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Development of the Technique of Sonic Boom Experimental Researches

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Annotation

The possibilities of the experimental-numerical technique improving of sonic boom estimation are analyzed. This technique is based on the gasdynamic flow characteristics measuring at near aircraft model flow-field in the wind tunnels (WT). First it is measured the disturbed pressure at the model near-field at the WT test section. Then obtained data are recalculated for the large distances by using the quasi-linear theory. For pressure definition at model near-field it was suggested the method that used the luminescent pressure transducers (pressure-sensitive paints, PSP). The PSP technique is based on the phenomenon of fluorescence quench of the organic luminophore by using the air oxygen. To realize this method the surface under investigation is covered with the special paint like thin polymer layer. This thin layer is permeable for the oxygen and contains the luminophore molecules. The luminophore becomes excited by the light of corresponding wave length and the fluorescence intensity or the life time of excited molecules were measured. The fluorescence intensity and molecule life time are inversely proportional to the pressure. It appears that at the distances from the body about 5-10 aircraft length the flow becomes significantly simplified. Three dimensional flow near the aircraft is transformed into a flow over the equivalent body of revolution. So the initial data for sonic boom calculation should be obtained using the Yu.L. Zhilin's theorem. According to it the derivative of cross-section area of equivalent body of revolution can be defined by the integral of the disturbed pressure coefficient at the aircraft near-field along the certain surface. This surface should be placed at the small distance about 0.1–0.5 aircraft length from it. The results of pressure distribution measuring at aircraft model near-field performed at TsAGI wind tunnels are presented. The measured pressure distributions as a rule possess the irregular perturbations caused by the experiment peculiarities. The last ones can distort the results for Zhilin's integral obtaining. It was supposed the special procedure of handling of pressure measuring results obtained by using the PSP. This procedure based on wavelet analysis methods allows to decrease the random noise influence and to rise the accuracy of pressure field integration in order to define the equivalent body of revolution form. The possibility of construction of measuring system based on the PSP method for the investigation of aircraft model near-field at the large industrial supersonic TsAGI wind tunnel T-109 was considered.

Keywords: *sonic boom, wind tunnel, pressure sensitive paints, numerical investigation*

1. Introduction

At the present time the leading countries in the world continue the works to design the second generation of supersonic passenger aircrafts. These works are performed for two directions: to create the supersonic transport of large seating capacity (SST) and to construct the supersonic business jets (SSBJ) for about 6-15 passengers.

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The SST/SSBJ development is connected with the number of ecology problems. These problems are the decreasing of engine emission, the diminution of the Earth land nose and the minimization of sonic boom effect. In accordance with the specialists opinion the last problem is the main obstacle at the way of designing the new generation of supersonic passenger aircrafts [1].

Negative effect of sonic boom (SB) on persons and animals (psychological and physiological) and on the buildings (destructive) is led to set the restriction on acceptable overpressure at the shock wave. In the USSR after the beginning of Tu-144 exploitation it was accepted the state standard (ST-ST № 23552-79) that allow the commercial supersonic aircraft flights above the populated areas. It was designated that the acceptable SB level was the level of 90 Pa with the possibility of about ±20 Pa deflection for the real atmosphere condition.

Now it is descended the toughening of ecology requests to the future generation of aviation technics.

International standards imposed the restrictions on sonic boom intensity are not accepted until now. This obstacle is complicated the situation. The discussions devoted to sonic boom problem are debated during some decades. Thus the International Civil Aviation Organization (ICAO) prescribed to the aviation companies to avoid «unacceptable situations for the population» connected with sonic boom (ICAO resolution A33-7, 1998). ICAO Council must take reasonable steps to elaborate the international convention for sonic boom measuring, to give the qualitative and quantitative definition of the expression «unacceptable situations for the population» and to set up the corresponding limits. Because of the supersonic aircraft certification concerning to airworthiness requirements can take place since 2020-2025 years the ICAO 2called on the countries that can be considered as the producers of supersonic aircrafts to present the suggestions for ICAO technical requirements fulfilling (ICAO resolution A39-1-G, 2016).

The requirements to the aircraft layout aimed at sonic boom minimization and the requirements aimed at lift-to-drag ratio increasing are usually at variance one with other. That lead to problem of aerodynamic and, correspondingly, economic efficiency of the aircraft.

At the present time there are investigated as traditional (passive) methods for sonic boom decreasing [2] based on aircraft shape changing so as active methods of immediately impact on the flow field [3].

There are different methods of sonic boom phenomenon investigation:

- numerical methods;
- experimental researches in terrestrial aeroballistic mountings and in the aerodynamic stands of short-time or periodic running;
- flying (natural) testing.

As the development of the calculation technique progresses the numerical simulation is widely spread. But it should be noted that during the numerical investigation aimed at sonic boom decreasing there are a number of difficulties can be arisen. These problems are connected with the calculation region peculiarities, with difficult grid construction for three dimensional aircraft configurations and with some assumptions that are used in numerical methods. In accordance to it the numerical simulation of sonic boom phenomenon must be conducted by experimental investigations in order to obtain the trough information for the bodies of difficult forms. It is necessary for validation of numerical calculation results also. Flying researches are rather expansive and their results depend on the atmosphere conditions at the testing moment. The wind tunnels (WT) are widely used for investigating of formation and spreading of sonic boom wave. With the aid of WT it is possible to consider the fundamental problem of interaction, propagation and diagnostic methods of weak shock waves at the near aircraft flowfield.

At the Institute of Theoretical and Applied Mechanics of Siberian Branch of Russian Academy of Sciences (ITAM SB RAS) it was developed the hybrid experimental-numerical technique for sonic boom phenomena modeling [4, 5]. Experimental part of this method is based on the simulation of near sonic boom zone in the WT by using aircraft models. As a result it was obtained the pressure distribution on the some control surface, remote from the test model at the distance about its length. The evolution of measured distributions of disturbed pressure during propagation for the large distances is defined by numerical methods based on the quasi-linear theory [6]. For the big azimuth gradients of disturbed flow at the model near-field the calculation is performed with taking into account the flow three-dimensional behavior [7].

At Central Aerodynamic Institute (TsAGI) there were fulfilled the works devoted to the problem of adoption and development of experimental-numerical technique for sonic boom phenomena simulation [4, 5] at large industrial WT. First this method was adopted at TsAGI T-113 and T-112 wind tunnels.

With this purpose, a special stand was designed and manufactured on T-113 and T-112 WT for the realization of experimental-numerical technique for sonic boom phenomena investigation. Instead of coordinate device with probe used in ITAM SB RAS or special detailed drained plate it was made an attempt to measure the static pressure fields on the measuring plate surface by using the luminescent pressure transducers. This plate is located near the lateral wall of WT working part and the measuring was performed PSP method [8, 9]. Using the PSP method must provide the continuous pressure distribution on the control surface at the model near-field. Then using this pressure distribution it should be calculated Zhilin's integral [10] and the profile of sonic boom wave at the large distances is to be defined also using the quasi-linear theory.

The present work focused on the development of experimental-numerical technique of sonic boom investigation at large industrial TsAGI supersonic WT. This paper, in turn, is the continuation of the articles [8, 9] devoted to utilizing of the high-informative technologies continuous data obtaining. These data are the results of distributed flow parameters at model near field.

2. The description of experimental equipment and testing technique

The most effective using of ITAM SB RAS experience in the region of sonic boom investigation is possible at two step method of approaching to the given problem. At first base service and development of this technique was performed at T-113 and T-112 TsAGI WT with the test section sizes of 0,6×0,6×2,0 m. Then for the experimental investigations of sonic boom spread from perspective SST/SSBJ the advanced technique is supposed to be used at large supersonic WT T-109 (working part sizes are 2,25×2,25×5,5 m).

For the wind tunnels T-113 and T-112 it was constructed and made the special stand for the realization of experimental-numerical technique for sonic boom phenomena investigation. The experimental researches were provided at WT working parts for two models generated the near field. One of them is a supersonic aircraft schematized model (the model has a delta wing) and another is the «cone-cylinder» axisymmetrical configuration. The tests were provided for Mach numbers M = 1,75; 2,0 and 2,25 at T-113 and for M = 1,75 at T-112.

At the present paper the others demonstrated the results of the experimental investigation of aircraft schematized model with delta wing performed at T-113 TsAGI WT. This wind tunnel is an analogue of T-313 WT for which the experimental-numerical technique in ITAM SB RAS was developed [4, 5]. The above mentioned technique then was used at the researches of different methods efficiency aimed at sonic boom level decreasing. The nozzle inserts of T-113 WT allow changing the Mach number from 1.77 to 6.0 discretely.

Experimental investigations were fulfilled as for the unloaded working part in order to define the background pressure fields and so for the model generating the waves (Fig.1). Model length L is 247 mm. For this supersonic aircraft model it was made the special bent holder that supported the model location at the fixed angle of 4° (without taking into account the deformations induced by the aerodynamic loading.

Pressure distributions are measured on the special flat plate that is located on the plugs of the optical windows using streamlined supporting arms. In order to obtain continues data of pressure flow field it was used the luminophore coating (PSP) painted on the plate surface.

The PSP technique is based on the phenomenon of fluorescence quench of the organic luminophore by using the air oxygen. To realize this method the surface under investigation is covered with the special paint like thin polymer layer. This thin layer is permeable for the oxygen and contains the luminophore molecules. The luminophore become excited by the light of corresponding wave length and the fluorescence intensity or the life time of excited molecules were measured. The fluorescence intensity and molecule life time are inversely proportional to the pressure.

Fig. 1. The supersonic aircraft schematized model

In order to excite the fluorescence it was used the ultra violet light-emitting diode (LED) lamp, but for recording time life of fluorescence there were used two CCD (device with charge connection) cameras with multi exposition mode «VIDEOSCAN-V2-285/P1-33». The UV LED lamp and two cameras were located at lateral WT window, bur measuring plate was mounted inside the flow at the opposite window at the distance of 100 mm from the lateral wall. The equipment location at WT is shown in Fig. 2.

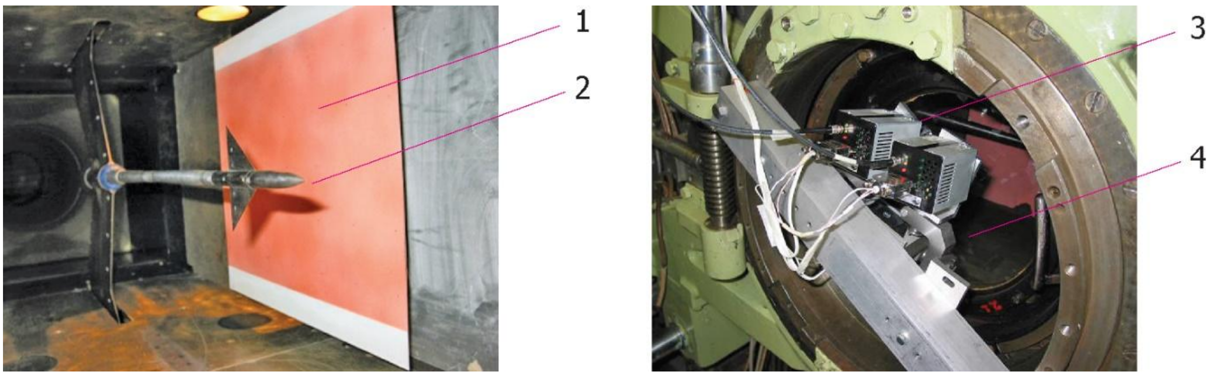

Fig. 2. The equipment location at WT. *1* — measuring plate, *2* — model, *3* — CCD cameras and UV LED lamp, *4* — WT lateral window

The device with charge connection transforms the light energy to the electric one. Two cameras are need for the fluorescence life time measuring. At the measuring moment the LED lamp scintillates during 100 mcs with the period of 220 mcs (4,54 kHz). At the same time one of the cameras accumulates the covering glowing intensity during the flash (first 50 mcs), but another camera does it after the flash during 110 mcs.

To control the measured pressure on the plate, nine static pressure holes were provided with the output of pneumatic lines to the standard measurement system. Such technique allow to obtain continues pressure distribution on the test plate that was located outside the viscous layer. Energetic expenses and the time needed for the experiment were sufficiently decreased also.

Because of the specific orientation of the optic equipment (the optical windows are located on the lateral walls only) the coordinate system was rotated on the angle $\gamma = 90°$. Thus the plate of nine static pressure holes and two temperature sensors with the bar indicating coating that was conditionally located under the model was mounted on the WT right wall. In this case instead of optic window it was inserted the special plug to which the supporting pylons were anchored (see Fig. 3).

The plate sizes are 510×500 mm. The static pressure holes are started from the section x = 166 mm from the plate beginning along the stream wise in the middle of the plate with the step of 21 mm. The pressure was measured by absolute pressure sensors and these data were duplicated by pressure difference sensors.

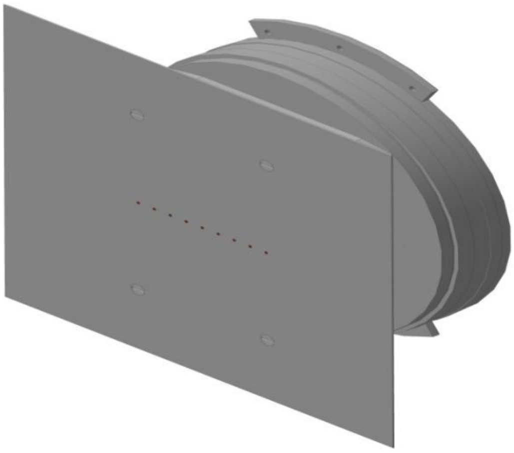

Fig. 3. Drainage plate on the plug

The design of the experimental stand in the T-113 WT was carried out with the use of modern software for computational gas dynamics. The main purpose of the mathematics simulation is the calculation of flow structure of near model flow field and acquisition of the data concerned to the overpressure distribution that was generated on the measuring plate. The modeling was fulfilled by using computational software complex ANSYS CFX [11] (TsAGI license №501024) based on the solution of the Reynolds averaged Navier-Stokes equations.

3. Numerical research method describing

The numerical methods are till now the main methods for the investigation of sonic boom propagation. In order to define sonic boom intensity you can calculate the whole process of disturbances propagation in the inhomogeneous medium using Euler or Navier-Stokes equations. It is possible theoretically but practical realization of such technique has no base at the present time. The reason of it is the scaled factor between the aircraft length L (about 50 m) and the flight height H (about 15 000 m). This factor as it seen equals to H/L = 300 for the typical supersonic aircraft. In this case the disturbances asymptotic behavior along the azimuthal plane coming through the longitudinal axis of the aircraft (see Fig. 4) will help to sonic boom calculator.

It appears that at the distances from the body about 5–10 aircraft length the flow becomes significantly simplified. Three dimensional flow near the aircraft is transformed into a flow over the equivalent body of revolution that is individual at each other azimuthal plane. The disturbed pressure field induced by this equivalent body of revolution can be considered as the initial data for sonic boom calculation using nonlinear acoustic method. Meanwhile these initial data will be different for each plane θ=const. The difference of these data contains the information of three dimensional flow over the aircraft.

Thus an idea of equivalent body of revolution is simplified the task of definition of sonic boom intensity. The problem now may be turned to the solving of the ordinary differential equation system for each azimuthal plane independently. The initial data for the plane θ = const can be obtained by calculating the three dimensional flow field at the distance of 5-10 aircraft length. Sometimes the sonic boom researchers go by this way especially at the stage of results final control. This task is more simple that the task of the three dimensional disturbed flow definition during it propagation from the aircraft to the Earth. But this procedure is rather expensive yet from the point of calculation time. Therefore it will be more effective to use the theorem, proved by Yu.L. Zhilin [10]. In accordance with this theorem the equivalent body of revolution can be defined by using the data of the aircraft near flow field (Fig. 5).

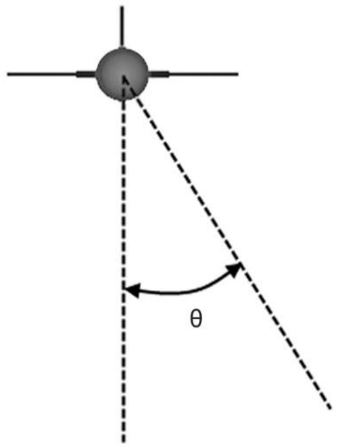

Fig. 4. Front view of the flow field disturbed by the supersonic aircraft

The parameters of the disturbed flow at the nearest zone were obtained by using the numerical code [12] based on the Euler equation system solution [13, 14]. During the numerical program development [12] there were used the algorithms that corrected the solving for the big values of aerodynamic parameters gradients. Then for the large distances the recalculation was made using Zhilin's method [10] based on the quasi linear theory [15]. In accordance to it the task of sonic boom intensity definition can be considered as the flow field near the certain equivalent body of revolution.

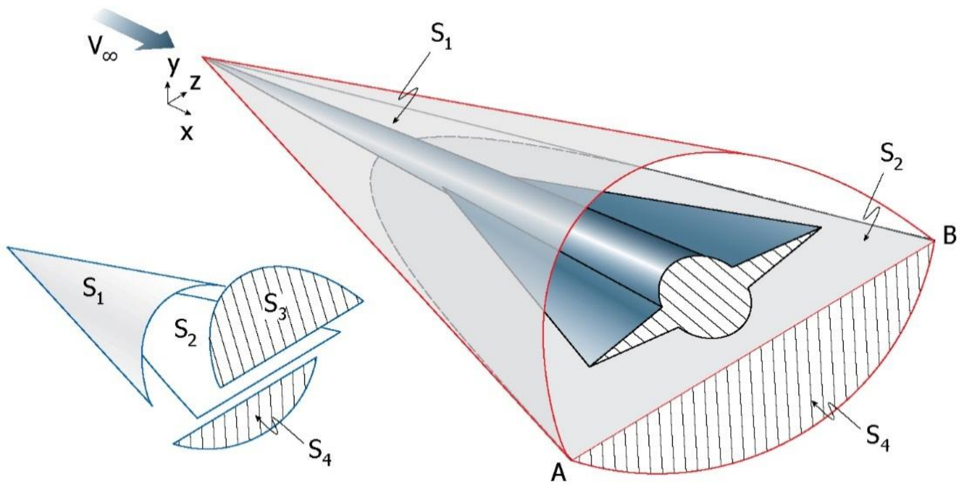

Fig. 5. Control volume for the supersonic flow about the aircraft

Yu.L Zhilin demonstrated [10] that the asymptotically far solution at some assumptions can be connected with the integral at the near zone on the certain surface S_2 located at 0.1-0.5 aircraft length L from the aircraft itself.

Zhilin's integral is defined by the formula:

$$\frac{dS_{\mathrm{eq}}(x_0)}{dx_0} = -\frac{1}{V_\infty}\int_A^B (\beta u + v)dz = -\frac{2\beta}{V_\infty}\int_A^B udz = -\frac{2}{V_\infty}\int_A^B vdz\ .$$

Here u and v are the components of disturbed velocity vector along the x and y axis. Taking into account that the disturbances are not large in accordance with the linear theory it formula can be rewritten using the pressure coefficient instead of velocity component:

$$C_p = -\frac{2u}{V_\infty}.$$

Now Zhilin's integral can be presented as:

$$\frac{dS_{\mathrm{eq}}(x_0)}{dx_0} = -\frac{2\beta}{V_\infty}\int_A^B u\,dz = \int_A^B \beta C_p\,dz\,.$$

Here x_0 is the coordinate of the cross-section of the S_3 plane with the x axis, V_∞ is the free stream velocity and parameter $\beta = \sqrt{\mathrm{M}_\infty^2 - 1}$.

For the random direction the coordinate system (x, y, z) can be chosen so as the S_2 plane and AB line segment would be perpendicular to the y axis.

Thus in order to define the derivative of cross-section area of the equivalent body of revolution you just need to integrate the pressure coefficient with some multiplier. The integration process can be exchanged by the summarizing on the uniform grid.

The general peculiarity of Zhilin's integral is the opportunity to change the distance between the S_2 plane and the x axis at fixed location of S_3 plane. For example, the S_2 plane can be located at the sufficiently far distance (R) from the body so that the main postulates of the linear theory can be valid (R ~ 0,3...0,5 L) but at the same time much closer than it is required to establish the asymptotic behavior of the solution (R ~ 3...5 L).

Sonic boom can be calculated on based of the cross-section area derivative of the equivalent body of revolution *dS/dx.* Another way is to use the pressure distribution at the initial sound wave. The first technique must give the more accurate results because the moving of about the 200 mm and more from the model can provide non valid pattern of the three dimensional flow.

4. Main results

The measured pressure distributions possess the essential irregular perturbations caused by the experiment peculiarities. The last ones can distort the results for Zhilin's integral [10] obtaining. As mentioned above, this integral is used to calculate the sonic boom in the middle and far zones. The main reasons of distortion of the useful signal are the shock waves interaction with boundary layer on the plate and flow nonuniformity proper to the WT. It is also should be noted the secondary disturbances caused by the model that are not appear without model at the empty WT test section and thus they are not fixed in «background» pressure distribution. In order to get more accurate results the data of the unloaded wind tunnel were subtracted from the results recording by the control measuring plate. That is assumed that the interaction of the disturbed flow generated by the model with «background» pressure distribution on the measuring plate (without model) occurs by the linear law. This assumption is rather accuracy in the case when the useful signal is by an order of magnitude more than «background» one [16].

Then, using Zhilin's integral, there were made the calculations of the derivative of the cross-section area of the equivalent body of revolution and of the pressure coefficient at the middle section of the measuring plate. Due to flow symmetry the integration process of the disturbed pressure distribution was performed for the upper half of the results only because at the lower half it is present the band of zero values conditioned by the holder shadow (Fig.6a). The procedure of «background» values of pressure distribution subtracting allows to improve the calculation results of the derivative of the cross-section area of the equivalent body of revolution.

In order to decrease the impact of random noise the based on signal wavelet analysis [17] algorithm for processing the data of pressure distribution measurements obtained by using the PSP was realized. At the present time the wavelet transformation is widely used for the tasks of handling and encoding of the different signals and images. Particularly the wavelet analysis gives to the researcher a number of unique possibilities to recognize the local and «thin» signal features. The given algorithm allowed to conserve the main local features of pressure signal (as shock waves). At this case it is succeed in decreasing sufficiently of the common noise level (Fig. 6b) and in improving the initial data for sonic boom calculation.

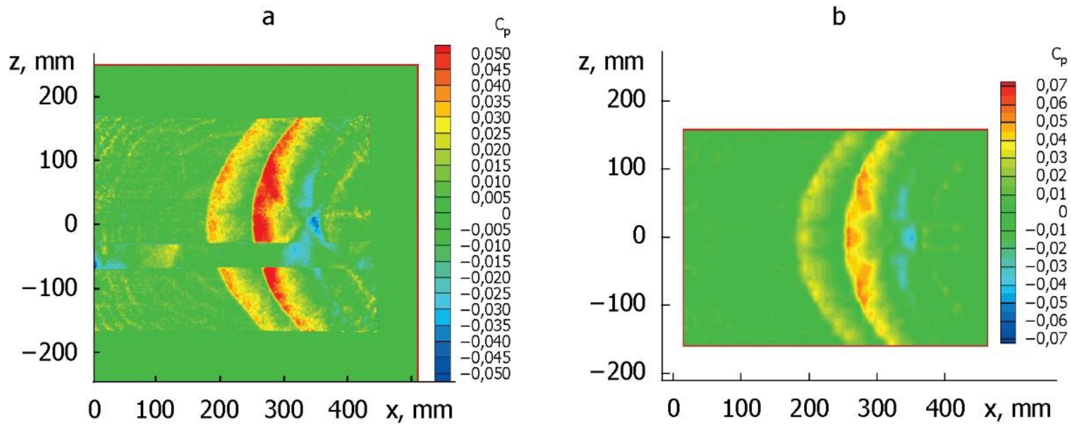

Fig. 6. Pressure coefficient distribution on the measuring plate at T-113 WT at the testing of aircraft schematized model at Mach number M = 2 with subtraction of unloaded WT data (a) and after using the smoothing procedure based on the wavelet analysis (b)

In the Fig. 7*a* it is shown the distribution of the derivative of the cross-section area $\frac{1}{L}\frac{dS_{eq}}{dx}$ of the equivalent body of revolution for aircraft schematized model at Mach number M = 2. This distribution is obtained by integrating of the initial one with subtracting of «background» data for the empty (unloaded) WT (blue curve) and by the integrating of the smoothing (red curve) pressure coefficient distribution.

The influence result of the given smoothing algorithm using on the ground sonic boom intensity is illustrated in the Fig. 7*b.* In this picture you can see two diagrams of sonic boom wave on the ground calculated for two variants of the derivative of the cross-section area of the equivalent body of revolution which were shown in the Fig. 7*a.* These calculations were made for the aircraft of geometric conformity to the given schematized model during the flight of the height of 16000 m. The results difference is corresponds to the difference in the initial pressure coefficient distributions (Fig. 6).

The described procedure of measuring data handling is used then at the definition of sonic boom intensity on the Earth ground.

The analysis of the series of performed tests allowed to state of the recommendations for changing a number of the bench geometric parameters. These changings are ought to take into account at the developing of this experimental-numerical method of sonic boom investigation into the large supersonic settings.

Let us consider the disturbed flow field on the control plate at Mach number M = 2 (Fig. 8). The model length L = 247 mm. Its silhouette is painted on the plate for the better obviousness. As it is seen at the mutual consideration of the plate and the model the large part of the disturbances zone goes beyond the scope of the measurements (blue zone in the Fig. 8 denoted by the digit *I*). This unpleasant problem can distort the sonic boom wave calculation [18]. For another's Mach number the situation is the same.

Zhilin's integral is calculated on the base of measuring plate data in the direction across to the stream wise. In this case it is necessary that the disturbances outgoing from the model will fall on the measuring plate in the interval of at least along the model length L. In preference this zone is to be about 1,5 L long.

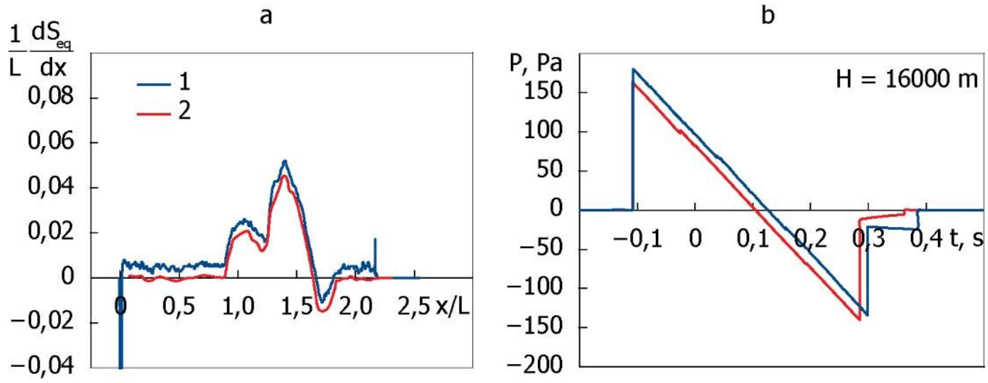

Fig. 7. The distributions of the derivative of the cross-section area of the equivalent body of revolution $\frac{1}{L}\frac{dS_{eq}}{dx}$ (*a*) and calculated on the base of these distributions profiles of sonic boom waves (*b*) for the aircraft model at Mach number M = 2: *1* — by using the initial C_p distribution with the subtracting data of the unloaded WT (Fig. 6*a*), *2* — by using the smoothing C_p distribution (Fig. 6*b*)

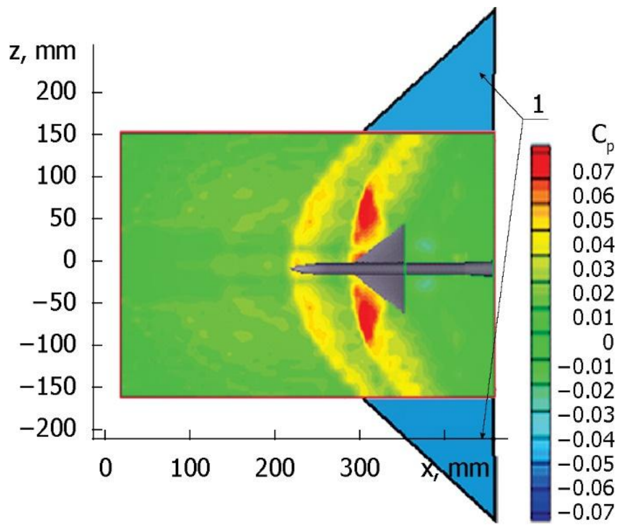

Fig. 8. WT working part and needed sizes of measuring plate; *1* — disturbances zones beyond the scope of the measurements

The optimal width of measuring plate is defined by the Mach cone coming from the model nose. Cone Mach section by the horizontal plane $y = y_0$ determines the disturbed region boundary equation on the measuring plate that is located at the distance of y_0 from the model. Now on the assumption of mentioned above the needed plate width Δz and optimal distance y_0 can be defined by the formulae:

$$\Delta z = 2\sqrt{(L + y_0\sqrt{M^2-1})^2 / (M^2-1) - y_0^2}\,,\ y_0 = \frac{\Delta z^2\sqrt{M^2-1}}{8L} - \frac{L}{2\sqrt{M^2-1}}\,. \quad (1)$$

These relations should be taken into account when designing an experimental stand for near-field pressure studies of promising SST/SSBJ models in large industrial WT.

Based on the results of methodological studies conducted in the wind tunnels T-113 and T-112, the scheme of the experiment with the measurements of the pressure in the near-field of the SST/SSBJ model in a large industrial supersonic wind tunnel TSAGI T-109 is elaborated (Fig. 9).

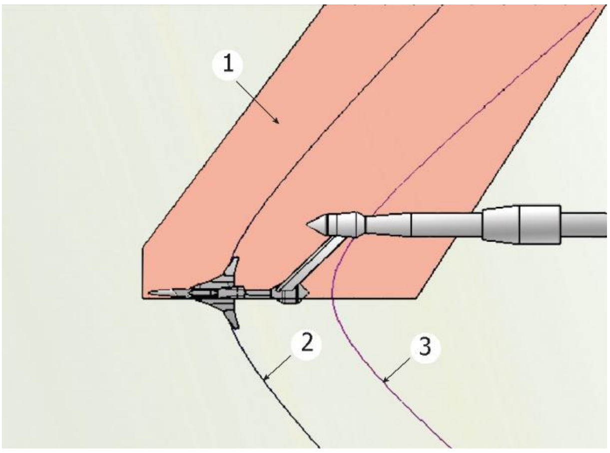

Fig. 9. The location of SST/SSBJ model, supporting devices and measuring surface of the test section (view from the left side): 1 — the area covered by the PSP; 2 — the front line of perturbations at M = 1.8; 3 — the tail line of the perturbation at M = 2.0

When developing a new experimental scheme, in addition to the imposed by the relationship (1), a number of additional requirements for the location of the SST/SSBJ model were also taken into account. The main of these requests are the absence of characteristic disturbed lines at the measuring zone and the providing of the «angles of view» of the cameras fixed the PSP-covering surface of not less than ±45°. At a new scheme it is considered the opportunity of using a lateral wall of the WT test section as a measuring plate.

For the previously estimations of boundary layers effect (on the control plate and on the WT wall) on the measuring results it was made the numerical simulation on the base of the program complex ANSYS CFX. The simulation was performed for the «cone-cylinder» supersonic model for the experiment condition at the T-113 WT (Fig. 10). The free stream Mach number M = 2, but the relative pressure is equal to one atmosphere. There were considered 3 variants: the basic configuration — the model placed at axis of symmetry of WT without measuring plate; the basic configuration with plate - the model placed at axis of symmetry of WT with the measuring plate located at 0.231 m from the model; and the configuration with the cone located at 0.231 m from the wall of the WT test section.

The results of numerical simulation are shown that the boundary layer on the WT wall does not led to the more sufficiently distortions of the pressure distributions than the boundary layer on the measuring plate. This effect of the boundary layer is displayed as a certain deviation of the shock wave reflection type from the measuring surface in comparison with the regular type (the reflection coefficient is a few different from 2, see Fig. 11).

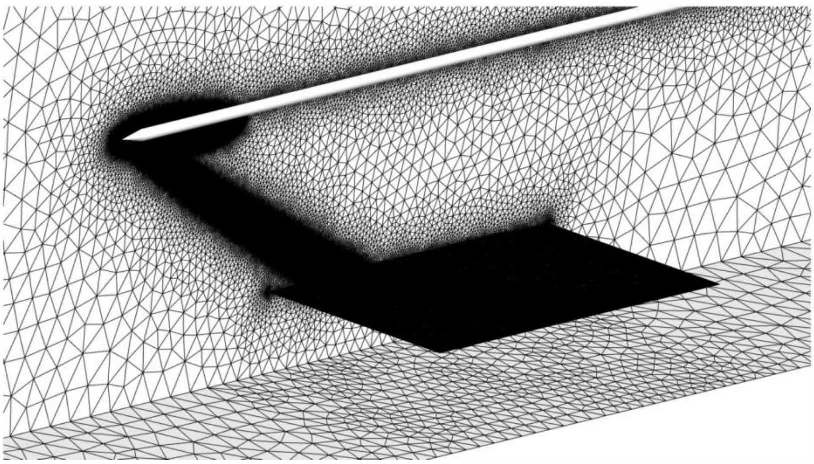

Fig. 10. Cone fixed by the holder at the WT T-113 and the calculation grid

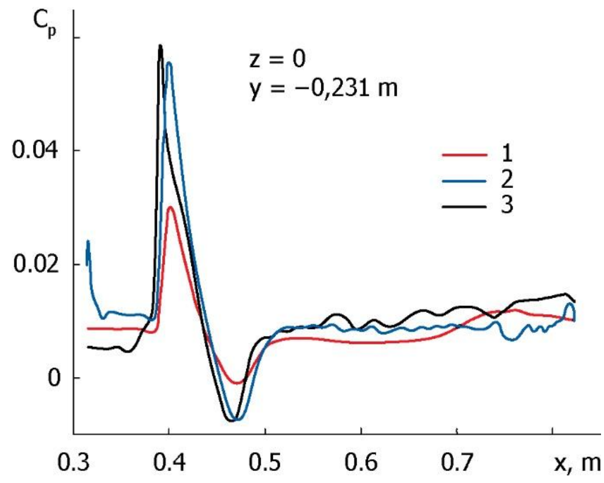

Fig. 11. The pressure coefficient distributions: 1 — basic configuration; 2 — basic configuration with plate; 3 — configuration with the cone located near the wall of the WT test section

5. Conclusions

There were analyzed the different possibilities of numerical-experimental technique improving. This technique is aimed at estimation of sonic boom based on flow gas dynamic characteristics measurements in the aircraft model near-field at the wind tunnel.

Using of the luminescent pressure transducers is provided in principle the getting the continues pressure distribution on control surface at the aircraft model near-field, permits to calculate Zhilin's integral on the base of this distribution and, finally, allows to define a profile of sonic boom wave at the large distances with the help of quasi linear theory.

It was developed the special procedure of handling of pressure distribution results obtained with the aid of pressure sensitive paints on the measuring surface at the wind tunnel. This procedure is based in turn on the method of the signal wavelet analysis. The utilizing of the above mentioned special procedure at the cases of the handling of the experimental results obtained at TsAGI T-113 WT allows to decrease the random noise effect and to improve the accuracy of pressure field integration in order to define the initial data (equivalent body of revolution form) for sonic boom calculation.

There were stated the recommendations for geometric parameters definition of the designing of bench for the investigation of supersonic aircraft model near-field using the pressure sensitive paints method. These recommendations would be taken into account at the stage of implementation of the experimental-numerical method of sonic boom investigation at the large industrial supersonic WT.

It is developed the experiment performing scheme aimed at measuring of the disturbed pressure field of SST/SSBJ model on the lateral wall of the WT test section.

Acknowledgments

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