

INVESTIGATION OF FLOW PATTERN NEAR SUPERSONIC
NOZZLE EXIT SECTION AT DIFFERENT
INCALCULABILITY FACTORS

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The pressure distribution on the nozzle wall near the exit cross section was studied experimentally for different incalculability factors. "Damping lengths" and internal pressure profiles in response to increases in the pressure drop across the shock discontinuity were determined up to a value close to the detachment value, for Mach numbers $M = 2$ to 3.5.

In some instances of investigation of flow patterns through supersonic wind tunnels with an Eiffel chamber, and also in the determination of the thrust characteristics of supersonic nozzles, the external pressure p_2 in the medium into which the nozzle jet is directed can deviate from the static pressure p_∞ of the supersonic stream in either direction. When $p_2 > p_\infty$, the disturbance is transmitted upstream through the subsonic part of the boundary layer, so that compression waves form and bring about an increase in the pressure along the nozzle wall. With increasing incalculability factor $\bar{p}_2 = p_2/p_\infty$, the extent of the region in which the shock interacts with the boundary layer increases and, as shown in [1, 2], the turbulent boundary layer detaches from the nozzle wall at the critical pressure ratio in the shock p_{2cr} . In that case the distance l_0 over which the disturbance is transmitted amounts to something on the order of 10 boundary-layer thicknesses. Information on the flow pattern in the neighborhood of the nozzle exit section and referable to other incalculability factors would also be of interest.

Experimental investigations were conducted at Mach numbers $M = 2.0, 2.5, 3.0,$ and 3.5 . Specially shaped supersonic nozzles with exit section diameter 110 mm were used in order to generate a uniform velocity field. The nonuniformity ratio $\Delta M/M$ of the velocity field was not greater than $\pm 1.5\%$. A diagram of the experimental arrangement appears in Fig. 1.

Abutting against the nozzles was a special attachment 1 featuring drainage ports 2 along the generatrix over a length of 80 mm. The first drainage port counting from the exit section of the spacer was located at a distance of 2 mm. The ports were spaced 2 mm apart over a length of 20 mm from the exit section. The spacing increased to 4-10 mm over the remaining length.

An ejector tube 3, 125 mm in diameter, was adjoined to the attachment in order to generate large pressure drops. Air was supplied to the arrangement from a battery of pressure cylinders, and was exhausted through a suction manifold.

The pressure distribution along the wall of the attachment was measured by a single mercury gage with the several points to be measured being connected to the gage in succession. The parameters of the boundary layer were measured with a total-pressure adapter with a lip height of 0.4 mm, and a static-pressure adapter 1.0 mm in diameter. The displacement of the adapters was monitored by an indicator gage accurate to within 0.01 mm.

Measurements taken in the boundary layer at the exit section of attachment showed that the boundary layer was turbulent in all cases, its thickness δ_∞ being 6.8, 6.9, 7.8, and 8.9 mm for the respective Mach numbers $M = 2.0, 2.5, 3.0,$ and 3.5 . The Reynolds number determined over the thickness of the boundary layer ranged from $3.36 \cdot 10^5$ to $2.6 \cdot 10^5$ for that range of Mach numbers.

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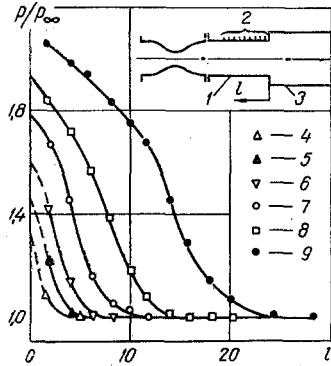


Fig. 1

Fig. 1. Pressure distribution in the zone where the condensation shock interacts with the boundary layer, at different incalculability factors for the Mach number $M = 2.5$ (l , mm): 4) $\bar{p}_2 = 1.67$; 5) 1.57; 6) 1.68; 7) 1.8; 8) 1.93; 9) 2.13.

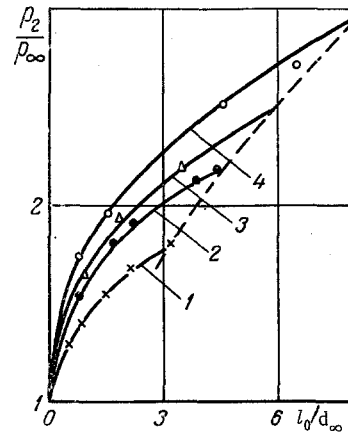


Fig. 2

Fig. 2. Dependence of the distance of the disturbance transfer on the incalculability factor: 1) $M = 2.0$; 2) 2.5; 3) 3.0; 4) 3.5.

Figure 1 also shows the distribution of the static pressure p along the wall of the attachment at different values of \bar{p}_2 for the Mach number $M = 2.5$. As \bar{p}_2 increases, the transfer of pressure is extended increasingly further upstream, i.e., the "damping length" increases. Since the first drainage point lies at distance $l/d_\infty = 0.3$ from the exit section, the change in the pressure distribution is detected only at $\bar{p}_2 \geq 1.2$.

It is clear from the pressure distribution plotted here that the profile of the internal pressure begins to change even when the conditions of outflow of the jet differ only insignificantly from ratings. At pressure drops \bar{p}_2 close to the detachment value, the "damping length" reaches its peak and the pressure on the nozzle wall increases continuously, within that length, from the pressure in the undisturbed stream to the pressure p_2 . A further increase in \bar{p}_2 entails detachment of the boundary layer and an advance of the condensation shock within the nozzle. Consequently, there are no fundamental differences, under reexpansion conditions, in terms of the effect of the incalculability factor on the internal pressure profile, including the state of the boundary layer when close to the detachment state. Only the scale of that effect undergoes a change.

Measurements taken in the boundary layer at $M = 2.5$ showed that the profile of total pressures under goes no changes in the range of \bar{p}_2 from 1 to 1.73, and the curve of the total pressures drops off abruptly in response to a pressure drop close to the detachment value. The static pressure in the direction normal to the nozzle wall varies continuously from the pressure at the wall itself to the pressure p_∞ in the undisturbed zone of the supersonic stream.

Measurements were taken of the velocity profile inside the nozzle at a distance of 1-2 mm from the nozzle exit section, for different values of \bar{p}_2 . The measurements revealed a deformation of the velocity profile observable even at $\bar{p}_2 = 1.26$. With increasing \bar{p}_2 , the velocity profile thins out somewhat, and the subsonic portion of the layer increases appreciably. Processing of the experimental data showed that the exponent n of the turbulent layer varies linearly as a function of the incalculability factor for \bar{p}_2 , and that $n = 7$ at $\bar{p}_2 = 1$, $n = 2$ at $\bar{p}_2 = 2$.

The results of investigations carried out at different Mach numbers and at different incalculability factors are plotted in Fig. 2 in the form of the dependence of the relative damping length $\bar{l}_0 = l_0/d_\infty$ on \bar{p}_2 . The damping length increases abruptly with increasing incalculability factor for a specified Mach number M . The broken curve corresponds to critical pressure drops in the condensation shock.

In the case of critical pressure drops, the length \bar{l}_0 increases with increasing M . For example, $\bar{l}_0 = 3.1$ at $M = 2.0$, and $\bar{l}_0 = 8.0$ at $M = 3.5$. If we consider the effect of the Mach number M on \bar{l}_0 when the value of \bar{p}_2 remains unchanged, we find that the damping length shortens with increasing Mach number.

In the investigation of the thrust characteristics of supersonic nozzles under the relevant incalculability conditions, the profile of the internal pressure acting on the walls of the nozzle in the vicinity of the nozzle exit section will differ from the pressure profile corresponding to the rated outflow of the jet ($p_2 = p_\infty$). If the stream is nonparallel upon exit from the nozzle, then the force being measured will include a component due to the increase in the pressure in the zone where the condensation shock interacts with the boundary layer. The greater the angle of the nozzle wall and the greater the incalculability factor, the larger that component will be. In order to estimate the effect of the incalculability factor on the thrust characteristics of supersonic nozzles, we have to know not only the distance over which the disturbance is transferred but also the pressure distribution near the exit section of the nozzle.

Detailed measurements at different Mach numbers M have made it possible to arrive at an empirical relationship between the variation in the static pressure on the nozzle wall and the value of \bar{p}_2 :

$$\frac{\bar{p} - 1}{\bar{p}_2 - 1} = \left[1 - \left(\frac{\bar{l}}{\bar{l}_0} \right)^{3/2} \right]^3.$$

Here $\bar{l} = l/d_\infty$ is the relative length from the exit section to the section of the nozzle in question. The value \bar{l} varies over the range $0 \leq \bar{l} \leq \bar{l}_0$. When $\bar{l} = 0$, $\bar{p} = \bar{p}_2$, and when $\bar{l} = \bar{l}_0$, $\bar{p} = 1.0$.

The proposed dependence yields excellent agreement between the pressure distribution and the experimental data over the range of Mach numbers $M = 2$ to 3.5 , and incalculability factors \bar{p}_2 from 1.0 to the value $\bar{p}_2 = 0.8 p_{2cr}$.

The effect of the ambient pressure p_2 on the internal pressure profile in the neighborhood of the nozzle exit section can also be manifested in the case where the pressure p_2 is below the pressure p_∞ in the supersonic stream. We might expect that the pressure along the nozzle wall will decrease downstream of some cross section. In order to estimate that influence, experiments were staged at $M = 2$ to 3.5 under conditions where the pressure p_2 was allowed to vary from some least value to the value $p_2 = p_\infty$. The pressure p_2 on the experimental arrangement could be 3-4 times lower than the stream pressure p_∞ . No appreciable change in the pressure distribution could be detected in the experiments over a length $l \geq (0.2 \text{ to } 0.3)d_\infty$. If such an influence is operative, it encompasses only a very small zone ($\bar{l} < 0.2 \text{ to } 0.3$) directly adjacent to the exit section of the nozzle at most.

Consequently, the internal pressure profile remains unchanged only under conditions $p_2 \leq p_\infty$. There is a zone of enhanced pressure on the nozzle wall near the nozzle exit section under reexpansion conditions. The dimensions of that zone and the degree of the pressure increase occurring there are determined by the size of the incalculability factor.

NOTATION

l	is the distance from nozzle exit section to point of interest on nozzle wall;
l_0	is the length of zone within which condensation shock interacts with boundary layer "damping length");
d_∞	is the thickness of boundary layer in undisturbed stream;
p_2	is the ambient pressure;
p_∞	is the static pressure in undisturbed supersonic stream;
p	is the static pressure on nozzle wall;
p_{2cr}	is the critical pressure ratio;
$\bar{p} = p/p_\infty$	is the relative pressure;
M	is the Mach number at nozzle exit section.

LITERATURE CITED

1. G. I. Petrov, V. Ya. Likhushin, I. P. Nekrasov, and L. M. Sorokin, *Trudy TsIAM*, No. 224, Oborongiz (1952).
2. G. Gedde, *Mechanics*, No. 5 [Russian translation], IL (1959).