A Rapid Analysis Tool for Aerodynamics/ Aerothermodynamics of Hypersonic Vehicles

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Abstract: A rapid engineering surface panel method to analyze aerodynamics and aerothermodynamics of hypersonic vehicles is developed. To obtain the surface pressure distribution of a hypersonic vehicle, the local surface inclination method is applied to calculate the pressure coefficient for each surface panel element, of which the normal vector is corrected first by using an efficient data structure and Rey-casting algorithm, local Reynolds numbers are calculated according to the geometric streamline method, then the aerodynamic heating flux is computed by both reference enthalpy relations and Reynolds analogy method. Several typical test cases are performed and the results indicate that, the developed tool is effective in predicting the aerodynamics/aerothermodynamics for complex geometry of hypersonic vehicle in a wide range of Mach numbers with a sufficient accuracy.

Key words: hypersonic; aerodynamics; aerothermodynamics; engineering prediction; reference enthalpy relation; Reynolds analogy

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0 Introduction

The aerodynamics and aerodynamic heating of hypersonic vehicle is the precondition of its aerodynamic geometry, flight trajectory and flight performance. At the concept research and preliminary design stage, various aerodynamics/aerothermodynamics configurations would be evaluated and optimized, it is difficult to carry out high precision numerical simulations and wind tunnel tests, since which are time-consuming and expensive in general. Thereby, a fast and reliable aerothermodynamic engineering prediction is more inclined.

The aerodynamics/aerothermodynamics engineering prediction approach, which could provide a medium-fidelity solutions in seconds and not hours or days required by Navier-Stokes solvers, is based on the surface panel method, calculates the aerodynamic forces with local surface inclination method, and uses Tauber's formulas[1] or reference enthalpy method (reference temperature method) to obtain aerodynamic heating distributions. Related literatures of hypersonic aerodynamics/aerothermodynamics engineering prediction began in the 1950 s, many researchers have conducted a lot of research on this subject and has made a lot of achievements. Zhang made a comprehensive introduction of different local surface inclination methods in his monograph[2], DeJarnette and Tauber made an authoritative review of approximate aerodynamic heat-transfer methods in their respective articles[3-4]. Moreover, a range of aerothermodynamic engineering prediction software (such as S/HABP[5], APAS[6], CBAERO[7]) has been developed and plays an important role in the conceptual and preliminary design of modern hypersonic vehicles.

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There are some issues when using the aerodynamics/aerothermodynamics prediction for practical application. For instance, in order to solve the pressure coefficient of a surface panel with local surface inclination method, the normal-vector direction of this panel must point outside of the vehicle′s body, and it is time-consuming to check and correct the normal-vector directions when the surface panel′s numbers are tremendous. According to Anderson′s analysis[8], the local surface inclination methods (such as Newtonian method and its modified, Tangent-Wedge and Tangent-Cone methods) have their own strengths and weakness, so it is required to choose different methods for application to different parts of hypersonic vehicles, and which is contrary to the purpose of rapid calculation, obviously. When using Reynolds analogy for aerodynamic heating, the local Reynolds numbers is difficult to measure for complex geometry vehicles.

Focusing on the aforementioned problems, an efficient data structure and Rey-casting algorithm is applied to check and correct the normal-vector directions of surface panels. The modified Newtonian method, based on a great quantity of experiment data, is used to compute the surface pressure coefficient distribution. The geometric streamline method is employed to calculate the local Reynolds numbers, which is needed in solving the aerodynamic heating. The results are compared with four typical hypersonic test cases to verify the fidelity and efficiency of our approach.

1 Surface Panel Method

The surface panel method, which is widely used in the aerodynamic characteristics of the engineering prediction, is to approximate the complex shape by using quadrilateral or triangular panel elements. Thus, the calculation of aerodynamic characteristics on vehicle surface is converted into the calculation on each panel element. Triangular elements are employed to approximate the geometrical surface in this paper as this type of elements is capable of treating complex geometry in an efficient and effective way.

In order to calculate the angle (θ) between outer-normal-vector and freestream velocity, the outward normal vector should be checked before aerodynamic calculation. Due to the defect of the method for calculating outward normal vector, the vector may point to the inside of the geometry model, so the correction procedure should be operated. Normally, the correction procedure is complex and time-consuming when the number of panels (grids) increases, therefore, the efficiency of the procedure is very important. Thus, a fast correction procedure is proposed here, which is given as follows:

Step 1 The alternating digital tree data structure[9] is applied to manage all grids so that the gird searching can be operated rapidly.

Step 2 For each grid, the point B, in the outside of the geometry model, is derived by translation of the panel centroid (Point A) with a tiny distance along the direction of outward normal vector, as shown in Fig. 1.

Step 3 Ray-casting algorithm[10] is applied as: a line segment BC is generated along a random direction, which is long enough to thread grids as many as possible, as shown in Fig. 1. ADT algorithm is used to calculate the number of panels intersected with BC.

Step 4 If the number is even, the normal vector points into the outside of the geometry model. If the number is odd, the normal vector points into the inside of the geometry model, which should be corrected.

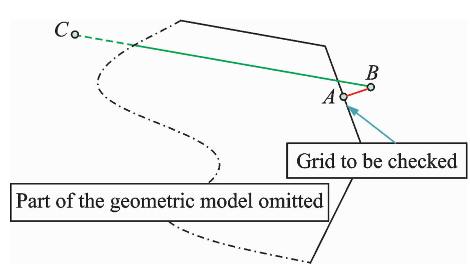


Fig. 1 Illustration of correction method for panel normal vector

A reentry capsule model is used to examine the correction procedure. In Fig. 2, the number of grids in this model is 900 000 approximately. Fig. 2(a) illustrates the circumstance before the correction procedure, many normal vectors points into the inside of the model. But after correction, all normal vectors points into the outside of the model, as Fig. 2(b) illustrates. The test is operated on the PC with Intel core i5-2310 CPU, and just cost 1.5 min, which indicates the efficiency of this correction procedure.

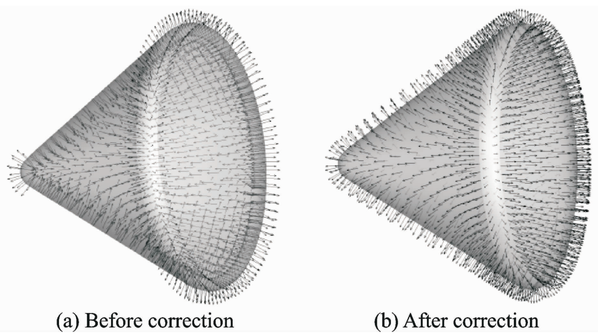


Fig. 2 Illustration of grid normal vectors before and after correction

2 Aerodynamics

The essentially ideas of calculating hypersonic aerodynamic characteristics with panel method is to assume that the aerodynamic characteristics of each grid is only related to the free stream parameters and the geometrical parameters (the area and outer normal vector direction) of current grid. Other grids have no interference with current grid. Therefore, as the pressure coefficient and geometrical parameters are known, the aerodynamic force coefficients (lift coefficient, drag coefficient, moment coefficient and even aerodynamic derivatives) could be integrated.

As we mentioned in the previous paper, it is necessary to select a suitable local surface inclination method which could adapt to different Mach numbers and complex shape vehicles. In 1979, DeJarnette[11] improved the modified Newtonian method after a large number of experiments. His method is more consistent with the experimental results at hypersonic aerodynamic engineering prediction, and suitable for different shapes of vehicle. In this paper, DeJarnette′s method was selected to calculate the pressure distributions. The surface pressure coefficient C_p can be written as

$$C_p = C_{p\max}(1 - D\cos^G\theta) \tag{1}$$

where the parameters D and G are the functions of Mach numbers.

And $C_{p\max}$ is the maximum value of the pressure coefficient, evaluated at a stagnation point behind a normal wave. The DeJarnette′s method is substituted by the modified Newtonian Law when Mach number is greater than 10.

After obtaining the pressure coefficient of every grid, the inviscid aerodynamic coefficients of three reference coordinate system C_x, C_y and C_z can be integrated.

$$\begin{cases} C_x = \sum C_p n_x \Delta A / S_{\text{ref}} \\ C_y = \sum C_p n_y \Delta A / S_{\text{ref}} \\ C_z = \sum C_p n_z \Delta A / S_{\text{ref}} \end{cases} \tag{2}$$

Hence, the aerodynamic force coefficients of vehicle (lift coefficient C_L, inviscid drag coefficient C_D, moment coefficient C_M) can be calculated in terms of the incidence (α) and slip angle (β):

$$\begin{cases} C_L = -C_x \sin\alpha + C_z \cos\alpha \\ C_D = C_x \cos\alpha \sin\beta + C_y \sin\beta + \\ \qquad C_z \sin\alpha \cos\beta \\ C_M = \dfrac{\left(\begin{array}{l} \sum C_p (x - x_{CG}) n_z \Delta A - \\ \sum C_p (z - z_{CG}) n_x \Delta A \end{array} \right)}{S_{\text{ref}} b_{\text{ref}}} \end{cases} \tag{3}$$

3 Aerothermodynamics

Here, the way to simulate aerodynamic heating is based on the geometric streamline method to calculate the local Reynolds number of each grid, obtaining the hypersonic vehicle surface heat flux with both reference enthalpy method and Reynolds analogy method.

3.1 Local Reynolds number and surface streamline pattern

The local Reynolds number Re_s can be written as

$$Re_s = \frac{\rho^* V_e x}{\mu^*} \tag{4}$$

where x is the distance measured along the body surface in meters, namely, is to search along the opposite streamline direction until stagnation grid. So, the first thing is to find stagnation girds in the aerodynamic heating calculation process. The stagnation grids are where the streamlines start, and the criterion of stagnation grid is: if all edges in a grid are outflow edges, the grid is a stagnation grid.

To provide the surface streamline pattern, the following approximation for surface velocity field is applied

$$\boldsymbol{V}_{\tan} = \boldsymbol{n} \times \boldsymbol{V} \times \boldsymbol{n} \tag{5}$$

where $\boldsymbol{V}$ is the free stream velocity vector and $\boldsymbol{n}$ the local grid outer-normal vector. Fig. 3 shows the surface streamline pattern of the STS Space Shuttle Orbiter, at Mach number of 24.87 and angle of attack (AOA) of 40.17°.

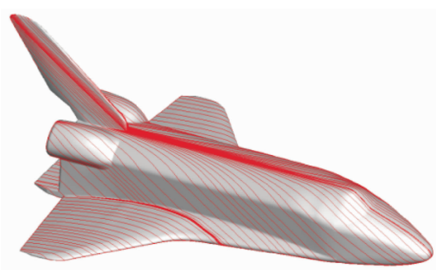


Fig. 3 Space Shuttle Orbiter, the CAD model and surface streamline pattern ($Ma = 24.87$, AOA = 40.17°)

3.2 High-temperature gas boundary-layer outer-edge parameters and transport properties

The surface aerothermal environments of hypersonic vehicle are closely related to boundary-layer edge parameters (such as velocity V_e, pressure P_e, viscosity coefficient μ_e, and so on), and these parameters are usually obtained by approximate method in engineering prediction. For perfect gas or equilibrium gas, only two of all the state parameters (pressure, density, temperature, enthalpy, acoustic velocity, entropy, viscosity coefficient, flow velocity) are independent, and the others could be solved by the appointed two parameters with corresponding thermodynamic relations. Accordingly, the remaining parameters can be solved by the isentropic conditions and the pressure of boundary-layer edge (which is equal to the pressure of wall[12]).

The thermodynamic characteristics and transport properties of high-temperature gas are frequently-used in hypersonic aerodynamic heating prediction. They can be directly found in relevant literatures of the physical form of high temperature air in local thermodynamic equilibrium[13-15], or obtained by fitting formulas[16].

3.3 Reference enthalpy method

The reference enthalpy approach is based on the idea of utilizing the formulas obtained from incompressible flow theory, where the thermodynamic and transport properties in these formulas are evaluated at some reference enthalpy indicative of the state parameters inside the boundary layer. The most popular choice for the reference enthalpy is proposed by Eckert[17]. The expression of reference enthalpy is given by the recovery enthalpy h_r, the boundary-layer edge enthalpy h_e and the wall enthalpy h_w

$$h^* = 0.19h_r + 0.23h_e + 0.58h_w \tag{6}$$

After obtained the reference enthalpy, based on the incompressible Blasius solution, using the Reynolds analogy and the reference enthalpy method, the aerodynamic heating to the surface is given by

$$q_w = 0.332 Pr^{-\frac{2}{3}} \rho_e V_e \left(\frac{\rho^* V^*}{\rho_e V_e}\right) \cdot Re_s^{-0.5} (h_r - h_w) \tag{7}$$

where these parameters with "*" are solved by reference enthalpy. When calculating the wall enthalpy h_w, the temperature of wall is needed. The majority of high-speed viscous-flow calculations assume one of two extremes, that is, they either treat a uniform, an adiabatic wall or constant-temperature wall[18]. In this paper, we use the latter, which is assumed that the wall temperature is 300 K when calculating the wall enthalpy.

For hypersonic vehicle, the aerodynamic heating causes the most damage to the flow stagnation position of the surface. Therefore, it is very important to calculate the stagnation position aerodynamic heating for thermal protection. According to Huo's analysis of different formulas[19], the Kemp-Riddell formula[20] is applied to calculate the stagnation position aerodynamic heating

$$q_{ws}=\frac{131\ 884.2}{\sqrt{R_N}}\left(\frac{\rho_\infty}{\rho_0}\right)^{0.5}\left(\frac{v_\infty}{v_c}\right)^{3.25}\left(1-\frac{h_w}{h_s}\right)\quad(8)$$

where q_{ws} is the aerodynamic heating to stagnation position, h_s the stagnation position enthalpy, $\rho_0=1.225\ \mathrm{kg/m^3}$, $v_c=7\ 900\ \mathrm{m/s}$, and R_N the curvature radius of stagnation position.

4 Tests

4.1 Test of aerodynamic characteristics

The first test case is to verify the surface pressure distributions of Apollo command module, concentrating on the orbital AS-202 mission. The geometrical dimension of AS-202 capsule is from Wright's paper[21], and the flight condition and aerodynamic characteristic data is referenced from Kinney's[22]. Specially, Kinney provided simulation data of Apollo command module with two aerodynamic codes: the CBAERO and the Data-Parallel Line Relaxation (DPLR). The DPLR code is a computational fluid dynamic (CFD) solver, also developed by NASA Ames Research Center.

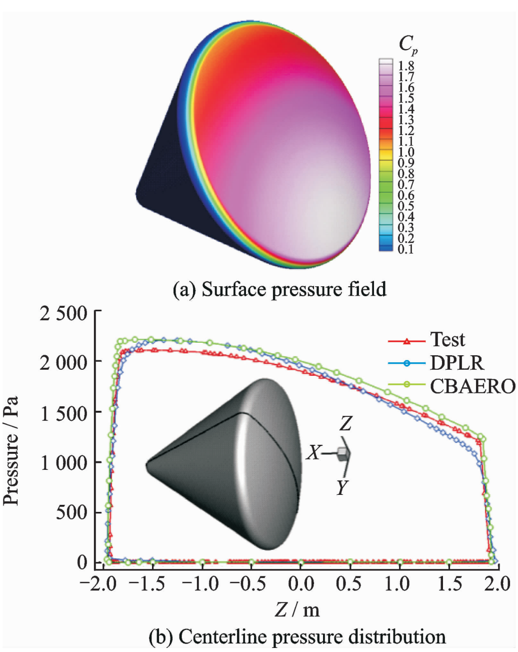


Fig. 4 Test of Apollo command module ($Ma=28.6$, AOA=18.2°)

Here, we compared our results with both CBAERO and DPLR at the flight conditions of free stream velocity being 8.24 km/s, Mach number being 28.6, and AOA being 18.2°. The Apollo surface contains about 40 000 triangular meshes. Fig. 4(a) is the surface pressure distribution. Fig. 4(b) shows the comparison results of our code's solution along the vehicle centerline with that of DPLR and CBAERO, exhibiting a perfect agreement.

The second test case is to test with the wind tunnel data of HL-20 lifting-body[23]. The HL-20 was selected to conduct a series of wind tunnel tests for supersonic to hypersonic (Mach 1.5—10) by Langley Researcher Center, NASA.

Fig. 5 presents the aerodynamic characteristic of HL-20 at $Ma=6$. Fig. 5(a) shows the surface pressure distribution of HL-20 at 30°. Fig. 5(b) is the lift-to-drag ratio with the change of AOA. The obtained results (red triangle-line) match the wind tunnel data (black quadrangle-line) well, with a maximum error (occurring at AOA=2.5°) no more than 7%. It can be considered that the method proposed in this paper can meet the requirements of engineering prediction accuracy.

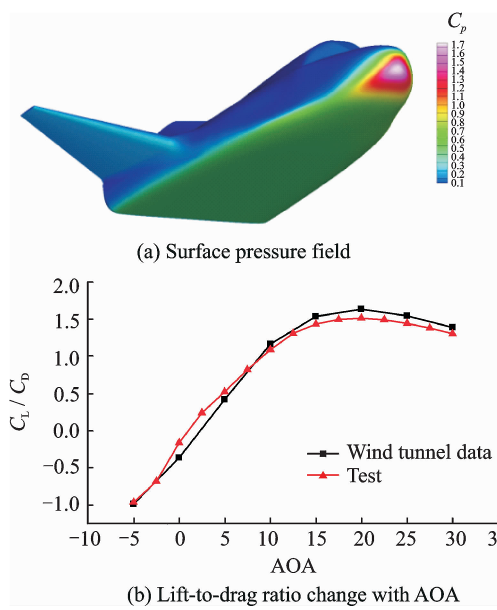


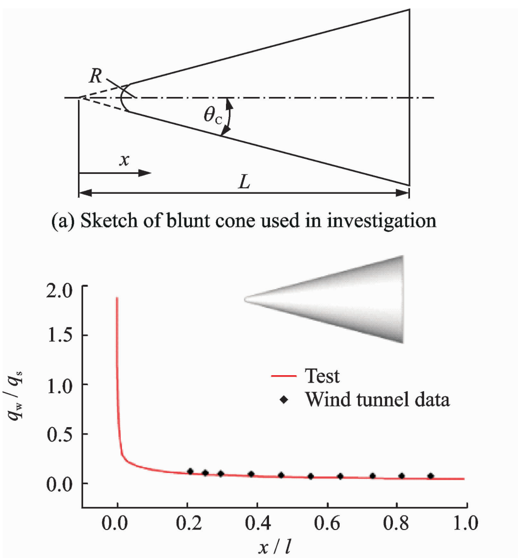
Fig. 5 Test of HL-20 lifting-body configuration ($Ma=6$)

4.2 Test of aerodynamic heating

Our third test case is to measure the aerodynamic heating distributions for a blunt cone, derived from NASA TN D-5450 report[24]. That report has extensive wind tunnel tests on blunt cone. Our operating condition is $Ma=10.6$, pressure being 132 Pa, velocity of free stream being 1 461.92 m/s, and wall temperature being

294.44 K. The schematic drawing of outer mold line of blunt as modeled in this study is shown by Fig. 6(a). The radius of curvature of blunt nosecone is 0.009 5 m, the half cone angle is 15°, and the overall length of pointed cone is 0.568 7 m.

For blunt cone, the windward aerodynamic heating distributions is much greater than leeward, which is a main emphasis of thermal protection. Therefore, a comparative analysis is made on the aerodynamic heating distributions of the center line at the windward bottom, as shown in Fig. 6(b), where q_0 is 215.91 kW/m^2, the red line is our results, and the black diamonds represent the wind tunnel data. The two results concur, thus verifying the reliability of our code.



(a) Sketch of blunt cone used in investigation

(b) Aerodynamic heating distribution of centerline, bottom

Fig. 6 Test of blunt cone (Ma=10.6, AOA=0°)

To explore the aerodynamic heating distributions at stagnation point, our last test case is to compare the obtained results with FIRE-II flight test data. FIRE-II, is one of American suborbital re-entry test program that used a subscale model of the Apollo Command Module to verify the configuration at a high reentry speed. The relevant dimensions of the Fire-II reentry vehicle outer mold line is originated from M. Wright[25]. The operating condition is at flight 1 651 sec, with air density of 6.05e−3 kg/m^3, velocity of 6.19 km/s, and temperature of 253 K. At this flight moment, in the flight test, the heat flux density of stagnation point is 405.4 W/cm^2, and our result is 431.454 W/cm^2 with an error of 6.45%, which meets the accuracy requirement of engineering prediction. Fig. 7 is the aerodynamic heating distributions of the surface.

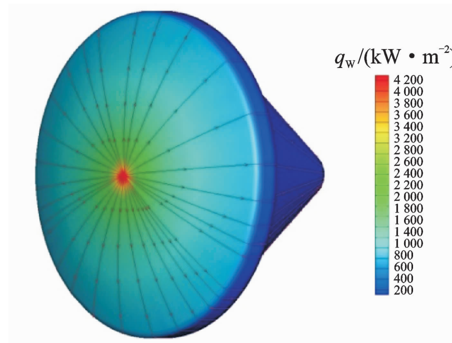


Fig. 7 Test of FIRE-II (Ma=19.414 5, AOA =0°)

5 Conclusions

This paper developed an engineering prediction approach and solver of hypersonic vehicle aerothermodynamic characteristics. The approach is based on panel method to approximate the hypersonic vehicle surface, and applies the local surface inclination method and the reference enthalpy method to calculate the aerodynamic characteristics and aerodynamic heating, respectively. Finally, several test cases are used to compare our results with wind tunnel test data, flight test data, CFD numerical simulation results, and other engineering code results. These comparisons reveal that our approach and solver are effective and reliable, and can be well used in the conceptual and preliminary design of hypersonic vehicles.

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