Computational grid adaptation for the sonic boom near field modeling

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Abstract. This paper discusses the numerical simulation of the 69-Degree Delta Wing Body for computing the sonic boom signature in the near field. For flow field calculation, the commercial CFD flow solver ANSYS Fluent is employed. The computational mesh in the perturbed flow regions was refined two times using the automatic gradient adaptation. The results are in agreement with the results obtained by other researchers.

1. Introduction
Sonic boom (SB) is one of the critical challenges for creating supersonic passenger aircraft [1-2]. Supersonic aircraft flying in the air generate sonic boom waves propagating on large areas, which annoys people. Finding the ways to reduce the sonic boom loudness is a major concern for the world's leading researchers.

Normally, the sonic boom is studied in the near and far fields. The near field of the SB is adjacent to the aircraft and has a complicated flow pattern with shock waves (SW), rarefaction and compression waves; its length is about the aircraft length ($K = H/L \sim L$, where $H$ is the radial distance from the aircraft, $L$ is its characteristic size). The far field, where the sonic boom parameters vary slightly in accordance with the asymptotic law, has the order of the distances of $K \sim L^2$. Numerical modeling is the most powerful research method, but it demands huge computational capability, requires the construction of complex grids to describe aircraft configurations, and, in addition, contains some assumptions. For modeling a real airplane model with complex geometry the time spent on meshing is important. For such simulation the compromise is a quick automatic meshing. The challenge here is the simulation of the flow in the near field of the sonic boom using mesh adaptation.

2. Test cases

2.1. Numerical simulation conditions
The flow around a body is numerically modeled in a three-dimensional formulation. The commercial CFD flow solver ANSYS Fluent is utilized for the numerical simulation. In this study the code is run in laminar mode excluding viscosity since viscous effects are negligible for sonic boom prediction. The calculation was carried out for a stationary flow of an ideal gas with the heat capacity ratio $\gamma = 1.4$. The implicit second-order scheme for temporal approximation and the AUSM flux vector...
splitting method with a second-order upwind scheme for spatial approximation were used. Initial conditions correspond to the flight altitude of 15,000 m and the flight velocity corresponding to the Mach number of 1.7.

2.2. Geometry of the model

The 69-Degree Delta Wing Body geometry taken from [3] is shown in figure 1. The fuselage is designed as a cylinder with a parabolic nose [4]. The wing has double-wedge 5-percent-thick sections with the ridge line located at mid-chord and is mounted on the cylindrical portion of the fuselage at the longitudinal plane of symmetry. The model has a fineness ratio \((L/d)\) of 16.2, where \(L\) and \(d\) are the length and the diameter of the model, respectively. This model was used for testing various prediction techniques in the framework of the AIAA Sonic Boom Prediction Workshops.

![Figure 1. 69-Degree Delta Wing Body geometry.](image)

2.3. Computational domain and mesh

The computational domain in \(x-y\) and \(y-z\) planes is shown in figure 2 and figure 3, respectively. The basic grid is generated by Fluent Meshing and contains 433 thousands cells. Due to the symmetry of the problem, only a quarter of the domain is considered. The unstructured mesh is condensed near the surface of the model. Structured mesh is aligned with the SW and the Mach angle at a distance.

![Figure 2. The computational domain (view from above).](image)  
![Figure 3. The computational domain (front view).](image)

3. Numerical results

Figure 4 shows static pressure disturbance contours around the 69-Degree Delta Wing Body in \(x-z\) plane. A series of SW and rarefaction waves is observed. Based on this solution, the mesh was adapted to the flow pattern. The static pressure gradient was chosen as a criterion for mesh adaptation.
Thus, the mesh was refined in the regions of perturbed flow, especially in shock wave fronts. The resulting computational grid in the middle region of the computational domain in the vertical symmetry plane (x-z) is shown in figure 5. After the first adaptation, the grid contained 7.5 million cells.

However, the use of the automatic gradient adaptation led to a significant increase in the number of cells near the surface of the model. The mesh near the nose of the model after the first adaptation is shown in figure 6. In addition, the number of cells has increased significantly in the wake area behind the aft of the model due to the large pressure gradients here (figure 7). To avoid this effect, it is necessary to divide the computational domain into subdomains (near the surface of the model and at a distance); the mesh adaptation within these subdomains should be carried out separately.

To optimize the calculations, mesh refinement and coarsening were carried out during the second gradient adaptation based on the results of the first adaptation. After the second adaptation, the grid consisted of 14.75 million cells.
Figure 6. Mesh near the nose of the model after the first adaptation.  

Figure 7. Mesh near the aft of the model after the first adaptation.  

Figure 8 shows the distribution of excess static pressure along the control line at a distance $K = H/L = 3.6$ on the vertical symmetry plane. The $x$-axis, passing parallel to model axis, is normalized to $L$. The origin of the axis is chosen arbitrarily. On the ordinate axis is the excess (relative to the incident flow pressure $-P_{\text{inf}}$) static pressure. The result is shown for the base grid and for the grids produced after the first and the second adaptation. It can be seen that the results of all three calculations are almost the same. The bow SW formed as a result of the flow around the nose of the model is observed with the intensity of $\sim 0.01$. The bow SW is followed by a rarefaction wave, which causes a decrease in pressure to a value lower than in the incident flow. Next, a SW ($x \approx 0.8$) formed by the wing is observed. The aft SW ($x \approx 1.35$) closes the profile. A closer look (figure 9) shows that an increase in the number of cells by 17 times relative to the base grid leads to an increase in the intensity of the SW from the wing. The difference between the results is about $6\%$. The next step (second adaptation) gives the same SW intensity as that on the base grid with a slight increase in the pressure gradient. This is mainly due to the mismatch of the fronts of the SW and the edges of the cells [5].

Figure 8. Distribution of excess static pressure along the control line in the near field of the model.  

Figure 9. Shock wave from the wing.  

Figure 10 shows a comparison with the results of numerical simulations obtained by researchers from Boeing, Dassault Aviation, JAXA, NASA, and ONERA [4]. The calculation results obtained at ITAM SB RAS are consistent with the results of other authors. The intensity of the shock waves is lower than the average value; however, it is at the level of the values obtained in NASA. The pressure gradient on the shock waves is lower too.
Figure 10. Comparison of the results of numerical modeling obtained by different researchers.

4. Conclusions

Numerical simulation of the flow in the near field of a sonic boom using mesh adaptation is presented. The mesh was refined in the regions of perturbed flow using the automatic gradient adaptation. This method allows refining mesh in the regions of perturbed flow, especially in shock waves. However, it led to an increase in the number of cells near the surface of the model. To avoid this effect, it is necessary to divide the computational domain into subdomains; the mesh adaptation within the subdomains should be carried out separately. The difference between the results obtaining on different meshes is about 6%.

Studies are done on a 69-Degree Delta Wing Body model used for testing various prediction techniques in the framework of the AIAA Sonic Boom Prediction Workshops. Numerical simulation of the near-field flow of the 69-Degree Delta Wing Body model was conducted. The results are in agreement with the results obtained by other researchers.

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