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AERODYNAMICS AND HEAT TRANSFER OF WAVERIDERS

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Abstract

Elements of flows behind a systems of plane and conical shocks, creating bodies which are named "wave-riders", are used for investigation and estimation of aerodynamic forces and heat fluxes to super- and hypersonic vehicles. Lift-to-drag ratio of such vehicles may be higher than for the wedge or simple wave-rider with one plane shock and the same lift.

Experiments with models of wave-riders confirmed the existence of the flow in vicinity of

1) the sharp leading edge, corresponding to the wedge flow with strong shock in the normal section;

2) the line of shocks intersection, corresponding to the strong shock, reflected from the wall, in the normal plane.

1. Introduction

G.I. Maikapar and T. Nonweiler have stressed the expediency of investigation of flow with attached plane shock waves near three-dimensional bodies.[1,2] This approach had been extended, and bodies formed by streamsurfaces of known two-dimensional and axisymmetrical flows were investigated.[3,4,5,6] With the help of this streamsurfaces it is possible to build a broad variety of bodies (fig. 1).

Aerodynamic characteristics and heat transfer rates to such bodies can be predicted simply and accurately and used for estimation of characteristics of supersonic and hypersonic vehicles.[7,8]

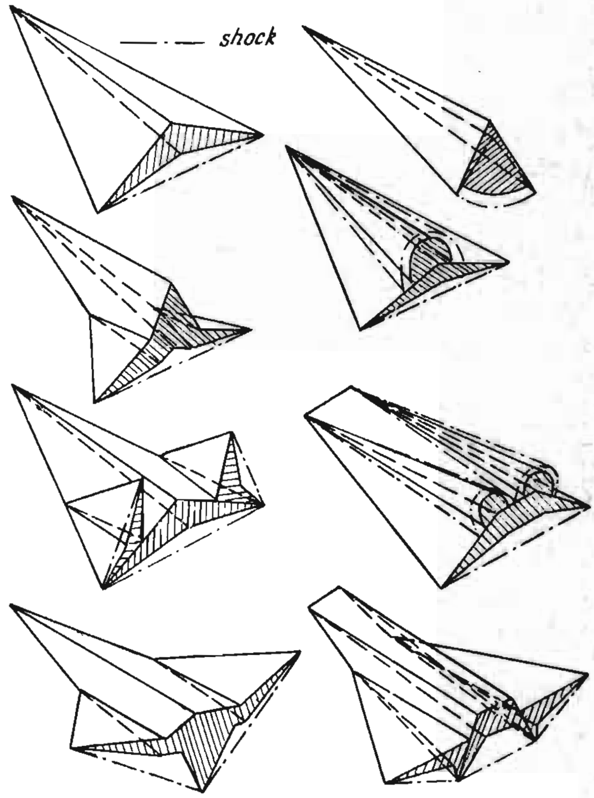


FIGURE 1. Bodies with plane and conical shocks.

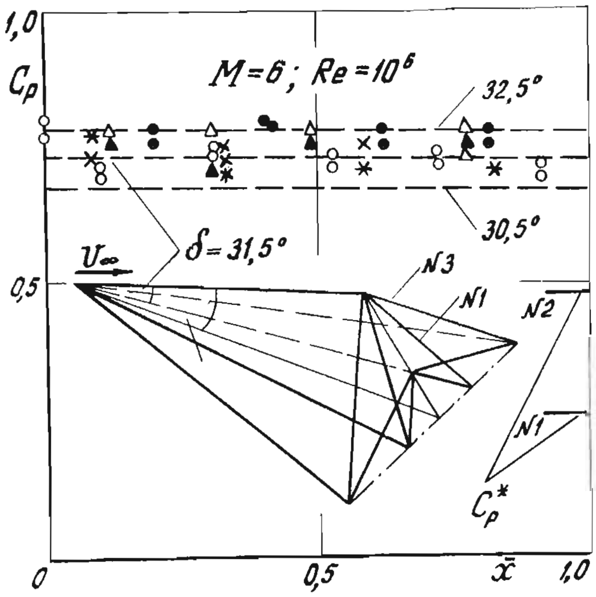
The simplest and basic element of flow with plane shocks about a body is a corner, the ridge line of which is parallel to the wedge, corresponding to the designed shocks. Recently there appeared a number of investigations of the flow in a corner.

In the present paper the realization of two exact solutions for corner flow is proved, and the problem of increase of lift-to-drag ratio of the wave-rider supporting two plane shocks in comparison with the wedge or wave-rider supporting one plane shock (caret wing) with the same lift is investigated.

2. Realization of the two Exact Solutions for Flow in a Corner

(1) The flow in a corner with plane shock is the simplest three-dimensional supersonic flow, similar to the flow near the wedge with an attached shock. This flow is sufficiently general because it is realized for a certain region of Mach numbers and wedge angles δ , corresponding to each corner. The mentioned region and flow parameters in the corner depend only on the angle between the ridge line and the plane of leading edges $\theta-\delta$ (θ - is the angle between the shock and the velocity vector of flow) and do not depend on the angle between panels of the corner V . By reducing it (strength of the shock unchanged) normal to the leading edge component of the velocity behind the shock also diminishes and beginning from certain V becomes subsonic. In normal to the leading edge plane such flows correspond to wedge flows with "strong" shocks; their realization was doubted by some authors (9,10,11).

Experimental investigations of the flow in a corner showed that threedimensionality and boundary conditions downstream leading edges realize such flows. In fig. 2 pressure coefficient (C_p) for inner surfaces of three corners with the same inclination of a ridge line to leading edges plane is shown. The corners are designed for a weak plane shock, corresponding to the wedge $\delta=31{,}5^\circ$ and M=6. Normal to leading edge component of the velocity behind the shock is subsonic for corners N 1 and N 2 ($U_{n2}<a_2$) and supersonic for corner N 3 ($U_{n2}>a_2$). Results of measurements for different panels are presented in fig. 2 in different dots. $\bar{x}$ - the conical coordinate.



corner N	1 ○ ○	2 △ ▲	3 × ✱
δ	31,5°	31,5°	31,5°
θ	43°	43°	43°
V	35°	56°	73°
U_{n2}/a_2	0,53	0,74	1,25

FIGURE 2. Comparison of measured and predicted (dotted lines) pressure in the corners (plane Shock).

In the limits of experimental scatter ($\Delta\delta=\pm1^\circ$) the pressure is the same for all corners and equal to the pressure on the wedge (dotted line). Pressure coefficient behind the shock on leading edge ($\bar{x}=1{,}0$) with supersonic normal component of the velocity ($U_{n2}>a_2$) is also shown in fig. 2 (C_p^*). It is much smaller than measured.

Limiting stream lines obtained with the help of smeared dots method for design conditions are parallel to the ridge line.

Corner flow with the plane shock is sufficiently stable to regime variation also in the case of the "strong" shock in normal to leading edge plane, if geometry is properly chosen. In fig. 3 pressure coefficient for inner panels of the corner at M=3 and different angles of attack is shown. At regimes when the shock for the wedge parallel to the ridge line departs from the leading edge plane not more than $\pm 1^{\circ}$ ($\delta = 3^{\circ} \div 24^{\circ}$) the measured pressure practically coincides with the pressure on the corresponding wedge (dotted lines). In the normal to leading edge section plane the shock corresponding to measured pressure is "strong".

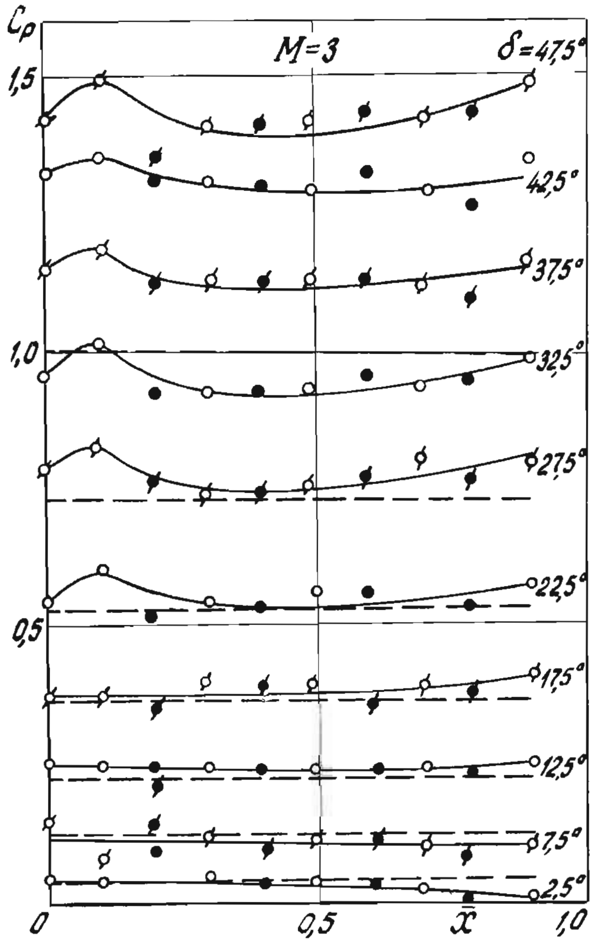


FIGURE 3. Comparison of measured and predicted (dotted lines) pressure in the corner (plane shocks).

For M=6 "corner flow" with a plane shock is realized for $\delta = 41^{\circ}$. Small departures from these conditions cause sharp variations of the flow. If angle of attack is decreased, there appear in a corner flow inner shocks inducting local separations and expansions (fig. 4).

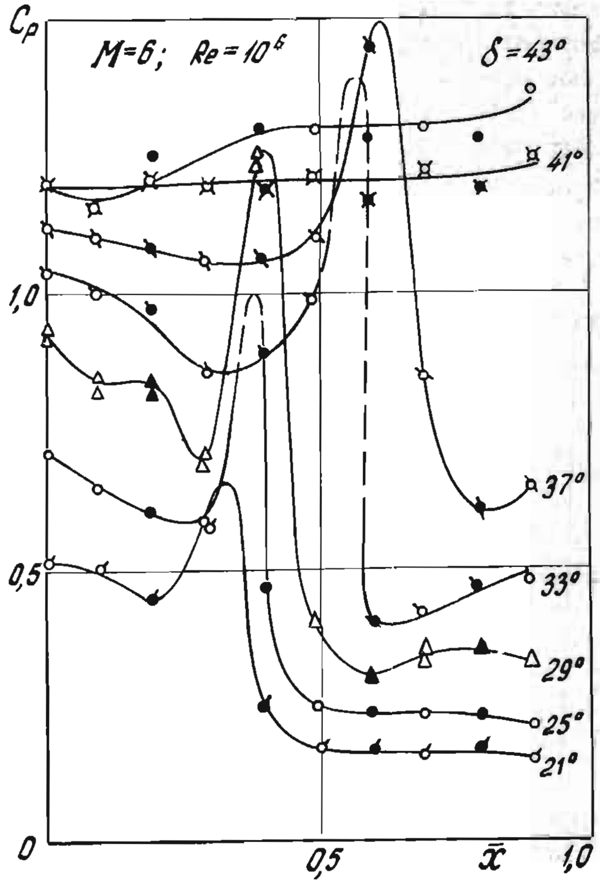


FIGURE 4. Pressure in the corner at off-design conditions.

(2) <u>The second exact solution for the corner flow</u> corresponds to the vicinity of the line of intersection in space of two oblique plane shocks. The shape of the reflected shocks and parameters of the flow behind them can be calculated in the plane normal to the intersection line. The problem is reduced to regular reflection of the oblique shock from the wall and has two solutions. The first solution realizing in two-dimensional flow corresponds to the weak reflected shock, the second corresponds to the "strong" shock in the above plane. In three-dimensional

flow for certain domain of defining parameters both these solutions correspond to the weak shock for supersonic velocity behind it.[12] For M=5 this domain is shown in fig. 5. θ_1 and θ_2 are angles between the incident and the reflected shocks and plane of symmetry, γ is the angle between the velocity vector ahead the shock and the intersection line,

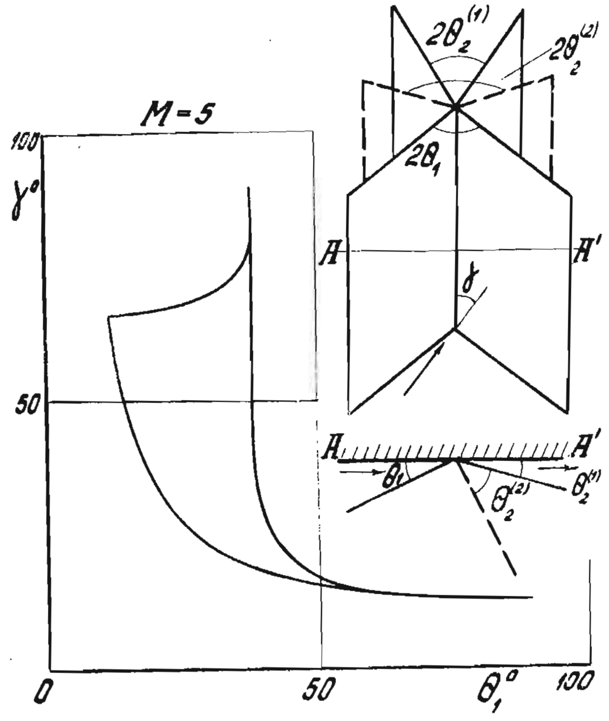


FIGURE 5. The domain of two solutions for intersecting in space plane shocks.

Pressure behind reflected shocks in the second solution is considerably higher (fig. 6).

In two-dimensional flow both solutions correspond to the same wall, on the contrary in three-dimensional flow the flow surfaces are different. If leading edges are rectilinear and their sweep - back satisfy the condition:

$$\cos\chi = \sqrt{1+\left[\frac{\cos(\theta_1+\theta_2)\cos(\theta_1-\delta')}{\cos\theta_1\cos(\theta_2+\delta')}ctg\gamma\right]^2}\sin\theta_1$$

$$\theta_1+\theta_2 > \pi/2 \qquad (1)$$

δ' - is the angle of flow deflection in shocks in calculation plane, than the second solution is the corner with panels normal to the reflected shocks.

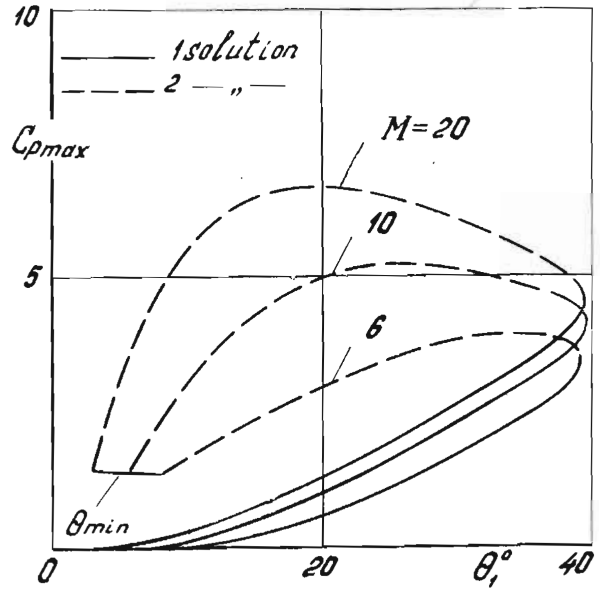


FIGURE 6. Maximum pressure behind four intersecting shocks.

Predicted pressure on inner panels for inviscid flow in such a corner (dotted line, fig. 7) correlates well with measured, though reflected shock induces separation ahead of it.

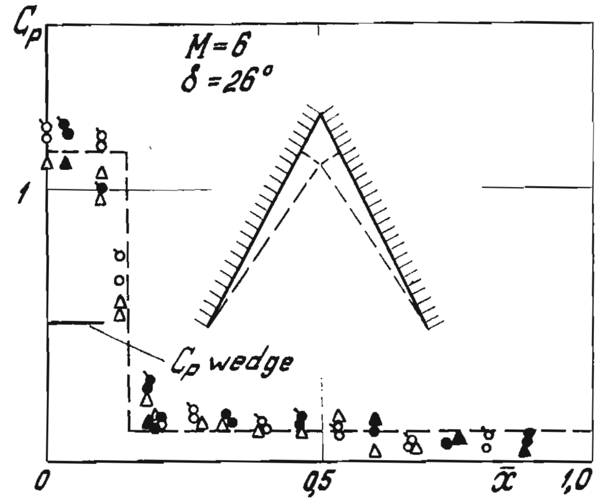


FIGURE 7. Comparison of measured and predicted (dotted line) pressure in the corner (four intersecting shocks).

The pressure in the vicinity of the ridge line of this corner is nearly twice that on the wedge with the same angle of incli nation ($\delta = 26^o$).

At angles of attack less than the designed $18^o \leq \delta \leq 26^o$ the flow with four three-dimensional intersecting shocks is mainteined (fig. 8). At $\delta > 26^o$, graphs of C_p and patterns of limiting streamlines show the second inner shock (fig. 8). Ahead of shocks on corner panels local separation occurs (dotted lines of divergence of limiting stream-lines, fig. 8), the beginning of which is the line of convergence (solid line). Flow pattern in fig. 8 is drawn from limiting streamlines obtained with the use of smeared dots.

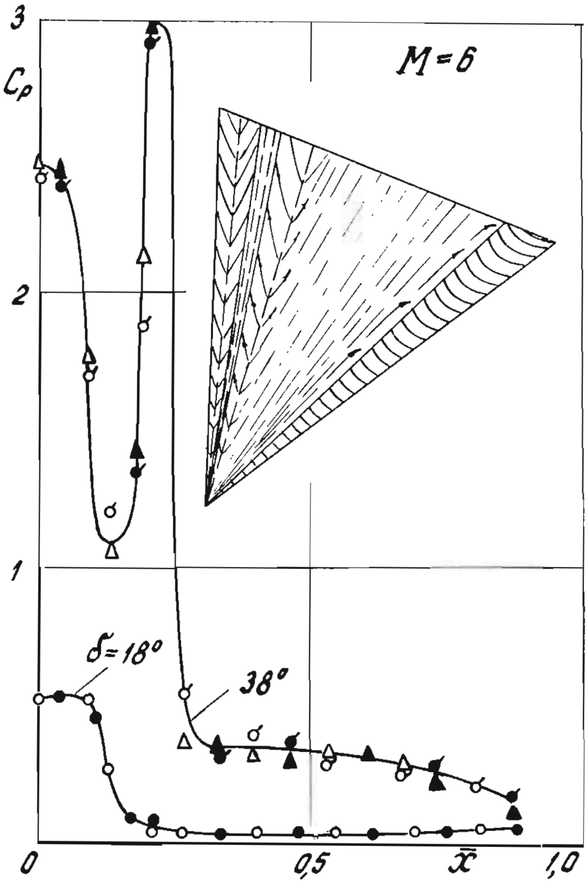


FIGURE 8. Pressure in the corner at off--design conditions.

Peaks of the heat transfer rate in the vicinity of inner shocks exceed the heat transfer rate near sharp leading edges (fig. 9). Heat transfer rate is maximum on leading edge and varies smoothly if there is only one shock in the flow and no inner shocks (fig. 9b). Heat transfer rate was measured with the use of the thermosensitive paint.

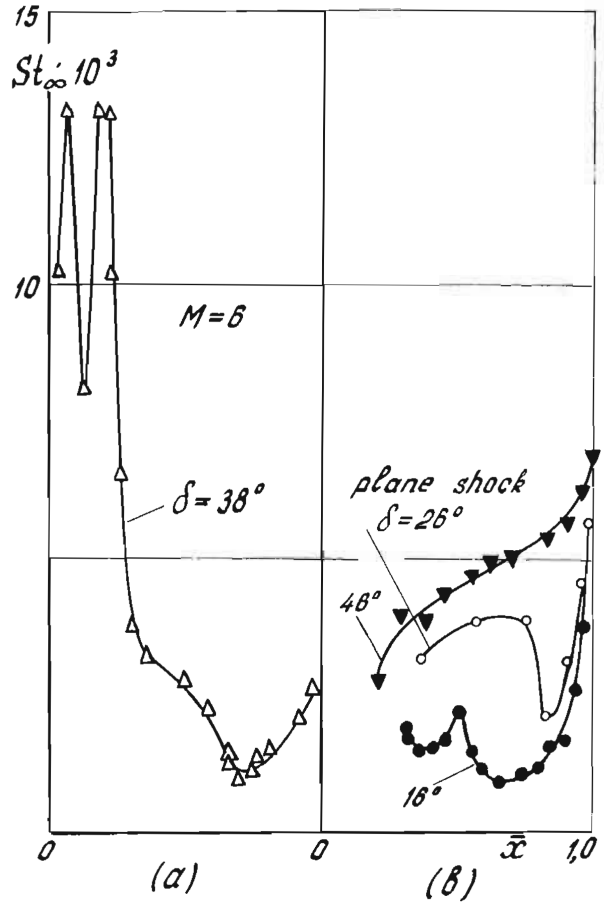


FIGURE 9. Heat transfer in the corner on the regime with (a) and with-out (b) inner shocks.

3. Lift-to-Drag Ratio of the Bodies Supporting Two Plane Shocks

Bodies supporting one plane shock have the same lift-to-drag ratio as the wedge. As a result of interference a body supporting two plane shocks intersecting on the outer edge can have lift-to-drag ratio higher, than that for the wedge with the same lift coefficient.

Attempts have been made previously to solve this problem using linear theory (13). Exact analysis of such flows with the aid of plane shocks is much simpler and gives results in general form. The body considered and shocks in the plane normal to middle edge are shown in fig.10

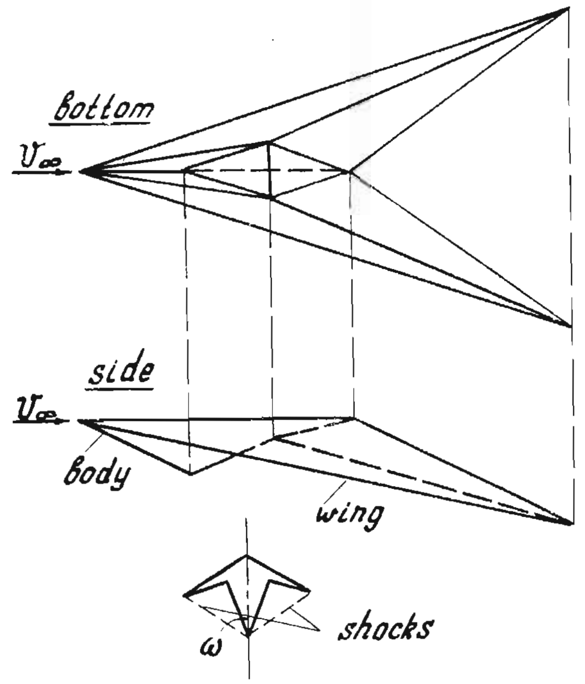


FIGURE 10. Body with two plane shocks.

Let us name the part of this body bounded by panels in the vicinity of symmetry plane a "body" and two lateral panels of lower surface - wings; in particular case wings may be placed in one plane. Upper surface of the body may be formed by the undisturbed flow streamsurfaces or may create expansion. Strength of the shocks (an angle of the equivalent wedge δ) and the angle between their planes 2ω given, panels of the body are determined, but leading edges of the wing can move in the planes of shocks and angle V between the panels of the body and the wing will vary. Minimum value of this angle corresponding to designed flow is attained when wing panel is normal to the shock plane:

$$ctg\, V_{min} = \sqrt{(tg\,\omega / tg\,\theta)^2 - 1}\; Sin(\theta - \delta) \qquad (2)$$

Designed flow is maintained if disturbances from trailing edge do not reach nearest panel. The boundary of influence region of the rectilinear trailing edge intersects the nearest panel on the straight line and angles between trailing edges of the body (σ) and the wing (σ') and the line of their intersection must satisfy unequalities symmetrical to σ and σ' .

$$ctg\,\sigma' \geq cos\,V\, ctg\,\sigma - \sqrt{ctg^2\beta - ctg^2\sigma}\; Sin\,V$$
$$if \quad cos\,V \geq -tg\,\beta\, ctg\,\sigma, \qquad (3)$$

or

$$ctg\,\sigma' \leq cos\,V\, ctg\,\sigma + \sqrt{ctg^2\beta - ctg^2\sigma}\; Sin\,V$$
$$if \quad cos\,V \geq tg\,\beta\, ctg\,\sigma, \qquad (4)$$
$$\sigma' \leq \pi - \beta, \quad if\; cos\,V < -tg\,\beta\, ctg\,\sigma$$
$$\sigma' \geq \beta, \quad if \quad cos\,V < tg\,\beta\, ctg\,\sigma, \qquad (5)$$

β - is Mach angle in flow behind the shock.

For a symmetrical wave rider with one shock

$$\beta \leq \sigma = \sigma' \leq \pi - \beta$$

For the case of the strong shock in a plane normal to the leading edge of the wing or the mean edge plane the domain of the designed flow on the panels is limited; for the case of the weak shock it is unlimited.

Lift-to-drag ratio of the considered bodies increases with an increase of the wing panel area related to the body panel area.(3) Maximum value of lift-to-drag ratio for bodies with upper surface parallel to undisturbed flow and corresponding angles of the trailing edges is equal.

$$\left(\frac{L}{D}\right)_{max} = \left[\sqrt{1-\left(\frac{cos\,\omega}{cos\,\theta}\right)^2} + \frac{cos\,\omega}{cos\,\theta\, cos\,\delta}\, ctg\,\lambda\right] ctg\,\delta \qquad (6)$$

$$\cos\lambda = tg(\theta-\delta)\,ctg\beta\,,$$

$$V_{min} \leq V \leq V_{min} + \lambda\,, \qquad (7)$$

$$ctg\,\sigma = \cos(V_{min}+\lambda)\,ctg\beta$$

$$ctg\,\sigma' = \cos(V_{min}+\lambda-V)\,ctg\beta\,. \qquad (8)$$

Unequality (7) is satisfied if the shock is strong in the plane normal to wing leading edge. For a given strength of the shock maximum value of lift-to-drag ratio is obtained when $\omega = \omega_o$:

$$\frac{\cos\theta}{\cos\omega_o} = \sqrt{1 + \frac{tg^2\beta - tg^2(\theta-\delta)}{tg^2(\theta-\delta)}\cos^2\delta} \qquad (9)$$

and is equal to

$$\left(\frac{L}{D}\right)_{\omega_o}^{max} = \sqrt{1 + \frac{tg^2(\theta-\delta)}{[tg^2\beta - tg^2(\theta-\delta)]\cos^2\delta}}\;ctg\,\delta \qquad (10)$$

and always exceeds lift-to-drag ratio of wave-rider supporting one shock of the same strength $(L/D = ctg\,\delta)$. In fig. 11 the relation of maximum lift-to-drag ratios of bodies supporting two shocks and one shock is shown as function of lift coefficient $C_L = C_p$ (lift is related to the projection of lower surface on the horizontal plane).

Decreasing strength of the shock

$$\lim \left(\frac{L}{D}\right)_{\omega_o} max.\,tg\,\delta = \infty\,.$$

If skin friction is taken into account, this limit is finite. Results of computations for M=6, δ =10° and laminar boundary layer are shown in fig. 11.

Minimum value of lift-to-drag ratio for bodies supporting two plane shocks is equal to

$$\left(\frac{L}{D}\right)_{min} = \left[\sqrt{1-\left(\frac{\cos\omega}{\cos\theta}\right)^2} - \frac{\cos\omega}{\cos\theta\cos\delta}\,ctg\,\lambda\right] ctg\,\delta \qquad (11)$$

$$for\quad \frac{tg\,\theta}{tg\,\omega} \leq \sqrt{tg^2\beta - tg^2(\theta-\delta)}\,\cos\beta\,,$$

or

$$\left(\frac{L}{D}\right)_{min} = \sqrt{(tg\,\omega/tg\,\theta)^2 - 1}\;\cos\omega \qquad (12)$$

for

$$\frac{tg\,\theta}{tg\,\omega} > \sqrt{tg^2\beta - tg^2(\theta-\delta)}\,\cos\beta\,.$$

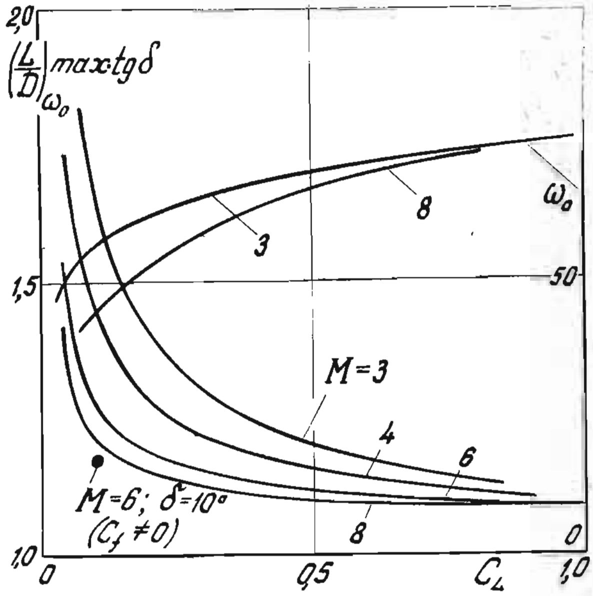


FIGURE 11. Maximum lift-to-drag ratio for bodies with two and one plane shock.

Region of lift-to-drag ratio variation for bodies considered with varying trailing edge is shown in fig. 12. Lift-to-drag ratio for trailing edge disposed in the plane normal to undisturbed flow direction is shown in a dotted line:

$$\left(\frac{L}{D}\right)_1 = \sqrt{1-\left(\frac{\cos\omega}{\cos\theta}\right)^2}\;ctg\,\delta\,. \qquad (13)$$

In computations base pressure was assumed equal to the pressure in undisturbed flow. Base drag reduction and increase of volume can be obtained by extending afterbody which does not influence the flow over the wing.

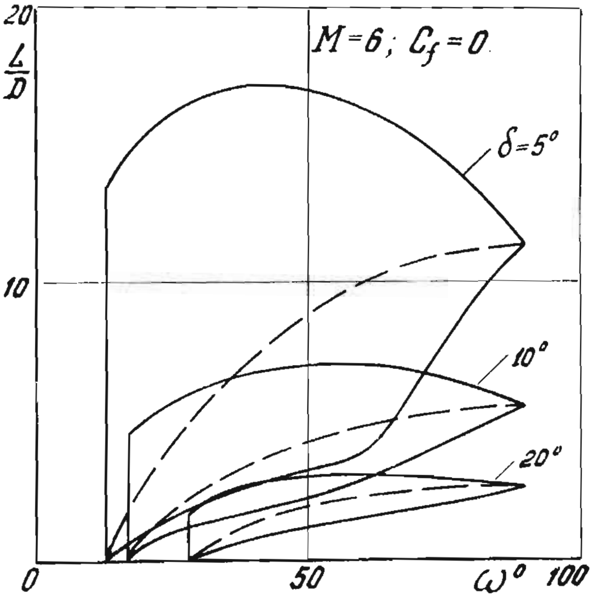


FIGURE 12. The rang of lift-to-drag ratio for bodies with two plane shocks.

Summary

Examples of supersonic flow in a corner show that:

(1) On leading edges of bodies in three-dimensional flow shocks, corresponding to strong shocks on the wedge in normal plane, are realized.

(2) Reflected shocks behind two plane shocks intersecting in space can be strong when considered in plane normal to the line of intersection. Pressure behind these reflected shocks is higher that that in two-dimensional flow.
Such flows can be realized if velocity behind shocks is supersonic. Realization depends on boundary conditions downstream.

(3) Inner shocks in the corner flow induce local separations and peaks of heat transfer rate.

(4) Lift-to-drag ratio of the bodies supporting two plane shocks intersecting on the outer edge can be higher that that of the wave-riders of the same lift supporting one plane shock.

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