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HYDRODYNAMICS IN TECHNOLOGICAL PROCESSES

# INFLUENCE OF LIFT REDISTRIBUTION ALONG THE LENGTH OF A SUPERSONIC AIRPLANE ON THE ACOUSTIC-SHOCK PARAMETERS

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*Results of numerical investigations of the influence of lift redistribution along the length of the airframe of an airplane arrangement on the parameters of the acoustic shock produced by this arrangement and its aerodynamic characteristics are presented. The airplane arrangements investigated were constructed by disposition of wings in tandem at definite values of the aerodynamic lift and the lifting-surface area. The gasdynamic parameters of the flow near an arrangement were calculated by the numerical scheme based on the integral Euler equations, and the disturbed-pressure distribution at a large distance from the arrangement was determined using the quasi-linear theory. It is shown that the acoustic-shock level is substantially decreased in the case where there arises a middle-zone effect and an excess-pressure profile with an N-like shape. The arrangement of a supersonic airplane of weight 40 t making a cruise at a height $H$ = 18 km and a Mach number $M_\infty$ = 2.0, which makes it possible to decrease the acoustic shock produced by this airplane with no decrease in its lift-drag ratio, was determined.*

**Introduction.** Since, in 1960s, a restriction was imposed on the level of the acoustic shock (50 Pa) produced by a supersonic passenger airplane (SPA), over the years a large number of investigations have been devoted to the solution of the problem on minimization of the acoustic shock [1–7]. In all this works, the quasi-linear Whitham theory [6] was used. In [7], this theory was extended to the wing + fuselage combination and, in [8], to the case of a nonstationary flight.

It was shown in [9, 10] that the problem on the acoustic shock produced by an airplane can be separated into two problems: the problem on the influence of the atmosphere and the flight conditions on the damping coefficient and the problem on the influence of the shape of the airplane and the regime of flow around it on the excess-pressure profile. This makes the solution of the problem on the acoustic shock produced by the airplane much simpler. It was shown in [1] that the intensity of the acoustic shock produced by an airplane can be substantially decreased by disposition of a wing at the tail of the fuselage as well as by decreasing the relative thickness of the wing and increasing the relative area of the fuselage midsection and the side chord of the wing.

In [2], the lower limit of the acoustic shock produced by a thin body moving in a homogeneous atmosphere was determined in the asymptotic approximation of the theory of [6], according to which, at a large distance from the body (far zone), the disturbed-pressure profile has an N-like shape. The features of the middle zone, in which, as evidenced by the disturbed-pressure profile, there arise individual shock waves, e.g., waves propagating from the fuselage and the wing, the intensity of which becomes smaller after their interaction than the intensity of the resulting shock wave, were considered in [3]. It was shown in this work that, by a corresponding redistribution of the aerodynamic lift along the length of an airplane, the middle-zone effect can be retained to a cruise flight altitude. In this case, the acoustic shock can be much smaller than the lower limit determined in [2].

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The minimum values of the intensity of the head shock wave (SW) and the maximum excess pressure in the wake of it, determined in [4] with account for the middle-zone effect manifesting itself as realization of an excess-pressure profile with a finite time of increasing pressure in the wake of the head shock wave, were smaller than those determined in [2]. In [5], an analogous approach was used for minimization of the intensity of the pressure jump and the maximum pressure in the wake of the head shock wave with regard for the trailing shock wave. The laws of distribution of the aerodynamic lift along the length of a body, providing a minimum intensity of the acoustic shock, were determined.

At the same time, according to the estimates made in [11], a Concorde SPA, whose configuration is considered as practically optimum from the standpoint of provision of a minimum acoustic shock, does not provide an acceptable acoustic-shock level; to decrease this level, it is necessary to increase the flight altitude or decrease the takeoff weight of the airplane. The results of investigations carried out at the Central Aerohydrodynamics Institute [12] also point to the fact that, at present, it is impossible to provide an acceptable level of the acoustic shock produced by an airplane with a large takeoff weight without a deterioration of its economic indices.

It should be noted that the available data on minimization of the acoustic shock produced by an airplane represent partial solutions of the problem obtained within the framework of the linear-theory assumptions without regard for the restrictions emposed on the aerodynamic characteristics of the airplane and, therefore, should be considered as recommendational in character. It is difficult to perform a more reliable minimization of the parameters of the acoustic shock produced by an airplane having a definite weight and lift-drag ratio with the use of numerical methods since the parameters of this shock are determined by a large number of factors. Because of this, at present, such investigations are carried out with the use of combined methods. In [13], the parameters of the disturbed flow near an airplane were determined from the solution of the Euler equations and were recalculated for larger distances from the airplane with the use of the quasi-linear theory [6]. More reliable results can be obtained using the complex experimental-calculation method [14] based on measurement of the parameters of the disturbed flow at the control surface of a wind tunnel near a model and further recalculation of the experimental data for larger distances.

In [2], it was theoretically shown that the intensity of the acoustic shock produced by a pointed thin body in the far zone will be decreased by bluntness of this body. This result was substantiated experimentally in the process of investigating the influence of the bluntness of different-shape bodies on the acoustic shock produced by them [15, 16]. It was established that the bluntness of a thin body leads to a decrease in the pressure drop in the wake of the head shock wave and a decrease in the positive-phase pulse, as compared to those of the initial thin body; however, this can be attained on condition that there is no an intermediate pressure jump that, depending on the configuration of the body, can arise near its surface as a result of the reexpansion of the flow around the bluntness. If this condition is fulfiffed for a certain degree of bluntness of a body, the intensity and the pulse of the shock wave reach minimum values in the far zone. The bluntness effect is explained physically as being due to the fact that the intensity of the shock wave near a blunted body and the intensity of the rarefaction wave in the wake of the shock wave increase, which leads to a nonlinear increase in the rate of damping of the shock wave, as compared to the thin body. In [16], it was established that, unlike the predictions of the quasi-linear theory, the parameters of the acoustic shock, produced by an equivalent rotation body moving with a moderate supersonic velocity in the far zone, depend substantially on the shape of the body and that bodies having a shape defined by an exponential function hold the greatest promise because a maximum disturbed pressure is provided on the surface of these bodies near there nose. The modification of the shape of a near-optimum exponential body with respect to drag by a spherical bluntness makes it possible to decrease the intensity of the head shock wave by two times as compared to that of the initial pointed body. Investigations of the influence of the load distribution over the wing of an airplane having a definite aerodynamic lift, design shape, and root-chord length [17] on the level of the acoustic shock produced by it also point to the fact that an optimum load distribution makes it possible to substantially decrease the parameters of the acoustic shock in the far zone.

**Formulation of the Problem.** In line with the foregoing, we investigated the possibility of decreasing the intensity of the acoustic shock produced by an airplane by increasing the disturbed pressure near its nose part with the use of a leading wing that is introduced into the airplane arrangement and creates a part of the prescribed aerodynamic lift. A schematic airplane arrangement with wings positioned in tandem was considered in [2], where the possibility of decreasing the level of the acoustic shock in the middle zone was demonstrated within the framework of the quasi-linear theory [6].

However, this problem calls for further investigations because the reliability of the estimates of the acoustic-shock parameters obtained with the use of the quasi-linear theory is low; the aerodynamic characteristics of the arrangements investigated, necessary for determining the applicability of the indicated method of control of the acoustic-shock parameters, are absent; and the number of parameters considered, which determine the problem, is insufficient.

In the present work, we investigated the influence of the distribution of the aerodynamic lift along the length of the airframe of an airplane on the parameters of the acoustic shock produced by this airplane and its aerodynamic characteristics by organization of an arrangement of leading and trailing wings, having different areas, in tandem. As a basis, we used the arrangement of an SPA of weight 40 t and length 45 m, making a stationary flight along a straight path at a height of 18 km under standard atmospheric conditions.

**Methods of Solution.** In the first stage, the problem on the supersonic flow around an airplane arrangement was solved. The gasdynamic parameters on the surface and in the neighborhood of the body were determined with the use of a numerical scheme based on the approximated integral Euler equations [18]. From the solution of this problem, the aerodynamic characteristics of the arrangement were determined, and the data obtained were used as the initial data for solution of the main problem.

In order that the condition of constancy of the prescribed aerodynamic lift be fulfilled, which is necessary for the flight of the airplane arrangements being considered at a definite height, the angle of attack was corrected such that its value did not exceed the initial angle of attack $\alpha = 5^{\circ}$ by more than 2–3%.

In the second stage, the parameters of the initial excess-pressure profile at a definite distance from the model being investigated were determined. According to the quasi-linear theory, the disturbed excess-pressure profile at a large distance $r$ from a disturbance source is determined from the relations [6]

$$\frac{\Delta p}{p_\infty} = \frac{\gamma M_\infty^2 \Phi(t, \theta)}{2(2\beta)^{1/2} r^{1/2}}, \tag{1}$$

$$x = \beta r - k\Phi(\tau, \theta)\, r^{1/2} + \tau\,. \tag{2}$$

For the airframe + wing combination, the function $\Phi(\tau, \theta)$ takes the form

$$\Phi(\tau, \theta) = \frac{1}{2\pi}\int_0^\tau \frac{S''(t)\, dt}{\sqrt{\tau - t}} + \frac{\beta \cos(\theta)}{2\pi q}\int_0^\tau \frac{Y''(t, \theta)\, dt}{\sqrt{\tau - t}}\,. \tag{3}$$

The first term in (3) defines the influence of the volume of the body on the intensity of the acoustic shock produced by it, and the second term defines the influence of the aerodynamic lift on this parameter. In the general case, relation (3) allows one to simulate the disturbed-gas motion at different distances from the disturbance source under conditions where several shock waves can propagate in the medium; the number of these waves is determined by the function $\Phi(\tau, \theta)$.

**Geometric Model of an Airplane Arrangement.** A complex aerodynamic arrangement was formed by the method of construction of a model with the use of high-level base objects. As applied to the aerodynamics problems, among these objects are the airframe of an airplane and its wings and stabilizers. A characteristic feature of this method is that, when a geometric object is defined by appropriately selected analytical relations, it provides the fulfilment of the necessary requirements, such as an interface convenience and openness. As applied to the numerical solution of the problems on flows around a flying vehicles, an interface convenience allows one, at a small number of determining parameters, to rapidly change the geometry of a configuration and perform parametric calculations for the purpose of determining the physical laws of the problem being considered.

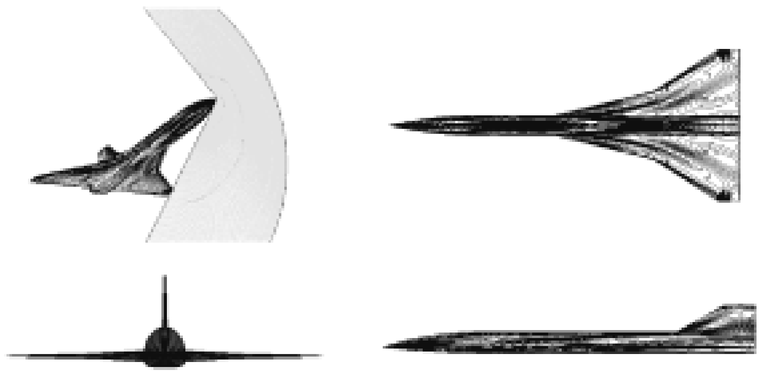

Fig. 1. Geometry of the initial arrangement and scheme of the flow around it ($M_\infty = 2$; $\alpha = 5^o$).

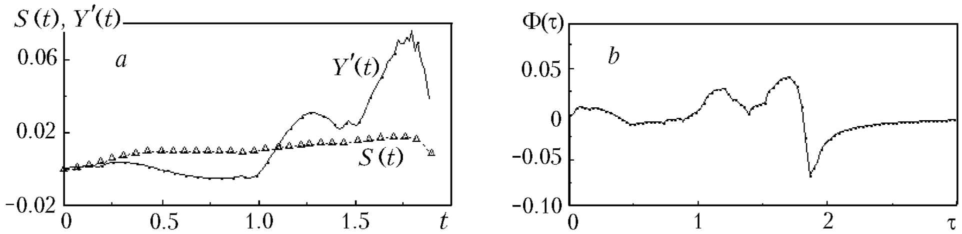


Fig. 2. Initial arrangement: a) $S(t)$ is the distribution of the cross-section area, $Y'(t)$ is the time dependence of the derivative of the lift; b) Whitham function.

In the present work, the outline of the cross sections of the geometric model of the airframe of an airplane was defined by a hyperelliptic function, and the geometric model of the wing of this airplane was a TU-144 SPA wing with a plane middle surface. In the airplane arrangement being considered, the airframe represents an ogival-cylindrical body consisting of an ogival head and a tail of elongation $\lambda = 4.5$, mated with the cylinder. The parameters $n$ and $\mu$ determining the outline of the airframe cross section take the following values: $n_1 = 2$, $\mu_1 = 4$ in the upper semiplane and $n_2 = 4$, $\mu_2 = 0.55$ in the lower semiplane [18].

The tandem arrangements, which make it possible to redistribute the aerodynamic lift along the airframe, were constructed such that the total areas of the leading and trailing wings were equal to the area $S_0$ of the initial wing. The positions of both wings, similar in geometry to the initial-arrangement wing, were determined by the longitudinal coordinates $\bar{x} = x/L_h$ of the origin of their side chords, determined relative to the nose of the airframe. The area of the front wing was determined from the relation $S = \bar{S}{\cdot}S_0$, where $\bar{S}$ is a predetermined parameter.

**Results of Calculations.** The initial arrangement of the SPA being considered is shown in Fig. 1. Figure 2 presents calculated distributions of the cross-section area $S(t)$ and the derivative of the aerodynamic lift $Y'(t, 0)$ along the length of the initial arrangement as well as the Whitham function $\Phi(\tau, \theta)$ determined by them with the use of relation (3). A characteristic features of this function is that it contains local maxima that are due to the influence of individual elements of the arrangement on the formation of the cross-section area of the equivalent body and the distribution of the aerodynamic lift. For example, in the intervals $\tau \in [0, 1]$, $[1, 1.5]$, and $[1.5, 1.7]$, the effects of the airframe, the roll, and the base wing appear.

The evolution of the disturbed flow propagating from the initial arrangement can be seen in Fig. 3 showing the profiles of the relative excess pressure in the wake of the head shock wave at different distances from the arrangement. For a real airplane the gauge $K = H/L = 400$ corresponds to the cruise altitude $H = 18$ km.

In the near zone, where $K = 3$, the head shock wave is a wave generated by the nose part of the airframe. In the zone of flow reexpansion, intermediate-frequency shock waves are formed by the roll and the base wing. The disturbed-pressure profile is closed by the trailing shock wave.

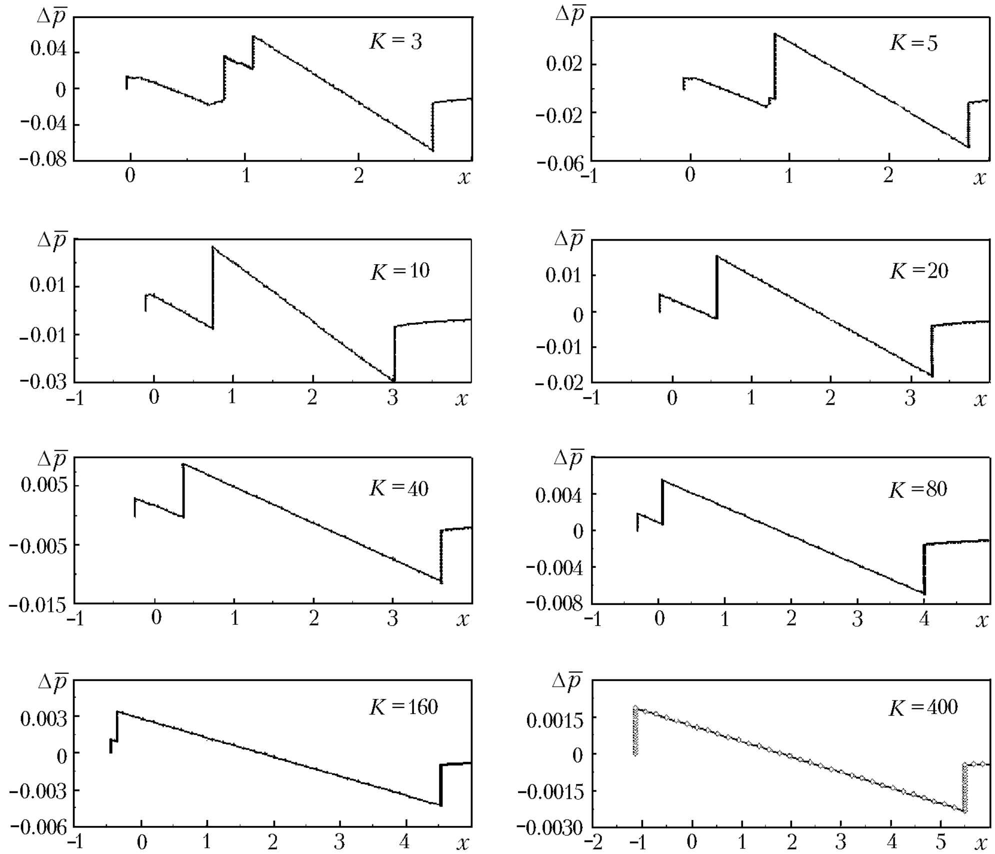


Fig. 3. Profile of the excess pressure at different distances from the SPA (arrangement with $S_0$).

At a distance $K = 5$, the profile shows an intermediate-frequency shock wave arising as a result of the interaction of the intermediate-frequency shock waves, formed by the roll and the base wing and having a correspondingly increased amplitude. As the distance from the arrangement increases ($K = 5$–$160$), the intermediate-frequency wave catches up with the head wave propagating from the airframe and, at a distance $K = 160$, they practically merge into one wave. Then, the pressure profile with a near N-like shape propagates with the head shock wave having a substantially increased intensity.

**Influence of the Leading-Wing Area on the Acoustic-Shock Parameters.** The redistribution of the aerodynamic lift along the length of an airframe was investigated at a fixed position of the leading wing $\bar{x}_1 = 0.25$. Near the arrangement with a leading wing of area $\bar{S} = 0.1$ (Fig. 4, $K = 400$), the disturbed-pressure profile shows the leading shock wave formed as a result of the interaction of the chock waves propagating from the airframe, the roll, and the base leading wing. In the negative phase of the wave there arises a small intermediate pressure jump caused by the interaction of the reexpanded flow downstream of the leading wing with the surface of the airframe. The next intermediate jumps on the pressure profile are due to the shock waves propagating from the roll and the base trailing wing. As the distance from the arrangement increases, the intermediate-frequency shock waves merge into one wave; therefore, the pressure profile of the head shock wave wake shows one intermediate-frequency shock wave (Fig. 4, $K = 400$). In this case, the intensities of the head and intermediate-frequency shock waves are much smaller than the intensity of the head shock wave of the initial arrangement.

The structure of the disturbed flow does not change with further increase in the area of the leading wing. At a cruise altitude ($K = 400$), the intensities of the leading and intermediate-frequency waves change insignificantly and the distance between them change nonmonotonically (Fig. 5, $K = 400$, $\bar{S} = 0.3$).

At $\bar{S} \geq 0.5$, the pressure profile at cruise altitude has an N-like shape that is retained as long as $\bar{S} = 0.6$ (Fig. 6a, $K = 400$). The pressure profile of the arrangement with a wing of area $\bar{S} = 0.7$ (Fig. 6b, $K = 400$) also

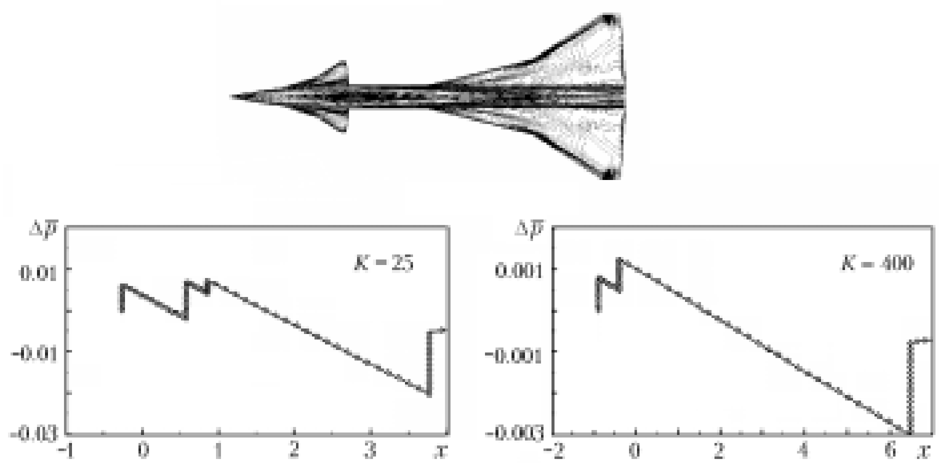


Fig. 4. Profile of the excess pressure at different distances from the SPA ($\bar{S}$ = 0.1, $\bar{x}_1$ = 0.25).

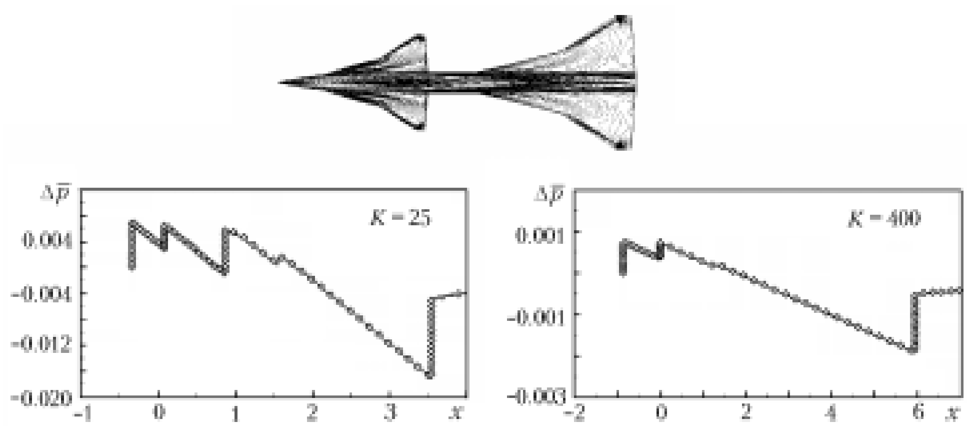


Fig. 5. Profile of the excess pressure at different distances from the SPA ($\bar{S}$ = 0.3, $\bar{x}_1$ = 0.25).

shows an intermediate-frequency shock wave, which persists with further increase in the area of the leading wing as long as $\bar{S} = 1$.

The results of investigation of the influence of the redistribution of the lifting surface along the length of the airplane flying at a cruise altitude are presented in Fig. 7a in the form of the dependences of the static pressure drops $\Delta p$ (kg/m$^2$) in the head and intermediate-frequency shock waves, the lift-drag ratio, and the dimensional pulse of the positive phase of the excess-pressure profile $\left(\bar{I}_+(\bar{S}) = \int_{x_{\text{s.w}}}^{x_0} \Delta\bar{p}d\bar{x}\right)$ on the relative area of the leading wing $\bar{S}$. The numerical values of these parameters are presented in Table 1. The pressure drops in the wakes of the reflected shock waves were determined for the reflection coefficient 2. An analysis of these data has shown that a decrease in the intensity of the acoustic shock, produced by the arrangement being considered, as compared to that of the initial arrangement, at a cruise altitude and at values of $\bar{S}$ varying in the range $0.1 \leq \bar{S} \leq 1.0$ can be both due to the middle-zone effect and the N-like pressure profile. In this case, the decrease in the acoustic shock depends substantially on the ratio between the areas of the leading and trailing wings.

In the range of $\bar{S}$ = 0.1–0.4, the excess-pressure in the head shock wave is substantially decreased by the middle-zone effect; it reaches 60% for $\bar{S}$ = 0.3–0.4. At the same time, in this range, the intensity of the intermediate-frequency shock wave decreases with increase in $\bar{S}$ and reaches a minimum at $\bar{S}$ = 0.3.

TABLE 1. Dependence of the Acoustic-Shock Parameters on the Leading-Wing Area at a Flight Altitude of 18 km and a Weight of 40 t ($\bar{x} = 0.25$)

| $\bar{S} = S/S_0$ | Head SW, $\Delta p_h$, kg/m$^2$ | Intermediate-frequency SW, $\Delta p_{int}$, kg/m$^2$ | Lift-drag ratio $Cy/Cx$ | Pulse $\bar{I}_+$ |
|---|---|---|---|---|
| 0 | 10.41 | 0 | 14.59 | 0.00270 |
| 0.1 | 4.51 | 4.48 | 15.16 | 0.00160 |
| 0.2 | 4.41 | 4.51 | 13.79 | 0.00142 |
| 0.3 | 4.18 | 2.15 | 13.21 | 0.00109 |
| 0.4 | 4.18 | 3.71 | 12.49 | 0.00122 |
| 0.5 | 8.50 | 0 | 11.37 | 0.00227 |
| 0.6 | 9.08 | 0 | 11.83 | 0.00231 |
| 0.7 | 6.04 | 2.42 | 10.71 | 0.00116 |
| 0.8 | 6.24 | 2.37 | 10.56 | 0.00138 |
| 0.9 | 6.57 | 2.18 | 10.58 | 0.00122 |
| 1.0 | 8.01 | 1.97 | 10.59 | 0.00180 |

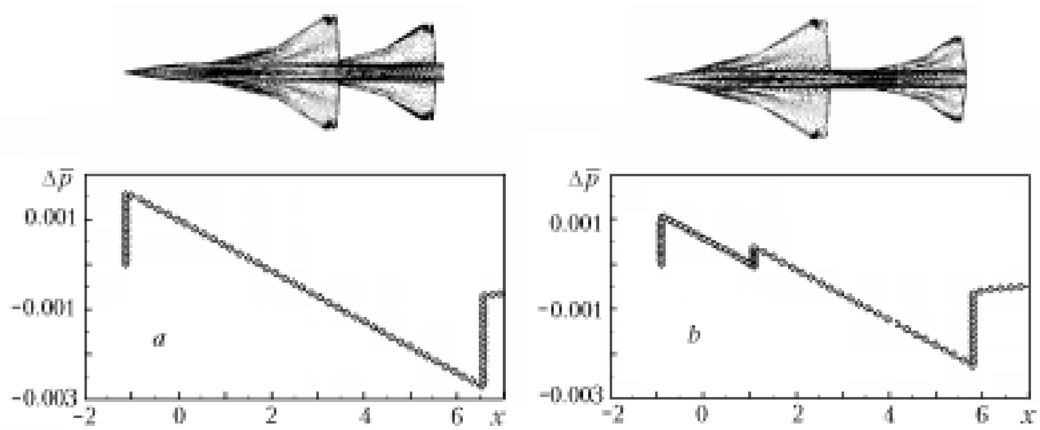


Fig. 6. Profile of the excess pressure produced by arrangements at $K = 400$: a) $\bar{S} = 0.6$; b) $\bar{S} = 0.7$. $\bar{x}_1 = 0.25$.

The parameters of the acoustic shock change substantially with change in the leading-wing area in the range $0.4 < \bar{S} < 0.7$. An increase in the leading-wing area to $\bar{S} = 0.5$ leads to an increase in the intensity of the leading shock wave and a decrease in the pressure drop in the intermediate-frequency shock wave, with the result that the pressure profile is transformed and takes an N-like shape that retains as long as $\bar{S} = 0.6$ (Fig. 6a). However, the increased intensity of the leading shock wave remains lower than that of the initial arrangement by 18% and 13% respectively at $\bar{S} = 0.5$ and 0.6.

Then, as is seen from the excess-pressure profile shown in Fig. 6b, an increase in the leading-wing area to $\bar{S} = 0.7$ leads to the formation of an intermediate-frequency shock wave. In this case, the acoustic-shock level is decreased due to the middle-zone effect. A change in the leading-wing area to $\bar{S} = 1$ causes the intensity of the leading shock wave to continuously increase and the intensity of the intermediate-frequency shock wave to decrease. A characteristic feature of the change in the acoustic-shock parameters in the range $0.7 \le \bar{S} \le 1$ is a marked decrease in the intensity of the intermediate-frequency shock wave, the level of which is much lower than that realized in the range $0.1 \le \bar{S} \le 0.4$, which is evidently due to the large decrease in the trailing-wing area. The decrease in the intensity of the head shock wave in this range is much smaller than that at $\bar{S} = 0.1$–$0.4$; it comprises 42–23%, which is due to the large increase in the leading wing area, as compared to that of the initial arrangement.

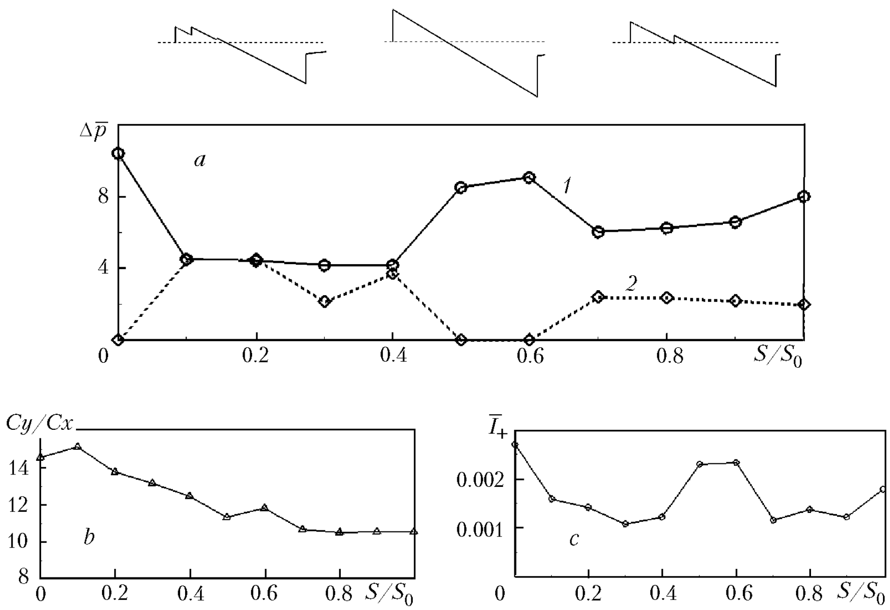


Fig. 7. Dependence of the acoustic-shock parameters on the ratio between the lifting-surface areas ($\bar{x}_1$ = 0.25): 1) head shock wave, 2) intermediate-frequency shock wave (a); b) change in the lift-drag ratio; c) change in the positive-phase pulse.

To determine the efficiency of control of the parameters of the acoustic shock produced by an airplane by changing its arrangement, it is necessary to know the changes srising in the aerodynamic characteristics of the new arrangements as compared to those of the initial configuration, in particular the changes in the lift-drag ratio.

The dependence of the lift-drag ratio on $\bar{S}$, presented in Fig. 7b points to the fact that this parameter changes insignificantly (≈4%) at $\bar{S}$ = 0.1 and then decreases nonmonotonically throughout the range of change in $\bar{S}$ being considered. The decrease in the lift-drag ratio ceases to be monotonic at $\bar{S}$ = 0.6, at which a local maximum of this parameter is realized. A further increase in the leading-wing area to $\bar{S} > 0.6$, leading to the appearance of an indication of an intermediate-frequency shock wave on the pressure profile, practically does not influence the lift-drag ratio, the value of which is smaller by approximately 27% than that of the initial arrangement. The largest gradient of decrease in the lift-drag ratio with increase in $\bar{S}$ is realized in the range $0.1 \le \bar{S} \le 0.4$ where the acoustic-shock level decreases by a maximum value due to the middle zone effect.

A characteristic determining the degree of action of the acoustic shock on the environment is the positive-phase pulse of the pressure profile (Fig. 7c). It is seen that, in the region of maximum decrease in the acoustic-shock level $0.1 \le \bar{S} \le 0.4$, the excess-pressure pulse decreases substantially and reaches, at $\bar{S}$ = 0.3, a local minimum, the value of which is smaller by 60% than that of the initial arrangement. In the range $0.5 \le \bar{S} \le 0.6$, in which the pressure profile has an N-like shape at a cruise altitude, the value of the increased pulse does not change practically and remains smaller by 15% than that of the initial arrangement. As a result of the middle-zone effect, at $\bar{S}$ = 0.7, the excess-pressure pulse decreases to a minimum level provided by the arrangement with $\bar{S}$ = 0.3. A further increase in $\bar{S}$ leads to an increase in this pulse; at $\bar{S}$ = 1, it reaches a maximum that is smaller by 33% than the maximum of the pulse formed by the initial arrangement.

According to the results obtained, the variant with $\bar{S}$ = 0.1 is the most promising arrangement in the problem being considered. This arrangement provides, due to the middle-zone effect, a decrease in the intensity of the leading and intermediate-frequency shock waves by 60% and a decrease in the positive-phase pulse of the acoustic-shock wave by 41% at a small increase in the lift-drag ratio, as compared to the initial arrangement. The arrangement with $\bar{S}$ = 0.3 exerts a minimum effect on the environment. This arrangement also provides, due to the middle-zone effect, a decrease in the intensity of the head and intermediate-frequency shock waves and in the positive-phase pulse of the acoustic-shock wave by, respectively, 60%, 79%, and 60%; however, in this case, the lift-drag ratio decreases by 9.5%

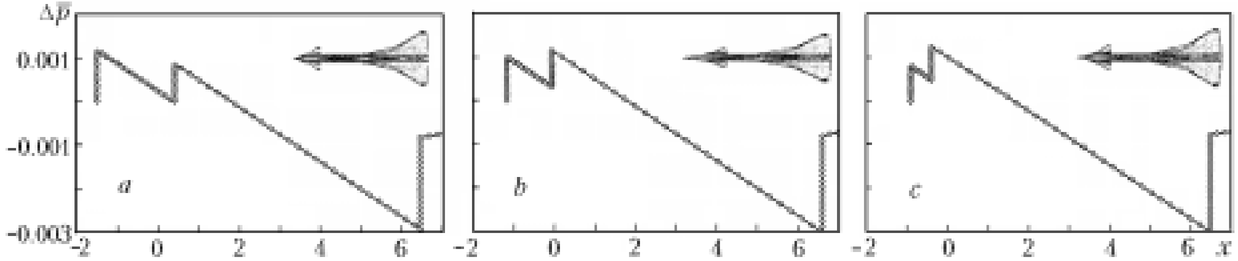

Fig. 8. Dependence of the excess-pressure profile on the position of the leading wing ($K$ = 400, $\bar{S}$ = 0.1): $\bar{x}_1$ = 0.065 (a), 0.125 (b), and 0.25 (c).

TABLE 2. Dependence of the Acoustic-Shock Parameters on the Area of the Leading Wing and Its Position along the Length of the Airframe

| $\bar{x}_1$ | Head SW, $\Delta p_h$, kg/m$^2$ | Intermediate-frequency SW, $\Delta p_{int}$, kg/m$^2$ | Lift-drag ratio $Cy/Cx$ | Pulse $\bar{I}_+$ |
|---|---|---|---|---|
| | | $\bar{S} = 0$ | | |
| 0 | 10.41 | 0 | 14.59 | 0.00270 |
| | | $\bar{S} = 0.1$ | | |
| 0.065 | 6.65 | 5.02 | 14.32 | 0.00170 |
| 0.125 | 5.71 | 4.80 | 15.20 | 0.00181 |
| 0.25 | 4.51 | 4.48 | 15.16 | 0.00160 |
| 0.375 | 10.41 | 0 | 16.68 | 0.00271 |
| 0.50 | 10.25 | 0 | 15.25 | 0.00262 |
| | | $\bar{S} = 0.3$ | | |
| 0.125 | 5.81 | 2.85 | 12.70 | 0.00135 |
| 0.25 | 4.18 | 2.15 | 13.21 | 0.00109 |
| 0.50 | 7.93 | 0 | 14.62 | 0.00177 |

as compared to that of the initial arrangement. The efficiency of decrease in the acoustic shock, provided by the arrangements with $\bar{S}$ = 0.5–0.6, forming an N-like wave in the cruise flight, is much lower. In this case, at $\bar{S}$ = 0.5, the maximum decrease in the intensity of the head shock wave and in the pulse is respectively 18% and 15% at a decrease in the lift-drag ratio of 22%, as compared to the initial arrangement.

**Influence of the Disposition of the Leading Wing along the Fuselage Length on the Acoustic-Shock Parameters.** To investigate the influence of the redistribution of the aerodynamic lift to the nose part of an arrangement on the acoustic-shock parameters in more detail, we have performed calculations for the most promising arrangements ($\bar{S}$ = 0.1, 0.3) with different positions of the leading wing relative to the nose of the airframe.

The pressure profiles (Fig. 8) realized at a cruise altitude of the arrangement with $\bar{S}$ = 0.1 show that a shift of the leading wing relative to its initial position ($\bar{x}_1$ = 0.25) to the nose of the airframe ($\bar{x}_1$ = 0.065) leads to an increase in the intensity of the head shock wave and in the distance between the head and intermediate-frequency shock waves. In this case, the intensity of the head shock wave remains much smaller than that of the initial arrangement and the pressure drop in an intermediate-pressure jump remains practically unchanged. A shift of the leading wing from its initial position downstream leads to the merge of the intermediate-pressure jump with the head shock wave ($\bar{x}_1$ = 0.375) and, consequently, to an increase in its intensity to the level produced by the initial arrangement. In this case, the shape of the disturbed-pressure profile is close to an N-wave. It should be noted that the lift-drag ratio of this arrangement (Table 2) is 16.5% larger than that of the initial arrangement, which represents a large reserve for taking additional measures on decreasing the acoustic-shock level.

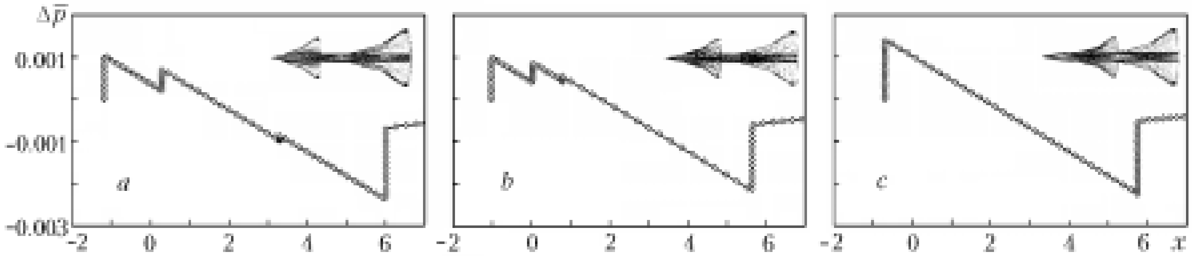


Fig. 9. Dependence of the excess-pressure profile on the position of the leading wing ($K$ = 400, $\bar{S}$ = 0.3): $\bar{x}_1$ = 0.125 (a), 0.25 (b), and 0.50 (c).

Analogous qualitative changes in the pressure profile arise when the leading wing having a larger area ($\bar{S}$ = 0.3) is displaced relative to its initial position (Fig. 9); however, in this case, there arise such quantitative changes as a decrease in the intensity of the intermediate pressure jump and the formation of an N-like profile. The pressure profile produced by the arrangement with $\bar{S}$ = 0.3 and $\bar{x}_1$ = 0.50 (Fig. 9c) takes an N-like shape after the interaction with the intermediate-pressure jump. The intensity of the head shock wave decreases by 28%, as compared to that of the initial arrangement, at a lift-drag ratio remained unchanged, and, as compared to the arrangement with $\bar{S}$ = 0.1 and $\bar{x}_1$ = 0.5, not only the intensity of the head shock wave decreases, but also the positive-phase pulse of the wave decreases by 40%. This result shows that the efficiency of decrease in the acoustic shock at an N-like pressure profile depends on the area and position of the leading wing relative to the nose of the fuselage. In this case, the indicated efficiency is much higher at small $\bar{S}$, as compared to the data obtained in the previous section for substantially larger $\bar{S}$.

The results obtained show that a head shock wave of minimum intensity is provided in the cruise regime due to the middle-zone effect in the case where a leading wing of definite area $\bar{S}$ is positioned at a certain distance from the nose of the airframe. For the leading-wing areas being considered, this position is $\bar{x}_1$ = 0.25, which is coincident with the leading wing position used in the previous section.

At an N-like pressure profile, the efficiency of decrease in the acoustic-shock level increases noticeably with increase in the area of the leading wing and increase in the distance between it and the nose of the airframe; in this case, the lift-drag ratio is retained practically at the level corresponding to that of the initial arrangement.

Thus, our numerical investigations have shown that the acoustic shock produced by an SPA can be substantially decreased by redistribution of the aerodynamic lift along the length of its airframe; in this case, a decrease in the acoustic shock is attained due to both the middle-zone effect and the formation of an N-like pressure profile. It was established that, as applied to an SPA of weight 40 t, making a cruise at a height of 18 km, the arrangement with $\bar{S}$ = 0.1 and $\bar{x}_1$ = 0.25 provides a maximum decrease in the acoustic shock level (by 60%), due to the middle-zone effect and an increase in the lift-drag ratio by 4%, as compared to that of the initial arrangement. A maximum decrease in the acoustic shock level at an N-like pressure profile and airplane parameters being considered is provided by the arrangement with $\bar{S}$ = 0.3 and $\bar{x}_1$ = 0.5; in this case, the intensity of the acoustic-shock wave is decreased by 28% and its positive-phase pulse is decreased by 40% at a lift-drag ratio equal to that of the initial arrangement.

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## NOTATION

$Cx$, drag coefficient; $Cy$, lift coefficient; $Cy/Cx$, wave aerodynamic characteristic; $D_m$, diameter of the midsection of the airframe, m; $H$, flight altitude, m; $\bar{I}_+$, positive-phase pulse of the excess-pressure profile; $K = H/L$, dimensional flight altitude; $k = (\gamma + 1)M_\infty^2 / 2^{\frac{1}{2}} \beta^{\frac{3}{2}}$, proportionality coefficient; $k_0$, reflection coefficient; $M_\infty$, Mach number of the incident flow; $n$, exponent of the hyperelliptic function; $p$, static pressure, kg/m$^2$; $L$, length of the arrangement, m; $p_\infty$, static pressure of the incident flow, kg/m$^2$; $\Delta\bar{p} = (p - p_\infty)/p_\infty$, relative excess pressure; $\Delta p = p - p_\infty$, absolute excess pressure, kg/m$^2$; $q = \frac{1}{2}\rho W_\infty^2$, kinetic head; $r$, distance from a disturbance source, m; $S_0$, area of the wing of the initial arrangement; $\bar{S} = S/S_0$, relative area of the leading wing; $S(t)$, function of distribution of the area of the arrangement cross section; $S''(t)$, second derivative of the function $S(t)$; $t$, longitudinal coordinate along the axis of the body; $dt$, length element; $W_\infty$, velocity of the incident flow; $\bar{x}_1 = x_1/L$, coordinate of the origin of the side chord of the leading wing on the airframe; $Y(t, \theta)$, distribution of local loads over the wing; $Y'(t, \theta) = \partial Y(t, \theta)/\partial t$; $Y''(t, \theta) = \partial^2 Y(t, \theta)/\partial t^2$; $\alpha$, angle of attack; $\beta$, $(M_\infty^2 - 1)^{1/2}$; $\Phi(\tau, \theta)$, Whitham function; $\gamma$, adiabatic index; $\lambda_{h.p} = L_h/D_m$, elongation of the head part of the airframe; $\mu$, relation between the semiaxes of the hyperelliptic function; $\theta$, azimuth coordinate; $\rho$, density; $\tau$, characteristic variable. Subscripts: h, head shock wave; h.p, head part; int, intermediate-frequency shock wave; r, reflection; m, middle; 0, initial arrangement; 1, origin of the side chord of the wing; $\infty$, parameters of the incident flow; s.w, shock wave.

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