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INVESTIGATION OF NACELLE DESIGN PARAMETERS FOR A SUPERSONIC AIRCRAFT WITH LOW FIDELITY APPROACHES

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ABSTRACT

Design optimization of complex aircraft structures require high computational time and power. Frameworks with multi-fidelity approaches aim to decrease the required computational time by integrating low-fidelity and high-fidelity solvers into these frameworks to overcome resource limitations. PANAIR is one of these low-fidelity tools which can be used for aerodynamic solutions in optimization studies. In this paper, for a supersonic aircraft, the effect of nacelle layout and sizing on the near-field pressure signature is investigated by using PANAIR. First, a geometric model is generated by algorithms developed in MATLAB. These algorithms generate geometry inputs to be used in PANAIR by taking parameters belonging to the nacelle, wing, and fuselage. Nacelle bodies have a circular shape through their length and are modeled by a surface of revolution to be placed on a certain wing-body configuration. The location and size of nacelles are varied by using a parameterization scheme. The near-field pressure signatures are obtained from PANAIR for different nacelle design cases. These solutions are compared with each other and with the results obtained from the baseline geometry without the nacelle. Finally, the near-field pressure signatures are employed for sonic boom calculations via aeroacoustic methods by NASA's sBOOM code, and the resulting ground signatures are assessed in terms of nacelle designs.

INTRODUCTION

Multi-fidelity optimization studies regarding the aerodynamic design of supersonic aircraft aim to bring tools with different fidelity levels together to obtain reliable results efficiently and reach objectives such as low drag or low sonic boom characteristics [Demiroglu, Yıldız and Nikbay, 2021; Choi, Alonso, Kroo and Wintzer, 2008; Giblette, 2019]. Since the aerodynamic design of a supersonic aircraft requires reliable tools in terms of accuracy, computational fluid dynamics (CFD) has been a commonly used method to examine the flow around an aircraft with high accuracy and increased computational performance today. On the other hand, the CFD method requires an extended processing time, which is a burden in design optimization studies. The multi-fidelity optimization studies aim to overcome this computational cost required by the CFD method by bringing tools with different fidelity levels together in a design environment.

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Although high-fidelity fluid analysis studies are widely employed for their accuracy in detailed design, it is also reasonable to use low-fidelity tools in conceptual design and multi-fidelity analyses where further design optimization studies can be built on. For flow solutions, it is seen that the panel methods are used in many studies to examine the near-field flow around an aircraft with their fast solutions provided by sacrificing the accuracy to an acceptable degree. The higher-order panel methods are based on linear potential supersonic theory. Due to the limitations of the theory, they provide sufficient accuracy on slender geometries exposed to non-transonic flow at small and moderate angles of attack. It should also be taken into account that the accuracy of near-field solutions to be obtained with panel methods decreases as they move away from the geometry. Despite these limitations, using panel methods is an option to consider in design optimization studies where many parameters need to be calculated since the solutions obtained by this method require much less time and computational power resources compared to the CFD results which are obtained by solving Euler or Navier-Stokes equations. If sufficient accuracy can be obtained by the panel method for the near-field pressure calculation, the computational convenience brought by the panel methods can be utilized. At this point, it is essential to investigate how efficiently panel methods can support the design process despite the limitations.

The current research aims to identify the effect of nacelle design parameters such as positioning on a supersonic aircraft by calculating the near-field and ground-level pressure signatures using panel methods. Within this research, a generic aircraft geometry is employed and the effect of nacelle layout on the pressure distributions is examined. The entire aircraft geometry, including the engines, is modeled by parsing it into rectangular panels with developed algorithms. Linearized supersonic equations are used to obtain flow solutions around the aircraft model by specifying boundary conditions for these panels. Near-field and ground-level pressure signatures are obtained for different layouts and sizes of the nacelle.

METHODS AND RESOURCES

This section explains algorithms and analysis tools used for this study, including geometry modeling, low-fidelity flow solutions, and obtaining pressure signatures.

Geometry Representation

To examine the flow around a certain geometry by the panel method, the entire geometry must be discretized to consist of rectangular panels. To achieve this, MATLAB scripts were developed in which the parameters constituting the body, planar wing, and engine configurations were provided as input. The tail geometry was not included for simplicity. These parameters consist of options such as airfoil, aspect ratio, taper ratio, body length, and diameter. An example of a wing-body configuration generated by using this code is shown in Figure 1. In this study, nacelles are modeled by

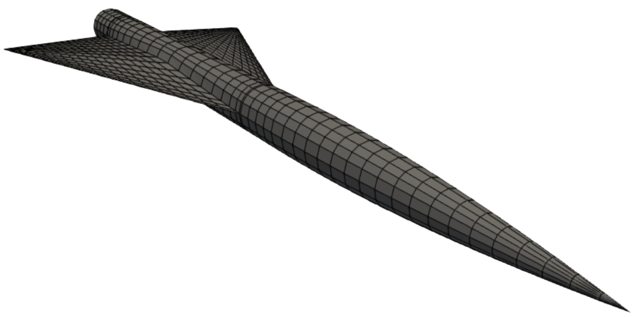


Figure 1: Example of body-wing configuration generated in MATLAB

surfaces of revolution. The parameterization applied while generating the nacelle geometry is shown in Table 1. Figure 2 shows an example of a nacelle geometry generated by this parameterization

Table 1: Nacelle parameterization

Parameter	Name
t_{max}	Nacelle maximum thickness
x_n/L	x location of nacelle base as fraction of fuselage length
y_n/L	y location of nacelle center line as fraction of fuselage length
z_n/L	z location of nacelle center line as fraction of fuselage length

scheme.

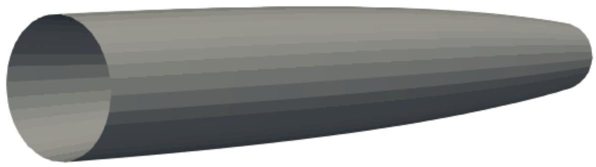


Figure 2: Example of nacelle geometry

Near-field Flow Solution

As a low-fidelity flow solver, PANAIR has been used to obtain the pressure distribution on the aircraft and in the near-field region. PANAIR solves the flow around arbitrary geometries using the higher-order panel method [Carmichael and Erickson, 1981]. By solving the Prandtl-Glauert equation, it gives solutions both in subsonic and supersonic conditions;

$$(1 - M_\infty^2)\phi_{xx} + \phi_{yy} + \phi_{zz} \quad (1)$$

where M_∞ is Mach number and ϕ is the perturbation velocity potential. PANAIR allows the modeling of slender geometries and non-transonic flows due to the limitations of the linearized potential flow theory. It provides pressure distribution around arbitrary geometries. PANAIR takes a geometry in parts called networks which consist of panels. Each network is placed into groups such as the base network and the wake network according to the boundary condition it must satisfy. Networks forming the geometry are given as input to PANAIR by using the methods mentioned in the previous section.

As previously stated, boundary conditions must also be specified while giving each network part to PANAIR. These boundary conditions are used to represent all of the models correctly according to their characteristics such as solid surfaces with impermeability conditions, flow regions separated from body base, or wakes shed from a configuration. In this work, these types of boundary conditions are used to model the aircraft and nacelle. One issue encountered during this process was regarding the flow through the nacelle geometry. At supersonic analysis using PANAIR, Mach waves forming in the internal volume of the nacelle can result in intense numerical errors. Most of the panels with different boundary conditions given to PANAIR requires to be inclined to the flow with an angle smaller than Mach angle. Super inclined panels, which have inclination angles greater than the Mach angle, are allowed when the upstream flow is not affected. PANAIR provides these super inclined panels with proper boundary conditions which can be used for swallowing incoming inlet flows through surfaces such as nacelle inlets and setting potential in the internal volume to zero. This capability of PANAIR is also utilized in this work to prevent numerical errors and enable a nacelle modeling with inlet and outlet. Specifying exhaust flows is also possible with super inclined panels by using PANAIR, but it is not applied in this study. It should be noted that performance criteria for the engine is not considered while determining the overall design of the nacelle.

Ground Signatures

Near-field pressure signatures obtained from PANAIR at a certain distance away from the aircraft are necessary to predict the ground signatures. These near-field pressure signatures should be propagated through the atmosphere to obtain sonic boom loudness solutions. For this purpose, sBOOM code which is developed by NASA is used [Rallabhandi, 2011]. Boom propagation process of sBOOM involves the solution of augmented Burger equations. The computational procedure applied is summarized in Figure 3.

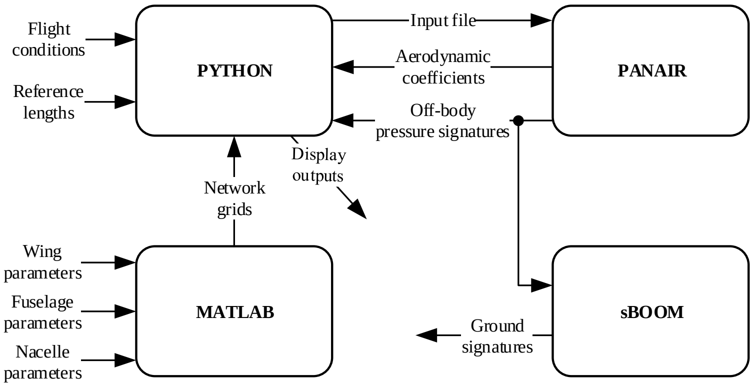
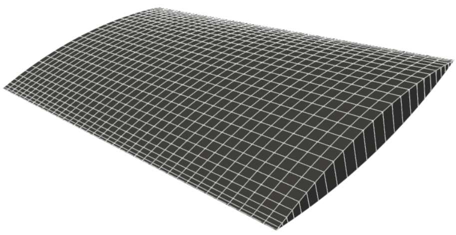


Figure 3: Computational procedure

NUMERICAL VALIDATION

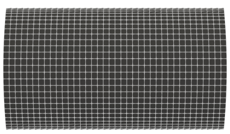
To obtain the flow solution around a certain geometry from PANAIR, firstly the model is discretized into panels. Comparisons with reference models were performed to verify the algorithms on which this procedure is based and the data obtained from PANAIR. In this section, flow solutions based on two different reference models are presented.

The first model is a rectangular wing configuration with a circular arc profile which is used in the experimental study carried out by National Advisory Committee for Aeronautics (NACA) [Coletti, 1955]. The geometric specifications of the wing are presented in Table 2 and its geometry is shown in Figure 4. In the experiment which was performed in the Langley 9-inch supersonic wind tunnel, the Mach number of the flow around the wing was 1.62. The reference wing is modeled to consist of 1640 rectangular panels in total.



(a) Isometric view

(b) Cross-section



(c) Top view

Figure 4: Model wing used for numerical validation

Table 2: Reference wing geometrical specifications

Parameter	Name	Value
c	Chord	6.967 cm
AR	Aspect ratio	1.8
t/c	Thickness ratio	0.059

The lift coefficient vs the angle of attack plot is presented in Figure 5. In the solutions obtained from PANAIR, it is seen that the lift coefficient values have a minor difference with respect to the experimental data. The measurements in the experiment conducted in the supersonic wind tunnel include the viscous effects created by the flow surrounding the wing. On the other hand, in linearized supersonic equations which PANAIR is based upon, viscous effects are neglected. However, the fact that the aerodynamic coefficient values and the slopes of the curves are close to each other shows that the reference wing was successfully modeled and its aerodynamic performance was achieved with very low errors.

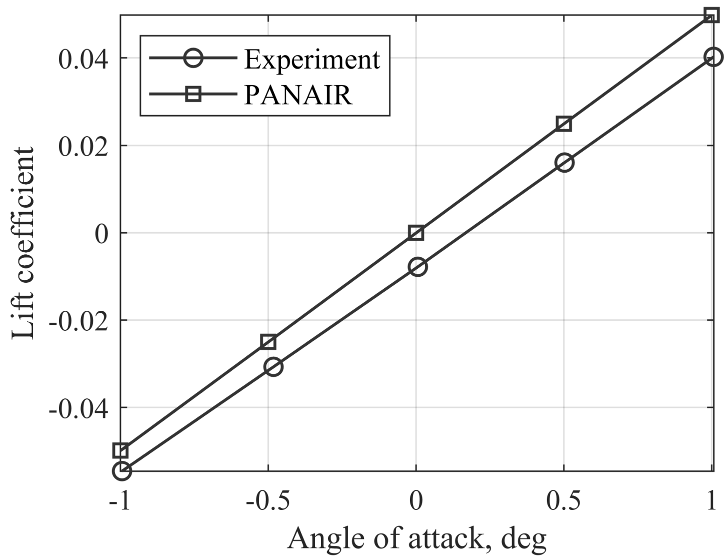


Figure 5: Change in lift coefficient with angle of attack

Another numerical investigation was performed for a wing-body configuration. The low-sweep configuration was studied by [Kroo et al., 2010] is used as the reference model. In that study, the variation of aerodynamic coefficients with angle of attack is calculated by using the specified configuration in PANAIR and other high-fidelity flow solvers. The wing has a leading edge sweep angle of 21.8 degrees, a reference area of 58.064 m2, and a biconvex airfoil with 2% thickness. Its aspect ratio is equal to 4. The fuselage of the aircraft is a parabolic geometry with a length of 30.48 meters and a fineness ratio of 20. Aerodynamic coefficients are obtained for Mach 1.5. The entire geometry is meshed into 2330 panels in total as shown in Figure 6.

(a) Isometric view (b) Top view

Figure 6: Body-wing configuration used for numerical validation

Figure 7 shows the results obtained from PANAIR and the data from the reference. Aerodynamic coefficients were obtained by changing the angle of attack from 0 to 1.8 degrees. It seems that the solutions obtained from PANAIR are highly compatible with the reference data.

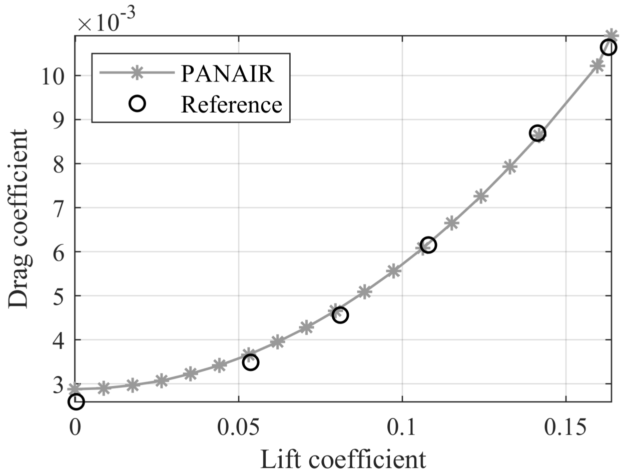


Figure 7: Variations in lift and drag coefficients

The PANAIR results are found to be satisfactory according to the comparisons made using wing and wing-body configurations. The path followed for panel discretization and current numerical approaches developed to obtain solutions using PANAIR are valid. The wing-body configuration examined in this section is used as the base geometry for nacelle investigation in the following sections.

APPLICATION

In this section, near-field and ground pressure signatures are examined by taking the wing-body geometry used for numerical verification as the base configuration. Pressure distribution graphs are obtained by considering various position and thickness values of the modelled nacelles based on the combinations of previously stated engine parameters. Nacelle length is set to 5 meter and diameters of inlet and outlet are set to 0.5 meter.

The fuselage, the wing, and the nacelle are modeled to consist of 738, 529, and 1887 panels, respectively. When wake networks are included, the entire model consists of 3843 panels. The flight conditions are 1.5 Mach number, 0 degree angle of attack, and 0 degree side slip angle. All near-field pressure signatures are obtained at 2 body lengths away from the aircraft for 0 degree azimuth angle. The pressure distribution which is obtained from PANAIR when nacelles are not included in the model is shown in Figure 8.

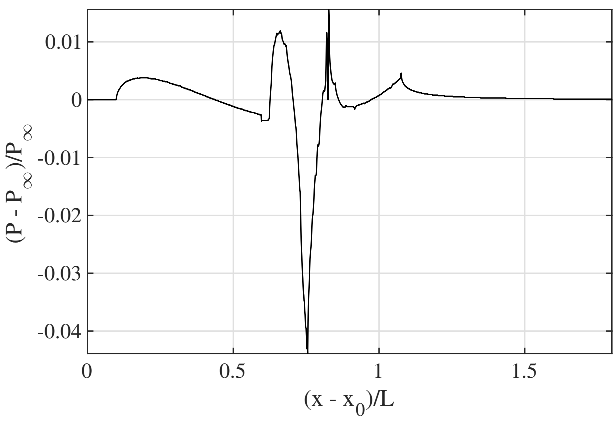


Figure 8: Near-field pressure signature without nacelles

For all sonic boom calculations presented in this paper, the standard atmosphere assumption was made and the atmosphere was considered without wind profile. Other input parameters given to sBOOM are presented in Table 3. The pressure signatures obtained at 2 body lengths away from the aircraft are propagated throughout the atmosphere. The ground signature which is obtained from sBOOM for the base geometry is presented in Figure 9.

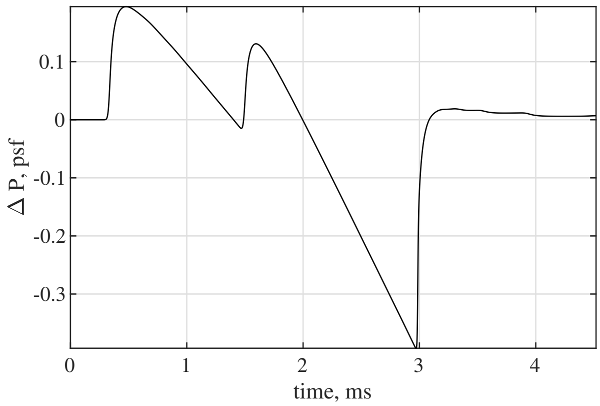


Figure 9: Ground signature without nacelles

Table 3: sBOOM input parameters

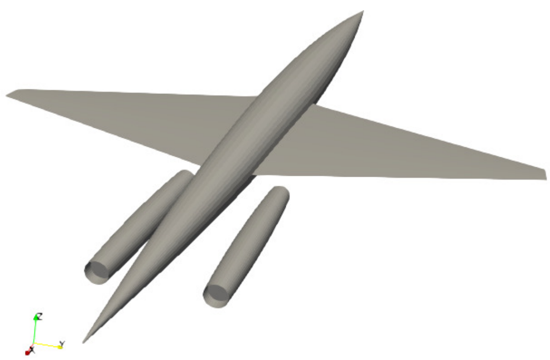
Parameter	**Value**
Mach	1.5
Altitude	55000 ft
Azimuth angle	0°
Reflection factor	1.9
Ground height	0

Location of Nacelle

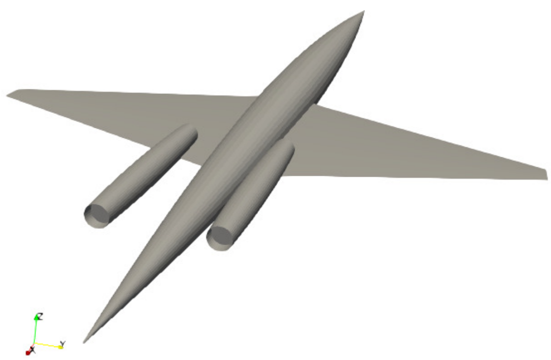
The effect of the nacelle position on the pressure distributions is investigated based on nacelle location parameters as shown in Table 4. These values belong to the engine in half of the aircraft geometry which is symmetrical with respect to the XZ plane. In Figure 10, the layouts of the nacelles on aircraft for these cases are shown.

Table 4: Inspected cases for location

Case	x_n/L	y_n/L	z_n/L
Case 1	0.10	0.04	-0.02
Case 2	0.10	0.04	0.02
Case 3	0.25	0.04	-0.02
Case 4	0.25	0.04	0.02
Case 5	0.25	0.12	-0.02

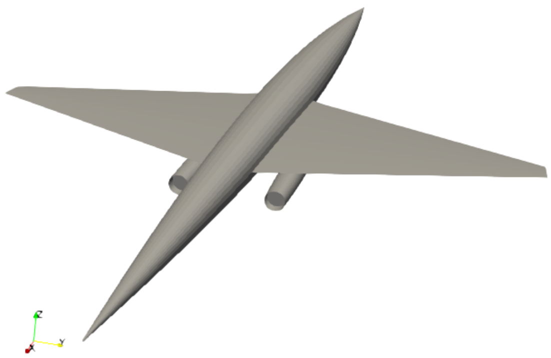


(a) Case 1

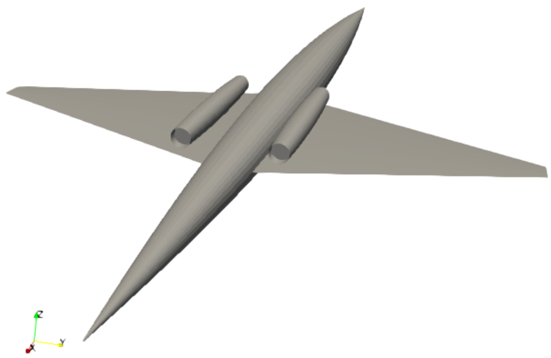


(b) Case 2

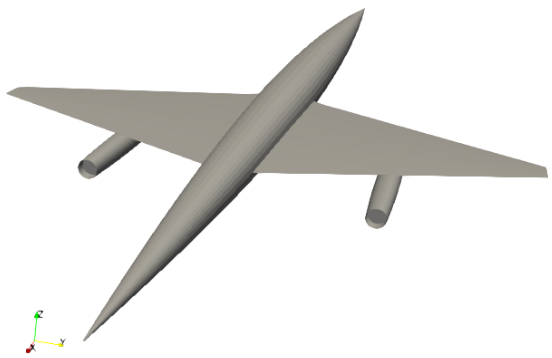
Figure 10: Inspected nacelle layout



(c) Case 3



(d) Case 4

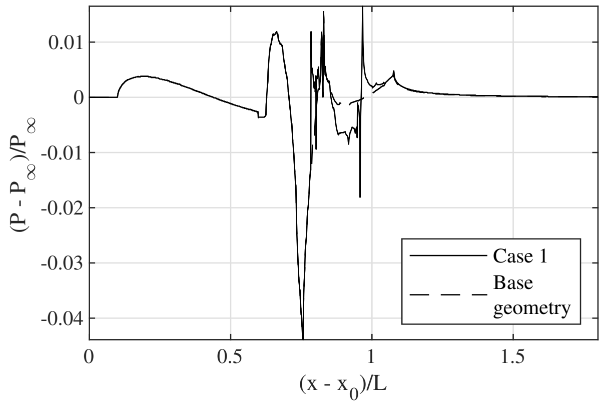


(e) Case 5

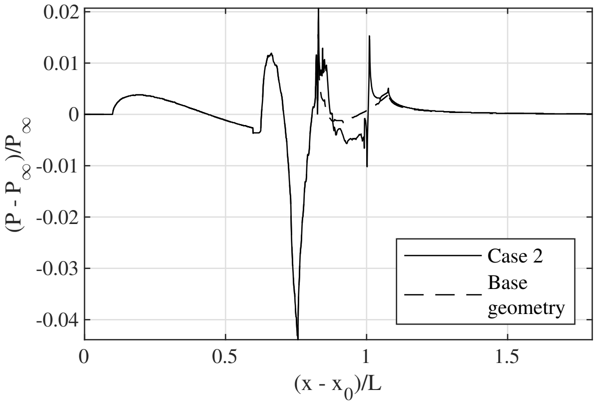
Figure 10: Inspected nacelle layout

The pressure distributions are obtained for cases 1, 2, 3, 4, and 5 which are given in Table 4. The pressure signatures for each case are shown in Figure 11 in comparison with the baseline geometry without the nacelle. As seen in each case, the addition of the engine causes fluctuations in the near-field pressure distributions obtained from PANAIR. The nacelle is located behind the wing in cases 1 and 2, while cases 3, 4, and 5 represent situations that the engine is under or above the wing. These three cases show similar deviations from the baseline geometry pressure distribution. When nacelles are located near the wing, the pressure distribution in the wing region is highly affected. In this region, there are more deviations from the pressure values obtained relative to the location of the nacelle behind the wing and an increase in the frequency of fluctuations is seen. As expected, these deviations also occur at the forward of the wing when the nacelle is located under or above the wing.

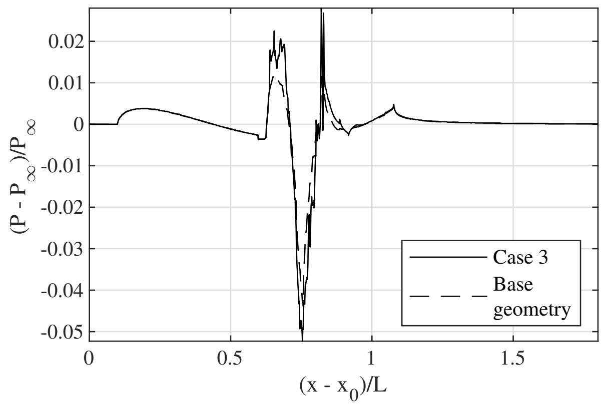
As in Case 2 and 4, when the engines are pulled behind the wing, the deviations from the pressure values seen in other cases decrease. The solution in the wing region remains almost the same for these two cases. In this respect, these two cases show similar results.



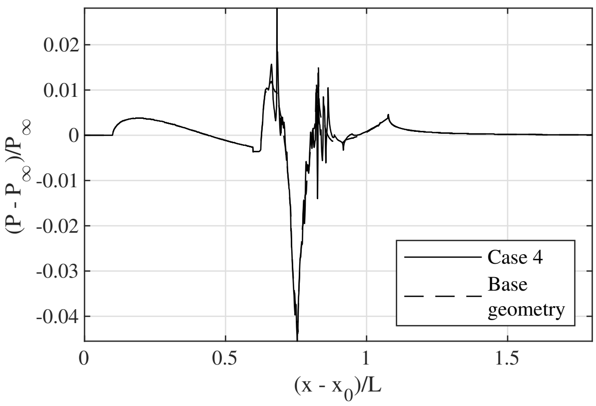
(a) Case 1



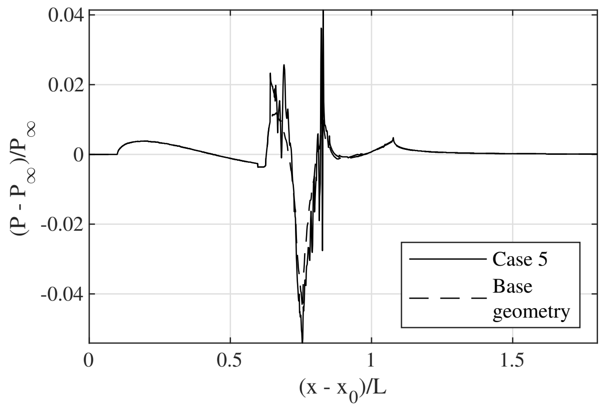
(b) Case 2



(c) Case 3



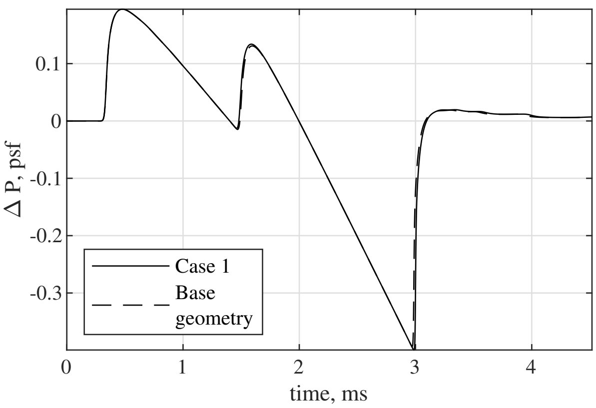
(d) Case 4



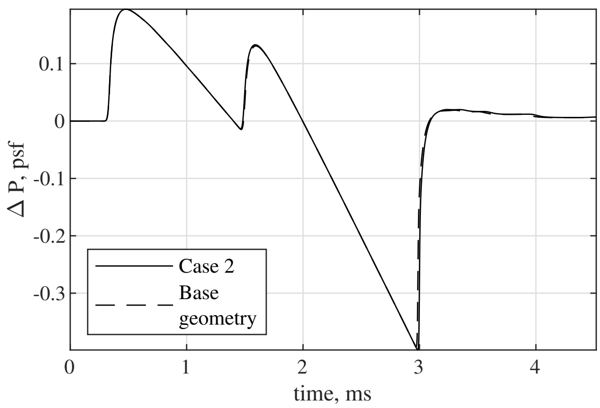
(e) Case 5

Figure 11: Pressure signatures for different locations of nacelle

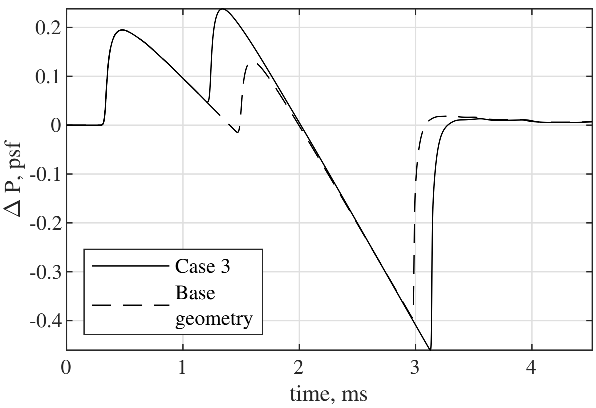
The ground pressure signatures for each case are shown in Figure 12 in comparison with the base geometry without the nacelle. As seen in each graph, fluctuations observed in near-field pressure signatures fade out and nearly the same ground signatures are observed for cases 1, 2, and 4. The most obvious differences relative to the distribution of base geometry are observed for cases 3 and 5. These cases represent the location of the nacelle under the wing. Shocks propagating from the nacelle are blocked by the wing in case 4; therefore, a different distribution from cases 3 and 5 is seen for this case. The differences between the cases can be seen in Figure 13.



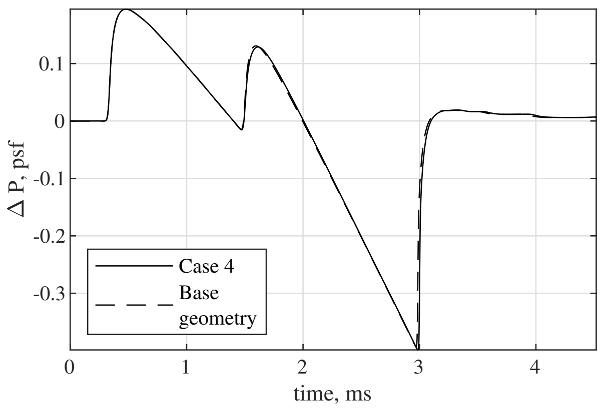
(a) Case 1



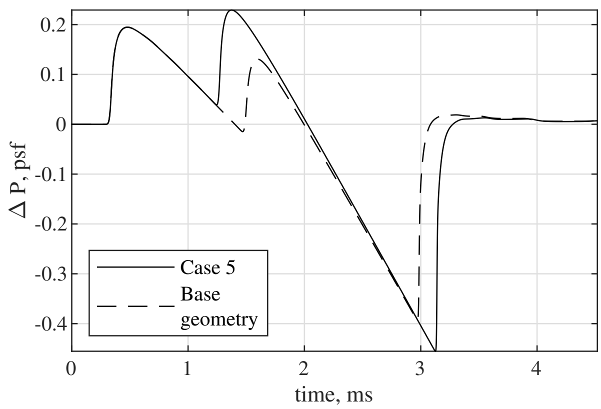
(b) Case 2



(c) Case 3



(d) Case 4



(e) Case 5

Figure 12: Ground signatures for different locations of nacelle

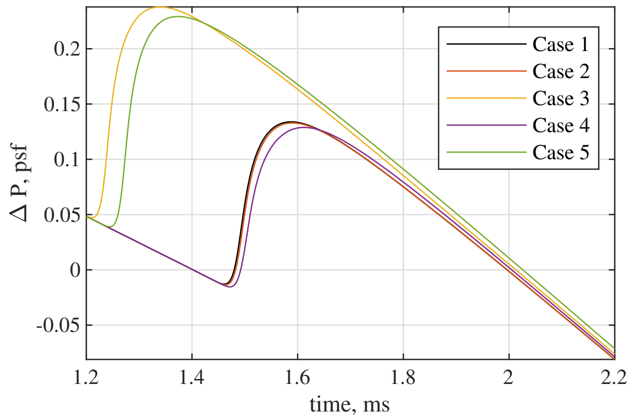


Figure 13: Comparison of ground signatures of all cases (zoomed)

Thickness of Nacelle

Pressure distributions are also obtained for different thicknesses of the nacelle. The thickness of the nacelle at the middle of its length is set to 125 mm while investigating the location of the nacelle. Nacelle location is set to the coordinate parameters of case 2 while increasing the thickness value from 125 mm to 250 mm and 375 mm. Figure 14 compares the nacelle section appearance for these values of thickness.

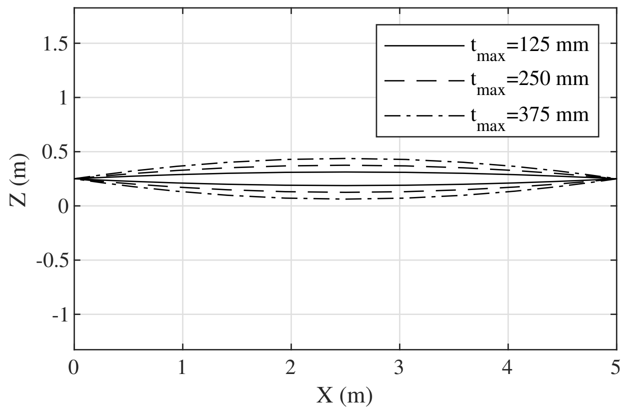


Figure 14: Nacelle thicknes comparison

The pressure signatures obtained for different thicknesses are shown in Figure 15. Each of the graphs in the figure represents a different thickness. Effects of increased thickness values compared to the default thickness value of 125 mm are shown.

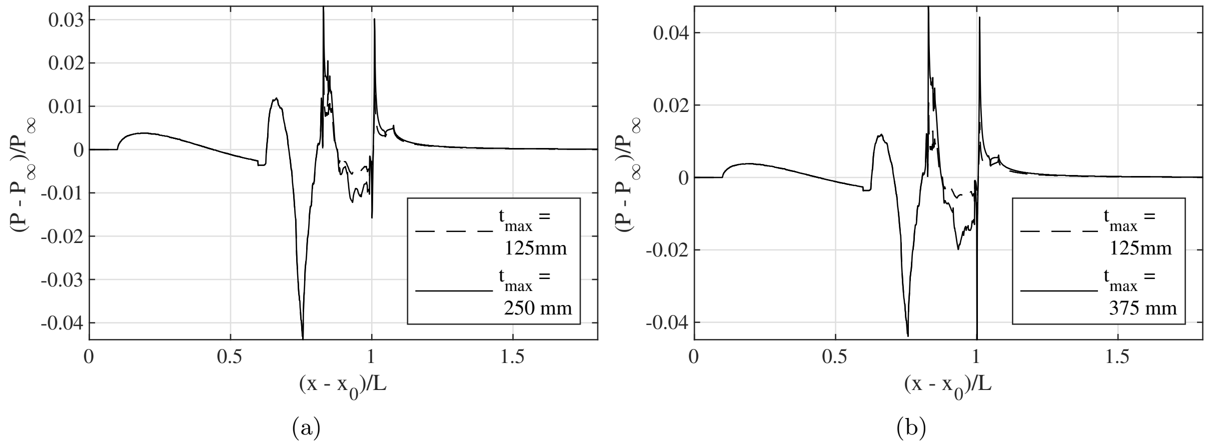


Figure 15: Comparison of pressure signatures for different thickness values of nacelle models when $x_n/L = 0.10,\ y_n/L = 0.04,\ z_n/L = 0.02$

The dotted lines in the graphs represent the engine model with relatively small thickness. As in the graphs in Figure 11, fluctuations are seen in the solutions obtained from PANAIR. As expected, an increase in thickness causes increase in magnitudes of pressure distribution around the nacelle. The effect of further increase of thickness on the pressure values is also observed from the solutions obtained from PANAIR.

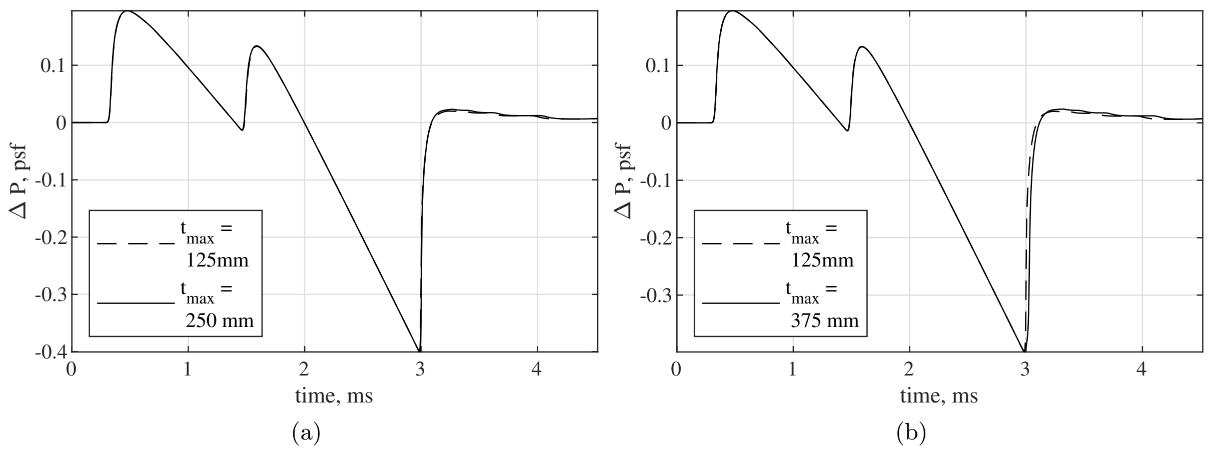


Figure 16: Comparison of ground signatures for different thickness values of nacelle models when $x_n/L = 0.10,\ y_n/L = 0.04,\ z_n/L = 0.02$

The ground pressure signatures obtained by propagating the near-field pressure signatures in Figure 15 are presented in Figure 16. For both cases, fluctuations fade out during propagation and resulting ground signatures for nacelles with different thicknesses at the same location show the same ground characteristics.

RESULTS AND DISCUSSION

For design optimization of supersonic aircraft with low sonic boom characteristics, it is important to combine low-fidelity and high-fidelity vehicles in order to mutually benefit from their efficiency and accuracy. It is quite common to use panel methods to solve the linearized supersonic equations for low-fidelity solutions in such optimization studies. The limitations of the panel methods make it impossible to examine the entire aircraft geometry using this method alone and require research for the areas where these tools may be suitable for use. For this purpose, a certain wing-body configuration is used as the baseline geometry and the effect of nacelle location and dimensions on the near-field and ground pressure distributions are investigated with PANAIR, which uses the higher-order panel method.

Since PANAIR requires the aircraft geometry to be discretized into a mesh consisting of rectangular panels for the flow solution, the algorithms to fulfill this task are generated in MATLAB. These algorithms take the parameters belonging to the wing, body, and ellipsoid-shaped nacelle as inputs and create the desired geometries to be used in PANAIR. In this way, nacelles with different thickness values are modeled at five different positions and the near-field pressure distribution solutions are obtained from PANAIR. For each of the ten cases presented, the entire geometry is modeled to consist of 3843 panels, including wake networks. The average duration of the solutions obtained from PANAIR for these cases is 57 seconds.

When nacelles are included, deviations from the values in the pressure distribution obtained from the baseline geometry occur. The magnitude of these deviations is higher when the nacelles are placed under or above the wing. Placing the nacelle behind the wing has a positive effect on the near-field pressure signature. In this case, the results are close to the results obtained for the baseline geometry without nacelles. Also, magnitudes of fluctuations are relatively small for this situation. However, fluctuations in the near-field pressure signatures always occur for each case. It can be assumed that this is due to the panel methods used by PANAIR. For all cases, these fluctuations fade out as the pressure signatures are propagated through the atmosphere using the sonic boom prediction methods. Whereas ground signatures obtained for the nacelle model under the wing which corresponds to cases 3 and 5 have some differences relative to the ground signature obtained for baseline geometry, the remaining ground signatures obtained show minor differences. Besides, the difference in the ground signatures of nacelles with different thickness values is not observed even if there are some minor differences in their near-field pressure distributions since these discrepancies fade out as the waves are propagated through the atmosphere.

In this study, the effect of the position and thickness of the engine nacelle model on the near field and ground pressure distributions is investigated for a supersonic aircraft by using a low-fidelity approach. The outcomes will be employed in multi-fidelity analysis and optimization of low-boom aircraft concepts. Furthermore, work on the specification of engine exhaust flow and analyzing engine models with PANAIR to evaluate the performance of the panel methods may be considered as future studies.

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References

Carmichael, R. and Erickson, L. (1981) *PAN AIR - A Higher Order Panel Method for Predicting Subsonic or Supersonic Linear Potential Flows About Arbitrary Configurations*, 14th Fluid and Plasma Dynamics Conference, American Institute of Aeronautics and Astronautics, 1981

Choi, S., Alonso, J. J., Kroo, I. M. and Wintzer, M. (2008) *Multifidelity Design Optimization of Low-Boom Supersonic Jets*, Journal of Aircraft, Vol 45, No. 1, p: 106–118, 2008

Coletti, D. H. (1955) *Investigation of the Effects of Model Scale and Stream Reynolds Number on the Aerodynamic Characteristics of Two Rectangular Wings at Supersonic Speeds in the Langley 9-Inch Supersonic Tunnel*, NACA-RM-L55D29, 1955

Demiroglu, Y., Yıldız, Ş. and Nikbay, M. (2021) *Multi-Fidelity Sonic Boom Minimization of a Supersonic Aircraft by Parametric Wing Shape Design*, AIAA Scitech 2021 Forum, 2021

Giblette, T. N. (2019) *Rapid Prediction of Low-Boom and Aerodynamic Performance of Supersonic Transport Aircraft Using Panel Methods*, M.Sc. Dissertation, Mechanical and Aerospace Engineering, Utah State Univ., Logan, UT, 2019

Kroo, I., Willcox, K., March, A., Haas, A., Rajnarayan, D. and Kays, C. (2010) *Multifidelity Analysis and Optimization for Supersonic Design*, NASA/CR-2010-216874, 2010

Rallabhandi, S. K., (2011) *Advanced Sonic Boom Prediction Using the Augmented Burgers Equation*, Journal of Aircraft, Vol 48, No. 4, p: 1245–1253, 2008