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CONTROLLING THE LEVEL OF THE SONIC BOOM GENERATED BY A FLYING VEHICLE BY MEANS OF CRYOGENIC FORCING. 1. COOLING OF THE VEHICLE SURFACE

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The possibility of controlling the sonic boom level by means of cooling the surface of a flying vehicle is discussed. The effect of surface cooling on the formation of the perturbed flow structure at large distances from the vehicle is demonstrated by an example of a modified power-law body of revolution. The intensity of the intermediate shock wave and the perturbed pressure pulse near the body are seen to decrease, which expands the altitude range of the region where the sonic boom is reduced (down to 50%). At larger distances from the body, cryogenic forcing ensures a 12% decrease in the bow shock wave intensity. The possibility of controlling the process of formation of wave structures near the surface, such as barrel shock waves, is demonstrated. An explanation of the cryogenic forcing mechanism is offered.

Key words: *supersonic transport, sonic boom, bow shock wave, cryogenic forcing, coolant, flow structure.*

Introduction. Restrictions on the intensity of the sonic boom (SB) generated by a supersonic transport (SST) inspired intense research in the field of SB minimization since the 1960s (see, e.g., [1–7]). The results of these studies based on the search for an optimal distribution of the volume and lift force along the plane failed to ensure an admissible pressure difference of 50 Pa on the bow shock wave (BSW) for a wide class of supersonic planes. For instance, Concorde in the cruising flight induced an excess pressure of more than 100 Pa on the ground level, even though its configuration was close to the optimal one [8]. Investigations of the possibility of designing an SST of the second generation, which were performed at the Central Aerohydrodynamic Institute (TsAGI, Zhukovskii), also show that it is currently impossible to ensure an admissible level of the sonic boom generated by an aircraft with a large take-off weight without deteriorating its performance [9]. The basic methods of reduction of SB intensity with the use of previously obtained solutions of minimization problems within the framework of the linear theory without restrictions on the drag imply realization of the effects of the middle zone of the sonic boom. This middle zone has a specific feature: in the cruising flight regime, the excess pressure profile near the Earth's surface does not acquire an N-shaped form and consists of individual shock waves, compression waves, and rarefaction waves separated in space [4, 10, 11]. An SST configuration with a weight of about 500 kN on the Earth's surface provides a certain distance between the BSW generated by the fuselage and the shock wave (SW) emanating from the wing, which allows the SB level to be reduced [12]. As the aircraft weight increases, however, it seems problematic to ensure such a pressure profile because of the greater contribution of the lift force to SB formation. Realization of the middle zone effect involves consideration of exotic configurations with an extended nose part of the fuselage and with arrow-shaped and jointed wings, which make it difficult to reach a lift-to-drag ratio necessary for the aircraft to be cost-efficient.

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Kussunose et al. [13] analyzed the possibility of using a configuration with the Busemann wing, which also shows the limitations of the traditional approaches. Following the Jones principle [2] of concentration of the perturbed pressure near the forebody, Fomin et al. [14] performed a numerical study within the framework of the Euler equations and demonstrated that a tandem arrangement of two lifting surfaces on the fuselage allows a significant decrease in the SB level for heavy aircraft as well. Naturally, this result has to be validated experimentally.

As it is difficult to satisfy the restriction on the SB level with the use of traditional (passive) methods and it seems fairly possible that the current restriction will become more severe (15–25 Pa), it is necessary to develop new unconventional active methods for controlling the SB parameters.

The first experimental research of active methods aimed at controlling the SB parameters by means of mass addition in the form of fan-shaped and axisymmetric air jets with different orientations with respect to the model and thermal energy released by a burning hydrogen–air mixture [15, 16] shows much promise of this method for the formation of a perturbed flow both near the body and at large distances from it. Indirect evidence of this fact is provided by results of research on drag reduction by adding thermal energy near the body; a detailed review of these papers can be found in [17]. Tretyakov et al. [18] obtained supersonic flow regimes with insertion of a powerful optical pulsed discharge by means of laser radiation ahead of the conical forebody. BSW dissipation changed the flow structure, and a significant decrease in the drag of the body was recorded with increasing pulse frequency. In [17, 18], the domain used to study the perturbed flow was restricted to a narrow zone near the model, where the aerodynamic characteristics of the body were formed.

The results of [15–18] showed that the characteristics of a flying vehicle and the parameters of the SB generated by the vehicle can be effectively controlled by means of appropriate addition of mass and energy, i.e., appropriate location of the sources and appropriate distribution of their power with respect to the vehicle surfaces. In view of improved technical capabilities of organizing energy supply into a supersonic flow with the help of laser or microwave radiation, electron guns, and arc discharges used in modeling flow control processes, it is obviously necessary to study the possibility of controlling the SB parameters by these active methods affecting the formation of a perturbed flow near the flying vehicle and its evolution with distance. The prospects of this approach are supported by [19–23].

In addition to available methods of solving the SB problem with the use of promising aerodynamic configurations and active forcing methods (addition of thermal energy and mass), it seems of interest to consider organization of energy removal by means of flow cooling, which can expand the possibilities of formation of a necessary structure of the perturbed flow around the flying vehicle for reduction of its drag and SB level.

Formulation of the Problem and Goal of Investigations. Chirkashenko and Yudintsev [24] studied the effect of the body shape on the parameters of the sonic boom generated by this body and determined a class of modified (with the use of spherical bluntness of the forebody) power-law bodies, which ensure significant [as compared with the initial body ($\bar{r}_{\rm bl} = 0$) with an identical aspect ratio] reduction of the BSW intensity (down to 50%) in the middle zone. The length of the SB middle zone is determined by the distance at which the BSW generated by the blunted forebody interacts with the intermediate SW propagating further downstream with a velocity greater than the BSW velocity. The intermediate SW is formed near the body surface owing to interaction of the accelerating flow (induced by spherical bluntness). The intensity of the intermediate SW and its position on the perturbed pressure profile for a given flight Mach number M_∞ are determined by the geometric parameters of the modified body: power index n, aspect ratio $\lambda = l/d_{\rm mid}$, and dimensionless bluntness radius $\bar{r}_{\rm bl} = 2r_{\rm bl}/d_{\rm mid}$ (l and $d_{\rm mid}$ are the length and the mid-section diameter of the body, respectively).

Figure 1 shows the behavior of the asymptotical parameter $\Delta\bar{p}_{\rm SW}K^{3/4}$ of the BSW intensity as it moves away from the body (in terms of the altitude H) for different values of the aspect ratio and bluntness radius [$\Delta\bar{p}_{\rm SW} = (P_{\rm SW} - P_\infty)/P_\infty$ is the SW intensity, where $P_{\rm SW}$ and P_∞ are the static pressure behind the SW and ahead of it, respectively, and $K = H/d_{\rm mid}$ is the dimensionless distance from the body].

A typical feature of modified power-law bodies is their capability to provide a greater decrease in drag than the initial power-law bodies ($\bar{r}_{\rm bl} = 0$) with identical aspect ratios for certain levels of spherical bluntness. It should be noted that power-law bodies ($\bar{r}_{\rm bl} = 0$) at moderate supersonic velocities are optimal in terms of the wave drag in the class of bodies of revolution with a given aspect ratio [25]. Reduction of the drag and SB intensity due to the bluntness effect indicates the prospects of using such bodies. As the aspect ratio of the body increases, the effect of bluntness on drag reduction becomes less pronounced and almost ceases at $\lambda = 6$ [25]. At the same time,

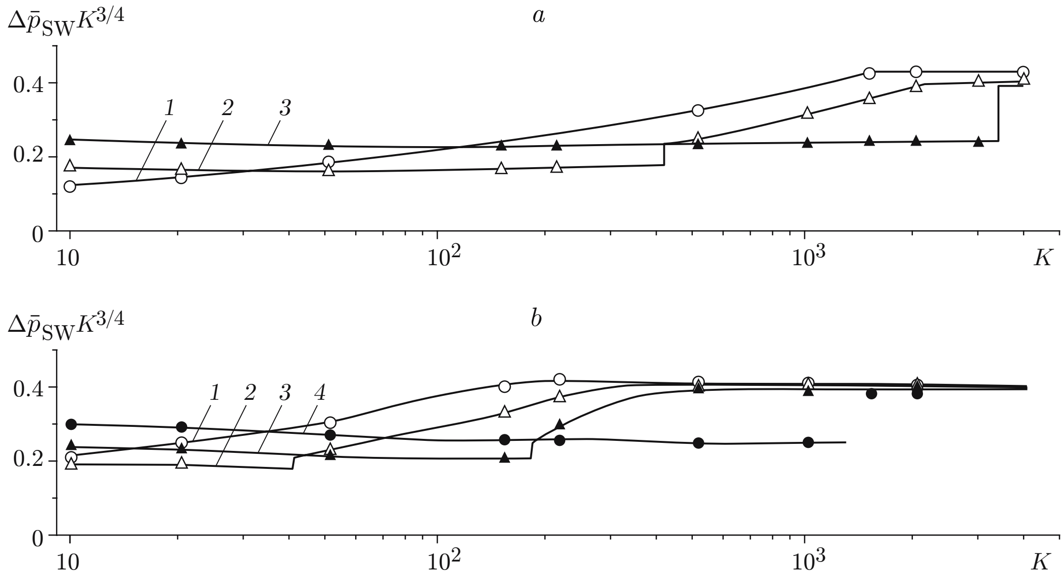

Fig. 1. Decay of BSW intensity with distance from a modified power-law body at $M_\infty = 2.03$ and different values of the aspect ratio and bluntness radius for $\lambda = 4$ (a) and 6 (b); $\bar{r}_{\rm bl} = 0$ (1), 0.1 (2), 0.2 (3), and 0.3 (4).

the length of the middle zone of the sonic boom increases with increasing aspect ratio and degree of bluntness (see Fig. 1). Thus, the requirements imposed on the body geometry to ensure reduction of the SB parameters and drag are contradictory.

At distances corresponding to the cruising flight altitude ($K = 6000$–7000), to increase the length of the SB middle zone with the minimum (or no) drag increase, it is necessary to prevent the interaction between the intermediate SW and the BSW. This can be done by several methods: by shifting the region of formation of the intermediate SW in the downstream direction, by reducing the intensity of the intermediate SW, or by preventing its formation. It should be noted that it is necessary to control the intermediate SW position if the SB level is reduced by means of injection of an upstream air jet from the nose part of a slender body. Chirkashenko and Yudintsev [15] described flow regimes with $M_\infty = 2$, which ensure reduction of the drag and SB parameters in the middle zone. The length of this zone is bounded by the intermediate SW formed near the surface owing to the flow around an effective body formed by the injected air jet. Fomin et al. [26] observed a similar flow structure in studying the method of drag reduction by means of injecting an upstream jet of a low-temperature plasma into a supersonic flow.

The intermediate SW is formed in an immediate vicinity of the surface of the modified power-law body. The flow expanding owing to spherical bluntness generates a system of compression waves behind the interface between the spherical and power-law surfaces; interaction of these waves leads to formation of a barrel shock wave.

For an oblique SW, the gas-dynamic relation between the SW intensity and the ratio of the static temperatures behind the SW (T) and ahead of the SW (T_∞) has the form [27]

$$\frac{T}{T_\infty} = (1 + \Delta\bar{p})\,\frac{\gamma - 1}{\gamma + 1}\,\frac{4\gamma}{(\gamma - 1)(\gamma + 1)[1 + \Delta\bar{p} + (\gamma - 1)/(\gamma + 1)]}. \qquad (1)$$

The results calculated by this dependence plotted in Fig. 2 show that a decrease in temperature behind the SW reduces the SW intensity. The temperature of the flow behind the intermediate SW can be reduced by mens of distributed injection of a supercooled gas from the body surface in the region of formation of this wave. In this case, reduction of the flow temperature behind the SW and, hence, the velocity of sound determining the velocity of propagation of perturbations are mainly provided by convective heat transfer between the injected gas and the flow incoming behind the SW. To eliminate the possibility of generation of additional shock waves by injected jets, it is necessary to organize an appropriate distribution of injection intensity over the body surface.

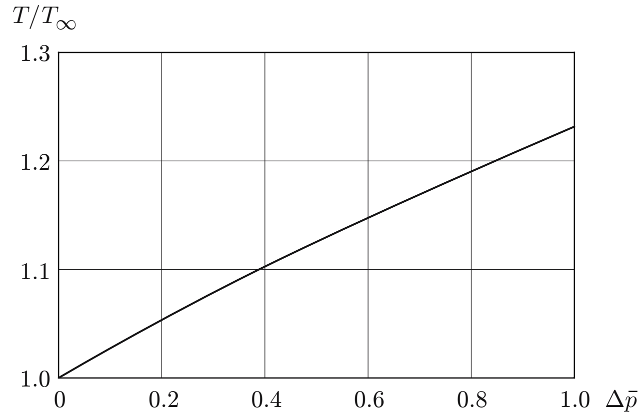

Fig. 2. Ratio of static temperatures on the oblique SW versus the SW intensity.

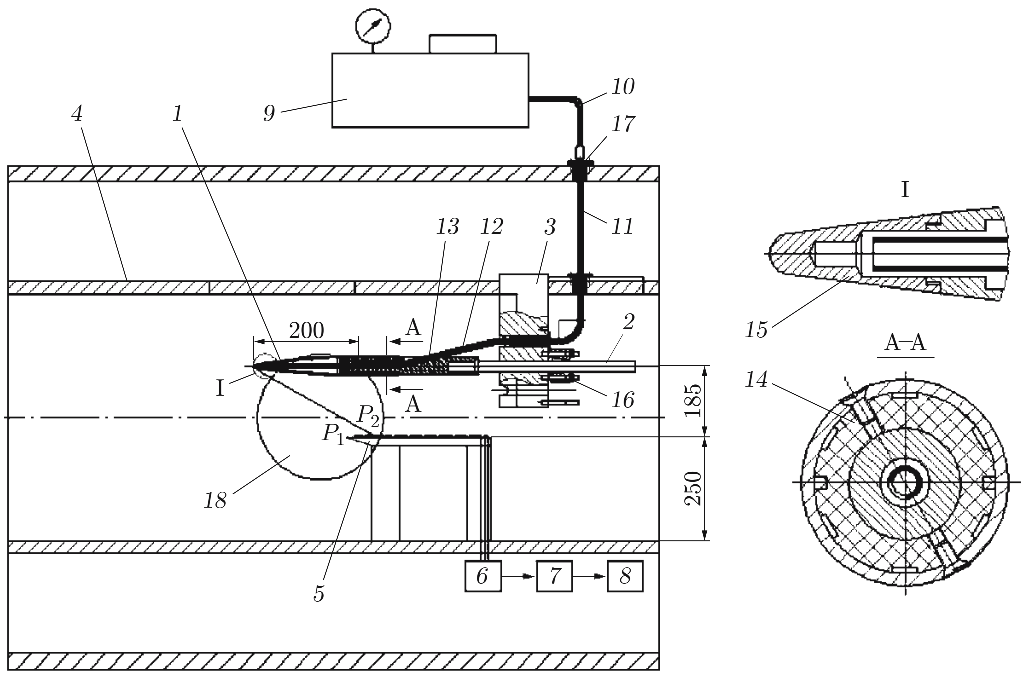

Fig. 3. Experimental setup: 1) model; 2) tail sting; 3) bracket; 4) test section of the T-313 wind tunnel; 5) plate with pressure taps; 6) pressure gauges; 7) registration system; 8) personal computer; 9) reservoir with liquid nitrogen; 10) pipeline for feeding liquid nitrogen into the pressure chamber; 11) pipeline 8 mm in diameter; 12) pipeline 6 mm in diameter; 13) intermediate sting; 14) fluoroplastic insert; 15) replaceable model tip; 16) collet mechanism; 17) heat insulator; 18) window.

As the intermediate SW is formed in an immediate vicinity of the body surface, we studied the possibility of controlling this process by means of surface cooling. A decrease in flow temperature is provided by heat conduction within the thermal boundary layer.

Experimental Technique. The study was performed in a T-313 supersonic wind tunnel of the Khristianovich Institute of Theoretical and Applied Mechanics of the Siberian Division of the Russian Academy of Sciences. The test parameters were $\mathrm{M}_\infty = 2.03$, $\mathrm{Re}_1 = 25 \cdot 10^6$ m^{-1}, and $T_0 = 258$ K. To obtain the full flow field of the flow perturbed by the body, we used a combined method including experiments and calculations [28, 29], which was based on measuring the profiles of the perturbed static pressure near the model mounted in the wind-tunnel test section and subsequent recalculation of these profiles to large distances with the use of a quasi-linear theory [5].

The layout of the experimental setup is shown in Fig. 3. The static pressure was measured with the help of a plate with pressure taps, with the measurement base being placed at a distance of 185 mm from the model axis, which corresponds to 3.7 diameters of the model mid-section. The root-mean-square error of pressure registration in the range $P \leqslant 0.1$ MPa by TDM9-A-0.1 and KPY42-A gauges was below 60 Pa. The useful signal [distribution of the dimensionless excess static pressure behind the reflected shock wave $\Delta\bar{p} = (P - P_\infty)/P_\infty$ over the measurement surface] was determined by the principle of superposition of perturbations, i.e., it was assumed that the interaction of the perturbed flow generated by the model with the background flow on the measurement plate (without the model) occurs in accordance with a linear law. The initial pressure distributions were smoothed to eliminate pressure fluctuations on the measured profiles, because their recalculation to large distances yields physically unjustified jumps in pressure.

The model was cooled by generating a coolant flow over the inner volume of the model. Liquid nitrogen used as a coolant was injected under an excess pressure from a reservoir through pipelines into the inner volume of the model forebody. After that, the coolant passed through rectangular grooves in the heat-insulation insert to the base part of the model and then was ejected into the incoming flow. Before wind-tunnel starting, the model mounted in the test section was cooled almost to the liquid nitrogen temperature (77 K) with operation of an injector providing removal of the coolant from the test section to the exhaust tank of the wind tunnel.

The model shaped as a modified power-law body of revolution ($\lambda = 6$, $n = 0.75$, and $\bar{r}_{\rm bl} = 0.2$) with a mid-section diameter $d_{\rm mid} = 50$ mm, mounted on a cylinder, was made of 12Kh18N10T steel.

The temperature on the model surface was measured by two thermoelectric transducers (thermocouples) with the hot junctions placed at distances of 5 mm (thermocouple T1) and 120 mm (thermocouple T2) from the model tip. T-type (copper–Constantan) thermoelectric converters with a wire diameter of 100 μm were used. The hot junction was placed onto the level of the outer surface of the model and was insulated from the model by a layer of a VS-9T heat-resistant glue 0.3 to 0.4 mm thick. The signals from the thermocouples were recorded by an NR34970A multichannel voltmeter, which provided temperature registration within 1.5%.

The pressure profiles were recalculated to large distances under the assumption of a homogeneous atmosphere by the method [30] based on the quasi-linear theory [5]. The relations determining the perturbed pressure on the characteristic and the position of this characteristic at an arbitrary distance from the initial profile have the following form in the second approximation with respect to perturbation intensity:

$$\Delta\bar{p} = \Delta\bar{p}_0 (r_0/r)^{1/2}, \qquad x = \beta r - k_1 \Delta\bar{p}\, r_0^{1/2} (r^{1/2} - r_0^{1/2}) + k_2 \Delta\bar{p}^2\, r_0 \ln(r/r_0) + x_0.$$

Here $\Delta\bar{p} = (p - p_\infty)/p_\infty$, $\beta = (\mathrm{M}_\infty^2 - 1)^{1/2}$, M_∞ is the Mach number, $k_1 = (\gamma + 1)\mathrm{M}_\infty^2/(\gamma\beta)$, $k_2 = (\gamma+1)^2 \mathrm{M}_\infty^2/(2\gamma\beta^{1/2})$, and γ is the ratio of specific heats.

In recalculating the initial pressure profiles to a distance r, the positions of discontinuities in the deformed profile were determined by the method developed in [5].

Results. Figure 4a shows the profiles of the dimensionless excess static pressure behind the reflected BSW, which were measured near the cooled and non-cooled model surface ($K = 3.7$). Figure 4b shows an upscaled fragment of these pressure distributions in the region of formation of the barrel shock and the confidence interval.

Figure 5 corresponding to Fig. 4b shows the pressures measured at different times and the confidence interval.

Model cooling does not involve any significant changes in the flow structure in the BSW region. In the region of formation of the barrel shock on the profile generated by the cooled model (see Fig. 4b), however, the excess pressure decreases behind the intermediate shock and increases ahead of the intermediate shock. As a result, the pressure difference on the intermediate shock near the cooled model surface is almost one half of the corresponding pressure difference on the non-cooled model. The reduced (as compared with the non-cooled model) level of pressure behind the intermediate shock persists in the downstream direction up to the rarefaction wave, which substantially reduces the positive pulse of the SB wave. Thus, cryogenic forcing reduces the intensity of the intermediate shock wave and the positive pulse of the SB wave with an unchanged BSW intensity.

Figure 6 shows the change in the flow temperature on the surfaces of the non-cooled and cooled models. The vertical bars on the abscissa axis correspond to the beginning and end of the supersonic regime of wind-tunnel operation.

During 1 min after the test time beginning (Fig. 6a), the temperature permanently decreases owing to unsteady heat transfer on the surface of the non-cooled model and reaches −5°C by the end of the test time in the region of the interface between the spherical and power-law surfaces (thermocouple T1 mounted at a distance

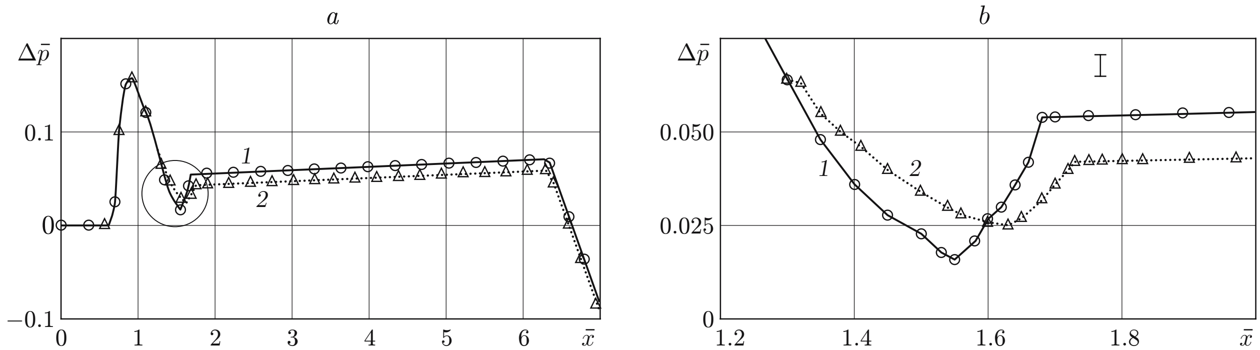

Fig. 4. Pressure profiles measured near the model surface (a) and upscaled fragment of these profiles (b) ($K = 3.7$): curves 1 and 2 refer to the non-cooled and cooled model, respectively.

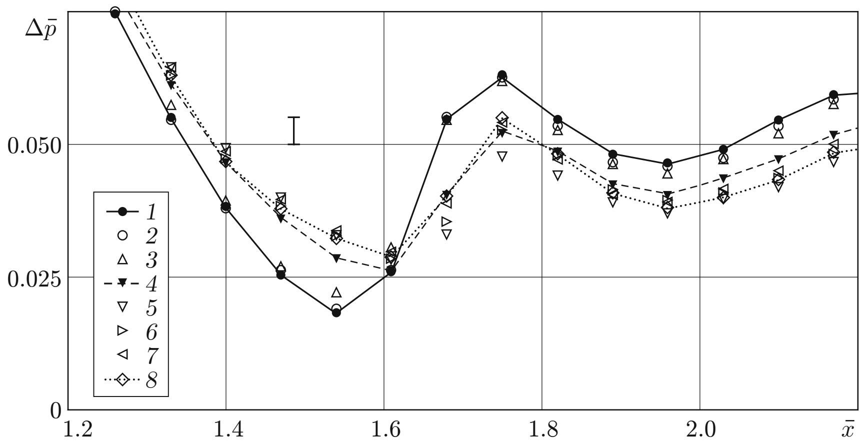

Fig. 5. Pressure distribution in the region of intermediate SW formation: curves 1–3 refer to the non-cooled model and curves 4–8 refer to the cooled model; $\tau = 324$ (1), 336 (2), 340 (3), 1505 (4), 1524 (5), 1540 (6), 1570 (7), and 1575 sec (8).

of 5 mm from the model tip). Steady heat transfer between the model and the incoming flow is established on the surface of the cooled model (Fig. 6b) soon after the test regime is stabilized. The temperature in the region of formation of the barrel shock reaches −50 to −55°C, which is substantially higher than the free-stream static temperature $T_\infty = 143$ K.

The temperature measured by the second thermocouple located on a less steep segment of the generatrix (thermocouple T2) is close to the free-stream static temperature. Thus, the flow temperature near the model surface in the region of formation of the barrel shock can be reduced approximately by 50°C by model cooling. The ratio of temperatures determining the barrel shock intensity changes from $T/T_\infty = 1.87$ to $T/T_\infty = 1.52$, i.e., by 18.5%.

The estimates calculated by Eq. (1) show that a 13.5% reduction of the static temperature behind the shock wave of intensity $\Delta\bar{p} \approx 0.1$ leads to a twofold decrease in intensity. As the calculations predict that the temperature ahead of the barrel shock is higher than the free-stream static temperature (approximately by 3–4%), we can assume that the ratio of SW intensities remains unchanged as the barrel shock waves (generated by the non-cooled and cooled models) propagate between the model and the measurement plate. Then, the reduction of the barrel shock intensity is caused by the decrease in temperature behind the shock wave due to model cooling.

The estimates obtained suggest that the decrease in intensity of the intermediate shock wave is caused by the lower velocity of propagation of perturbations in the region of their formation. To reach the temperature ratio $T/T_\infty = 1$, which prevents the barrel shock formation under the test conditions, the flow temperature near the model surface has to be additionally reduced by 61°C by means of increasing the mass flow of the coolant, decreasing the thickness of the model wall, or replacing the model material by a material with a higher thermal conductivity.

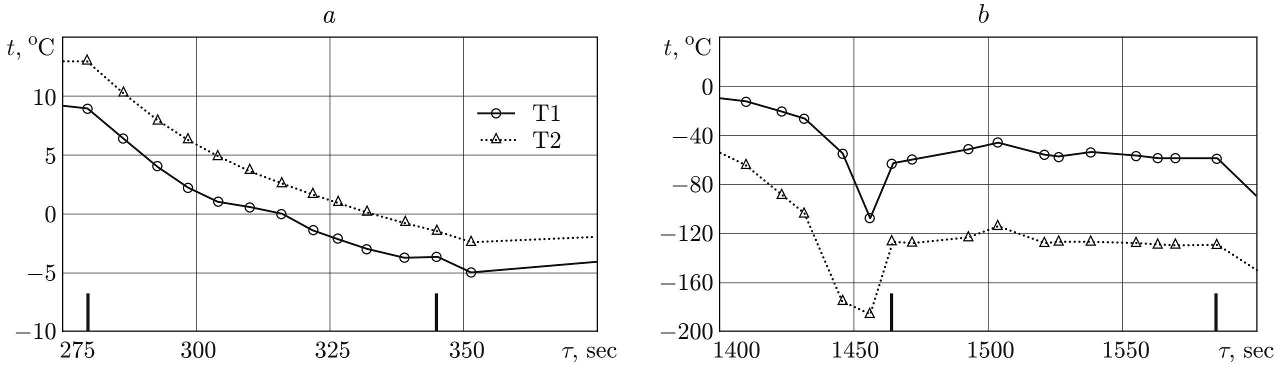

Fig. 6. Temperature on the model surface: (a) non-cooled model; (b) cooled model.

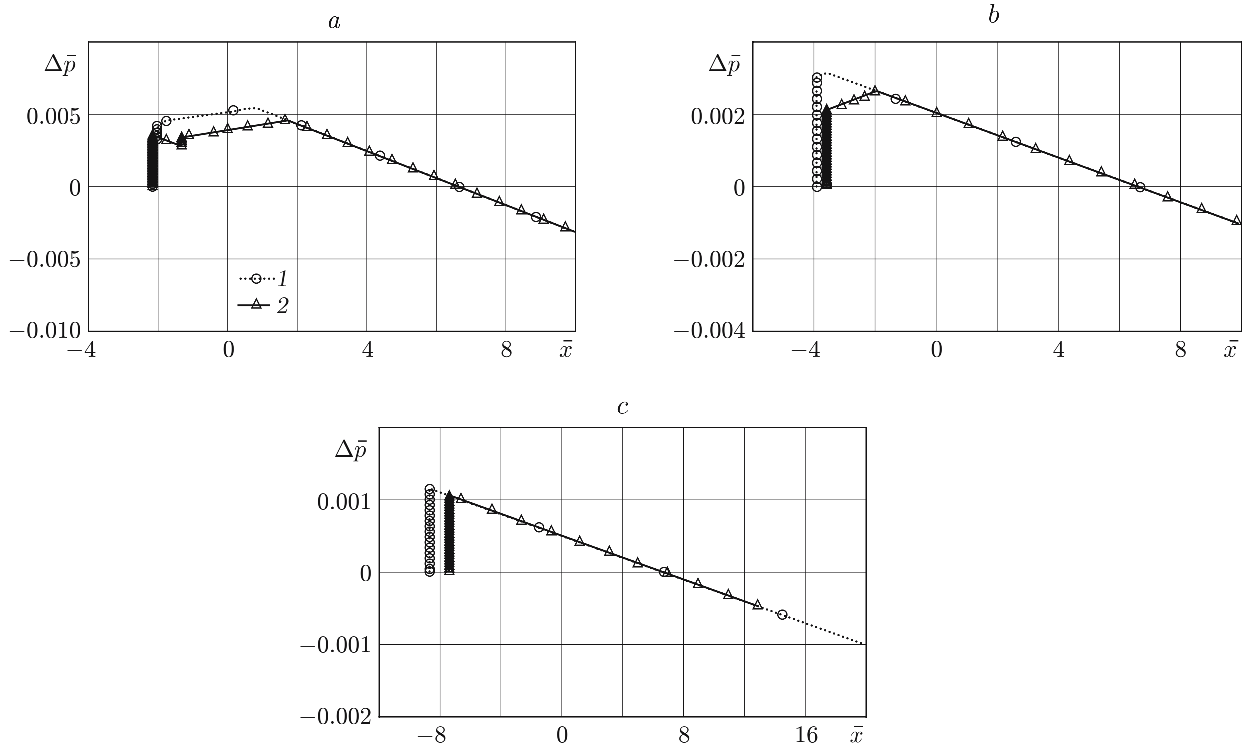

Fig. 7. Pressure profiles behind the bow shock wave at different distances from the model: $K = 500$ (a), 1500 (b), and 6000 (c); curves 1 and 2 refer to the non-cooled and cooled model, respectively.

Under real conditions, the SST cruising flight proceeds at an altitude of 18,000 m at $T_\infty = 217$ K, which exceeds the static temperature of the flow under the test conditions by a factor of 1.52. Therefore, a given temperature ratio is reached under real conditions owing to a more significant decrease in surface temperature. Thus, to obtain reliable information about the influence of perturbation velocity on the barrel shock formation, it is necessary to model real temperature conditions. Schlieren pictures taken during flow visualization showed that the flow regimes are identical for the model without coolant supply and for the cooled model.

The results obtained show that it is possible to control the process of formation of wave structures, such as the barrel shock, by means of surface cooling in the region of nucleation of these structures.

The evolution of the perturbed pressure profiles with distance from the non-cooled and cooled models is illustrated in Fig. 7. In the case with a distance $K = 500$ from the non-cooled model (Fig. 7a), the BSW is formed owing to interaction of the intermediate shock with the shock wave generated by spherical bluntness, which leads to a higher BSW intensity. The intermediate shock generated by the cooled model and propagating with a lower velocity because of its decreasing intensity persists on the pressure profile. On the cooled model, the intensity of the BSW generated by spherical bluntness is 25% lower than that on the non-cooled model. The recalculated results predict that the intermediate shock on the pressure profile persists up to $K = 1400$, with the BSW intensity decreasing to 50%.

At a distance $K = 1500$ from the cooled model (Fig. 7b), the BSW on the pressure profile is formed owing to interaction with the intermediate shock wave. The BSW intensity is significantly lower than the intensity of the BSW generated by the non-cooled model. Further decay of this wave induced by the positive pressure gradient behind the wave occurs much slower, as compared with the BSW decay on the non-cooled model. As a result, at a distance $K = 6000$ (Fig. 7c) practically corresponding to the asymptotic law of decay, the intensity of the BSW generated by the cooled model is approximately 12% lower than the intensity of the BSW generated by the non-cooled model.

Conclusions. The possibility of controlling the process of formation of wave structures, such as the barrel shock, in an immediate vicinity of the surface inducing these structures by means of surface cooling in the region of nucleation of these structures is demonstrated. The effect of surface cooling persists in the perturbed flow at large distances from the body.

Reduction of intensity of the intermediate shock wave and the positive pulse of the SB wave near a modified power-law solid by means of surface cooling allows the zone with a significantly reduced level of the sonic boom (down to 50%) to be substantially extended, with an additional 12% decrease at large distances.

The calculated estimates suggest that the main method of changing the flow structure in the vicinity of the intermediate SW is the reduction of the velocity of perturbations near the surface generating this wave.

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