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SUBSTANTIATION OF SCIENTIFIC AND METHODOLOGICAL APPARATUS FOR CALCULATING THE AERODYNAMIC CHARACTERISTICS OF UNMANNED AERIAL VEHICLE DURING FLIGHT AT SUPERSONIC SPEEDS IN ITS CONCEPTUAL DESIGN

The article presents the results of analyzing and generalizing the theoretical data on the method of disturbed pressures for performing estimates of the main aerodynamic characteristics of an unmanned aerial vehicle at supersonic speeds. The presented scientific and methodological apparatus are proposed for the calculated data on the main aerodynamic characteristics. The mathematical model presented is built for an ideal environment, so to take into account the viscosity forces, the paper synthesizes the mentioned mathematical model of linear aerodynamics of supersonic velocities, which is implemented by combining the method of disturbed pressures and engineering methods.

The basic relations and equations of the closed system of integral equations, the features of the proposed numerical implementation, and some results of test calculations are presented.

Keywords: *aerodynamic characteristics, disturbed pressure method.*

INTRODUCTION

The rapid development of unmanned aerial vehicles in recent years is directly related to the use of more and more advanced technologies in the war of the russian federation against sovereign Ukraine. It should be noted that this has become an obvious phenomenon not only in the aggressor country and our country, but also in other countries of the world. The rapid expansion of the nomenclature of unmanned aerial vehicles (UAVs) has made it possible to solve a wide range of combat and special tasks on the front line and in the depths of both territories. However, all currently known domestic and foreign models are subsonic aircraft, most of which are quadcopters with a maximum flight speed of about 100 km/h and extremely limited time and range of use, which is a consequence of the use of electric motors and limited capacity batteries.

At the same time, the experience of using UAVs in combat operations has shown that one of the main indicators from the list of technical characteristics among the determining ones, such as range, duration, weight of the warhead, effective scattering plane, hit probability, etc. is the flight speed of an unmanned aerial vehicle (UAV). High speeds allow UAVs to spend less time in the area of enemy air defense fire and reach their targets more quickly and in general, increase the effectiveness of their combat use.

An example of this is the positive experience of our side using the Tu-141 Strizh reconnaissance UAV as a carrier platform, equipped with a high-explosive fragmentation warhead and used as a kamikaze UAV at the airfield near the city of Engels, Saratov region of the russian federation. As a result of a surprise attack on the airfield, several strategic enemy aircraft were destroyed and damaged. And although the maximum speed of the mentioned UAV does not exceed the Mach number $M \le 0,9$ at low flight altitude, the efficiency of such a relatively high-speed aircraft was obvious.

It should be emphasized that in the technical literature and scientific publications, information has recently appeared regarding the intensification of work on the development of supersonic UAVs in some countries, including Ukraine [1, 2] for the performance of various tasks in operational-tactical and strategic operational areas zones of the depth of the enemy's defense construction. This formulation of the question implies in-depth research on the design of the optimal aerodynamic layout of future prospective models, and this is directly related to the need to study aerodynamic characteristics (ACs) at both subsonic and supersonic speeds, which must be available already at the stage of conceptual design and the possibility of operational to receive them to work out the composition of the sample with the required qualities.

If a sufficiently developed scientific and methodological apparatus (SMA) has been created for preliminary estimates of ACs at subsonic speeds [3−6], then for estimation calculations of ACs at supersonic speeds, which would allow numerical methods to obtain not only integral results [7], but also the spectrum of the main ACs at the maximum, including distributed pressures and derivatives of kinematic parameters [8] approaches, extremely limited.

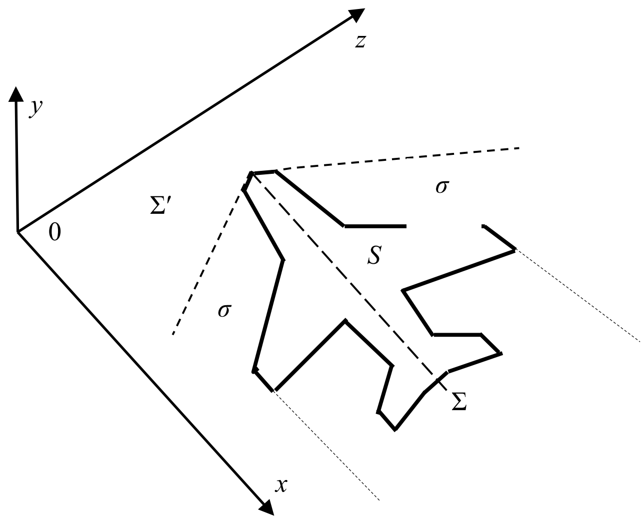
The aim of this work is to substantiate, based on the generalization of theoretical data from various authors, a numerical method for evaluating the main ACs of UAVs at supersonic speeds for prompt design and layout decisions at the stage of conceptual design and testing of the above UAV using the developed methodology for the numerical implementation of the constructed MM.

RESEARCH RESULTS

In this paper, based on the analysis of known approaches to research and analysis of the main aerodynamic characteristics of UAVs at supersonic speeds, we propose an SMA based on the synthesis of a mathematical model (MM) of linear aerodynamics for an ideal continuous medium, which is numerically realized by the method of disturbed pressures and engineering methods for calculating aircraft and their components based on generalized analytical materials based on the results of multivariate experimental and analytical studies by well-known authors. The basic relations and equations of the closed system of integral equations, features of numerical realization and some calculation results are presented.

The problem of UAV flowing around an ideal fluid with a constant supersonic velocity is considered U_0. It is believed that the angle of attack of the UAV α is small ($\alpha << 1$), therefore, it can be taken into account in the calculations $\sin\alpha \approx \alpha$.

Taking into account the mentioned UAV, we will present it in the form of a system of schematic basic elements i with an area S_i of each of them. Collectively, all elements from S_i must represent the shape of the UAV in plan, for example, in the horizontal plane, fig. 1 [8].



F i g . 1. Schematization of the UAV. Coordinate system

For the convenience of the solution and subsequent numerical implementation, we choose a rectangular left coordinate system $Oxyz$ (Fig. 1), where

$$x = Xb^{-1},\ y = Yb^{-1},\ z = Zb^{-1}, \qquad (1)$$

where x, y, z – dimensionless coordinates; X, Y, Z – dimensional coordinates; b – characteristic linear size (average aerodynamic chord − b_a).

Let's assume that the fluid motion is vortexless and not vulnerable to external forces.

For convenience, let us introduce the disturbed pressure coefficient

$$\overline{p} = 0{,}5(p - p_\infty)(\rho V_0^2)^{-1}, \qquad (2)$$

where p – pressure at the point under consideration, p_∞ – undisturbed pressure, ρ – medium density, V_0 – is the transport velocity of the medium.

For the stationary flow of an ideal fluid, the wave equation satisfied by the distribution of the function $\overline{p}$, is as follows:

$$(1 - M^2)\frac{\partial^2 \overline{p}}{\partial x^2} + \frac{\partial^2 \overline{p}}{\partial y^2} + \frac{\partial^2 \overline{p}}{\partial z^2} = 0. \qquad (3)$$

In accordance with the task under consideration, we will take into account the areas in the general scheme, Fig. 1:

$S = \sum_i^n S_i$ – the surface of the base element is the discontinuity surface for the feature $\overline{p}$;

Σ – the surface of the vortex sheet, which descends from the rear edges of the aircraft;

σ – the area disturbed by the aircraft, excluding the surface Σ;

Σ' – undrilled area.

The following boundary conditions and assumptions are used for the marked areas:

on S $\left(\frac{\partial \phi}{\partial y}\right)_S = -\alpha$ – condition of nonleaking surface S; (4)

on Σ $\Delta\overline{p} = 0$ – no pressure drop, (5)

where $\Delta\overline{p} = \overline{p}_н - \overline{p}_в$ – difference of dimensionless pressures from below $\overline{p}_н$ and above $\overline{p}_в$ at a specific point with coordinates (x, y, z) on the surface S;

in the area of σ – continuity of function $\overline{p}$ and $\left(\frac{\partial \overline{p}}{\partial y}\right)_\sigma$;

in the area of Σ' – no pressure $p = 0$ and $\left(\frac{\partial \overline{p}}{\partial y}\right)_{\Sigma'} = 0$; (6)

the surfaces of compression jumps appearing at the supersonic edges coincide with the surfaces created by the characteristics;

the Chaplygin-Zhukovsky condition is fulfilled at the subsonic trailing edges

$$\lim_{x \to x^*} \overline{p}(x, y, z) = \overline{p}(x^*, y, z), \qquad (7)$$

where x^* – is the coordinate of the trailing edge.

Output relations

For the continuity equation (3), we will use the solution [9] in the form:

$$\Delta\overline{p}(x, y, z) = -\frac{2}{\pi}\iint_{S,\sigma,\Sigma} \frac{H}{R}\left(\frac{\partial \overline{p}}{\partial y}\right)_{x_1, y_1, z_1} dS - \frac{4}{\pi}\int_{l_S, l_\sigma} \frac{H}{R}\left(\frac{\partial \phi}{\partial y}\right)_{x_1, y_1, z_1} dz_1, \qquad (8)$$

where (x_1, y_1, z_1) − coordinates of the point − source of excitations;

l_S, l_σ – contours of the base of the sealing strip on the wing edge of the UAV when the surface of the sealing strip is selected from the side S and σ accordingly.

If the source point (x_1, y_1, z_1) lies inside the selected Mach cone, then $H = 1$, if it is the other way around, i.e. the point is the source (x_1, y_1, z_1) lies outside the selected Mach cone, then $H = 0$ [9, 10].

In equation (8), from the boundary conditions (4), the values of $\left(\frac{\partial \phi}{\partial y}\right)_S$, and derivatives $\left(\frac{\partial \overline{p}}{\partial y}\right)$ and $\left(\frac{\partial \phi}{\partial y}\right)$ are related by the Cauchy-Lagrange integral as [8]:

$$\left(\frac{\partial \overline{p}}{\partial y}\right) = 2\frac{\partial}{\partial x}\left(\frac{\partial \phi}{\partial y}\right). \qquad (9)$$

From the analysis of the above, it turns out that in the right-hand sides of equations (8) the unknowns are $(\frac{\partial \overline{p}}{\partial y})_{\sigma,\Sigma}$ and $(\frac{\partial \phi}{\partial y})_{l_b}$.

Using the materials of theoretical studies [8, 11], we can obtain expressions for determining the values of $(\frac{\partial \overline{p}}{\partial y})$ in the form:

$$(\frac{\partial \overline{p}}{\partial y})_{\sigma,\Sigma} = \frac{1}{2\pi}\sum_{i=1}^{r}\int_{z}\frac{H}{(z-z_1)^2}\Delta\tilde{p}(x_1,y_1,z_1)\frac{(x-x_1)}{R}dx_1, \quad (10)$$

where r − the number of rows of cells along the axis Ox, in which it is assumed $\Delta\overline{p} = const$; $\Delta\tilde{p}$ − the difference in relative loads (pressures) between adjacent cells along the axis Ox.

To obtain an expression for the definition $(\frac{\partial \phi}{\partial y})_{l_b}$ we will use the results of work [11] regarding the obtained ratios for the function $(\frac{\partial \overline{p}}{\partial y})$

$$(\frac{\partial \phi}{\partial y})_{l_b} = -\frac{\kappa^2}{2\pi}\iint_{S} H\Delta\overline{p}(x_1,y_1,z_1)\frac{1}{R^3}dS, \quad (11)$$

here the inner integral is understood as its main value [8].

Based on the fact that

$$-\frac{\kappa^2}{2\pi}\iint_{S} H\Delta\overline{p}(x_1,y_1,z_1)\frac{1}{R^3}dS = \\ = \frac{1}{2\pi}\frac{\partial}{\partial x}\iint_{S} H\Delta\overline{p}(x_1,y_1,z_1)\frac{(x-x_1)}{(z-z_1)}\frac{dS}{R}, \quad (12)$$

we will get

$$(\frac{\partial \phi}{\partial y})_{l_b} = \frac{1}{4\pi}\iint_{S} H\Delta\overline{p}(x_1,y_1,z_1)\frac{(x-x_1)}{(z-z_1)^2}\frac{1}{R}dS. \quad (13)$$

Thus, the given expressions (10), (13) together with (8) form a closed system of equations (14), which allows to determine the values $\Delta\overline{p}$ in the calculation points of the surface S the aircraft or on its entire surface:

$$(\frac{\partial \overline{p}}{\partial y})_{\sigma,\Sigma} = \frac{1}{2\pi}\sum_{i=1}^{r}\int_{z}\frac{H}{(z-z_1)^2}\Delta\tilde{p}(x_1,y_1,z_1)\frac{(x-x_1)}{R}dx_1,$$

$$(\frac{\partial \phi}{\partial y})_{l_b} = \frac{1}{4\pi}\iint_{S} H\Delta\overline{p}(x_1,y_1,z_1)\frac{(x-x_1)}{(z-z_1)^2}\frac{1}{R}dS, \quad (14)$$

$$\Delta\overline{p}(x,y,z) = -\frac{2}{\pi}\iint_{S,\sigma,\Sigma}\frac{H}{R}(\frac{\partial \overline{p}}{\partial y})_{x_1,y_1,z_1}dS - \frac{4}{\pi}\int_{l_S,l_\sigma}\frac{H}{R}(\frac{\partial \phi}{\partial y})_{x_1,y_1,z_1}dz_1.$$

The mentioned system of equations (14) is proposed to be solved by the method of successive integration [8−17].

For the first approximation, we will set the distribution of values $\Delta p_1(x,y,z)$ from the following equation:

$$\Delta\overline{p}_1(x,y,z) = -\frac{2}{\pi}\iint_{S}\frac{H}{R}(\frac{\partial \overline{p}}{\partial y})_{x_1,y_1,z_1}dS - \\ -\frac{4}{\pi}\int_{l_S}\frac{H}{R}(\frac{\partial \phi}{\partial y})_{x_1,y_1,z_1}dz \quad (15)$$

By distribution $\Delta p_1(x_1,y_1,z_1)$ we will further define $(\frac{\partial \overline{p}}{\partial y})_{\sigma,\Sigma}$ and $(\frac{\partial \phi}{\partial y})_{l_b}$ for which expressions (10), (13) are used.

For the second approximation, we obtain the following expression:

$$\Delta\overline{p}_2(x,y,z) = \Delta\overline{p}_1(x,y,z) - \\ -\frac{2}{\pi}\iint_{\sigma,\Sigma}\frac{H}{R}(\frac{\partial \overline{p}}{\partial y})_{x_1,y_1,z_1}dS - \frac{4}{\pi}\int_{l_b}\frac{H}{R}(\frac{\partial \phi}{\partial y})_{x_1,y_1,z_1}dz_1 \quad (16)$$

To determine $\Delta p_3(x,y,z)$, calculate it again $(\frac{\partial \overline{p}}{\partial y})_{\sigma,\Sigma}$ and $(\frac{\partial \phi}{\partial y})_{l_b}$ by distribution $\Delta p_2(x,y,z)$, and then by equation (16) we find $\Delta p_3(x,y,z)$.

The number of approximations is determined by the given accuracy of the calculation. The structural block diagram of the algorithm for the numerical implementation of the technique is presented in the form shown in Fig. 2.

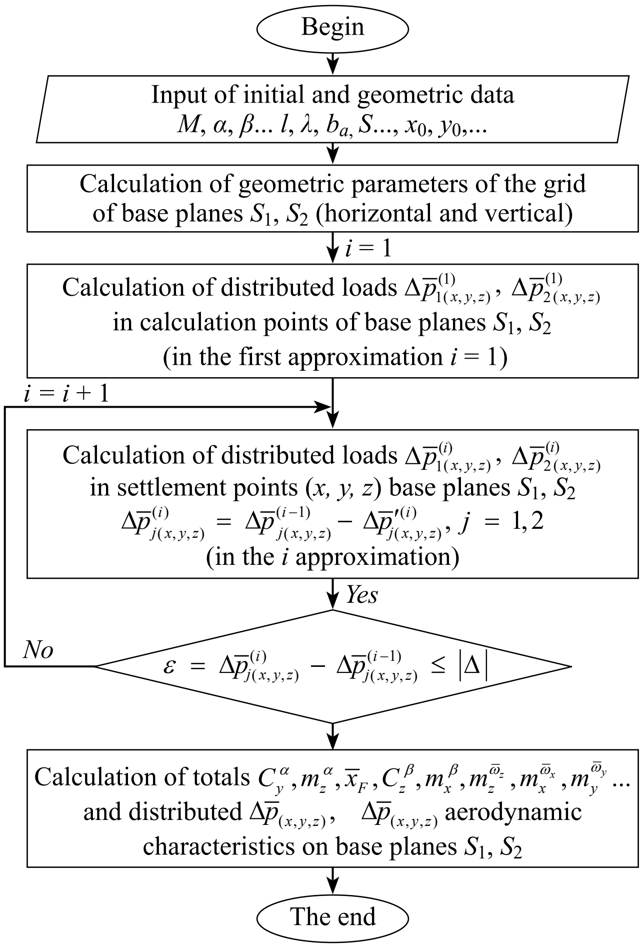


Fig. 2. Structural block diagram of the algorithm for the numerical implementation of the methodology for calculating the aerodynamic characteristics of UAVs at supersonic speeds

During the numerical implementation of the given SMA, the integrals in expressions (8), (10), (13) are represented by finite sums:

$$J_1 = \frac{4\Delta z_1}{\pi}\sum_{i=1}^{n}(\frac{\partial \phi}{\partial y})_i\frac{H_i}{R_i}; \quad (17)$$

$$J_2 = \frac{2\Delta z_1}{\pi}\sum_{i=1}^{n}\Delta x_i \sum_{j=1}^{m}(\frac{\partial \overline{p}}{\partial y})_{ij}\frac{H_{ij}}{R_{ij}}; \quad (18)$$

$$J_3 = \frac{1}{2\pi}\sum_{j=1}^{m}\Delta z_j \sum_{i=1}^{n}\Delta \overline{p}_{ij}\frac{(x - x_1)_{ij}}{(z - z_1)^2_{ij}}\frac{H_{ij}}{R_{ij}}; \quad (19)$$

$$J_4 = \frac{\Delta z_1}{4\pi}\sum_{i=1}^{n}\frac{\Delta x_i}{(z - z_1)^2_i}\sum_{j=1}^{m}\Delta \overline{p}_{ij}(x - x_1)_{ij}\frac{H_{ij}}{R_{ij}}. \quad (20)$$

It should be noted that in expression (16) two integrals have integrand functions $(\frac{\partial \overline{p}}{\partial y})$ and $(\frac{\partial \phi}{\partial y})$ with singularities near the subsonic front edges.

In the future, we will use the method, proposed in [9] to eliminate the peculiarities in the numerical implementation of mathematical models.

The linear formulation of the problem implies the possibility of using the superposition method when calculating the disturbed pressure by representing it as a linear sum of solutions obtained in partial problems:

$$\overline{p} = \sum_{j=1}^{N}\overline{p}_{\varepsilon_j}, \ \varepsilon_j \in \alpha, \beta, \omega_x, \omega_y, \omega_z \ldots . \quad (21)$$

Solution features problems α, β have been discussed above, as for non-stationary problems (with rotation, etc.), the aircraft is modeled here also by flat basic elements in the horizontal and vertical planes.

Let's introduce dimensionless angular velocities:

$$\overline{\omega}_x = \omega_x l V_0^{-1}, \ \overline{\omega}_y = \omega_y l V_0^{-1}, \ \overline{\omega}_z = \omega_z b_a V_0^{-1}, \quad (22)$$

where V_0 − undisturbed flow velocity, l – wingspan, b_a – *CAX*.

In the future, we will mark the base surfaces in the horizontal plane with the index 1, and the index 2 − in the vertical plane, leaving the zone markings with the same symbols. The general statement of the problem was considered above, which concerns the specifics of determining the boundary conditions of non-flow on the surfaces of the basic elements S_1, S_2 then this condition has the form:

$$(\frac{\partial \phi}{\partial n_{1,2}})_{S_{1,2}} = \alpha_M, \quad (23)$$

where α_M local angle of attack, ϕ – dimensionless potential of perturbed velocities, $n_{1,2}$ − normal to the surface $S_{1,2}$, respectively.

For the case of ω_x the problem, condition (23) takes the form:

$$(\frac{\partial \phi}{\partial y})_{S_1} = \overline{\omega}_x(z - z_0), \ (\frac{\partial \phi}{\partial z})_{S_2} = \overline{\omega}_x(y_0 - y); \quad (24)$$

for the task case ω_y:

$$(\frac{\partial \phi}{\partial y})_{S_1} = 0, \ (\frac{\partial \phi}{\partial z})_{S_2} = \overline{\omega}_y(x - x_0); \quad (25)$$

for the task case ω_z:

$$(\frac{\partial \phi}{\partial y})_{S_1} = \overline{\omega}_z(x - x_0), \ (\frac{\partial \phi}{\partial z})_{S_2} = 0. \quad (26)$$

Here x_0, y_0, z_0 – are the coordinates of the center of mass of the aircraft.

Taking into account (9), conditions (24−26) can be written as:

for the task case ω_x: $(\frac{\partial p}{\partial y})_{S_1} = 0, \ (\frac{\partial p}{\partial z})_{S2} = 0;$ (27)

for the task case ω_y: $(\frac{\partial p}{\partial y})_{S_1} = 0, \ (\frac{\partial p}{\partial z})_{S2} = 2\overline{\omega}_y;$ (28)

for the task case ω_z: $(\frac{\partial p}{\partial y})_{S_1} = 2\overline{\omega}_z, \ (\frac{\partial p}{\partial z})_{S2} = 0.$ (29)

It follows from (8) that the nonleakage boundary condition should be used both in the form of (24)−(26) and in the form of (27)−(29), with the base element S_1 $n = y, \ l = z$, and for the S_2 $n = z, \ l = y$.

In the first approximation, the load $\Delta\overline{p}$ with (15) is determined by integrating only the information known from the boundary conditions, taking into account the above. During numerical integration, instead of integrals, expressions (17, 18) for the final sums are used in the form:

$$J_1 = \frac{4\Delta z_1}{\pi}\sum_{i=1}^{n}\frac{1}{R_i}(\frac{\partial \varphi}{\partial y})_i,$$
$$J_2 = \frac{2\Delta z_1}{\pi}\sum_{i=1}^{n}\Delta x_i \sum_{j=1}^{m}\frac{1}{R_{ij}}(\frac{\partial p}{\partial y})_{ij}. \quad (30)$$

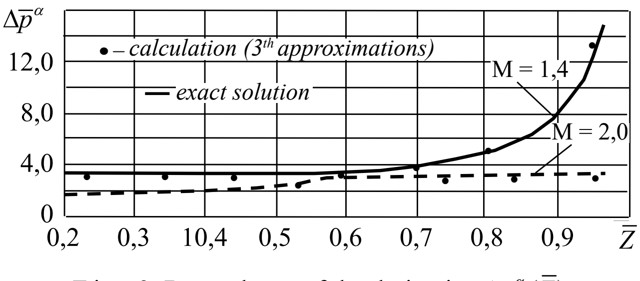
RESEARCH RESULTS

Numerical studies of the total and distributed aerodynamic characteristics according to the above methodology were carried out on a triangular wing with an extension λ = 4 for cases of subsonic and supersonic leading edges for numbers $1{,}1 \le M \le 2{,}0$.

The maximum number of approximations was set to at least 10, the number of design points on the wing surface where the loads were determined was at least 100, parameter values n and m in the studies varied from 2 to 50.

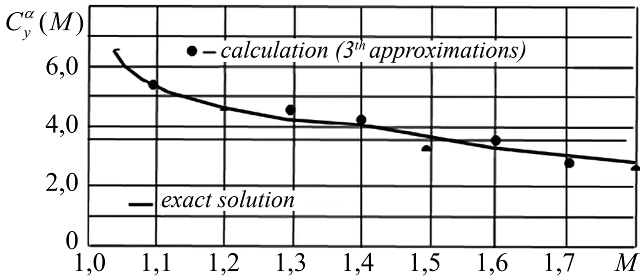
The parameters of the calculation grid cells were selected in accordance with the recommendations of [8−10] and were as follows *0,42−0,735* and *0,5* relative to the axes *x*, *z* accordingly.

Figs. 3−5 show the results of the calculations in comparison with the exact solutions [8]. Thus, Fig. 3 shows the load distribution $\Delta\overline{p}^{\alpha}$ along the relative span $\overline{Z}$ in a certain section *x = const* for different Mach numbers in the case of a subsonic leading edge (*M=1,4*) and in the case of a supersonic leading edge (*M=2,0*).



F i g . 3. Dependence of the derivative $\Delta\overline{p}^{\alpha}(\overline{Z})$ of the triangular wing with $\lambda = 4$ for numbers M=1,4 and M=2,0

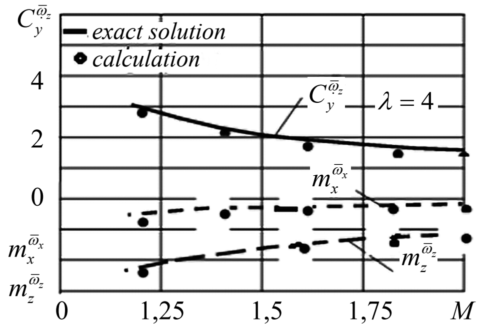
Fig. 4 shows the results of calculations of the dependence of the derivative coefficient of the lifting force on the angle of attack $C_y^\alpha = f(M)$ at Mach numbers for the third approximation.



F i g . 4. Dependence of the derivative $C_y^\alpha(M)$ f the triangular wing from $\lambda = 4$ in the supersonic speed range

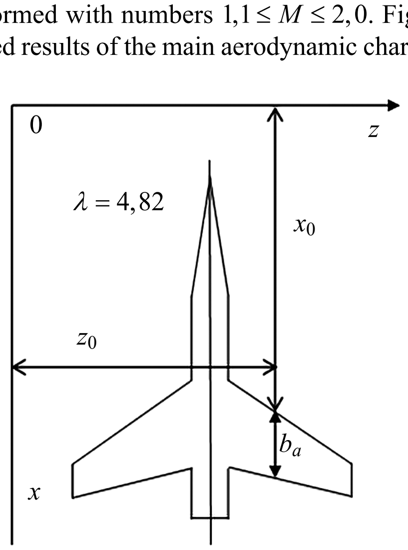
It should be noted that during calculations already after the third approximation, the obtained results coincided very well with the results of the exact solution.

Calculated non-stationary aerodynamic derivatives $C_y^{\bar{\omega}_z}$, $m_z^{\bar{\omega}_z}$, $m_x^{\bar{\omega}_x}$ for triangular wing with $\lambda = 4$ at supersonic speeds in comparison with the results [8] are shown in Fig. 5.

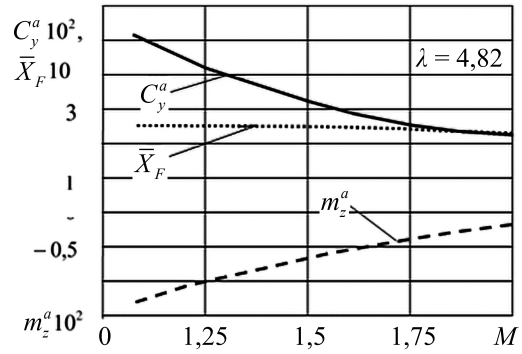


F i g . 5 Calculated aerodynamic derivatives ($C_y^{\bar{\omega}_z}$, $m_z^{\bar{\omega}_z}$, $m_x^{\bar{\omega}_x}$) of a triangular wing

For example, Fig. 6 shows the geometry of a hypothetical supersonic UAV with a swept wing of average elongation, which in the calculations was equal to $\lambda = 4,82$. Calculations were performed with numbers $1,1 \le M \le 2,0$. Fig. 7 shows the obtained results of the main aerodynamic characteristics.



F i g . 6. Diagram of a hypothetical supersonic UAV



F i g . 7. Calculated main aerodynamic characteristics $\left(C_y^a, m_z^a, \bar{X}_F\right)$ of a hypothetical UAV at supersonic speeds

Sufficient consistency of the results of the test calculations, compared with the exact solution for the above examples, indicates the efficiency of the given mathematical apparatus and calculation methods and the possibility of its use for applied research.

It should be noted that, in aggregate, the presented SMA for the computational assessment of supersonic aerodynamics is built by analogy with the SMA of subsonic aerodynamics [18, 19], where, along with the methods of numerical realization of aerodynamic characteristics in an ideal environment, engineering calculation methods [20, 21] are used to take into account the influence of the viscosity of the real environment on the basis of generalized analytical materials based on the results of multivariate experimental and analytical studies by well-known authors.

CONCLUSIONS

1. Based on the analysis of known approaches to the calculation of the main aerodynamic characteristics at supersonic speeds, the method of disturbed pressures is chosen as a variant of the numerical implementation of the mathematical model. The general formulation of the problem of supersonic aerodynamics, boundary conditions, constraints, integral relations and features of numerical realization are presented.

2. The proposed SMA belongs to the same class of problems - boundary integral equations, as well as the SMA of subsonic aerodynamics, the mathematical models of which are implemented by the method of discrete features. This approach is aimed at universalizing the general methodology for a comprehensive assessment of the aerodynamic characteristics of UAVs to obtain the necessary calculation data using a common algorithm.

3. Methodological studies were carried out and corrected initial data were selected for further numerical studies. As examples, the results of numerical calculations of the total and distributed main aerodynamic characteristics of bearing surfaces of different geometries are presented in comparison with the exact solutions at supersonic speeds. As an example, we present calculations of the main aerodynamic characteristics of a hypothetical supersonic UAV.

Sufficient consistency of the results of the test calculations compared to the exact solution indicates the efficiency of the presented mathematical apparatus and calculation methods and the possibility of using SMA in general for applied research.

4. In total, the given SMA for the calculation of supersonic aerodynamics is built in full analogy with the SMA of subsonic aerodynamics, where, along with the methods of numerical implementation of aerodynamic characteristics in an ideal environment, engineering methods are used for calculating the aerodynamics of aircraft and their components in the supersonic speed range based on generalized analytical materials based on the results multivariate experimental and analytical studies by well-known authors, which makes this SMA universal and suitable for automating the tasks of complex studies on working out the aerodynamic layouts of UAVs.

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ОБГРУНТУВАННЯ НАУКОВО-МЕТОДИЧНОГО АПАРАТА РОЗРАХУНКОВОЇ ОЦІНКИ АЕРОДИНАМЧНИХ ХАРАКТЕРИСТИК БЕЗПІЛОТНОГО ЛІТАЛЬНОГО АПАРАТА ПРИ ПОЛЬОТІ НА НАДЗВУКОВИХ ШВИДКОСТЯХ ПІД ЧАС ЙОГО КОНЦЕПТУАЛЬНОГО ПРОЄКТУВАННЯ

У статті наведено результати аналізу та узагальнення відомих теоретичних даних щодо чисельного методу збурених тисків для здійснення оціночних розрахунків основних аеродинамічних характеристик БпЛА при польоті на надзвукових швидкостях.

Досвід застосування БпЛА у ході бойових дій показав, що одним з головних показників з переліку технічних характеристик серед визначальних, таких як: дальність, тривалість, маса бойової частини, ефективна площина розсіювання, є і швидкість польоту. Реалізація високих швидкостей дозволяє мінімізувати перебування БпЛА у зоні дії вогневих засобів проти-повітряної оборони противника та більш оперативно досягати цілі.

Останнім часом з’являється інформація щодо активізації робіт з розроблення в деяких країнах, у тому числі і в Україні, надзвукових БпЛА для виконання різноманітних задач в оперативно-тактичній та стратегічній операційних зонах глибини побудови оборони противника. Така постановка питання передбачає дослідження з вибору оптимального аеродинамічного компонування зразків, що безпосередньо пов’язане з необхідністю дослідження аеродинамічних характеристик як на дозвукових так і на надзвукових швидкостях.

Наведений науково-методичний апарат пропонується для отримання розрахункових даних про основні аеродинамічні характеристики з метою послідуючого прийняття або коригування конструктивно-компонувальних рішень в ході концептуального проєктування. Наведена математична модель побудована для ідеального середовища, тому для врахування сил в’язкості в роботі здійснено синтез згаданої математичної моделі лінійної аеродинаміки надзвукових швидкостей, що чисельно реалізована за рахунок комбінації методу збурених тисків та інженерних методик розрахунку літальних.

Наведено основні співвідношення та рівняння замкненої системи інтегральних рівнянь, особливості запропонованої чисельної реалізації та деякі результати тестових розрахунків.

Ключові слова: *аеродинамічні характеристики, метод збурених тисків, граничні умови, інтегральні рівняння.*

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