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THE APPRAISAL OF CONTENT IS TENTATIVE.
(FOR KEY SEE REVERSE)

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Outline of Contents

(This outline is not exactly adhered to, but its points are covered in the report.)

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The Problem: The Diffusor in Aerodynamic Engines.

In all aerodynamic engines the following processes take place independently or in combination with others:

a. Nozzle Processes: With a decrease in pressure there is an increase in the velocity of the gaseous medium.

b. Diffusor Processes: The pressure is increased when the velocity of the medium is decreased.

c. Mechanical energy is either supplied to or derived from the medium by moving engine parts. It can always be proved whether such a process is combined with a nozzle process or with a diffusor process.

d. By means of combustion or a heat exchanger, energy in the form of heat is either supplied to or derived from the medium.

The quality of an aerodynamic engine depends on whether it can carry out the above processes while keeping the entropy increase very small.

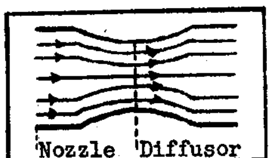
a. Nozzle processes can be carried out in such a way that the entropy increase is limited to an insignificant and unavoidable amount caused by surface friction.

b. Diffusor processes have up until now been carried on with sufficiently small entropy increases only at subsonic velocities. No matter how good the diffusor flow may be, the entropy increase is always greater than in the case of nozzle flow. Diffusors which work with an efficiency equal to that of nozzles have been used at subsonic velocities only with special dissipative operations (boundary layer control, Magnus effect).

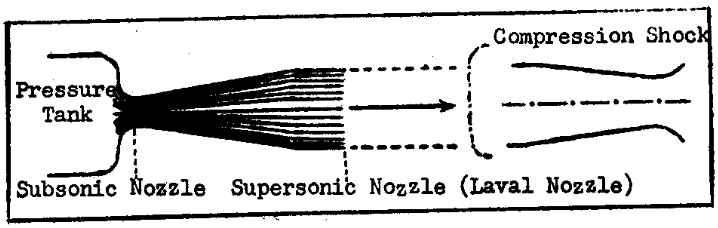
In the case of supersonic flow, diffusors are not possible as reverse Laval nozzles.

Figure 1

Diffusor as the Reverse of a Nozzle



a. In the case of subsonic flow.



b. In the case of supersonic flow.

a. In the case of subsonic flow, a diffusor as the reverse of a nozzle is theoretically possible (assuming a careful flaring).

b. In the case of supersonic flow, a diffusor as the reverse of a nozzle is theoretically impossible. A plane shock wave will always be formed in front of the intake of a conically constricted tube. Shock waves are always produced when the attempt is made to retard a supersonic flow with pressure increase by means of a diffusor. With an increasing Mach number the inefficiency, caused by the entropy decrease in the plane shock wave, always increases very rapidly.

c. The energy exchange between moving engine parts and the flowing gases as the power medium always shows a slight, unavoidable amount of entropy increase as a result of surface friction. A considerable limitation in the form of aerodynamic engines of all types is imposed not only by the fact that the flow at moving engine parts is accelerated with a decrease in pressure, but also by the fact that there are always areas in which the flow is retarded with an increase in pressure. The greater the retardation, the steeper the pressure increase will be, and the greater the losses and entropy increase will be as a result of flow separation, shock waves, and turbulence. Thus, in designing aerodynamic engines, and especially in the case of compressors, the attempt is always made to distribute the pressure increase on the turbine wheel and the diffusor so that a too rapid local increase of pressure can be avoided. Up until now supersonic velocities in compressors have been avoided because of the great difficulties of diffusor operation in supersonic flow. This limited considerably the operational efficiency of compressors.

d. The processes of heat input and heat exchange are not directly influenced by the problems connected with the diffusor, as far as efficiency is concerned. On the other hand, if a solution to the problems connected with the diffusor can be obtained so that aerodynamic engines can be operated with considerably higher pressures, then much smaller heat exchangers can be built. In addition, the processes of combustion will be able to be carried on with greater efficiency at higher pressures, combustion chambers will be able to be built smaller, and the dissociation of combustion products in the case of some combustion reactions will be able to be avoided through increased pressure.

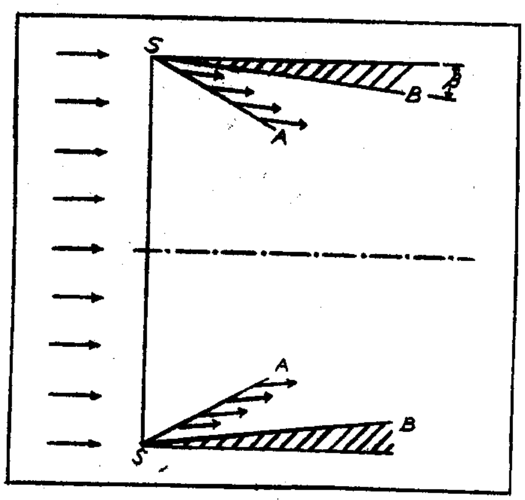
This survey of the principle processes in aerodynamic engines is intended to present and interpret the chief significance of the diffusor problem, the transformation of any velocity into pressure.

Diffusor Processes in the Case of Supersonic Velocities

Subsonic diffusors have been so well studied, it appears that no appreciable progress in their development is to be expected in the foreseeable future. To be sure, the processes in conically extended tubes need further study in the range of the Mach numbers between 0.5 and a little below 1.0. It appears that the most favorable angle of divergence with these Mach numbers is smaller than the eight degree angle generally considered to be ideal. In contrast to the well investigated processes in subsonic diffusors, little besides the plane shock wave has become known about diffusors used at supersonic velocities. Busemann and his co-workers in Braunschweig from 1938 to 1942 attempted to explain, by theory and by experiments in the wind tunnel, what possibilities exist for building supersonic diffusors, and especially whether it is possible to force a flow pattern which represents the reverse of the flow in a Laval nozzle. Research on these flows was facilitated by the process (hodographic process) of Guderlei (Busemann's co-worker in Braunschweig in 1942), which is a further development of Prandtl's and Busemann's process for the production of two-dimensional supersonic flows.

Figure 2

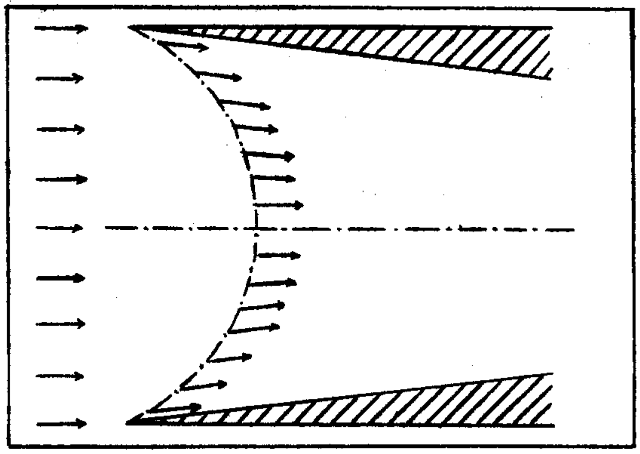
Flow within a Short, Convergent, Conical Tube



The supersonic flow coming from the left strikes at S at the edge of a conical surface SB. The streamlines close to the edge are deflected from the axis by the angle β. The deflection of the streamlines propagates along the conical surface SA of an oblique shock wave. Behind the oblique shock wave the pressure has increased. If the course of flow is extended from the edge S in the direction of the oblique shock wave SA, all the parameters of flow behind the shock, including the shock angle σ and the compression ratio P_1/P_0, remain variable along the oblique shock wave in the direction of the axis.

Figure 3

Flow within a Convergent, Conical Tube



The above sketch (Figure 3) shows the flow within a convergent, conical tube up to the axis. As the distance from the axis becomes smaller, the angle σ of the oblique shock wave increases until, at the axis, it becomes 90 degrees. The oblique shock wave at the edge has thus become a plane shock wave at the axis. This flow pattern is, however, not stable and can only be produced in the wind tunnel with short conical tubes. Besides, there is no regeneration of the full adiabatic pressure. In the case of longer conical tubes, and in the case of the slightest disturbance in the short conical tubes, the plane shock wave moves upstream along the axis of the cone and stands in front of the conical tube.

Figure 4

Plane Shock Wave in front of Tube

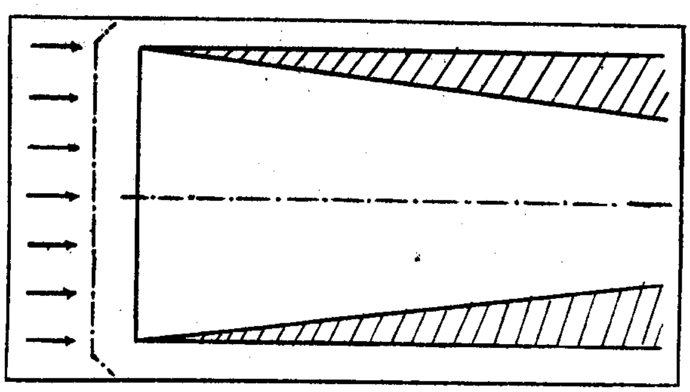
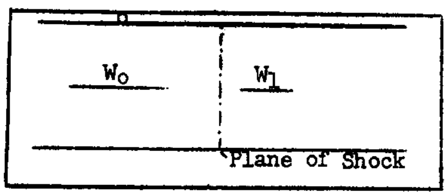


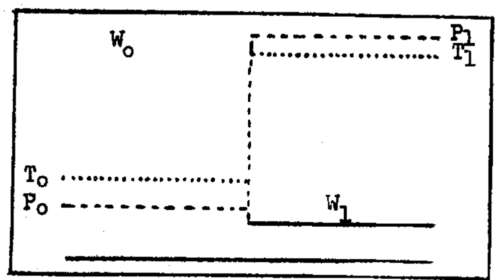
Figure 4 shows a plane shock wave in front of a tube. A variant of this flow, the flow around the Busemann ring, will not be discussed further here, for it produces only a slightly better compression than the simple plane shock wave.

A plane shock wave always occurs in the supersonic range when the velocity is retarded.

Figure 5 Plane Shock Wave



a. Streamline Pattern

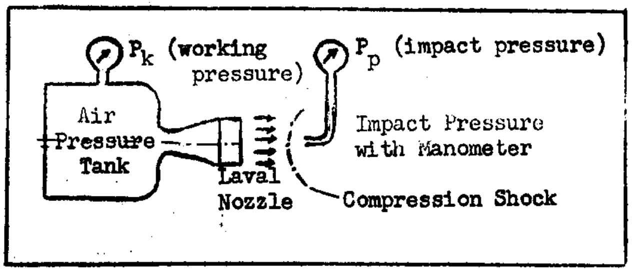


b. Variables of State

The plane shock wave takes place within a plane perpendicular to the direction of flow. In the shock plane the velocity decreases to subsonic velocity, while the pressure and the temperature both increase.

[redacted]

Figure 6



From Enclosure 1 it is apparent that the efficiency of compression in the plane shock wave drops rapidly with an increase in the Mach number. The plane shock wave can be utilized technically only when the Mach number is slightly over 1.0.

The Multiple-Shock Supersonic Diffusor

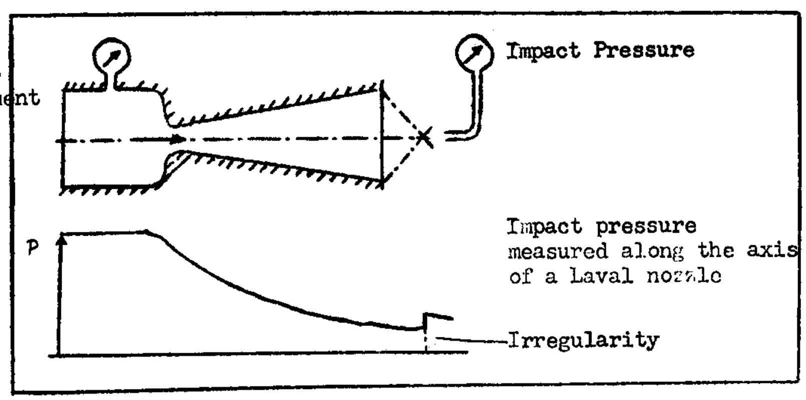
The Simple Two-Dimensional Multiple-Shock Supersonic Diffusor in Theory

Discovery of the multiple-shock supersonic diffusor: [redacted]

[redacted] The invention of the multiple-shock supersonic diffusor was preceded by an observation by Stodola, which was correctly interpreted by Prandtl who thus pointed out the way to the invention of the multiple-shock supersonic diffusor. Stodola had measured impact pressures in a Laval nozzle through which steam was flowing and struck upon an inexplicable irregularity. (Stodola: Gas Turbines)

Figure 7

Stodola's Experimental Arrangement



Impact pressures measured along the axis of a Laval nozzle through which steam was flowing showed a regular decrease in the direction of higher Mach numbers of flow. The regularity of this pressure drop was interrupted by a sudden pressure rise close behind the mouth of the Laval nozzle. Prandtl interpreted this as proof of an oblique shock wave moving out from the mouth. The pitotube showed a higher pressure behind this oblique shock wave. With the same Mach number of approach flow, therefore, a combination of an oblique shock wave (oblique shock moving from the mouth of the nozzle) and a plane shock wave (in front of the impact tube) produces a greater regeneration of pressure than a simple plane shock wave alone.

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[redacted]

When in 1940 the question of a high pressure regeneration from a supersonic flow with greater Mach numbers became more and more pressing on the Trommsdorff missiles (Lorin engines), and when the slight pressure regeneration in the plane shock wave was such a handicap to development that the entire project seemed to be jeopardized, Prandtl recalled the observation of Stodola, and urged that the combination of an oblique and a plane shock wave be investigated. 25X1

In pursuing these investigations, which were carried on simultaneously in three different places, by Oswatitsch, Seiffarth, and Walchner in Goettingen, Busemann and his co-workers in Braunschweig, [redacted] the fact was established that such a combination of two shocks actually does provide for considerably greater pressure regeneration than could be attained with the single plane shock wave. 25X1

[redacted] 25X1

The oblique shock wave: If a supersonic flow is deflected by the angle β from its direction of flow by a cutting edge, this unsteady deflection continues in the flow in the form of a shock wave at the angle $\sqrt{}$.

[redacted] 25X1

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The combination of several shocks in the multiple-shock diffusor: If the flow still has supersonic velocity after the first oblique shock wave, it can pass through a second oblique shock wave, and so on, until after a final shock wave, which is mostly a plane shock wave, the flow finally is decelerated to subsonic velocity. In the question of what combinations of shocks at a given Mach number and a given number of shocks produces an optimal efficiency of supersonic compression, it can be derived and proved that the optimal efficiency is reached when $D_1 = D_2 = D_3 = ... = D_{n-1} = C_n$, whereby C_n is the square of the Mach number before the final plane shock wave which terminates the supersonic compression. [redacted] Up until now these deviations have not been formed successfully into a mathematical presentation; they can be determined only from case to case through time consuming numerical and graphical operations.

The optimal compressions in these three-dimensional multiple-shock diffusors are always a little better than the optimal compressions in the corresponding two-dimensional multiple-shock diffusors, so that the values of the two-dimensional diffusors give a very good indication of the operational performance of all multiple-shock diffusors.

In Enclosure 2 the degrees of efficiency of optimal two-dimensional multiple-shock diffusors, depending on the square of the Mach number of the flow, are plotted, and in each case all the values for optimal diffusors with two, three, and more shocks are combined into a single curve. The degrees of efficiency for the two-dimensional shock wave, already presented in Enclosure 1, are also plotted as a single shock diffusor.

From an observation of the curves in Enclosure 2 it is apparent that the degree of efficiency in the multiple-shock supersonic diffusor at all Mach numbers can be brought close to the maximum value of 1.0, if the number of shocks is high enough.

The inverse Meyer-Prandtl flow as a diffusor with infinite shock number: If the number of shocks is very high, whereby the value E becomes smaller and smaller and finally differs only very little from 1.0, then a very interesting flow pattern is produced, which we shall call the "inverse Meyer-Prandtl flow". In his dissertation (1905) Meyer investigated, upon Prandtl's suggestion, the constant deflection of a supersonic flow with a simultaneous transformation of velocity into pressure. (A nozzle process in the sense of the introductory section.)

(See Figure 9.) A gas flows along a solid wall with supersonic velocity. There is, on the other side of the wall, an evacuated space. The wall ends in a cutting edge, after which the gas flow is unconfined. The individual gas particles now flow, while cooling and expanding, with increasing velocity on curved paths into the evacuated space on the other side of the solid wall.

a. All streamlines are similar to one another only in respect to the center of similarity at the point of the cutting edge S.

b. On a straight line (Mach line) drawn through the point of the cutting edge (S) variables of state (pressure, density, temperature, velocity and Mach number) are always the same.

c. A straight line drawn through the point of the cutting edge S bisects all other streamlines at the same angle α. This angle is also the Mach angle.

$$\sin\alpha = \frac{1}{Ma}$$

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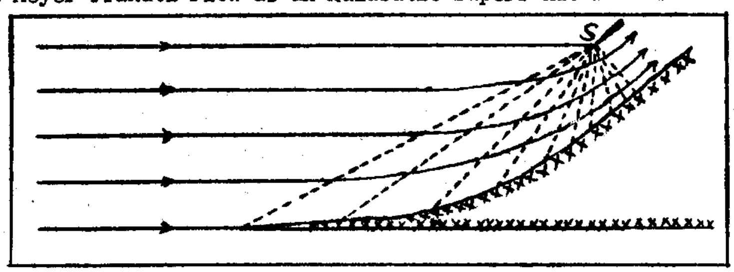
If a parallel homogeneous supersonic flow strikes a curved surface, a deceleration and adiabatic compression of the gas takes place along the curved surface with a simultaneous deflection. This process is, however, adiabatic only for a short distance. The Mach waves going out from the curved surface interfere with one another and are integrated to form an oblique (non-adiabatic) shock wave. If one now attempts to adapt the curvature of the surface to the construction processes well known in the literature in such a way that this interference of the Mach waves does not produce an oblique shock wave, then one produces a contour which corresponds to the course of the streamline of the Meyer-Prandtl flow. One important construction specification (which will later by applied to similarly constructed flows) is that the contour of flow should be such that all Mach waves converge in a single point. The inverse Meyer-Prandtl flow is at the same time a multiple-shock diffusor with a very great number of shocks, which, however, have such slight compression that they may become Mach waves.

Figure 11 Inverse Meyer-Prandtl Flow as an Adiabatic Supersonic Diffusor



The compression in such a Meyer-Prandtl diffusor takes place without any adiabatic loss. [redacted] there are a great number of possible applications of the Meyer-Prandtl flow and of the inverse Meyer-Prandtl flow in engineering which had not been used at all until recently. This is especially true in the case of nozzles of steam- and gas-turbines, and in blades of constant-pressure gas turbines, whose improved flow application through proper design could result in greatly increased efficiency of one-pressure stages. [redacted] 25X1

Geometric Arrangement of Two-Dimensional Multiple-Shock Diffusors

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In the preceding section the multiple-shock supersonic diffusors were characterized by the Mach number of flow, the number of shocks, and the index D of the shocks. No explanation was given on how these shocks are released, and no mention was made of technical application.

The simplest theoretical forms are the so-called two-dimensional multiple-shock diffusors. (The designation is not accurate, for the two-dimensional diffusors are also three-dimensional.) Two-dimensional flows (also called plane flows) are those which can be completely represented by a two-dimensional cross section. The plane of cross section lies in the direction of flow, all edges of bodies are perpendicular to the plane of cross section, and the flow is assumed to be extended arbitrarily on both sides of the plane of cross section. The same flow pattern is produced in all further planes of cross section parallel to the plane of projection. The two-dimensional presentation is used for cutting edges and profiles with perpendicular approach flow. Two-dimensional multiple-shock diffusors have two basically different possibilities of arrangement, one-sided and two-sided.

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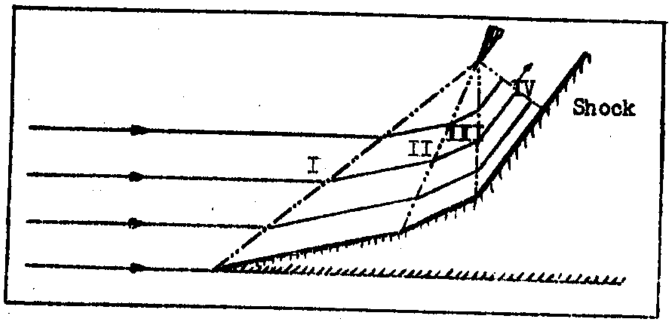
Figure 12

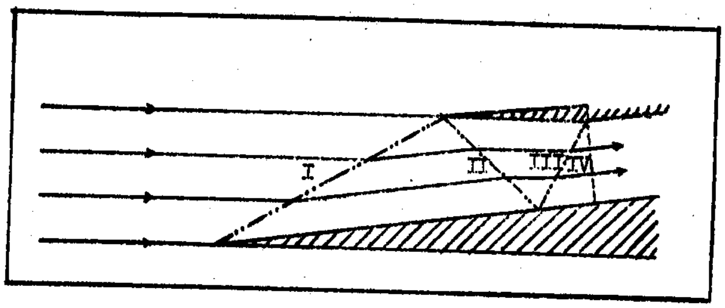
One-Sided Four-Shock Diffusor



The one-sided arrangement has the disadvantage that the flow, after passing through the diffusor, is more or less deflected from the original approach flow direction. Designing such one-sided, two-dimensional diffusors for installation in aerodynamic engines is often difficult. On the other hand, the flow in one-sided, two-dimensional diffusors is much more stable and less subject to interference than in two-sided diffusors.

Figure 13

Two-Sided Four-Shock Diffusor

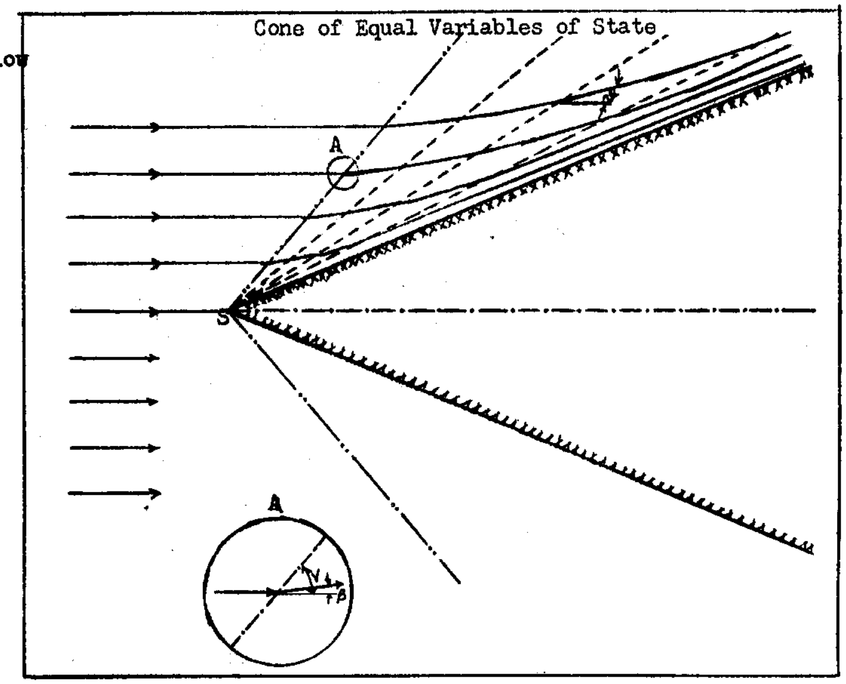


In the two-sided arrangement the flow is only slightly deflected from its original direction. The design of the installation of such a diffusor is therefore much easier. Unfortunately, the two-sided diffusors in practical research show a considerable tendency toward sensitivity to interference and instability of flow. The final plane shock wave has the tendency to move upstream and stand in front of the diffusor.

Rotation-symmetrical multiple-shock diffusors: Of all research done on three-dimensional flows up until now, rotation-symmetrical flows have been best investigated and have the greatest technical importance. In the case of a rotation-symmetrical flow, the flow around or in the rotation-symmetrical solid body is investigated. The direction of the parallel homogeneous flow lies in the direction of the axis of symmetry of the rotation-symmetrical solid body. The case of the oblique approach flow, in which a body presenting rotation symmetry is subject to flow at an angle to the axis, has so far presented the greatest difficulty in theoretical treatment. The simplest case of rotation symmetrical supersonic flow is the flow around a straight circular cone.

Figure 14

Conical Flow



A conical oblique shock wave extends from the point of a cone which is subjected to flow in the axial direction. For a small area A in the immediate vicinity of this shock wave, the same considerations and formulas apply as in the case of the two-dimensional oblique shock wave already derived. The same conditions exist on all points of the conical shock wave. Shock number D, compression ratio, angle of deflection β, and impact angle γ are everywhere equal.

In the area between the conical shock wave and the cone, the variables of state and the direction of flow (ϑ angle of direction of flow with the axis) change along a streamline. On coaxial cones of flow through the point of the solid cone the variables of state and direction of flow are always equal.

The flowing gas is decelerated along a streamline with simultaneous (more or less insignificant) adiabatic compression and simultaneous deflection. The streamlines run asymptotically to the solid cone.

The conical flow can be calculated and illustrated without difficulty. Indeed, no consistent formula can be given for the streamlines, but the form of the streamlines is established by two simultaneous differential equations, which can be solved numerically by the well-known Runge-Kutta or Adams-Stoermer method, or graphically by the Busemann-Guderlei method. To be sure, the amount of calculation or drawing necessary for each conical flow is considerable. A second solid cone, and so on, can be attached to the first solid cone.

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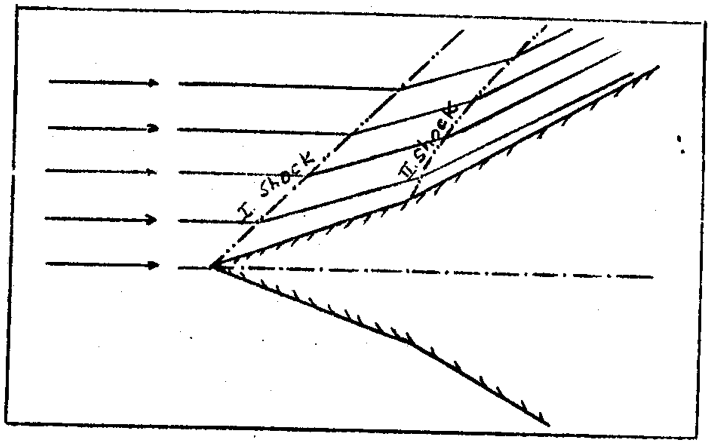
Figure 15

Flow Around Two Subsequent Cones

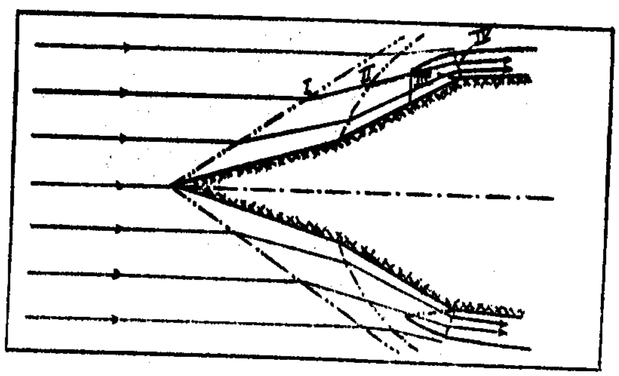


The flow in the vicinity of the second cone can be replaced with great approximation by a simple conical flow. The deviations amounted to only a few minutes of angle in cases thus far investigated.

The path of the second shock wave in the plane of projection is no longer a straight line and can be determined only through laborious graphics. Just as a two-dimensional multiple-shock diffusor can be constructed from a sequence of oblique shock waves, a three-dimensional rotation symmetrical multiple-shock diffusor can, by analogy, be constructed from a series of conical shock waves and conical flows.

Figure 16

Three-Dimensional Rotation-Symmetrical Four-Shock Diffusors



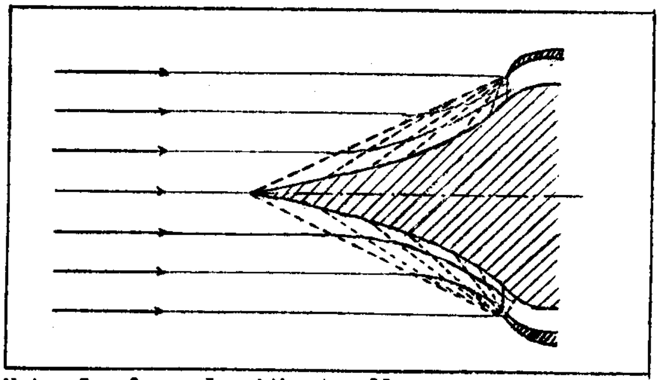
The theoretical efficiency of such three-dimensional diffusors is a little bit higher than the efficiency given in Enclosure 2 for two-dimensional diffusors, because there is an additional compression without adiabatic loss in the areas between the shock waves and the solid cones. This deviation likewise changes the requirement of equal D for all shock waves. The optimal values are then to be determined only through very time-consuming comparative calculations. (See also Enclosure 15 - Flow pattern of a five-shock diffusor at Mach 4, compression 1: 102, $\vartheta = 0.7$.)

The construction of a diffusor requires a special, time-consuming calculation (determining the most favorable D-sequence), if an efficiency less than 1 is assumed for the subsonic portion.

Three-dimensional Meyer-Prandtl diffusor, Laval nozzle without outer wall: Just as in the case of two-dimensional multiple-shock supersonic diffusors, where the diffusor with several shocks which degenerated into Mach waves produced the greatest efficiency of compression when the diffusor was designed as an inverse Meyer-Prandtl flow with Mach waves converging to a single point, it is also possible to design a three-dimensional rotation symmetrical supersonic diffusor with several shocks which have degenerated into Mach waves, if the Mach waves all converge to a single point. The construction of such three-dimensional diffusors with Meyer-Prandtl flow requires an enormous amount of graphics work and calculations.

Figure 17

Three-Dimensional Meyer-Prandtl Diffusor

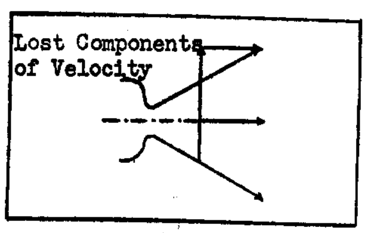


Note: Laval nozzle without walls.

The Laval nozzles of powder rockets and other jet engines involve the following problems: For complete pressure release a definite ratio between the narrowest cross section in the Laval nozzle and the largest cross section at the end of the Laval nozzle is necessary for every gas pressure in front of the nozzle. This ratio of cross section can be attained by Laval nozzles of varying angles of divergence. A short Laval nozzle with a large divergence angle produces a strongly divergent jet.

Figure 18

Short Laval Nozzle with a Large Angle of Divergence

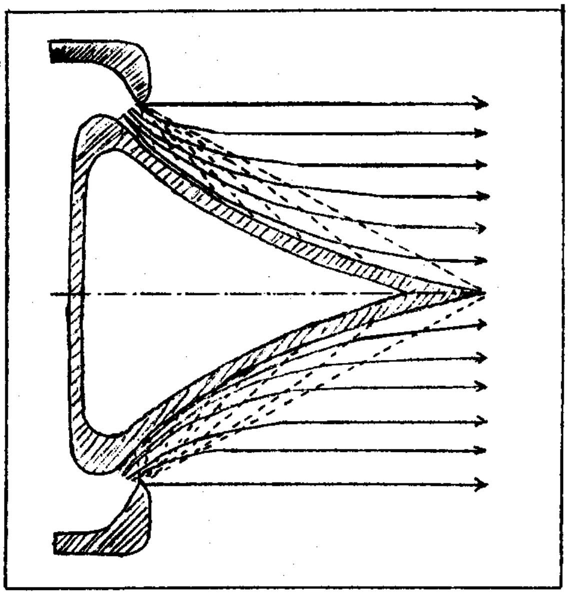


At the edge of the emitting gas jet the vectors of the gas velocity are inclined at a rather large angle to the axis. The velocity component perpendicular to the axis is lost for the purpose of producing thrust; only the axial components of the velocity vector determine the desired thrust. Therefore, jet engine Laval nozzles with too great an increasing angle of divergence produce a decrease of specific thrust. Long Laval nozzles with slight divergence have an almost parallel ejecting jet, but they have great losses because of friction along the wall of the nozzle. Such long nozzles are also too heavy.

The Laval nozzle without external wall with parallel jet exhaust has the advantages of lesser weight, insignificant losses as a result of friction, and no losses caused by divergence of flow.

Figure 19

Flow in a Laval Nozzle without an External Wall



Practical Results and Experimental Techniques

The most favorable forecast which could be made for the use of multiple-shock diffusors would be worthless unless it could be followed by practical experimentation. These practical experiments were conducted in the supersonic wind tunnels in Goettingen, Braunschweig, and Kummersdorf and produced in addition to a splendid confirmation of the theory a whole series of theoretically unforeseeable facts, which alone made the actual operation of supersonic diffusors possible. Ram-jet engines (Lorin engines) equipped with supersonic diffusors were also tested in free flight. From the functioning of these engines conclusions could be drawn on the functioning of the supersonic diffusors installed in them. The experiments were carried on through a series of methods which showed common characteristics. The method of diffusor research can be illustrated by a single example.

Example: Installing a three-dimensional multiple-shock diffusor.

Figure 20

Experimental Method of Measuring the Parameter of a Multiple-Shock Supersonic Diffusor

(Arranging the model in the wind tunnel)

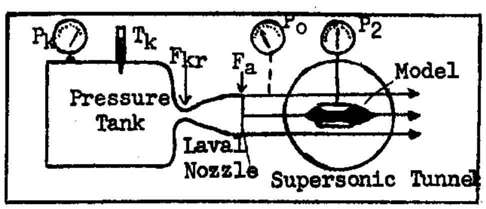


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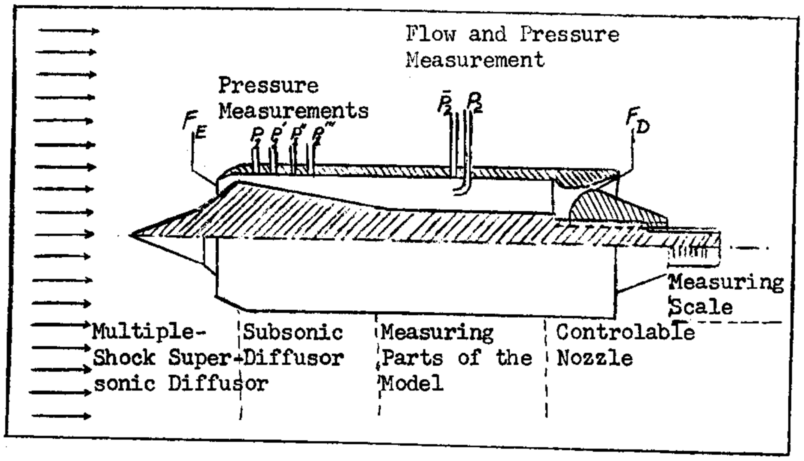
Figure 21

Details of Model



In front of a Laval nozzle (pressure tank or external air according to the arrangement in the wind tunnel) the pressure (P_k) and temperature (T_k) of the air measured. Behind the Laval nozzle the static pressure (P_o) is measured. The ratio F_{kr} (cross section at the narrowest critical point of the nozzle) to F_a (cross section of exit of the Laval nozzle) determines the Mach number behind the Laval nozzle. The so calculated Mach number is checked with the measured ratio P_o / P_k.

A third check of the Mach number can be made by means of the evaluation of Schlieren photographs of the flow behind the Laval nozzle. The measured pressure-chamber temperature is entered in relation with the Mach number in order to calculate the velocity in the supersonic tunnel. In the free air flow of the tunnel the diffusor model is suspended and equipped. Pressure gauges P_1, P_1', P_1'', and P_1''' record the processes in the subsonic portion of the diffusor.

The measurements of the static pressure $\bar{P}_2$ and the impact pressure P_2 in the measuring section of the model produce these results:

a. The portion of the air flows through the "flow" (Durchfluss) of the diffusor (from the difference between impact pressure and static pressure).

b. The regeneration of pressure in the diffusor caused by the impact pressure P_2.

c. The efficiency of the diffusor ϑ is from the ratio P_2/P_k . $\vartheta = \frac{P_2}{P_k}$.

The measurement area is closed by a nozzle with an adjustable narrowest opening F_D (throttling orifice).

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The carrying out of an experiment will serve as a practical example and at the same time illustrate the behavior of the multiple-shock supersonic diffusor. A three-dimensional rotation-symmetrical diffusor designed for Mach number 3.2 is to be tested. The theoretical efficiency factor from

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Enclosure 2 amounts to $\vartheta = 0.787$ ($\vartheta = 0.276$ in the case of a plane shock wave). The adjustable nozzle was set at the greatest value of F_D (throttle cross section). A test period of about six seconds, long enough for the instruments to settle down and record a fixed value, was used. The pressure regeneration or efficiency factor and the flow were plotted against F_D / F_E as an independent variable. An efficiency factor of 0.23 was measured.

In the next experiment the throttling cross section was reduced somewhat. The measured efficiency factor amounts to 0.25. The measured flow remained the same. In this manner, a series of 20 or 30 experiments were conducted with a stepwise reduction of the throttling cross section F_D. As the throttling cross section narrowed, the efficiency factor of the diffusor-compression rose to 0.72. In every experiment, the flow at the point of the diffusor was checked through Schlieren photographs and found to be sound. From pressure gauges P_1', P_1'', P_1''', etc., the fact was established that the plane shock wave, which stood far to the rear of the measurement section in the first few experiments, gradually moved forward until it stood close to the narrowest point of the supersonic diffusor.

In the next experiment an efficiency factor of 0.74 was obtained. When throttling down a little more, a complete change in the flow pattern became outwardly apparent through a roaring, howling noise at the model. The Schlieren photographs used as a check showed alternately pictures of sound flow with oblique shock waves at the diffusor spike, and the deflection angles, which corresponded to the previously calculated values, and pictures of a disturbed flow with a plane shock wave in front of the model and a surging back current of air coming out of the model and moving upstream in the tunnel. At the same time the efficiency factor dropped to 0.23 and less. The instruments in the measurement section P_2, P_2 showed strong fluctuations and did not come to rest. The flow dropped considerably. With a further throttling, the flow dropped further, the efficiency factor remained poor, and the flow was still disturbed.

Figure 22 Efficiency Factor ϑ and Flow G of a Four-Shock Supersonic Diffusor Depending on Throttle Orifice

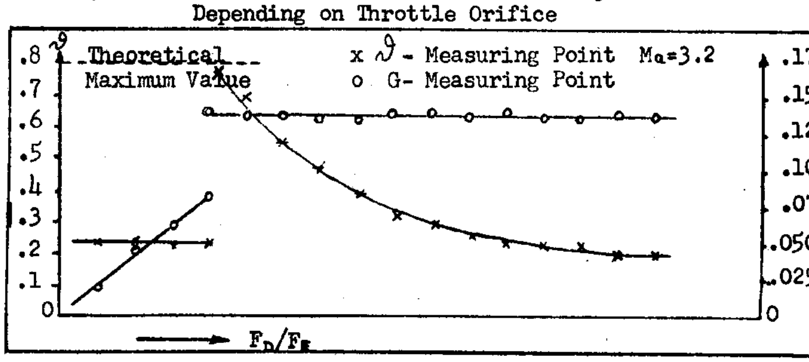


Figure 23 Schlieren Photograph of the Undisturbed Flow at the Diffusor Spike

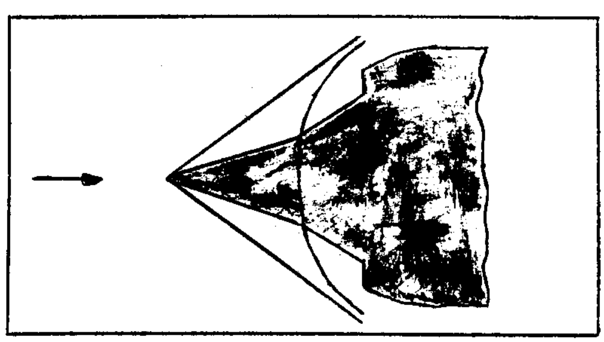
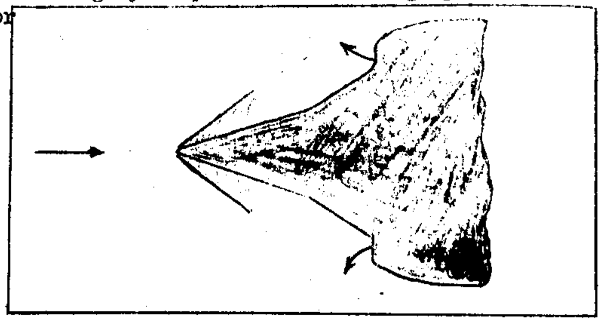


Figure 24 Schlieren Photograph of the Disturbed Flow at the Diffusor Spike with Back Current of Highly Compressed Air Moving Upstream out of the inside of the Diffusor



The following are interpretations of the results of the experiments:

a. The diffusor reaches its maximum compression value only if a correctly set throttle position regulates the flow to the subsonic section.

b. The flow becomes more and more unstable in the vicinity of the theoretical maximum values of compression.

c. If the throttle is set at maximum compression, the last shock wave, which terminates the subsonic section of the diffusor, stands so close behind the narrowest portion of the supersonic diffusor that it can move quickly upstream and stand in front of the diffusor. The resultant disturbed diffusor flow causes the diffusor compression to drop considerably. The pressure inside the diffusor is now higher than in front of the diffusor; the air cannot flow through the throttle fast enough and flows forward against the stream in a surge out of the diffusor. This surge of air completely destroys the flow pattern in front of the diffusor and further decreases its compressibility. The flow of air out of the diffusor continues until the pressure inside the diffusor has decreased to the point where the normal undisturbed diffusor flow can restore itself. The pressure within the diffusor again rises to a critical point. The plane shock wave moves rapidly out of the interior of the diffusor and upstream to the position in front of the narrowest part. The process repeats itself.

This change from diffusor flow to backward surge takes place with a frequency which depends on the Mach number and the throttle setting of the diffusor. In especially small two-dimensional three-shock diffusors frequencies up to 40,000 oscillations per second were measured (ultrasonic pipe).

The two-dimensional diffusors show similar instability in the vicinity of the highest theoretically possible compression. Their sensitivity to disturbance is even greater than that of the three-dimensional rotation symmetrical diffusors. The boundary layer in the two-dimensional diffusor is a source of disturbance. The decelerated material of the boundary layer in a multiple-shock diffusor is not able to overcome the pressure jump to the second or third shock wave, but stands in front of the other shock and disturbs the flow. Klaus Oswatitsch in 1942 showed that a slight boundary-layer control in front of a shock always removes completely the special sensitivity to disturbance of the two-dimensional multiple-shock diffusor.

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The factors which make a diffusor vulnerable to disturbance are the high shock number, irregularities of approach flow in time and space, operation of the diffusor too close to the theoretically maximum compression, poorly machined surfaces on the diffusors, dull spikes and cutting edges, and oblique approach flow. The vulnerability to disturbance of a diffusor flow decreases upon operation of the diffusor below the theoretical maximum values of compression, with decreased shock number, with boundary-layer control, with sound, parallel, stationary approach flow, and with careful mechanical treatment of the diffusor.

The design of a diffusor, especially in regard to limits of stability, is always a matter of research. Only a practical test with a model can answer the question of how far the compression can be increased in each case without the process becoming unstable. Years of experience were necessary for a designer to learn what liberties he may take with the compression factor without fear of instability.

Summary

In the multiple-shock supersonic diffusor a method is given of transforming the energy of velocity of gases flowing at supersonic velocity into pressure with a high degree of efficiency. The degree of efficiency of this pressure transformation is limited only by the more or less great sensitivity to disturbance and instability of the flow. This limit of the efficiency factor, in each case, can be determined on the basis of experience and test-stand experiments. There are also additional methods of decreasing the sensitivity to disturbance even with a high efficiency of compression. The theoretical estimated performance of the multiple-shock supersonic diffusor has been confirmed by years of actual experiment.

Multiple-Shock Supersonic Diffusors in Aerodynamic Engines

The invention of the multiple-shock supersonic diffusor must now be put to use in such a way as to test its technical and economic advantages when installed in aerodynamic engines. First, the general basic advantages of its use in aerodynamic engines, and what alterations in the design of such engines are necessary, must be presented. Up until now the designer of such engines had to attempt to avoid high relative velocities of the medium, which limited greatly the freedom of the designing engineer. The extent of this limitation can only be realized by a comparison between the older engines with low relative velocities of medium and the newer engines with maximum velocities of medium. Furthermore, with the introduction of the multiple-shock supersonic diffusor a whole series of aerodynamic engines could be built economically, which before had been too costly or technically unfeasible.

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In the case of compressors, the peripheral velocity u_2 cannot be increased arbitrarily at the widest diameter of the rotor, because the stress of the material increases through centrifugal force approximately with the square of the peripheral velocity.

With the materials on hand at the present time, the maximum peripheral velocities should not be over 520 $m.s^{-1}$, in the case of mine compressors not over 360 $m.s^{-1}$. The amount of $u_2^2 - u_o^2$ therefore has an upper limit. The amount of $v_2^2 - v_o^2$ is generally not very large. The value of w_2^2 is decisive for the efficiency, since w_o^2 usually remains small compared to w_2^2.

Up until now in radial gas-turbine compressors in which the question of maximum efficiency in one stage is important, w_2 was limited by the sonic velocity which is about 400 $m.s^{-1}$ in the air behind the compressor as a result of the previous warming.

In the case of other compressors w_2 was limited by the fact that the poor efficiency in the subsonic diffusor attached to the rotor prevented a high w_2, and the compression of the fixed diffusor was displaced into the rotor. If these limitations are removed, and if the diffusor is capable of handling maximum exit velocities w_2, the rotor efficiency per stage can be increased, both in the case of axial compressors and in the case of radial compressors. For example, the maximum compression in the case of single-stage axial compressors built in 1949 for gas turbines is $P_5 / P_o = 4$ to a maximum of 6, with a peripheral velocity of the rotor of approximately 520 $m.s^{-1}$.

If the rotor is built for maximum exit velocity (blades bent forward) with the same peripheral velocity, a compression of $P_5 / P_o = 20$ can be obtained with an exit velocity of $w_2 = 935$ $m.s^{-1}$ in the diffusor. It was further determined through experimentation that compressions up to 23 could be reached in single-stage axial compressors without increasing the peripheral velocity of the rotor.

Almost all aerodynamic engines allow an increase of efficiency with constant peripheral velocity, if the limitations on the maximum velocity of the medium, conditioned by sonic velocity on the one hand and by the inadmissibility of high compressions in the subsonic diffusors on the other hand are eliminated.

History of the Lorin Engine

The Lorin engine is an example of the second possibility afforded by the invention of the multiple-shock supersonic diffusor, that of developing completely new types of aerodynamic engines. The Lorin engine was presented in principle in 1908 by the Frenchman, Pierre Lorin. [redacted]

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Theory of the Lorin Engine

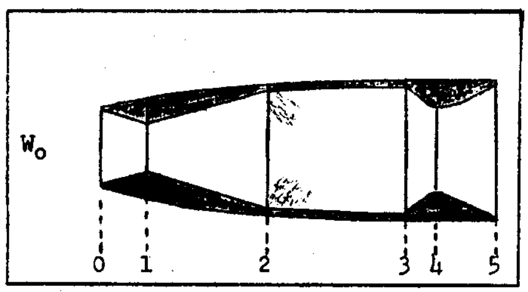
In order to get a view of the magnitude of the Lorin effect, the following simplified assumptions are made:

a. All processes of compression and expansion are adiabatic ("adiabatic Lorin engine").

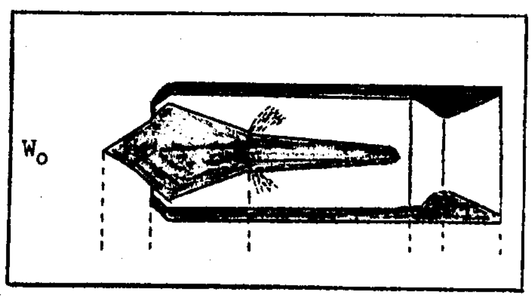
b. The gas constant $\underline{R}$ and the ratio of the specific heat of the medium $\underline{\eta}$ remain constant at all temperatures and at all pressures. (The actual change in the gas constant $\underline{R}$ in the investigated temperature and pressure range, even in the case of combustion, amounted to 2.5 percent at the highest. The change of $\underline{\eta}$ is greater; in the investigated ranges $\underline{\eta}$ changed from 1.28 to 1.4.)

On the basis of these simplified assumptions, the process in the Lorin engine is as follows: A tubular model open at both ends is subjected to an approach flow of air at a high velocity w_o.

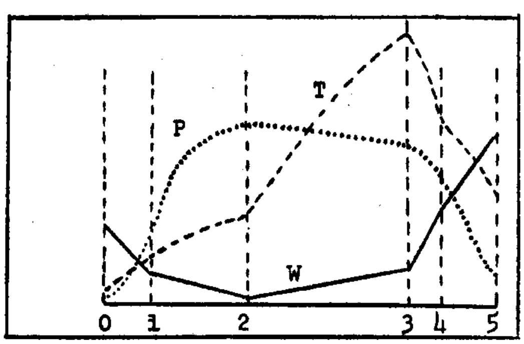
Figure 25 Lorin Engine



a. Tubular Type



b. Circular Type



c. Variables of State

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From 0 to 1 in the supersonic diffusor, and from 1 to 2 in the subsonic diffusor, the air is decelerated adiabatically to negligible velocity w_2. ($w_2 \approx 0$) There is an increase of both temperature and pressure of the medium.

If C_o is the square of the Mach number of the flowing air at 0, then the temperature of the air at 2 is $T_2 = T_o \left(\frac{\eta - 1}{2} C_o + 1\right)$ and the pressure of the air at 2 is $P_2 = P_o \left(\frac{\eta - 1}{2} C_o + 1\right)^{\frac{\eta}{\eta - 1}}$.

Heat is supplied to the air at 2 (injected fuel is burned). The heat supply is ended at 3. Between 3 and 5 in the Laval nozzle the medium expands to pressure P_5 ($P_5 = P_o$); the temperature drops to T_5, and the velocity of the medium rises to w_5. Through the process a thrust against the approach flow of air is exerted on the engine. 25X1

η' represents for each Mach number the maximum value of the attainable efficiency of the Lorin engine. This is shown in Enclosure 3. In Enclosure 4 the efficiency factor η of the Lorin engine is represented as depending on the Mach number and on the heat supply (limited by material reasons) in the Lorin engine. From Enclosure 4 it is clear that the low efficiency factor of the Lorin engine at low Mach numbers precludes the use of this type engine for low flight velocities. For greater flight velocities the Lorin engine is superior to all other types because of its light weight and simple construction.

Although the Lorin engine (ram-jet engine) in practice does not attain the efficiency of the theoretical adiabatic ram-jet engine, a summary and a survey of the properties of the ideal adiabatic Lorin engine are presented. Practice has shown that these formulas, when used correctly, offer the first foothold for the presentation of a new design. For that reason these formulas are summarized in the following:

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of the gas would be a little higher than the outside pressure. Cross sections F_3 and F_2 are disregarded. If the expansion ratios are too great, with the result that the velocity W_2 and W_3 remain below 70 m/s and 100 m/s respectively, the loss in thrust and efficiency as well as the combustion difficulties remain insignificant. F_4 and F_1 could be kept in correct ratio to one another by means of a correct quantitative regulation of the fuel. The entire experiment could be arranged in such a way that, with a fixed F_4, variable intake cross sections could be computed and set so that an approximately optimal performance of the engine is attained over the entire range of Mach numbers.

Ram-Jet Engines (Tr-Géschosse) with Variable Intake Cross Section

A detailed analysis of the process showed that a proportionately larger intake cross section was necessary with increasing flight velocities. The example of C3 in Enclosure 6 shows the imperically determined pattern which the intake cross section follows with the increasing Mach number. Such a change of intake cross section by means of moveable, remote-controlled parts in operation would involve considerable, exorbitant technical work and expense. It was necessary to find a mechanism which would make this increase of F_o possible without moveable parts. Such a mechanism may be explained by means of a simple example: Figure 26 shows the axial approach flow of a two-dimensional circular cone combined with an input edge S at different Mach numbers. From the edge S, whose geometrical position to the cone determines the amount of air intake, the streamlines leading to the points of the edge S are traced back to their position at the point of the cone, in an example of a low and a high Mach number. On a plane of projection situated perpendicular to the axis of the cone through the point of the cone, the streamlines traced back from the edge S always describe a circle whose cross-sectional area is the actual intake cross section F_o. It is also evident that this actual intake cross section increases with an increased Mach number. The curves which represent the increase of F_o with an increased Mach number, are different for different positions of the point S. This condition determined on a simple straight circular cone can be established also in the case of combinations of different circular cones. If one wants to determine the correct geometric position of the diffusor (position of the cones to one another and the edge S) for an engine in the prescribed Mach range, one must construct the streamlines for every geometric position at various Mach numbers and plot the graphically discovered dependence of Mach number and intake cross section F_o. At the same time one can observe the efficiency and determine the curve of the desired ratio of F_o to the Mach number. The construction is always based on the geometric position which is closest to the desired position, whereby the F_o-values must always lie a little under, and never above, the desired F_o-values. If the diffusor conveys too much air into the combustion chamber, the combustion gases can not any longer pass through the critical cross section F_4, if the already mentioned maximum pressure is reached. A back flow, which increases periodically in intensity, moves through the diffusor and impairs the efficiency of the diffusor to such an extent that the thrust drops to a fraction of its optimal value. In experiments with free flying models the drop in thrust is accompanied by a characteristic noise which sounds much like the howl of racing cars equipped with superchargers. The correct design and choice of point S and the necessary constructional drawings of the flow patterns requires three months' work of six people, which is a great expenditure of time, labor, and money. This great expenditure is justified, however, by the solution which can be had by adapting the engine to the optimal operational conditions without moving parts for any given time, on a purely aerodynamic basis. In the course of time a designer would probably develop the ability to judge the most favorable conditions at a glance.

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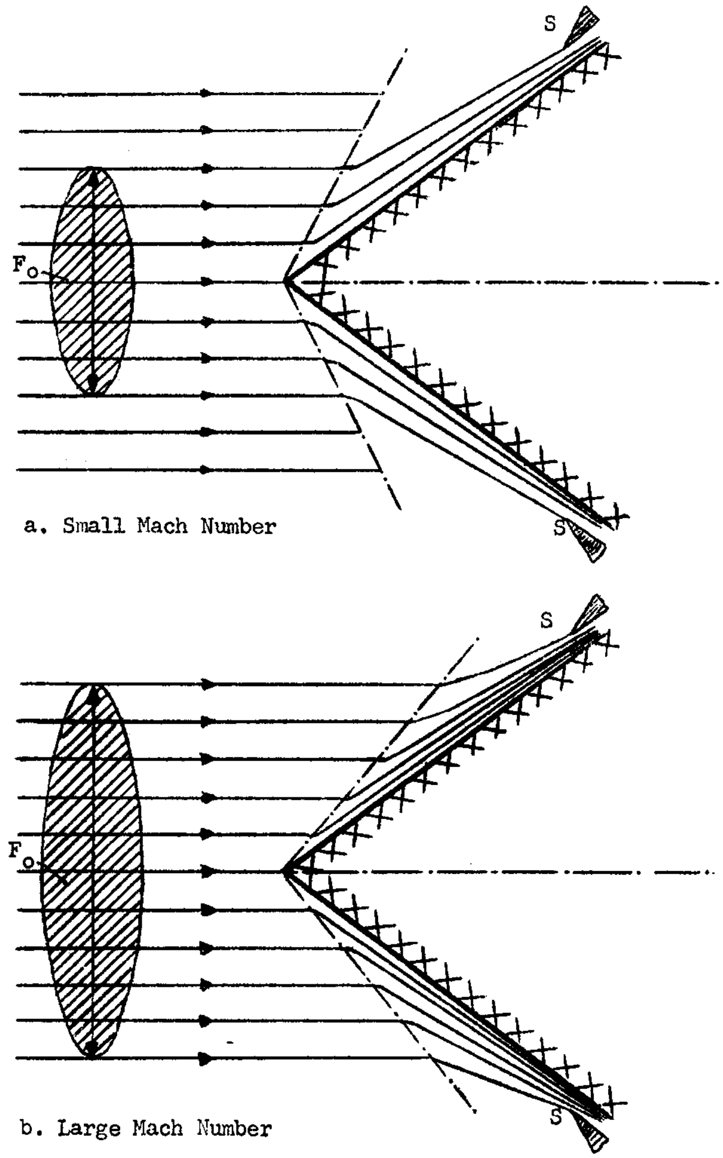
Figure 26 Axial Supersonic Flow at a Three-Dimensional Two-Shock Diffusor at Different Mach Numbers and Increase of Intake Cross Section F_o with an Increasing Mach Number



Thermodynamic Calculation of the Propulsive Process

The calculation of the thermodynamic process in the combustion chamber can begin, if the diffusor conditions are established, and if for the entire range of Mach numbers, the course of F_o (intake cross section), w_o (approach flow velocity), ρ_o (air density in front of the engine), T_o (initial temperature), P_o (air pressure in front of the engine), and η (efficiency of the diffusor) are given, from which are derived $M = F_o \cdot \rho_o \cdot w_o$ (the weight of the air mass supplied per second), T_2 (temperature of the air behind the diffusor), and P_2 (pressure behind the diffusor). This calculation is not as exact as the calculations for the diffusor and must be continuously checked by combustion tests. For various amounts of fuel (change of ratio of amount of fuel to amount of air supplied per second) the combustion temperature and the composition of the power-plant gases are determined. The calculated values are corrected through experiments on the amount of dissociation, of heat loss through radiation, and of the dissipation of heat through the heat conductance of the wall of the combustion chamber. For a consideration of the expansion of the power-plant gases, the dependence of the ratio of specific heat (κ) on temperature and pressure is calculated. With that the situation is established at point 3, after the termination of the combustion. The expansion of this gas mixture is pursued theoretically up to point 4 (critical, narrowest cross section of the Laval nozzle) and up to point 5 (exit of the gases out of the end cross section F_5 of the Laval nozzle). The velocity F_5 and the remaining variables of state at 5 determine the calculated thrust. 25X1

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G_B = fuel weight per second.

H_u = net calorific power.

A = mechanical heat equivalent.

The efficiency factor here is of theoretical interest only and can be used as an interesting comparative value to other engines. For the purpose of engine design, another efficiency factor η_b is defined as $\eta_b = \frac{S - W}{G_B \cdot H_u \cdot A}$

whereby W is the external air resistance and is calculated as
$W = \frac{\rho_o}{2} \cdot w_o^2 \cdot c_w \cdot (F - F_o)$

c_w = coefficient of resistance of the external resistance of the engine. For the engine to be designed, let F_5, F and c_w be given; then all characteristic values at (5), at (4), the thrust S, acceleration force B = S - W, and η_B can be calculated for freely chosen parameter ξ and $\frac{F_o}{F}$ (always smaller than 1.0).

The result of such a calculation is represented in Enclosure 14, whereby points of equal ξ are joined by curves. This representation gives the key to the correct design of all cross sections of a ram-jet engine. The individual η_B curves are combined by a common envelopment. Where this envelopment indicates a maximum value for η_B is the most favorable operational state for the engine. The values of F_o and ξ belonging to this maximum of the envelopment are considered the most favorable values.

If an engine is to operate satisfactorily beyond a certain range of Mach numbers, this construction should be repeated for an analogously selected sequence of Mach numbers of this Mach-number range. The diffusor is to be designed in such a way that in the total Mach number range this F_o-value is approximately reached. At the same time the required narrowest (critical) cross sections F_4 of the Laval nozzle should be represented graphically for all these Mach numbers, F_o and ξ values. Since F_4 is not variable, F_o and ξ must be balanced with the constant F_4 for a simultaneous most favorable efficiency factor at any given time.

Since the additional constructions of the streamlines for different geometrical positions of the diffusor must be made, and since additional corrections must be made on this coarse calculation for the dissociation, the friction in the combustion chamber, and for the final w_3, several months of computation are necessary for the accurate design of an optimal ram-jet engine.

The most important operational conditions must be checked through experiments on models. These involve only minor changes in the computed values. This calculation represents only the beginning. The following steps are to be taken: With the above given formulas the approximate position of the most favorable parameter is determined. In the vicinity of the most favorable point all calculations for more closely chosen Mach- and ξ-values are repeated and the following corrections made:

Pressure loss because of friction in the combustion chamber,
Heat loss by means of heat conductance at the walls of the combustion chamber,
Heat loss through dissociation,
Heat loss through premature completion of the combustion process (The short durations in the combustion chamber do not allow the reactions to be completed, especially in the case of higher hydrocarbons. Considerable portions of unburned gases are always observed along with free oxygen in the combustion gases.),
Heat loss in the nozzle through the relaxation-period of the degrees of freedom. In the case of smaller engines, the duration in the nozzle can amount to less than 10^{-4} seconds.
Loss through friction within the nozzle,
Loss through non-parallel jet expulsion from the nozzle.

The results so obtained must be tested on the test stand. With logical and rational treatment of the corrections, the test-stand experiment will coincide with the corrected calculations.

Fuels for Ram-Jet Engines

A whole series of fuels have been discussed and tested for ram-jet engines operating at high Mach numbers. The most favorable fuel has not yet been agreed upon; experience has helped in selecting the more favorable fuels, however.

In many cases a high calorific value of the chosen fuel is of importance. The calorific value is not the only determinative factor however; other factors can often be of greater importance for actual operation. In all ram-jet engines, up until now, the fuel has been stored in the engine itself. The overall size of the engine must be kept as small as possible for propulsion at high Mach numbers; the space required for the fuel is, therefore, of prime importance. Indeed, the space required by the fuel is even more important than the weight of the fuel, for all fuels tried so far have amounted to only six percent to 12 percent of the total weight. A decrease in the weight of fuel does not mean much in the total efficiency of the engine. A saving in space is extremely important in the total efficiency of the engine; indeed, the space required for fuel limits the efficiency of the engine and determines the size of the pay load for a missile propelled with a ram-jet engine. The most favorable choice of fuel is, therefore, not the fuel with the highest calorific value per unit of weight, but the fuel with the highest calorific value per unit of volume. The generally used gasolines of the hydrocarbon series from about hexane on up are unfit for other reasons. Indeed, the calorific value of hexane, 10,630 Kcal/kg, is favorable in relation to the unit of weight. The low specific weight of hexane, 0.66 kg/l, produces only 7016 Kcal/l. Tetralin produces a little more than 9500 Kcal/l, and even some medium diesel oils are favorable with 9200 Kcal/l. Carbon disulphide, with only 3404 Kcal/kg, has as a consequence of its specific weight of 1263 kg/l a calorific value of 4300 Kcal/l and is, because of its other favorable properties, above all because of its splendid ignitibility, a fuel which must be seriously considered.

The desired boiling point of a liquid fuel is not under 40°C and not over 300°C. The fusion point should be not over -30°C. The heat of vaporization should be as low as possible. The spontaneous-ignition temperature of the fuel should be as low as possible. The ignition process in the combustion chamber would then take place as follows: Even in the high air layers there are in the combustion chamber behind the diffusor, at high Mach numbers (over 2.0), air temperatures which are high enough to cause a spontaneous ignition of the injected fuel. All the difficulties of the applied ignition are avoided in the self-ignition of the fuel. It is especially important that an applied ignition always requires a certain amount of time for the ignition process to progress through the entire mixture. With the very short time which is available for the mixture of fuel and air to mix in the combustion chamber (mostly only a few thousandths of a second), any type of applied ignition will require an intolerable increase in the volume of the combustion chamber. The self-ignition process can, on the other hand, take place in a minimum of space. The fuel is injected at high pressure through several fine fuel nozzles and is thus immediately prepared for ignition. Ignition and combustion occur immediately at the point of injection. The entire volume of the combustion chamber, and therewith the entire duration, is available for the combustion reaction. The saving in combustion-chamber space under these conditions can be as high as sixty to eighty percent. Reports on tests on chemical compounds for suitable ignitibility, which have appeared in the literature up until now, cannot be used. The apparatus used for such tests on self-ignition were designed and operated without consideration of the time factor. In such an apparatus a drop of fuel is introduced into heated air. A self-ignition temperature is, in such a case, that temperature at which the first ignition of the introduced drop can be observed. The fact that an appreciable time passes from the introduction of the drop of fuel until the appearance of the first combustion reaction is not considered at all in the case of such an apparatus. In the fast ram-jet engine, however, the ignitible mixture may have already passed the Laval nozzle and have passed out into the open before the ignition process has taken place. In suitable apparatus, therefore, the chosen fuels were tested for their ignition qualities during a very short period of mixing of air and fuel in the combustion chamber. The results of such tests were quite paradoxical. For example, the compound zinc-ethyl normally has a self-ignition temperature which lies far below the freezing point. A drop of zinc-ethyl ignites spontaneously upon contact with even extremely cold and even thin air. In the ram-jet engine at air temperatures of over 500°C and pressures over 10 atmospheres zinc-ethyl did not ignite, that is, not within the period of three-thousandths of a second which was available for ignition. Zinc-ethyl could, therefore, not be used as a fuel for ram-jet engines. Carbon disulphide proved to be the most readily ignitible fuel and was, therefore, alone or mixed with medium diesel oils, the only universally applicable fuel for ram-jets. As far as the extraordinarily difficult experimental techniques with acetylene permitted the use of this extremely pressure-sensitive substance (mostly dissolved in other fuel components), acetylene proved to be just as readily ignitible. The experiments with ethylene dichloride showed promise at the beginning but have not yet been completed. There were also tests made on ethyl ether and a series of medium diesel oils (solar oil), which are obtained from coal-tar distillation. Even butadiene and its derivatives, ethylene, and others showed promise. Benzol and other saturated aromatic compounds proved to be especially poor.

The combustion velocity of fuels was likewise of extreme importance in the operation of small fast ram-jet engines. The concept of combustion velocity is not simple to define and must be determined for every individual ram-jet engine. Air temperature and air pressure serve as initial values in the determination, (for example, 500°C and 20 atm). If this air is led past a fuel injection position at a velocity of about 60 or 80 meters per second and observed on its course through a tube which acts as a combustion chamber, the moment of ignition and the course of the combustion can be observed through quartz windows by means of a spectroscope. The full course of the combustion reaction up to the final stable end products requires a considerable time, which is represented in this type of experiment by a distance along the axis of the tube which is used as a combustion chamber. In such an arrangement six centimeters of tube-length correspond to about one meter per second.

The time required for 80 percent, 90 percent, or 95 percent of the total heat of combustion to be released is chosen as a measure for the combustion velocity. Let us designate these combustion velocities as 80 percent-combustion-velocity, 90 percent-combustion-velocity, etc. A 100 percent-combustion-velocity has so far never been reached in ram-jet tests. Experiments have shown that, even after 100 meters per second, final reactions take place. For the smallest engines with very short combustion periods 80 percent combustion was established as a requirement; for larger ones the requirement was 90 percent, and for very large ones (A_4) 95 percent was required. The necessary combustion velocity for each fuel was determined. With this definition, there is no distinction between ignition, velocity, and combustion velocity, and there should not be any distinction. The main idea is, once more, to attain small combustion chambers through high combustion velocities. The size of the fuel molecule and the strength of the bond within the molecule determine the combustion velocity, if other conditions (variables of state of the supplied air, combustion-chamber temperature, etc.) are equal; small molecules produce high combustion velocities, and very large and stable molecules produce low combustion velocities. The experimental series was as follows: hydrogen gas, carbon disulphide, acetylene, ethylene, various diesel oils, mostly with unsaturated bonds, methane, tetralin, and saturated aliphatic and aromatic hydrocarbons. The experiments need to be continued. In comparative experiments, the shape, as well as the size of the combustion chamber should always be the same, because the shape of the combustion chamber can influence the conditions of ignition and the combustion velocity. The variables of state of the supplied air should also be equal in comparative tests, otherwise the data obtained will not be reliable.

A fuel should be easy to handle, to pour, and to measure; it should not be too readily combustible, and should not endanger the health of the men who must handle it. The otherwise so promising acetylene is so difficult to handle, that it can only be used in small amounts dissolved in other fuels. Acetylene dichloride cannot be used because it is extremely poisonous. Another excellent fuel, carbon disulphide, is an extraordinarily unpleasant, poisonous, and inflammable material. The diesel oils and tetralin are especially easy to handle.

The fuel must not, either alone or in combination with the elements of the atmosphere, attack the inner walls of the fuel containers too violently. Unfortunately, the otherwise so excellent carbon disulphide forms metallic compounds of thiocyanic acid through reaction with the iron of the inner wall of the fuel container, the air, and the water vapor in the air. A container which contains carbon disulphide corrodes very rapidly. Acetylene dichloride is especially dangerous. The diesel oils are completely harmless.

The fuel or the fuel mixture must not have the tendency to dissociate readily but should be reasonably stable. The tendency of otherwise readily and quickly burning unsaturated hydrocarbons toward saponification and resinification often causes stoppages of the nozzles.

Some of the most promising fuels can be detonated by a shock. Acetylene, which cannot be stored under pressure, is one of the most dangerous in this respect. An accident led to the discovery that a sufficiently strong initial shock will cause an exothermic disintegration of carbon disulphide. Caution should be exercised in the handling of fuels. Carbon disulphide is not dangerous when mixed with diesel oils.

Price and technical possibility of manufacture will play an important role in the selection of fuel in the future. Diesel oils mixed with carbon disulphide, and tetralin and dekalin for very fast engines (Mach numbers greater than 3.0) represent cheap fuels which are available in great quantity.

To get comparative data on various fuels, a card index was established for fuels and mixtures of fuels. This card index showed the behavior of the fuels and mixtures on the basis of the conditions given above. About 2600 tests on fuels and mixtures have been carried out so far. Other test series on corrosion, self-ignition, etc., were conducted. Although a fuel which would be ideal for all purposes has not yet been found, the card index will give the best fuel for any particular use.

For small Mach numbers and small engines, in which ignitibility and combustion velocity determine the size of the combustion chamber, carbon disulphide is the best fuel, in spite of its extremely unpleasant characteristics. The self-ignition of injected carbon disulphide will take place even at Mach numbers in the vicinity of 2.0, down to 1.5 at low altitudes, if the combustion chamber has the correct shape. A mixture of carbon disulphide and diesel oils is best used for Mach numbers from 2.0 to 3.0. For Mach numbers above 3.5, and for larger engines, only tetralin and dekalin, without addition of carbon disulphide, can be used. All of the collected test data needs to be checked by new experiments. Perhaps newer metallic organic compounds can bring surprises and improve the performance of ram-jet engines.

Individual Parts of the Ram-Jet Engine

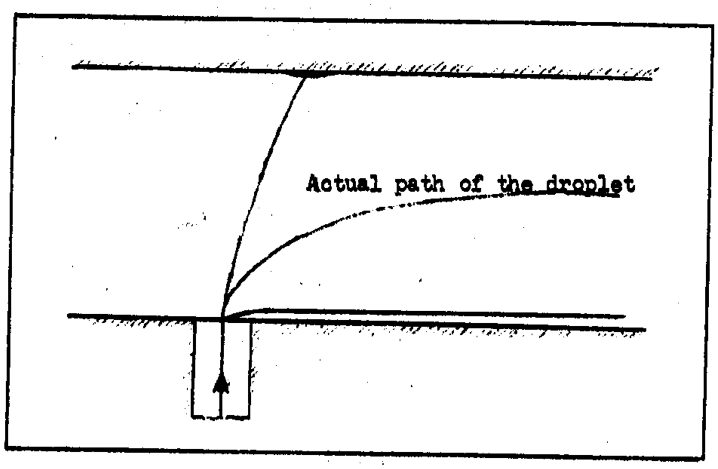
It has already been pointed out in the discussions of ignition and combustion that the geometrical proportions of the combustion chamber are significant. The tests on fuels should not be separated from the tests on the optimal shape and size of the combustion chamber. The first factor in the design of the combustion chamber is the absolute value of the chamber, the volume. The specific volume of the combustion chamber ($\frac{\text{volume of chamber}}{\text{amount of fuel consumed per second}}$) can be reduced with increased Mach number and improved efficiency. Too small a combustion chamber will prevent the complete course of the combustion reaction; too large a combustion chamber makes an almost complete combustion possible but produces increased heat losses at the walls of the combustion chamber and through frictional resistance within the combustion chamber. When the gas velocity in the combustion chamber remains below 100 meters per second, it makes no difference what the ratio of combustion-chamber length to combustion-chamber cross section is. A short, wide combustion chamber is just as good as a long, narrow combustion chamber, if the volume is the same in both. The short combustion chamber with low gas velocity is only a little better than the long, thin combustion chamber, when the average combustion-chamber velocity does not exceed 100 meters per second. The designer, therefore, has an almost free hand in the choice of the shape of the combustion chamber, as far as the ratio of length to cross section is concerned.

Since the temperatures of the gases in complete combustion sometimes are quite high (up to 1700°C), it is important that the combustion chamber be constructed in such a way that the actual flames make the least possible contact with the walls of the combustion chamber. For example, a cylindrical combustion chamber can be designed in such a way that the flames fill only the center area of the combustion chamber and reach the wall of the combustion chamber only toward the end shortly before entering the Laval nozzle. This reduces heat losses and precludes many of the difficulties of cooling the wall of the combustion chamber. In a great many engines the combustion chamber was circular in shape. The wall of the fuel container was cooled by the fuel itself. The outer wall of the combustion chamber was cooled in all cases by the air streaming by in flight. The fact that the surface friction of hot walls is greater than that of cold walls had to be taken into consideration, of course. The streamlines in the area of the combustion chamber should not be smooth and turbulence free. Rather large areas of still air and turbulence should be provided within the combustion chamber so that a rich and concentrated mixture of fuel and air can be had. These still-air and turbulent areas facilitate the combustion reaction. For example, the ram-jet missiles of the E-series, E_6 and E_7 , had a smooth construction of the combustion chamber which was favorable for aerodynamic reasons; but the combustion in these engines was so poor and irregular that the missiles were a complete failure. The confused, and unfavorable, aerodynamic construction of the engine E_5, on the other hand, produced excellent combustion conditions. Details of this geometric construction of the combustion chambers are considered in the discussion of the individual engines. No general principles for the shaping of the combustion chamber could be considered binding; the performance of the engine had to be established by tests in every single case.

The choice of injection nozzle and injection process is important in the effort to keep the combustion chamber as small as possible. In the combustion chambers of gas turbine engines and in the combustion chambers of ram-jet engines with small Mach numbers (under 1 Leduc, Schmidt-tube) only a few injection nozzles are used. The thorough mixture of fuel and air is left up to the local turbulence, which requires considerable space. The need for economy of space in rapid ram-jets makes other methods necessary. A great number of small injection nozzles should be arranged so that the mixture will be prepared in the shortest time and in the smallest area. The number of necessary injection positions varies from 72 to 480.

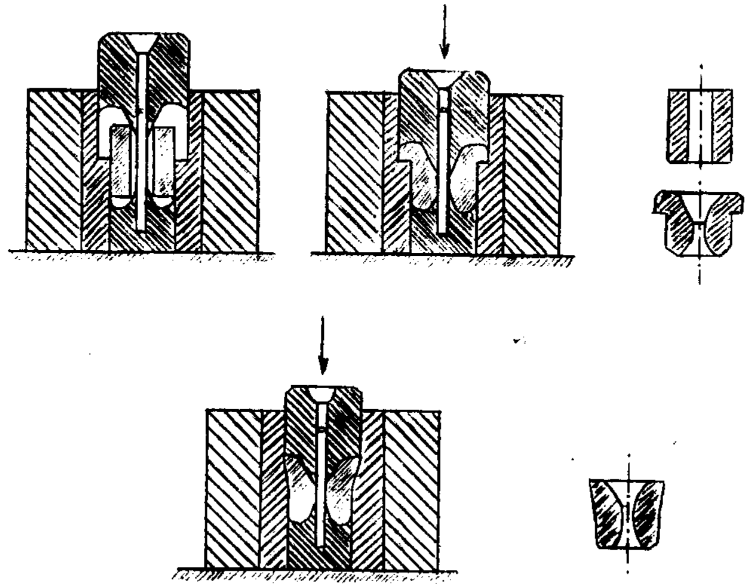
The direction of fuel injection is either perpendicular to the direction of air flow or slightly against the air flow (up to 30°). The size of the injected droplets determined by diameter of the nozzles, and the velocity of the droplets, determined by the injection pressure, were adjusted so that the droplets reached the optimal depth in the air stream. Such an adjustment could only be made on the basis of actual experimentation. In all cases the size and velocity of the drops had to be small and low enough to prevent the droplets from reaching the opposite wall of the chamber. Too small drops and too low velocity, on the other hand, produced an insufficient preparation of the fuel-air mixture. The average penetration of the droplets of fuel into the combustion chamber was set at a point where there would be an excess of fuel of about 10 to 20 percent.

Figure 27



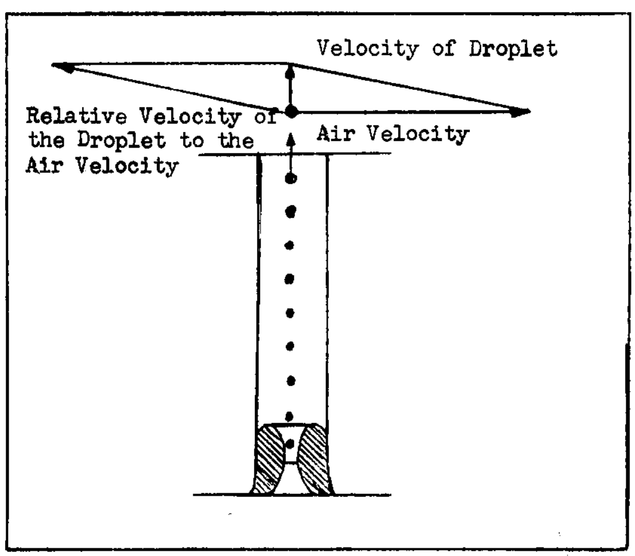
The great number of injection nozzles to be used in a single ram-jet engine necessitated a cheap method of producing them. The best method of manufacturing the nozzles is by pressing small conical bored pieces of about 2.00 mm outer diameter and about 0.15 to 0.35 mm inner diameter at the narrowest point of the boring over a mold. Annealed copper and rough-pressed circular sheets of ceramic soft iron are the best material. The unfinished piece in the shape of a small hollow cylinder is pressed in a tool in such a way that the boring is kept open by means of a short thin steel wire. (See Figure 28.) After pressing, the pieces show a surprisingly good dimensional stability, especially in the narrow boring. The size of the boring is checked in a flow meter. The nozzles are then inserted into holes of 2.00 mm inner diameter in the wall of the fuel container in such a way that the nozzle is positioned at the bottom of a "flash hole" which is about 8,00 or 10.00 mm deep. (See Figure 29.)

Figure 28



This flash hole, which is filled with a very unctuous mixture of ignitor liquid and air, was very important for producing a satisfactory self-ignition of the fuel. The flash hole should be made deep enough to guarantee that the droplet will in every case be burning when it leaves the flash hole.

Figure 29



The absolute value of this narrow zone has an influence on the process of self-ignition in the engine. If it is assumed that there is a very small zone where self-ignition is guaranteed by the correct concentration of both components and a sufficiently high combustion-chamber temperature, a certain finite number of molecular collisions which lead to a reaction can be determined for every cubic millimeter of this area for a small unit of time (10^{-5}). The amount of energy released by an effective molecular collision and the thereby occurring chemical reaction can either be radiated outward through heat conductance (impulse radiation outward) or remain within the zone of the mixing area and there increase the temperature and thus improve the conditions of ignition for subsequent reactions. It is obvious that in a very small zone of the mixing area the amount of energy which escapes the zone through heat- or impulse radiation is greater than in a larger zone of mixing area. To be sure, there will be a succession of effective molecular collisions in the small zone, and individual reactions will occur, but the temperature of the zone will not rise significantly. In the center of a larger zone of the mixing area, on the other hand, there will be an accumulation of heat, and the temperature will increase according to an exponential law. As a consequence, the number of effective collisions per unit of time will increase rapidly, and self-ignition will take place.

This train of thought, which was well verified by the experimental behavior of droplets of fuel traveling at various speeds, led to the conclusion that droplets which travel too fast ignite poorly. A certain velocity of fuel droplets cannot be exceeded, if self-ignition is to be maintained.

Conditions are altogether different for a droplet which is already burning. The demand for a high combustion velocity means that the products of combustion on the burning droplet must be removed again and again as quickly as possible and replaced by fresh air.

In order to keep the total area for ignition and combustion small, the droplet of fuel must fly slowly in relation to the surrounding air during the ignition process, and more rapidly after the combustion stage has been reached. This initially slow and subsequently rapid movement of the droplet is determined by the use of the flash hole. (See Figure 29.) In the flash hole the air is to be considered at rest. The relative velocity of the droplet to the air is equal to the actual velocity of the droplet. When the burning droplet leaves the flash hole it strikes a stream of air which is travelling at about 80 meters per second. The relative velocity of the droplet is then equal to the velocity of the air.

These conditions were verified and improved in a series of experiments. The result of these extensive experiments, based on the above considerations, was that the total factor (ignition period plus combustion period) was reduced to values which were lower than had ever before been considered possible. The result was that extremely small combustion chambers were able to be used with success.

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