4500
 4875 11.0 580.0 1.11 5912 21 1.11 4500
 4890 11.0 585.0 1.11 6112 21 1.11 4500
 4905 11.0 590.0 1.11 6312 21 1.11 4500
 4920 11.0 595.0 1.11 6512 21 1.11 4500
 4935 11.0 600.0 1.11 6712 21 1.11 4500
 4950 11.0 605.0 1.11 6912 21 1.11 4500
 4965 11.0 610.0 1.11 7112 21 1.11 4500
 4980 11.0 615.0 1.11 7312 21 1.11 4500
 4995 11.0 620.0 1.11 7512 21 1.11 4500

4800 11.0 555.0 1.11 2512 11 1.11 4500
 4815 11.0 560.0 1.11 5120 21 1.11 4500
 4830 11.0 565.0 1.11 5312 21 1.11 4500
 4845 11.0 570.0 1.11 5512 21 1.11 4500
 4860 11.0 575.0 1.11 5712 21 1.11 4500
 4875 11.0 580.0 1.11 5912 21 1.11 4500
 4890 11.0 585.0 1.11 6112 21 1.11 4500
 4905 11.0 590.0 1.11 6312 21 1.11 4500
 4920 11.0 595.0 1.11 6512 21 1.11 4500
 4935 11.0 600.0 1.11 6712 21 1.11 4500
 4950 11.0 605.0 1.11 6912 21 1.11 4500
 4965 11.0 610.0 1.11 7112 21 1.11 4500
 4980 11.0 615.0 1.11 7312 21 1.11 4500
 4995 11.0 620.0 1.11 7512 21 1.11 4500

TABLE VI

TABLE VII

VELOCITY CALIBRATION -- FOLLOWING THE CONTACT SURFACE

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Predicted Mach Number M _{PC}	Time After Triggering Microseconds	Bow Wave Angle Degrees	Measured Mach Number M _M	Remarks
238	89.5	96	2.38	2175	32	1.90	none
239	89.0	191	2.92	2175	22	2.75	none
240	88.5	117	2.53	2175	28	2.20	none
241	89.0	130	2.60	2540	26.5	2.30	none
242	89.5	80	2.24	3100	29.0	2.05	none
243	91.0	53	1.94	3500	33.0	1.84	none
244	85.5	100	2.44	1980?	29.5	2.05	question time. appears short.
245	86.5	152	2.71	1980?	29	2.07	question time. appears short.
246	87.5	125	2.55	2360	34	1.81	none
247	89.0	70	2.13	2920	41	1.58	none
265	71.0	30	1.60	3500	46	1.46	none
266	63.0	62.3	2.05	3100	44	1.50	none

IV ELEV

EDUCATION - LITERATURE AND ARTS

NAME	AGE	SEX	EDUCATION	ARTS	REMARKS	DATE	INITIALS	REMARKS	DATE	INITIALS
JOHN	25	M	High School	Painting	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	22	F	High School	Music	Member of Music Club	1925	M.A.	Member of Music Club	1925	M.A.
JOHN	20	M	High School	Art	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	18	F	High School	Art	Member of Art Club	1925	M.A.	Member of Art Club	1925	M.A.
JOHN	15	M	High School	Art	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	12	F	High School	Art	Member of Art Club	1925	M.A.	Member of Art Club	1925	M.A.
JOHN	10	M	High School	Art	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	8	F	High School	Art	Member of Art Club	1925	M.A.	Member of Art Club	1925	M.A.
JOHN	5	M	High School	Art	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	3	F	High School	Art	Member of Art Club	1925	M.A.	Member of Art Club	1925	M.A.
JOHN	1	M	High School	Art	Member of Art Club	1925	J.H.	Member of Art Club	1925	J.H.
MARY	0	F	High School	Art	Member of Art Club	1925	M.A.	Member of Art Club	1925	M.A.

TABLE VII (continued)
VELOCITY CALIBRATION --- FOLLOWING THE CONTACT SURFACE

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Predicted Mach Number M _{PC}	Time After Triggering Microseconds	Bow Wave Angle Degrees	Measured Mach Number M _m	Remarks
276	74.0	39.3	1.77	3400	36	1.74	none
277	73.5	39.3	1.77	3580	35.5	1.76	none
279	73.0	56.1	1.97	2175	32	1.92	none
288	77.0	54.4	1.96	3675	39	1.62	none
289	77.0	44.0	1.83	3675	40	1.60	none
290	77.0	24.5	1.48	3675	58	1.23	none

TABLE VIII

EARLY STRIATION FLOW MEASUREMENTS

Round Number	Air Temperature	Pressure Ratio	Predicted Mach Flow	Bow Wave Angle	One-half		Striation Measured Angles	Remarks
					Mach Flow	Mach Flow		
33	80	51.6	1.91	60	1.42	1.39 1.34 1.31	46 48 50	Flow following the contact surface.
34	80	46.7	1.86	---	---	1.15 1.21 1.27	60 56 52	Flow following the contact surface. Detached bow wave.
35	84	55.2	1.96	70	1.34	1.27 1.34 1.44	52 48 44	Flow following the contact surface.
36	84	49.7	1.90	---	---	1.27 1.34 1.44	52 48 44	Flow following the contact surface. Detached bow wave.

NOTE: Interference between Mach wavelets caused varying striation angle measurements. These results indicate the spread as measured.

ИЛИ СКАТ

ВНЕШНЕУСЛОВИЯ ВОЛН КОЛЛЕКЦИИ УЧЕБ

ВНЕШНЕУСЛОВИЯ ВОЛН КОЛЛЕКЦИИ УЧЕБ

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ВНЕШНЕУСЛОВИЯ ВОЛН КОЛЛЕКЦИИ УЧЕБ

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TABLE IX
EXPERIMENTAL DATA ON STRIATIONS

Round Air Tem- Num- perature ber Deg. F.	Pressure Ratio P_2/P_0	Time After Triggering Micro- seconds	Bow Wave Angle Degrees	Measured Mach Flow M_∞	Mach Wavelet Angle (1/2°)	Measured Mach Flow (1/2°)	Mach Angle (1°)	Measured Mach Flow (1°)
252	72.0	167	56.0	1.22	no data	.	.	.
253	73.0	167	58.0	1.21	no data	.	.	.
254	77.0	98.2	71.5	1.12	no data	.	.	.
255	78.0	100.0	70.0	1.13	no data	.	.	.
256	79.0	24.0	55.5	1.25	no data	.	.	.
257	79.0	127.0	27.0	2.28	no data	.	26.0	2.28
258	74.0	150.0	30.0	2.02	no data	.	.	.
259	75.0	43.0	37.0	1.70	no data	.	.	.
260	70.0	185	33	1.86	no data	.	.	.
261	71.0	185	33	1.86	obscured		33.5	1.82
262	78.0	183	30.5	2.03	29.5	2.03	30.0	2.00
263	77.0	183	32.5	1.90	32	1.89	32.5	1.91
264	71.0	183	distorted	distorted	34	1.82	not visible	
265	71.0	95.2	distorted	distorted	32	1.89	32.0	1.89

THE END

TABLE IX (Continued)

EXPERIMENTAL DATA ON STEADIATIONS

Round Num- ber	Air Tem- perature Deg. F.	Pressure Ratio P_2/P_0	Time After Triggering Micro- seconds	Bow Wave Angle Degrees	Mach		Measured		Measured	
					Wavelet Angle (1/2")	Flow M_∞	Mach Flow (1/2")	Mach Angle (1")	Mach Flow (1")	Mach Flow (1")
266	63.0	62.3	3100	44		1.50	1.50	42.0		1.50
267	63.0	73.3	3100	none						
268	64.0	85.7	3100	observed by cellophane	28.0		2.13	28.0		2.13
269	64.0	118.0	3100	27	26.0	2.25	2.28	26.0		2.28
270	68.0	142.0	2175	62.5	not enough density	1.18				
271	69.0	98.0	2350	69.0	not enough density	1.13				
272	74.0	79.8	2175	35.0	double exposure obscured	1.80				
273	74.0	91.5	1975	42.0	40.0	1.54	1.56		not enough intensity	
274	74.0	91.5	2360	44.5	42.0	1.48	1.49	42.0		1.49
275	74.0	91.5	3100	47.5	44.0	1.41	1.44		observed	
276	74.0	39.3	3400	36.0	none	1.74				
277	73.5	39.3	3580	35.5	none	1.76				
278	73.5	56.1	1975	none						
279	73.0	56.1	2175	32.0	none	1.92				

TABLE IX (Continued)

EXPERIMENTAL DATA ON STRIATIONS

Round Number	Air Tem- perature Deg. F.	Pressure Ratio P_2/P_0	Time After Triggering Micro- seconds	Bow Wave Angle Degrees	Measured		Measured		Measured	
					Mach Flow M_m	Wavelet Angle (1/2")	Mach Flow (1/2")	Mach Angle (1")	Mach Flow (1")	Mach Angle (1")
280	69.0	101.8	1975	69.0	1.13	not enough density				
281	70.0	101.8	2175	33.0	1.88	not enough density	32.0	1.89		
282	71.0	101.8	2640	28.5	2.10	28.5	2.10	28.5	2.10	
283	71.5	101.8	2830	34.0	1.80	not dense enough				
284	73.5	99.6	2260	32.0	1.90	not dense enough				
285	74.0	99.6	2360	32.0	1.90	32.0	1.90	not dense enough		
286	74.5	99.6	2450	32.0	1.90	not dense enough				
287	75.0	99.6	3500	35.0	1.78	35.0	1.74	not dense enough		
288	77.0	54.4	3675	39.0	1.62	not dense enough				
289	77.0	43.9	3675	40.0	1.60	40.0	1.60	40.0	1.60	
290	77.0	24.5	3675	58.0	1.23	not dense enough				

TABLE X

SHOCK TUBE CHOKING EXPERIMENTS

Round Number	Air Temperature Degrees F.	Pressure Ratio P_2/P_0	Shock Strength	Predicted Mach Number Mps	Time After Triggering Microseconds	Remarks
31	89.0	77.8	5.90	1.12	695	none
32	89.0	77.8	5.90	1.12	900	none
33	89.0	77.8	5.90	1.12	300	none
34	89.0	77.8	5.90	1.12	500	none
3	79.5	60.2	5.50	1.07	0	pickup located in the center of the test window.
4	79.5	60.2	5.50	1.07	200	pickup located 6 inches behind the test window.

Index	Date	Time	Lat	Long	Alt	Wind	Temp	Humid	Press	Vis	Clouds	Remarks
1	10-1	1000	12.2	13.5	500							Light clouds, 1000m
2	10-2	0900	12.0	13.4	0							Light clouds, 1000m
3	10-3	0800	11.8	13.3	200							Light clouds, 1000m
4	10-4	0700	11.6	13.2	100							Light clouds, 1000m
5	10-5	0600	11.4	13.1	300							Light clouds, 1000m
6	10-6	0500	11.2	13.0	500							Light clouds, 1000m
7	10-7	0400	11.0	12.9	200							Light clouds, 1000m
8	10-8	0300	10.8	12.8	100							Light clouds, 1000m
9	10-9	0200	10.6	12.7	300							Light clouds, 1000m
10	10-10	0100	10.4	12.6	500							Light clouds, 1000m
11	10-11	0000	10.2	12.5	200							Light clouds, 1000m
12	10-12	2300	10.0	12.4	100							Light clouds, 1000m
13	10-13	2200	9.8	12.3	300							Light clouds, 1000m
14	10-14	2100	9.6	12.2	500							Light clouds, 1000m
15	10-15	2000	9.4	12.1	200							Light clouds, 1000m
16	10-16	1900	9.2	12.0	100							Light clouds, 1000m
17	10-17	1800	9.0	11.9	300							Light clouds, 1000m
18	10-18	1700	8.8	11.8	500							Light clouds, 1000m
19	10-19	1600	8.6	11.7	200							Light clouds, 1000m
20	10-20	1500	8.4	11.6	100							Light clouds, 1000m
21	10-21	1400	8.2	11.5	300							Light clouds, 1000m
22	10-22	1300	8.0	11.4	500							Light clouds, 1000m
23	10-23	1200	7.8	11.3	200							Light clouds, 1000m
24	10-24	1100	7.6	11.2	100							Light clouds, 1000m
25	10-25	1000	7.4	11.1	300							Light clouds, 1000m
26	10-26	0900	7.2	11.0	500							Light clouds, 1000m
27	10-27	0800	7.0	10.9	200							Light clouds, 1000m
28	10-28	0700	6.8	10.8	100							Light clouds, 1000m
29	10-29	0600	6.6	10.7	300							Light clouds, 1000m
30	10-30	0500	6.4	10.6	500							Light clouds, 1000m
31	10-31	0400	6.2	10.5	200							Light clouds, 1000m
32	10-32	0300	6.0	10.4	100							Light clouds, 1000m
33	10-33	0200	5.8	10.3	300							Light clouds, 1000m
34	10-34	0100	5.6	10.2	500							Light clouds, 1000m
35	10-35	0000	5.4	10.1	200							Light clouds, 1000m
36	10-36	2300	5.2	10.0	100							Light clouds, 1000m
37	10-37	2200	5.0	9.9	300							Light clouds, 1000m
38	10-38	2100	4.8	9.8	500							Light clouds, 1000m
39	10-39	2000	4.6	9.7	200							Light clouds, 1000m
40	10-40	1900	4.4	9.6	100							Light clouds, 1000m
41	10-41	1800	4.2	9.5	300							Light clouds, 1000m
42	10-42	1700	4.0	9.4	500							Light clouds, 1000m
43	10-43	1600	3.8	9.3	200							Light clouds, 1000m
44	10-44	1500	3.6	9.2	100							Light clouds, 1000m
45	10-45	1400	3.4	9.1	300							Light clouds, 1000m
46	10-46	1300	3.2	9.0	500							Light clouds, 1000m
47	10-47	1200	3.0	8.9	200							Light clouds, 1000m
48	10-48	1100	2.8	8.8	100							Light clouds, 1000m
49	10-49	1000	2.6	8.7	300							Light clouds, 1000m
50	10-50	0900	2.4	8.6	500							Light clouds, 1000m
51	10-51	0800	2.2	8.5	200							Light clouds, 1000m
52	10-52	0700	2.0	8.4	100							Light clouds, 1000m
53	10-53	0600	1.8	8.3	300							Light clouds, 1000m
54	10-54	0500	1.6	8.2	500							Light clouds, 1000m
55	10-55	0400	1.4	8.1	200							Light clouds, 1000m
56	10-56	0300	1.2	8.0	100							Light clouds, 1000m
57	10-57	0200	1.0	7.9	300							Light clouds, 1000m
58	10-58	0100	0.8	7.8	500							Light clouds, 1000m
59	10-59	0000	0.6	7.7	200							Light clouds, 1000m
60	10-60	2300	0.4	7.6	100							Light clouds, 1000m
61	10-61	2200	0.2	7.5	300							Light clouds, 1000m
62	10-62	2100	0.0	7.4	500							Light clouds, 1000m
63	10-63	2000	-0.2	7.3	200							Light clouds, 1000m
64	10-64	1900	-0.4	7.2	100							Light clouds, 1000m
65	10-65	1800	-0.6	7.1	300							Light clouds, 1000m
66	10-66	1700	-0.8	7.0	500							Light clouds, 1000m
67	10-67	1600	-1.0	6.9	200							Light clouds, 1000m
68	10-68	1500	-1.2	6.8	100							Light clouds, 1000m
69	10-69	1400	-1.4	6.7	300							Light clouds, 1000m
70	10-70	1300	-1.6	6.6	500							Light clouds, 1000m
71	10-71	1200	-1.8	6.5	200							Light clouds, 1000m
72	10-72	1100	-2.0	6.4	100							Light clouds, 1000m
73	10-73	1000	-2.2	6.3	300							Light clouds, 1000m
74	10-74	0900	-2.4	6.2	500							Light clouds, 1000m
75	10-75	0800	-2.6	6.1	200							Light clouds, 1000m
76	10-76	0700	-2.8	6.0	100							Light clouds, 1000m
77	10-77	0600	-3.0	5.9	300							Light clouds, 1000m
78	10-78	0500	-3.2	5.8	500							Light clouds, 1000m
79	10-79	0400	-3.4	5.7	200							Light clouds, 1000m
80	10-80	0300	-3.6	5.6	100							Light clouds, 1000m
81	10-81	0200	-3.8	5.5	300							Light clouds, 1000m
82	10-82	0100	-4.0	5.4	500							Light clouds, 1000m
83	10-83	0000	-4.2	5.3	200							Light clouds, 1000m
84	10-84	2300	-4.4	5.2	100							Light clouds, 1000m
85	10-85	2200	-4.6	5.1	300							Light clouds, 1000m
86	10-86	2100	-4.8	5.0	500							Light clouds, 1000m
87	10-87	2000	-5.0	4.9	200							Light clouds, 1000m
88	10-88	1900	-5.2	4.8	100							Light clouds, 1000m
89	10-89	1800	-5.4	4.7	300							Light clouds, 1000m
90	10-90	1700	-5.6	4.6	500							Light clouds, 1000m
91	10-91	1600	-5.8	4.5	200							Light clouds, 1000m
92	10-92	1500	-6.0	4.4	100							Light clouds, 1000m
93	10-93	1400	-6.2	4.3	300							Light clouds, 1000m
94	10-94	1300	-6.4	4.2	500							Light clouds, 1000m
95	10-95	1200	-6.6	4.1	200							Light clouds, 1000m
96	10-96	1100	-6.8	4.0	100							Light clouds, 1000m
97	10-97	1000	-7.0	3.9	300							Light clouds, 1000m
98	10-98	0900	-7.2	3.8	500							Light clouds, 1000m
99	10-99	0800	-7.4	3.7	200							Light clouds, 1000m
100	10-100	0700	-7.6	3.6	100							Light clouds, 1000m

STATION DATA SHEET

DATE

TABLE XI (Continued)
DATA ON NACA 0009 AIRFOIL EXPERIMENTS

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Mach Flow Taken from Fig. 8	Time After Triggering Microseconds	Remarks
207	93.5	167.0	1.20	2175	Detached shock 0 degree angle of attack
208	94.5	119.0	1.15	2175	Detached shock 0 degree angle of attack
209	94.5	96.4	1.13	2175	Detached shock 0 degree angle of attack
210	94.5	80.9	1.10	2175	0 degree angle of attack
211	95.5	159.0	1.19	2175	0 degree angle of attack
212	95.5	120.8	1.16	2540	Detached shock 0 degree angle of attack
213	98.0	97.7	1.13	2540	Detached shock 0 degree angle of attack
214	99.0	81.3	1.10	2540	Detached shock 0 degree angle of attack
215	89.0	61.1	1.05	2175	Flow some cello- phane 0 degree angle of attack
216	89.0	49.1	1.01	2175	No flow indi- cations 0 degree angle of attack
217	85.0	60.7	1.05	2175	No flow indi- cations 0 degree angle of attack
218	86.0	55.4	1.03	2175	0 degree angle of attack
219	87.0	44.5	0.98	2175	No flow indi- cations 0 degree angle of attack
220	88.5	50.9	1.02	2540	0 degree angle of attack

TABLE XI (Continued)

DATA ON NACA 0009 AIRFOIL EXPERIMENTS

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Mach Flow Taken from Figure 8	Time After Triggering Microseconds	Remarks
221	89.0	54.2	1.03	2540	Large cellophane 0 degree angle of attack
222	86.5	53.8	1.02	2540	None 0 degree angle of attack
223	86.5	109.0	1.14	2540	None 0 degree angle of attack
224	82.0	43.7	0.98	2540	None 0 degree angle of attack
225	82.0	37.2	0.95	2540	None 0 degree angle of attack
226	79.5	41.4	97.0	2540	Detached shock 4 degree angle of attack
227	79.5	52.7	1.02	2540	None 4 degree angle of attack
228	86.0	25.0	0.87	2540	None 4 degree angle of attack
229	86.0	25.0	0.87	3100	None 4 degree angle of attack
230	93.0	25.10	0.87	2440	Cellophane in flow stream 4 degree angle of attack
231	93.0	25.10	0.87	3675	Cellophane in flow stream 4 degree angle of attack
232	84.5	39.9	0.97	2540	Detached shock 4 degree angle of attack
233	84.0	39.9	0.97	2925	Detached shock 4 degree angle of attack
234	85.5	15.9	0.75	2540	No flow indication 4 degree angle of attack

TABLE XI (Continued)

DATA ON NACA 0009 AIRFOIL EXPERIMENTS

Round Number	Air Temperature Deg. F.	Pressure Ratio P_2/P_0	Mach Flow Taken from Figure 8	Time After Triggering Microseconds	Remarks
235	85.5	15.9	0.75	2925	No flow indication 4 degree angle of attack
236	85.5	15.9	0.75	2740	None 4 degree angle of attack
237	85.5	24.5	0.87	2920	None 4 degree angle of attack
238	89.5	96.0	1.12	2175	None 4 degree angle of attack

Number	Def.	Angle	Time	Angle	Angle of vision
528	88.2	60.0	1.13	51.2	Angle of vision
529	82.8	54.2	0.94	53.2	Angle of vision
530	82.2	72.8	0.12	54.0	Angle of vision
532	82.2	72.8	0.12	53.2	Angle of vision
No 1704					
Number	Def.	Angle	Time	Angle	Angle of vision
533	82.2	54.0	1.13	51.2	Angle of vision
534	82.2	54.0	1.13	51.2	Angle of vision
535	82.2	54.0	1.13	51.2	Angle of vision

DATA ON MVGV 0003 VIBROIT EXPERIMENTAL

TABLE XI (continued)

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APPENDICES

The following definitions will be used whenever it is convenient:

$$\gamma \equiv \frac{P}{P_0}$$

where P_0 and P are the pressures ahead of and behind a shock front, respectively.

$$\lambda \equiv \frac{1+\gamma}{1-\gamma}, \quad \text{where} \quad \gamma \equiv \frac{C_p}{C_v}$$

The numerical values of γ and λ are very nearly 1.4 and 6, respectively, when the gas under consideration is air, and these values will often be substituted into the final formula.

APPENDIX

The following relations will be used:

It is convenient:

where γ and β are the angles between the x and y axes and the z axis, respectively.

$$\frac{p}{h} \equiv \frac{1}{2}$$

$$\frac{q}{h} \equiv \frac{1}{2}$$

$$\frac{1+\gamma}{1-\gamma} \equiv \frac{1}{2}$$

The following relations will be used:

The modified version of the γ and β angles, respectively, with the new metric tensor, is also, and these values will also be substituted into the final formula.

APPENDIX I

DEVIATION OF THE RANKINE-HUGONOT RELATION
FOR AN IDEAL GAS

The conservation laws of mass, momentum and energy, applied to the gas contained in a cylinder of unit cross section which passes through a plane, stationary shock front are:

$$\text{mass: } \rho_0 V_0 = \rho_1 V_1 \quad (\text{AI-1})$$

$$\text{momentum: } P_0 + \rho_0 V_0^2 = P_1 + \rho_1 V_1^2 \quad (\text{AI-2})$$

$$\text{energy: } \frac{1}{2} V_0^2 + C_p T_0 = \frac{1}{2} V_1^2 + C_p T_1$$

$$\text{or } V_0^2 + \frac{2\gamma}{\gamma-1} \frac{P_0}{\rho_0} = V_1^2 + \frac{2\gamma}{\gamma-1} \frac{P_1}{\rho_1} \quad (\text{AI-3})$$

where ρ is density, V is velocity, P is pressure, γ is the ratio of specific heats (assumed constant), and the subscripts 0 and 1 apply to the states on the upstream and downstream sides of the shock, respectively.

Division of the momentum equation by the mass or continuity equation gives

ARTICLE 1

DEVIATION OF THE ALKALINE-ACIDITY RELATION FOR THE IDEAL GAS

The following law of mass, assuming the change
of the gas constant is constant as well as
the reaction which passes through a state, is usually shown
from the

$$p \cdot V = R \cdot T$$

$$p \cdot V = R \cdot T$$

$$\frac{1}{2} V + 2T = \frac{1}{2} N + 0.75 N$$

$$V + \frac{2T}{2} = V + \frac{1}{2} T$$

where p is density, V is volume, R is the
ratio of specific heats, T is temperature, and
the ratio of specific heats is 1.4 for air.
The ratio of specific heats is 1.4 for air.

Division of the reaction equation by the
constant relation gives

$$\frac{P_0}{\rho_0 V_0} + V_0 = \frac{P_1}{\rho_1 V_1} + V_1$$

or

$$V_0 - V_1 = \frac{P_1}{\rho_1 V_1} - \frac{P_0}{\rho_0 V_0} \quad (\text{AI-4})$$

and multiplication of (AI-4) by $V_0 + V_1$ gives

$$V_0^2 - V_1^2 = (V_0 + V_1) \left(\frac{P_1}{\rho_1 V_1} - \frac{P_0}{\rho_0 V_0} \right) \quad (\text{AI-5})$$

Equating this value of $V_0^2 - V_1^2$ to that obtained directly from the energy equation gives

$$(V_0 - V_1) \left(\frac{P_1}{\rho_1 V_1} - \frac{P_0}{\rho_0 V_0} \right) = \frac{2\gamma}{\gamma-1} \left(\frac{P_1}{\rho_1} - \frac{P_0}{\rho_0} \right) \quad (\text{AI-6})$$

Making use of the continuity equation, this may be written

$$\frac{P_1}{\rho_0} - \frac{P_0}{\rho_0} + \frac{P_1}{\rho_1} - \frac{P_0}{\rho_1} = \frac{2\gamma}{\gamma-1} \left(\frac{P_1}{\rho_1} - \frac{P_0}{\rho_0} \right)$$

or

$$\frac{1}{\rho_0} \left(P_1 - P_0 + \frac{2P_0\gamma}{\gamma-1} \right) = \frac{1}{\rho_1} \left(P_0 - P_1 + \frac{2P_1\gamma}{\gamma-1} \right) \quad (\text{AI-7})$$

from which

$$\frac{\rho_1}{\rho_0} = \frac{V_0}{V_1} = \frac{P_0 + \left(\frac{2\gamma}{\gamma-1} - 1 \right) P_1}{P_1 + \left(\frac{2\gamma}{\gamma-1} - 1 \right) P_0} = \frac{P_0 + \frac{\gamma+1}{\gamma-1} P_1}{P_1 + \frac{\gamma+1}{\gamma-1} P_0} = \frac{\gamma\gamma + 1}{\gamma + \gamma}$$

$$\frac{\rho_1}{\rho_0} = \frac{V_0}{V_1} = \frac{6\gamma + 1}{\gamma + 6} \quad (\text{AI-8})$$

which is the Rankine-Hugoniot relation for air.

V_0 and V_1 are velocities of flow relative to the shock front. For later reference, the transformation to a coordinate system in which the gas ahead of the shock front is stationary is affected by

$$\frac{p}{\rho V} + V = \frac{p}{\rho V} + V$$

or

$$V - V = \frac{p}{\rho V} - \frac{p}{\rho V} \quad (11-4)$$

and multiplication of (11-3) by $V + V$ gives

$$V^2 - V^2 = (V + V) \left(\frac{p}{\rho V} - \frac{p}{\rho V} \right) \quad (11-5)$$

Evaluating this value of $V - V = V$ as the speed of sound from the energy equation gives

$$(V - V) \left(\frac{p}{\rho V} + \frac{p}{\rho V} \right) = \frac{p}{\rho} \left(\frac{1}{V} - \frac{1}{V} \right) \quad (11-6)$$

Making use of the continuity equation, $\rho_0 V_0 = \rho V$, we get

$$\frac{p}{\rho_0 V_0} + \frac{p}{\rho_0 V_0} = \frac{p}{\rho_0 V_0} + \frac{p}{\rho_0 V_0} \quad (11-7)$$

$$\frac{1}{\rho_0} \left(p - p + \frac{p}{V_0} + \frac{p}{V_0} \right) = \frac{1}{\rho_0} \left(p - p + \frac{p}{V_0} + \frac{p}{V_0} \right) \quad (11-8)$$

from which

$$\frac{p}{\rho_0 V_0} = \frac{p}{\rho_0 V_0} = \frac{p}{\rho_0 V_0} \quad (11-9)$$

which is the Rankine-Hugoniot relation for the

ρ_0 and V_0 are the values of the variables before the

shock front. For a given substance, the relation between

a compressed system is given by the equation of state

from which the relation is obtained by

$$\left. \begin{aligned} V_0 &= U \\ V_1 &= U - u \end{aligned} \right\} \quad (\text{AI-9})$$

where U is the velocity of the shock front and u is the velocity of flow behind the shock front. (AI-8) is then

$$\frac{U}{U - u} = \frac{\mu\gamma + 1}{\gamma + \mu} = \frac{6\gamma + 1}{\gamma + 6} \quad (\text{AI-10})$$

from which

$$u = U \left(1 - \frac{\gamma + \mu}{\mu\gamma + 1} \right) = U \frac{(\gamma - 1)(\mu - 1)}{\mu\gamma + 1} = U \frac{(\gamma - 1) 5}{6\gamma + 1} \quad (\text{AI-11})$$

(6-1A)

$$V_0 = U$$

$$V = U - u$$

where U is the velocity of the shock front and u is the velocity of flow behind the shock front. (A1-3) is then

(6-1A)

$$\frac{1 + \gamma \frac{u}{U}}{1 + \gamma} = \frac{1 + \gamma \frac{u}{U}}{1 + \gamma} = \frac{U}{U - u}$$

from which

(6-1A)

$$\frac{1 + \gamma \frac{u}{U}}{1 + \gamma} = \frac{1 + \gamma \frac{u}{U}}{1 + \gamma} = \frac{U}{U - u} \Rightarrow U = \frac{1 + \gamma}{1 + \gamma \frac{u}{U}} u = u \left(\frac{1 + \gamma}{1 + \gamma \frac{u}{U}} \right) = u$$

APPENDIX II

DERIVATION OF PRANDTL'S RELATION $V_0 V_1 = a_{cr}^2$

If a_{cr} is the critical sound speed, defined as the value at which the sound speed and the flow speed become equal in an expansion from a reservoir, the energy equation (AI-3) may be written

$$\left. \begin{aligned} \frac{\gamma}{\gamma-1} \frac{P_0}{\rho_0} &= -\frac{1}{2} \frac{\gamma+1}{\gamma-1} a_{cr}^2 - \frac{1}{2} V_0^2 \\ \frac{\gamma}{\gamma-1} \frac{P_1}{\rho_1} &= -\frac{1}{2} \frac{\gamma+1}{\gamma-1} a_{cr}^2 - \frac{1}{2} V_1^2 \end{aligned} \right\} \quad (\text{AII-1})$$

from which

$$P_0 = \rho_0 \left(-\frac{\gamma+1}{2\gamma} a_{cr}^2 - \frac{\gamma-1}{2\gamma} V_0^2 \right)$$

$$P_1 = \rho_1 \left(-\frac{\gamma+1}{2\gamma} a_{cr}^2 - \frac{\gamma-1}{2\gamma} V_1^2 \right)$$

Putting these values for P_0 , P_1 into the momentum equation (AI-2),

$$\rho_0 \left(\frac{\gamma+1}{2\gamma} V_0^2 + \frac{\gamma+1}{2\gamma} a_{cr}^2 \right) =$$

$$\rho_1 \left(\frac{\gamma+1}{2\gamma} V_1^2 + \frac{\gamma+1}{2\gamma} a_{cr}^2 \right)$$

$$\rho_0 (V_0^2 + a_{cr}^2) = \rho_1 (V_1^2 + a_{cr}^2)$$

APPENDIX II

DERIVATION OF PRANDTL'S RELATION $V_0 V_1 = a_{cr}^2$

It a_{cr} is the critical sound speed, defined as the value at which the sound speed and the flow speed become equal in an expansion from a reservoir, the energy equation (A1-2) may be written

$$(A1-2) \quad \begin{cases} \frac{\gamma}{\gamma-1} \frac{P_0}{\rho_0} - \frac{1}{2} V_0^2 = \frac{\gamma+1}{\gamma-1} \frac{P}{\rho} - \frac{1}{2} V^2 \\ \frac{\gamma}{\gamma-1} \frac{P}{\rho} - \frac{1}{2} V^2 = \frac{\gamma+1}{\gamma-1} \frac{P_{cr}}{\rho_{cr}} - \frac{1}{2} V_{cr}^2 \end{cases}$$

from which

$$\begin{aligned} P_0 &= P \left(\frac{\gamma+1}{2\gamma} - \frac{\gamma-1}{2\gamma} \frac{V^2}{a_{cr}^2} \right) \\ P &= P_{cr} \left(\frac{\gamma+1}{2\gamma} - \frac{\gamma-1}{2\gamma} \frac{V_{cr}^2}{a_{cr}^2} \right) \end{aligned}$$

Putting these values for P_0 and P into the momentum equation (A1-2),

$$\begin{aligned} \rho_0 \left(\frac{\gamma+1}{2\gamma} V_0^2 + \frac{\gamma+1}{2\gamma} a_{cr}^2 \right) &= \\ \rho \left(\frac{\gamma+1}{2\gamma} V^2 + \frac{\gamma+1}{2\gamma} a_{cr}^2 \right) &= \\ \rho_0 (V_0^2 + a_{cr}^2) &= \rho (V^2 + a_{cr}^2) \end{aligned}$$

or, using the continuity equation (AI-1),

$$V_1 (V_0^2 + a_{cr}^2) = V_0 (V_1^2 + a_{cr}^2) \quad (\text{AII-2})$$

Solving (AII-2) for a_{cr}^2 ,

$$a_{cr}^2 = \frac{V_0 V_1^2 - V_0^2 V_1}{V_1 - V_0} = V_0 V_1 \quad (\text{AII-3})$$



or, using the constraint (A1-1),

$$\left(\frac{1}{V_0} + \frac{1}{V_1} \right) V_0 = \left(\frac{1}{V_0} + \frac{1}{V_1} \right) V_1$$

Solving (A1-1) for V_1

$$V_1 = \frac{V_0 V_1 - V_0 V_1}{V_1 - V_0}$$

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APPENDIX III

DERIVATION OF THE FORMULAE FOR THE SPEED OF A SHOCK OF PRESSURE RATIO γ , PROCEEDING INTO A STATIONARY GAS, AND FOR THE FLOW SPEED BEHIND THE SHOCK

The energy equation,

$$\frac{1}{2} V^2 + \frac{1}{\gamma-1} a^2 = \frac{\gamma+1}{2(\gamma-1)}$$

on division by $1/2 v^2$, becomes

$$1 + \frac{2}{\gamma-1} \left(\frac{a}{V} \right)^2 = \frac{\gamma+1}{\gamma-1} \left(\frac{a_{cr}}{V} \right)^2 \quad (\text{AIII-1})$$

If v is considered the flow speed upstream of a stationary shock, as in Appendix I, this is

$$1 + \frac{2}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 = \frac{\gamma+1}{\gamma-1} \left(\frac{a_{cr}}{V_0} \right)^2$$

or

$$\left(\frac{a_{cr}}{V_0} \right)^2 = \frac{\gamma-1}{\gamma+1} \left[1 + \frac{2}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 \right] = \frac{1}{\gamma} \left[1 + (\gamma-1) \left(\frac{a_0}{V_0} \right)^2 \right] \quad (\text{AIII-2})$$

An alternate expression for $(a_{cr}/v_0)^2$ is obtained from the Rankine-Hugoniot relation (AI-7), and Prandtl's Relation (AII-3):

$$\frac{V_1}{V_0} = \frac{V_0 V_1}{V_0^2} = \left(\frac{a_{cr}}{V_0} \right)^2 = \frac{\gamma + \mu}{\gamma \mu + 1} \quad (\text{AIII-3})$$

APPENDIX III

DERIVATION OF THE FORMULAS FOR THE SPEED OF A SHOCK OF
THE RANKINE-HUGONIO TYPE, PROCEEDING INTO A STATIONARY
GAS, AND FOR THE FLOW SPEED BEHIND THE SHOCK

The energy equation

$$\frac{1}{2} V^2 + \frac{1}{\gamma-1} a^2 = \frac{1}{2} V_0^2 + \frac{1}{\gamma-1} a_0^2$$

on division by V_0^2 , becomes

$$1 + \frac{1}{\gamma-1} \left(\frac{a}{V} \right)^2 = 1 + \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 \quad (AII-1)$$

If v is considered the flow speed upstream of a shock, as in Appendix I, this is

$$1 + \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 = \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 + \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2$$

or

$$\left(\frac{a_0}{V_0} \right)^2 = \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 \left[1 + \frac{1}{\gamma-1} \left(\frac{a_0}{V_0} \right)^2 \right]$$

An alternate expression for $\left(\frac{a_0}{V_0} \right)^2$ is obtained from
the Rankine-Hugoniot equation (AII-7), the pressure relation

(AII-3):

$$\frac{V_0}{V} = \frac{V_0}{V} = \frac{V_0}{V} = \frac{V_0}{V} \quad (AII-2)$$

setting these expressions for $(a_{cr}/v_0)^2$ equal,

$$\frac{\gamma + \mu}{\gamma\mu + 1} = \frac{1}{\mu} \left[1 + (\mu - 1) \left(\frac{a_0}{V_0} \right)^2 \right] \quad (\text{AIII-4})$$

from which

$$\left(\frac{V_0}{a_0} \right)^2 = \frac{\gamma\mu + 1}{\mu + 1}$$

or, transforming to coordinates in which the medium ahead of the shock front is at rest,

$$U = a_0 \sqrt{\frac{\gamma\mu + 1}{\mu + 1}} = a_0 \sqrt{\frac{6\gamma + 1}{7}} \quad (\text{AIII-5})$$

(AIII-5) with (AI-11) gives, for the flow speed behind the shock wave

$$u = a_0 \frac{(\mu - 1)(\gamma - 1)}{\sqrt{(\mu + 1)(\gamma\mu + 1)}} \quad (\text{AIII-6})$$

setting these expressions in (2.1) equal,

$$(2.11) \quad \left[\frac{1}{\gamma} \left(1 - \frac{v^2}{c^2} \right) \right] = \frac{1 + \frac{v^2}{c^2}}{1 + \frac{v^2}{c^2}}$$

from which

$$\left(\frac{v}{c} \right)^2 = \frac{1 + \frac{v^2}{c^2}}{1 + \frac{v^2}{c^2}}$$

or, transforming to coordinates in which the motion is of the shock front is at rest,

$$(2.12) \quad U = c \sqrt{\frac{1 + \frac{v^2}{c^2}}{1 + \frac{v^2}{c^2}}}$$

(2.12) with (2.1) gives for the time needed to pass the shock wave

$$(2.13) \quad U = c \sqrt{\frac{1 + \frac{v^2}{c^2}}{1 + \frac{v^2}{c^2}}}$$

APPENDIX IV

DERIVATION OF THE EXPRESSION FOR THE MACH NUMBER OF THE
FLOW BEHIND A SHOCK WAVE WHICH MOVES INTO
A STATIONARY MEDIUM

The ratio of the sound speed ahead of a shock front to that behind it is given by

$$\frac{a_0}{a_1} = \sqrt{\frac{T_0}{T_1}} = \sqrt{\frac{P_0 \rho_1}{P_1 \rho_0}} \quad (\text{AIV-1})$$

By means of Equation (AI-7) this may be written

$$\frac{a_0}{a_1} = \sqrt{\frac{\gamma \mu + 1}{\gamma(\gamma + \mu)}} \quad (\text{AIV-2})$$

With the flow speed given by Equation (AIII-6), the Mach Number is given by

$$\begin{aligned} M = \frac{u}{a_1} &= \frac{u}{a_0} \frac{a_0}{a_1} = \frac{(\mu - 1)(\gamma - 1)}{\sqrt{(\mu + 1)(\gamma \mu + 1)}} \cdot \sqrt{\frac{\gamma \mu + 1}{\gamma(\gamma + \mu)}} \\ &= \frac{(\mu - 1)(\gamma - 1)}{(\mu + 1)\gamma(\gamma + \mu)} = \frac{5(\gamma - 1)}{\sqrt{7\gamma(\gamma + 6)}} \quad (\text{AIV-3}) \end{aligned}$$

APPENDIX IV

DERIVATION OF THE EXPRESSION FOR THE WAVE NUMBER OF THE
 PION BEHIND A SHOCK WAVE WHICH MOVES INTO
 A STATIONARY MEDIUM

The ratio of the pion speed ahead of a shock front
 to that behind it is given by

$$\frac{C_0}{C_1} = \sqrt{\frac{D_0}{D_1}} = \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} \quad (AIV-1)$$

By means of Equation (AII-7) this may be written

$$\frac{C_0}{C_1} = \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} = \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} \quad (AIV-2)$$

With the pion speed given by Equation (AIII-2), the Mach
 Number is given by

$$M = \frac{U}{C_1} = \frac{U}{C_0} \frac{C_0}{C_1} = \frac{U}{C_0} \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} = \frac{U}{C_0} \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} \quad (AIV-3)$$

$$M = \frac{U}{C_0} \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} = \frac{U}{C_0} \sqrt{\frac{D_0}{D_1} \frac{D_1}{D_0}} \quad (AIV-4)$$

APPENDIX V

DERIVATION OF THE EXPRESSION FOR THE MACH NUMBER IN THE
FLOW BEHIND THE CONTACT SURFACE IN THE SHOCK TUBE,
WHEN AIR IS USED IN BOTH CHAMBERS

The sound speed in the air behind the contact surface is given by

$$a_{ic} = a_0 \left(\frac{P_1}{P_2} \right)^{\frac{\gamma-1}{2\gamma}} = a_0 \left(\frac{P_1}{P_2} \right)^{\frac{1}{\mu+1}} \quad (\text{AV-1})$$

From Equation (AIII-6)

$$\frac{P_0}{P_2} = \frac{1}{\gamma} \left[1 - \frac{\gamma-1}{\sqrt{(\mu+1)(\gamma\mu+1)}} \right]^{\mu+1}$$

or

$$\frac{P_0}{P_2} \gamma = \frac{P_0}{P_2} \frac{P_1}{P_0} = \frac{P_1}{P_2} = \left[1 - \frac{\gamma-1}{\sqrt{(\mu+1)(\gamma\mu+1)}} \right] \quad (\text{AV-2})$$

with (AV-1), this gives

$$\frac{a_{ic}}{a_0} = 1 - \frac{\gamma-1}{\sqrt{(\mu+1)(\gamma\mu+1)}} \quad (\text{AV-3})$$

Therefore the Mach number is given by

$$M_{\text{Predicted Contact Surface}} = \frac{u}{a_{ic}} = \frac{\sqrt{\frac{(\mu-1)(\gamma-1)}{(\mu+1)(\gamma\mu+1)}}}{1 - \frac{\gamma-1}{\sqrt{(\mu+1)(\gamma\mu+1)}}} = \frac{(\mu-1)(\gamma-1)}{\sqrt{(\mu+1)(\gamma\mu+1)} - (\gamma-1)} \quad (\text{AV-4})$$

$$= \frac{5(\gamma-1)}{\sqrt{7(6\gamma+1)} - (\gamma-1)}$$

APPENDIX V

DERIVATION OF THE EXPRESSION FOR THE MAGNITUDE OF THE
FROM WHICH THE CONTACT STRESS IS THE STRESS
WHEN AIR IS USED IN BOTH CHANNELS

The contact stress in the air bearing is given by
Equation (17-1) is given by

$$Q_{10} = Q_0 \left(\frac{P}{P_0} \right)^{\frac{1}{n-1}} = Q_0 \left(\frac{P}{P_0} \right)^{\frac{1}{n-1}} \quad (17-1)$$

From Equation (17-1) is

$$\frac{P_0}{P} = \frac{1}{\left(\frac{Q_{10}}{Q_0} \right)^{n-1}} = \frac{1}{\left(\frac{Q_{10}}{Q_0} \right)^{n-1}} \quad (17-2)$$

$$\frac{P_0}{P} = \frac{1}{\left(\frac{Q_{10}}{Q_0} \right)^{n-1}} = \frac{1}{\left(\frac{Q_{10}}{Q_0} \right)^{n-1}} \quad (17-3)$$

With (17-2) and (17-3) is

$$\frac{Q_{10}}{Q_0} = \frac{1}{\left(\frac{P_0}{P} \right)^{\frac{n-1}{n}}} = \frac{1}{\left(\frac{P_0}{P} \right)^{\frac{n-1}{n}}} \quad (17-4)$$

Therefore the contact stress is given by

$$\frac{Q_{10}}{Q_0} = \frac{1}{\left(\frac{P_0}{P} \right)^{\frac{n-1}{n}}} = \frac{1}{\left(\frac{P_0}{P} \right)^{\frac{n-1}{n}}} \quad (17-5)$$

APPENDIX VI

DERIVATION OF THE EXPRESSION FOR THE CHOKING
MACH NUMBER IN THE FLOW IN THE SHOCK

Sibert¹ develops the following equation stating that A_1 is for the flow in the tube and A_2 is for the flow at the model.

$$\frac{A_2}{A_1} = \frac{M_1}{M_2} \left(\frac{1 + \frac{\gamma-1}{2} M_2^2}{1 + \frac{\gamma-1}{2} M_1^2} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \quad (\text{AVI-1})$$

for $\gamma=1.4$

$$\frac{A_2}{A_1} = \frac{M_1}{M_2} \left(\frac{1+0.2 M_2^2}{1+0.2 M_1^2} \right)^3 \quad (\text{AVI-2})$$

M_1 becomes M_{ch} and M_2 becomes unity since the flow at the model will become unity Mach.

$$\frac{A_2}{A_1} = \frac{1.728 M_{ch}}{(1+0.2 M_{ch}^2)^3} \quad (\text{AVI-3})$$

since the area of the model, A_m , taken from the area of the tube is equal to the area at the model

$$A_2 = A_1 - A_m$$

$$\frac{A_1 - A_m}{A_1} = \frac{1.728 M_{ch}}{(1+0.2 M_{ch}^2)^3} \quad (\text{AVI-4})$$

$$\frac{A_m}{A_1} = 1 - \frac{1.728 M_{ch}}{(1+0.2 M_{ch}^2)^3}$$

¹Sibert, loc. cit., pp. 85-86.

APPENDIX II

DERIVATION OF THE EXPRESSION FOR THE CHANGING
WATER LEVELS IN THE TIDE IN THE SHOCK

Since the tide is a function of time, the water level in the shock is a function of time. Let A be the water level in the shock at time t . Then the flow at the shock is

$$(A-1) \quad \frac{dA}{dt} = \frac{M}{M_1} \left(\frac{1}{2} + \frac{1}{2} \frac{M}{M_1} \right)$$

for $A < A_0$

$$(A-2) \quad \frac{dA}{dt} = \frac{M}{M_1} \left(\frac{1}{2} + \frac{1}{2} \frac{M}{M_1} \right)$$

M_1 denotes the water level in the shock at time t . The model will be used to find A_0 .

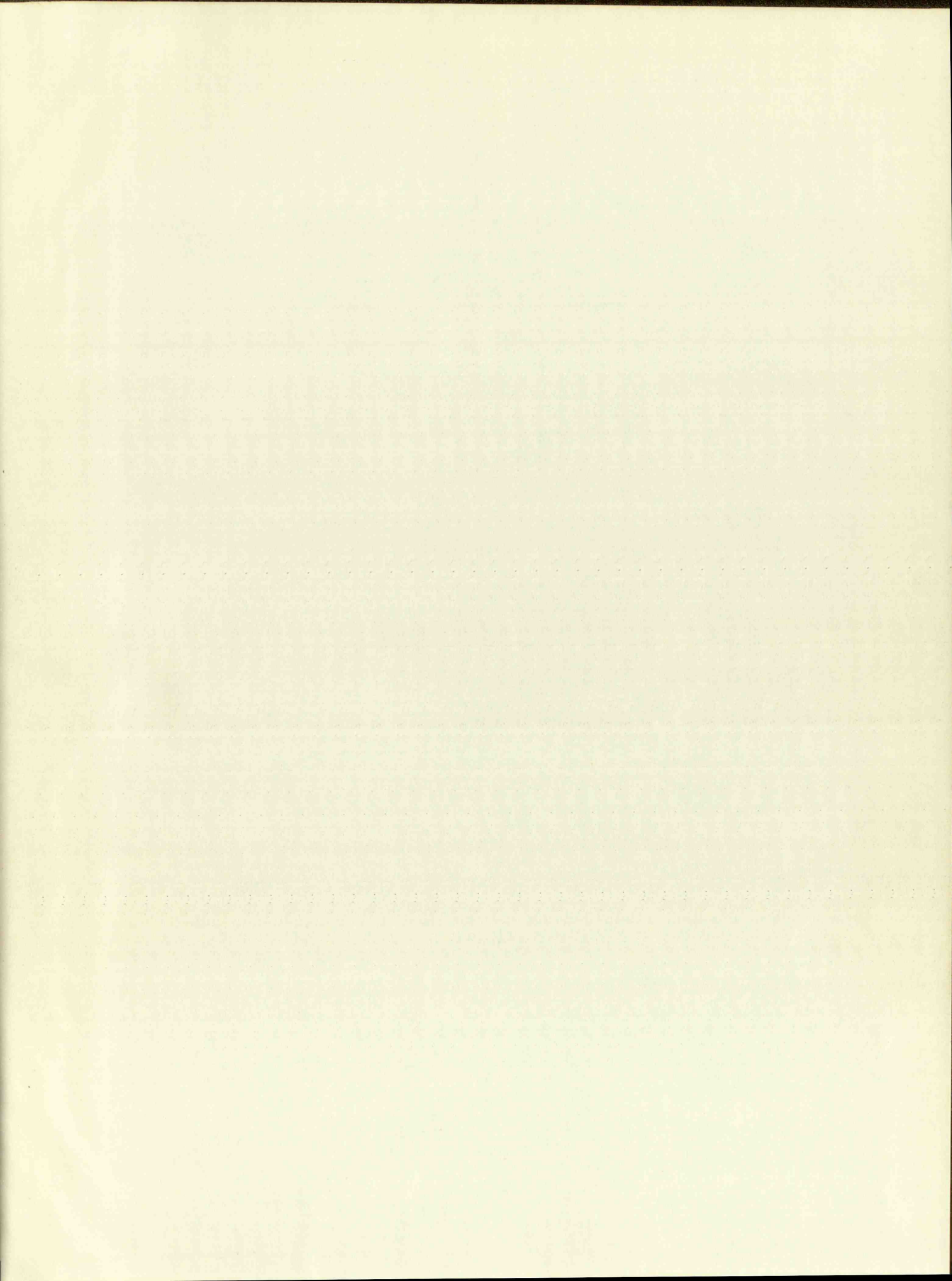
$$(A-3) \quad \frac{dA}{dt} = \frac{M}{M_1} \left(\frac{1}{2} + \frac{1}{2} \frac{M}{M_1} \right)$$

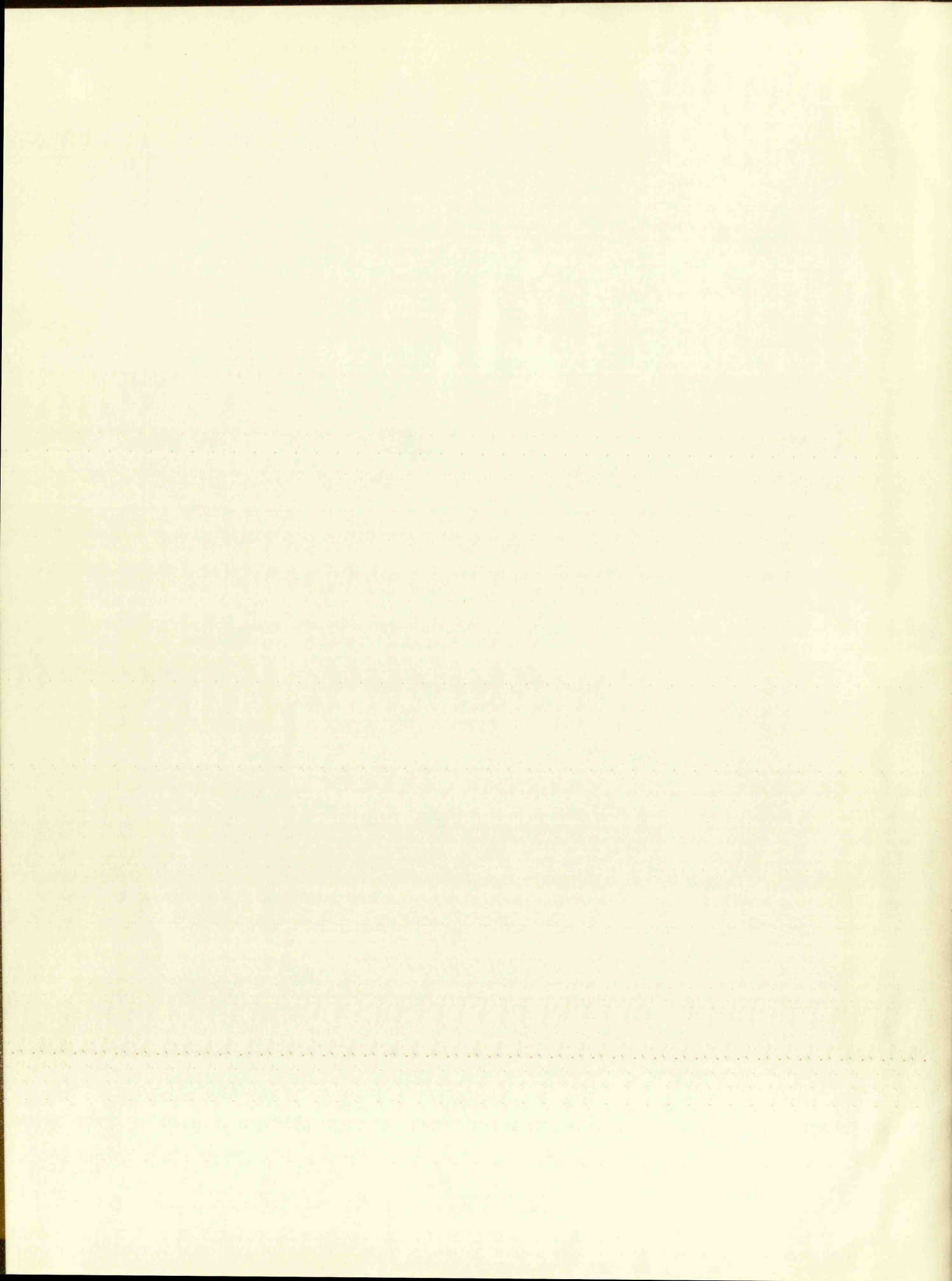
since the water level in the shock is A_0 when the tide is at the peak. The water level in the shock is A_0 when the tide is at the peak.

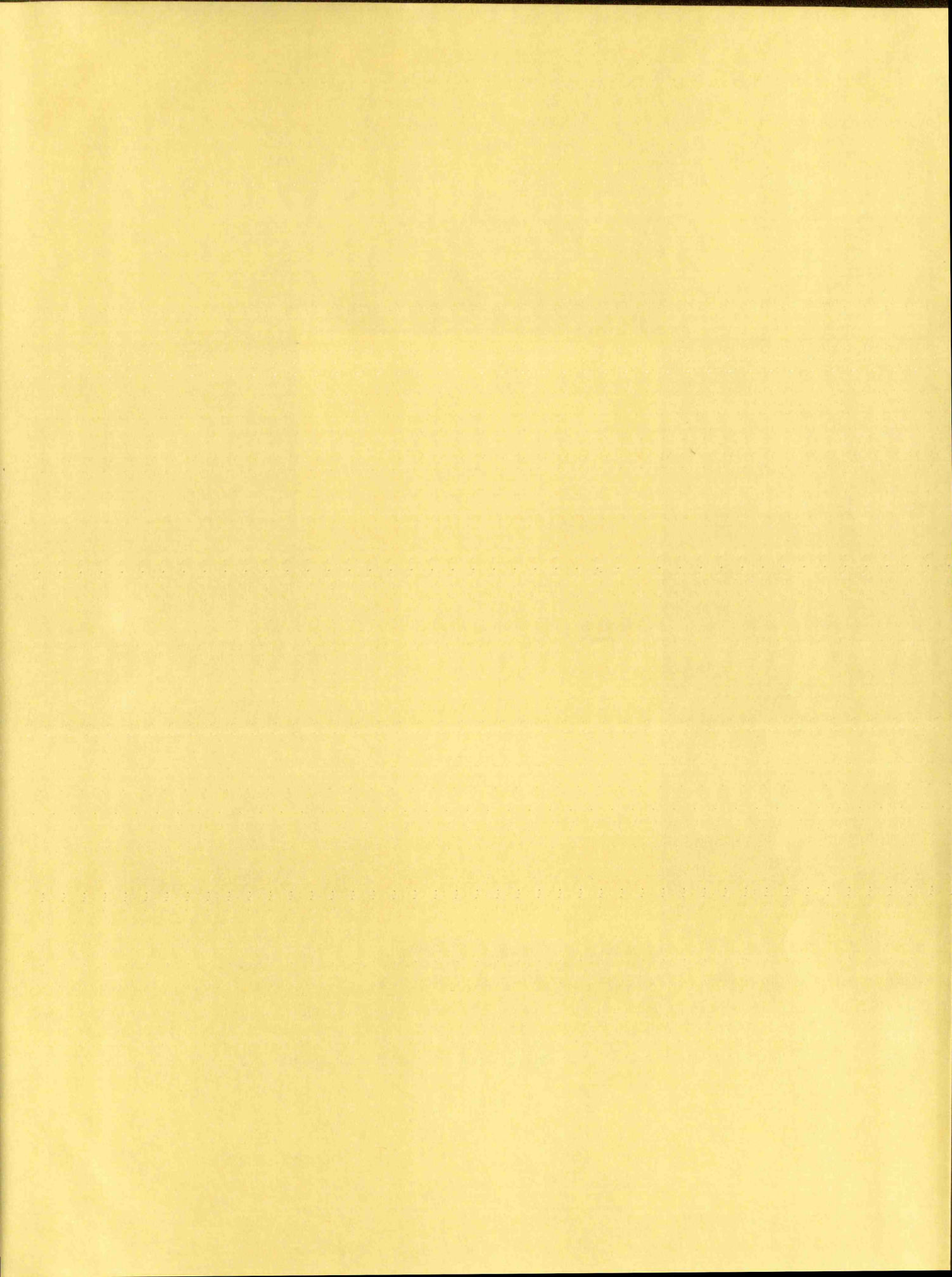
$$A_0 = A_0 - A_0$$

$$(A-4) \quad \frac{dA}{dt} = \frac{M}{M_1} \left(\frac{1}{2} + \frac{1}{2} \frac{M}{M_1} \right)$$

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