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by

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13. ABSTRACT

An increase in the Mach number of airborne vehicles using air-breathing jet engines leads to a substantial decrease in the performance of the air intakes used in these engines. The main reason for the deterioration in performance is an increase in the influence of air viscosity. Despite the fact that the Mach number following compression in a hypersonic air intake intended for engines with supersonic combustion is greater than 1, the pressure ratio in it reaches $\bar{p} = 100$ and higher. In principle, such a pressure ratio can be obtained in a system of weak compression shocks with relatively low levels of total-pressure losses. However, the realization of such compression involves the occurrence of separation of the boundary layer, which causes a sharp increase in the losses associated with eddy formation, and in some instances makes it impossible to obtain the calculated flow scheme. In particular, e.g., the geometric flow area at the exit of a hypersonic air intake does not equal the calculated value of the throat area due to separation regions arising in the inlet. Therefore, to prevent separation of the boundary layer we must set up a system for controlling the boundary layer.

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An increase in the Mach number of airborne vehicles using air-breathing jet engines leads to a substantial decrease in the performance of the air intakes used in these engines. The main reason for the deterioration in performance is an increase in the influence of air viscosity. Despite the fact that the Mach number following compression in a hypersonic air intake intended for engines with supersonic combustion is greater than 1, the pressure ratio in it reaches $\overline{p}$ = 100 and higher. In principle, such a pressure ratio can be obtained in a system of weak compression shocks with relatively low levels of total-pressure losses. However, the realization of such compression involves the occurrence of separation of the boundary layer, which causes a sharp increase in the losses associated with eddy formation, and in some instances makes it impossible to obtain the calculated flow scheme. In particular, e.g., the geometric flow area at the exit of a hypersonic air intake does not equal the calculated value of the throat area due to separation regions arising in the inlet. Therefore, to prevent separation of the boundary layer we must set up a system for controlling the boundary layer.

At relatively low supersonic velocities we control the boundary layer by suction or bleed. In this manner we can, with a low flow of bleed air (1-3% of the total flow through the air intake), increase the value of the total-pressure coefficient σ, decrease the nonuniformity of the velocity field, and expand the range of stable operation of the air intake.

With a number of simplifying assumptions, a theoretical analysis of flow in the boundary layer in the region of its interaction with the compression shock with suction or bleed of the boundary layer can be made. This analysis, and extensive experimental studies, have shown that as a result of bleed or suction of the wall boundary layer there is an increase in fullness of the velocity profile in the boundary layer and, as a result, a rise in the so-called critical pressure ratio $\bar{p}_{cr}$, i.e., the pressure ratio across the shock for which flow in the boundary layer can be kept unseparated. In this way we can design air intakes with nonseparated flow in the greater part of the duct. However, upon transition to hypersonic flight speeds the Mach number on the surface of the central body becomes rather high, i.e., $M \geq 3$. As a result, for suction of the boundary layer there must be rather large openings, which leads to structural difficulties. However, with boundary-layer bleed through slotted channels, the extent of the region of effective operation of such channels is limited to lengths of the order of 10 times the thickness of the boundary layer behind the point of bleed. Therefore, for air intakes operating under a wide range of conditions, where the points of intersection of shocks with the boundary layer shift with a change in flight regime, this method of control either becomes ineffective or requires the setting up of several bleed channels which, in addition to structural difficulties, leads to an increase in flow of the bled air. Nevertheless, this method is highly recommended at supersonic speeds and, considering the specific inertia in the structural solutions, it ca apparently also be used at hypersonic velocities. In addition to this method of controlling the boundary layer, for such purposes we can also use other very effective methods to increase the stability of the boundary layer with respect to separation. These methods use the natural features of hypersonic power plants. Such methods for controlling the boundary layer include tangential injection of gas into the boundary layer and cooling of surfaces in the flow, i.e., a reduction of the temperature factor $\overline{T}_w = T_w/T_0$, or a combination of both methods. Here $\overline{T}_w$ is the surface temperature, T_0 is the stagnation temperature.

Just as with suction or bleed of the boundary layer, tangential injection or cooling results in an increase in energy in the wall region of the boundary layer, and thus an increase in the value of the critical pressure ratio. Knowing the dependence of $\overline{p}_{cr}$ on the parameters which define the characteristics of the injected stream and on the temperature factor $\overline{T}_w$ allows us to shape the body of the hypersonic air intakes to maintain nonseparated flow in the boundary layer. We should note that the system for injecting gas tangentially into the boundary layer can serve the dual role of feeding the gaseous fuel. To determine the required dependences we performed, in general form, calculations and experimental studies of the influence of heat transfer and tangential injection on the value of the critical pressure ratio, and the data obtained were used to set up criteria for non-separation of flow in hypersonic air intakes.

The flow formed with tangential injection to the boundary layer can be approximately divided into three regions (Fig. 1).

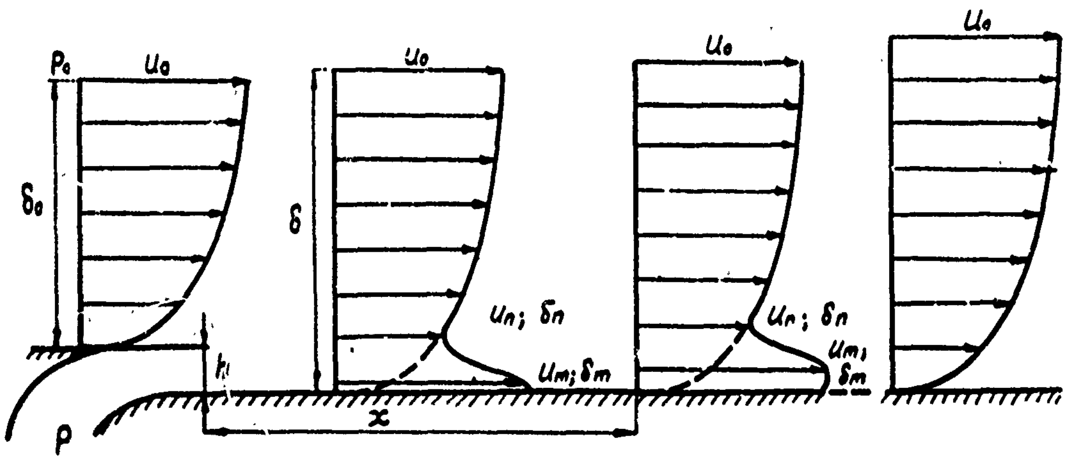


Fig. 1.

1. The external "unperturbed" region, in which the velocity profiles retain their previous form.

2. The region of mixing of the injection stream with the boundary layer, in which the velocity profile is described by the law for jet flow.

3. A thin wall region, a new boundary layer formed as a result of flow of the injected jet along the surface. The velocity profile in this boundary layer changes as a function of distance from the point of injection but, as experimental studies have shown, it is described quite well by the exponential dependence $u/u_0 = (y/\delta)^{1/n}$; u and u_0 are the velocities in the boundary layer and in the unperturbed flow, δ is the boundary-layer thickness. Because of the thinness of this boundary layer near the injection point, and to obtain agreement between the calculation and experimental data, we can use the value $n = 9$.

Introducing a number of assumptions, we can describe the velocity profile obtained with tangential injection. The continuity equation will have the form

$$\left(\int_0^\delta \rho u \, dy\right) + G_{inj} + \Delta G = \left(\int_0^\delta \rho u \, dy\right)_{X=X_S} \qquad (1)$$

Here G_{inj} is the flow of injected air and ΔG is the flow occurring through the external boundary of the boundary layer at distance $X = X_0$ from the point of injection to the examined section $X = X_S$. Flow through the external boundary of the boundary layer with and without injection is assumed to be identical.

It is also assumed that a change in the boundary of the "perturbed" region (point "n" on the velocity profile) occurs due to entrainment, by the injected jet, of flow from the wake boundary layer. The entrainment properties of the injected jet are described quite well by the relationship

$$\int_0^{\delta_n} \rho u \, dy = G_{inj}\gamma\sqrt{x/h} \qquad (2)$$

where $\gamma = 0.26$ is an empirical coefficient. The shape of the velocity profile in the mixing region can be given in the form

$$\frac{u - u_n}{u_m - u_n} = \left(1 - \eta^{3/2}\right)^2, \text{ where } \eta = \frac{y - \delta_m}{\delta_n - \delta_m}$$

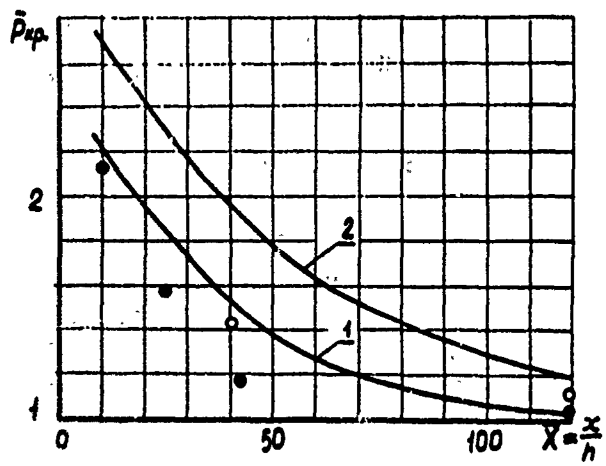
By calculating the change in the local friction coefficient we can estimate the increase in stability of the boundary layer to separation with tangential injection. By using assumptions on the self-similarity of the change in characteristics of the boundary layer ahead of the injection point, we can obtain the dependence of

$$C_{p\ cr} = \frac{p_{cr} - p_\infty}{(1/2)\rho_\infty u_\infty^2}$$

on the local friction coefficient

$$\frac{C_{p\ cr.inj}}{C_{p\ cr}} = \sqrt{\frac{C_{f\ inj}}{C_{f\ \infty}}}$$

Here $C_{p\ cr}$ and $C_{p\ cr.inj}$ are the coefficients of critical pressure at which there is separation of the ordinary boundary layer and the boundary layer with injection, respectively (the subscript ∞ pertains to parameters of unperturbed flow ahead of the interaction region). Figure 2 gives the change in critical pressure ratio $\bar{p}_{cr} = p_{cr.inj}/p_{cr}$ with injection as a function of $\bar{x} = x/h$ for two values of the relative flow of injected air $\bar{G} = G_{inj}/G_0 = 0.05$ and 0.1. The same figure shows experimental points for $\bar{G} = G_{inj}/G_0 = 0.04$ and 0.006. We see that despite the approximate nature of the examination, the coincidence between calculated and experimental data is satisfactory.



$1 - \bar{G} = 0.005$; $2 - \bar{G} = 0.1$; $\bullet - \bar{G} = 0.04$; $\circ - \bar{G} = 0.06$.

Fig. 2.

For a calculation analysis of the influence of heat transfer on boundary-layer separation, let us write in general form the dependence of the friction coefficient on the specific parameters:

$$\frac{\tau_w z}{\rho_0 u_0^2} = \varphi\left(M_0,\ \bar{T}_w,\ \frac{dp_0}{dx}\cdot\frac{z}{\rho_0 u_0^2},\ \frac{\rho_0 u_0 z}{\mu_0}\right) \qquad (3)$$

We know that the closeness of the turbulent boundary layer to separation for an adiabatic wall is defined by the parameter $\xi = dp_0/dx \times$ $\times\ z/\mu_0$, where dp_0/dx is the pressure gradient in the external flow and z is the characteristic dimension of the boundary layer. At the separation point, parameter ξ reaches its critical value ξ_{cr}. Solving Eq. (3) at the separation point relative to $dp_0/dx \cdot z/\rho_0 u_0^2$, and expanding the obtained expression into a series with respect to parameter $\mu/\rho_0 u_0 z$, we obtain at the limit, as $\mu \to 0$,

$$\frac{dp}{dx}\cdot\frac{z}{\rho_0 u_0^2} = \psi(M_0,\ \bar{T}_w).$$

In an incompressible fluid, the value of function $\psi(0,\ 1) \approx 0.015$, if we take as the characteristic dimension the thickness of boundary-layer displacement δ^*. Let us approximately represent the friction stress in the separated section in the form of a 4-th order polynomial in $\bar{y} = y/\delta$. To determine the coefficients let us use the boundary conditions which result from the equations of motion at the separation point. As a result, in the separated region we get

$$\frac{\tau}{\rho_0 u_0^2} = \frac{dp_0}{dx}\cdot\frac{\delta}{\rho_0 u_0^2}(\bar{y} - \bar{y}^4). \qquad (4)$$

Disregarding molecular friction in the separated section and representing turbulent friction in the form proposed by Prandtl, we get

$$\bar{\rho}\bar{l}^2\left(\frac{\partial \bar{u}}{\partial \bar{y}}\right) = \frac{dp_0}{dx}\cdot\frac{\delta}{\rho_0 u_0^2}\cdot(\bar{y} - \bar{y}^4)$$

$$\bar{\rho} = \frac{\rho}{\rho_0},\ \bar{u} = \frac{u}{u_0},\ \bar{l} = \frac{l}{\delta} \qquad (5)$$

Integrating differential Eq. (5), assuming retention of the similarity of the velocity profile and the excess temperature profile, we get

$$\psi(M_0,\ \bar{T}_w) = \frac{dp_0}{dx}\,\frac{\delta^*}{\rho_0 u_0^2} = \frac{2}{k-1}\,\frac{1}{}\,\frac{\delta^*}{\delta}\left(\frac{\delta}{\delta^*}\right)_H(\alpha - \beta)^2\psi(0,\ 1) \qquad (6)$$

where α and β are known functions of M_0 and $\bar{T}_w$.

With unrestricted increase in M_0 the separation parameter asymptotically vanishes. Cooling of the surface helps to increase the stability of the boundary layer to separation. This is physically explained by the decreased sensitivity of the boundary layer to the pressure gradient due to increased gas density in the wall region.

Let us examine the conditions of flow separation occurring during the interaction of a compression shock with the boundary layer. For the flow of an inviscid gas whose boundary is defined by the displacement thickness of the boundary layer, let us use the linearized Prandtl-Meyer relationship

$$\frac{p - p_1}{(1/2)\rho_0 u_0^2} = \frac{1}{(M_0^2 - 1)^{1/2}} \cdot \frac{d\delta^*}{dx} \tag{7}$$

whereas for a viscous gas near the separation point let us use (6). From a combined examination of conditions in a viscous and inviscid gas we get

$$\bar{p}_{cr} = n\frac{kM_0^2}{2} \cdot \xi^{1/2}(M_0^2 - 1)^{-1/4} + 1. \tag{8}$$

The value of the proportionality factor n = 4.2 was found from experiments on the interaction of a compression shock with a turbulent boundary layer on a heat-insulated surface. Figure 3 shows the results of calculating the value of

$$c_{p\ cr} = \frac{c_{p\ cr}(\bar{T}_w)}{c_{p\ cr}(1)} = \left[\frac{\xi_{cr}(\bar{T}_w)}{\xi_{cr}(1)}\right]^{1/2} \tag{9}$$

using this method, and experimental data obtained on various models. We see the good coincidence of the experimental and calculated results.

These control methods were used to improve the characteristics of hypersonic air intake models. The experimental models were very simple plane air intakes with a central body in the form of a flat wedge with angle β_{we} = 19°. The relative throat area was taken as

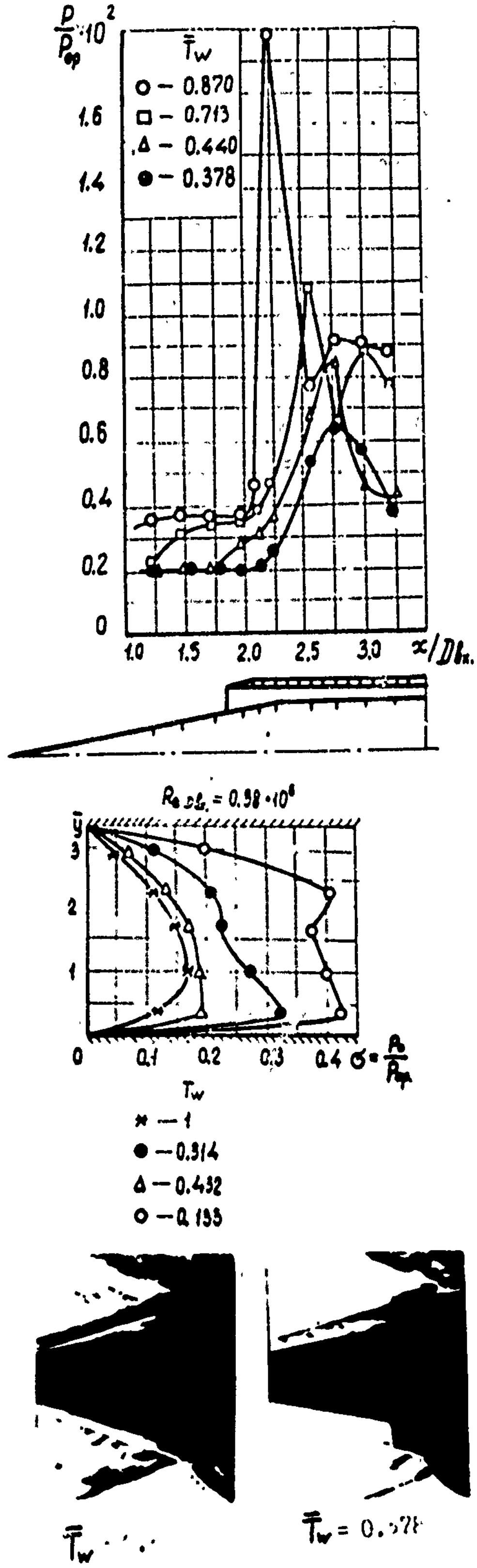

$\frac{p}{p_{op}} \cdot 10^2$
$\bar{T}_w$
○ — 0.870
□ — 0.713
Δ — 0.440
● — 0.378
$x/D_{вх}$
$Re_{вх} = 0.38 \cdot 10^6$
$\bar{y}$
$\sigma = \frac{p_0}{p_{op}}$
T_w
× — 1
● — 0.314
Δ — 0.432
○ — 0.135
$\bar{T}_w = 0.378$

Fig. 5

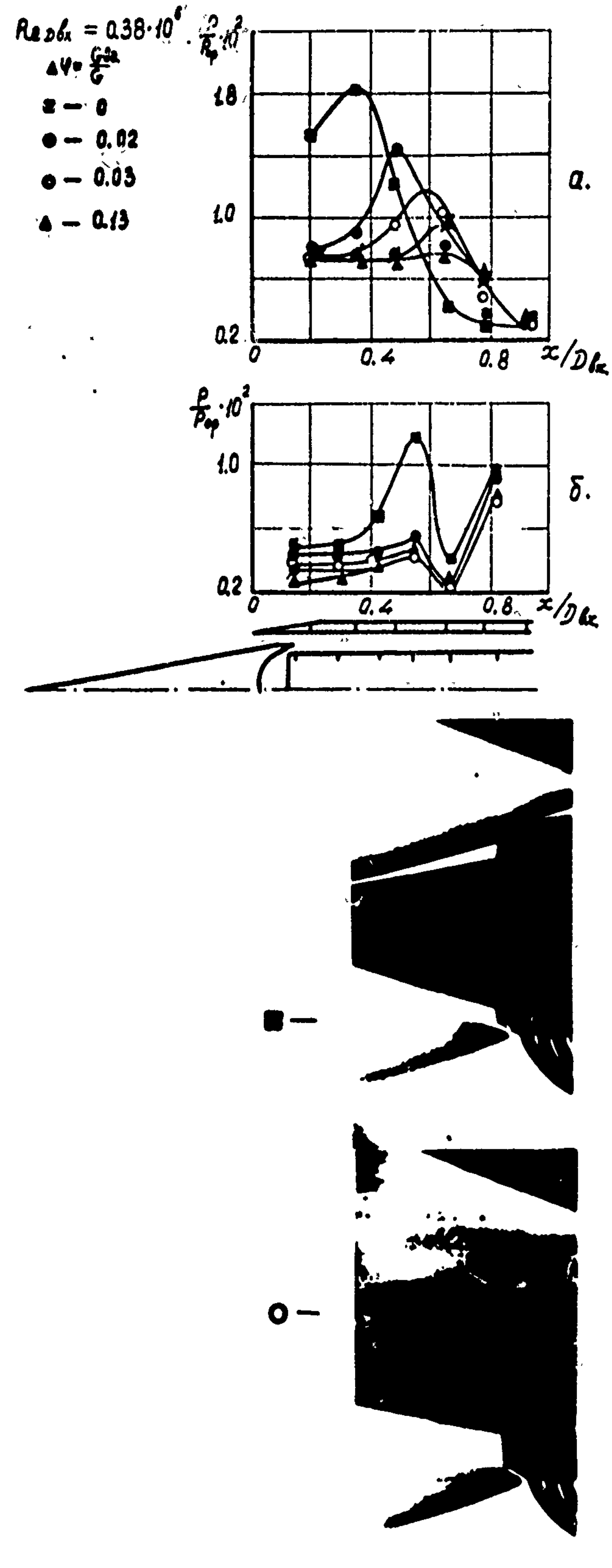

$Re_{вх} = 0.38 \cdot 10^6$
$\Delta\psi = \frac{G_{сл}}{G}$
■ — 0
● — 0.02
○ — 0.03
▲ — 0.13
$\frac{p}{p_{op}} \cdot 10^2$
a.
$x/D_{вх}$
б.

Fig. 6

experimental data are given for an axisymmetrical model with $M = 6$, $\beta_{we} = 10°$, $\bar{F}_{th} = 0.34$, and $Re = 0.86 \cdot 10^6$.) This same figure gives optical flow photographs. We see the disappearance of a separation region ahead of the air intake inlet with cooling of the central body. The total-pressure field in the inner duct of the air intake, shown in this figure, indicates that with cooling there is a substantial increase in the total-pressure level in the flow core. Similar results are also obtained when using tangential injection ahead of the region of incidence of the compression shock reflected from the cowl. Figure 6 gives the distribution of static pressure along the central body and the cowl, obtained on a model axisymmetrical air intake with $M = 6$, $\beta_{we} = 10°$, $\bar{F}_{th} = 0.34$ and $Re = 0.86 \cdot 10^6$. This same figure also shows optical flow photographs. The inclusion of tangential injection causes the disappearance of the separation region ahead of the air-intake inlet ($\bar{G}_{inj} = 3\%$).

Thus, the examined systems for controlling the boundary layer are effective ways for substantially improving the characteristics of hypersonic air intakes.