Sonic Boom Reduction using an Adjoint Method for Supersonic Transport Aircraft Configurations

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Introduction

The objective of this work is to develop the necessary methods and tools to facilitate the design of low sonic boom aircraft that can fly supersonically over land with negligible environmental impact. Traditional methods to reduce the sonic boom signature were targeted towards reducing aircraft weight, increasing lift-to-drag ratio, improving the specific fuel consumption, etc. Traditional adjoint implementations were aimed at reducing a cost function computed from the pressure distribution on the surface that is being modified. In this case, however, we would like to obtain sensitivity derivatives of pressure distributions that are not collocated at the points where the geometry is being modified. This type of sensitivity calculation has not been attempted before and will be necessary for the sonic boom minimization problem. In order to include the tailoring of the ground pressure signatures, it becomes necessary to compute sensitivity derivatives of the sonic boom signature with respect to a large number of design variables that affect the shape of the airfoil or aircraft.

For typical cruise altitudes required for aircraft efficiency, the distance from the source of the acoustic disturbance to the ground is typically greater than 50,000 ft. A reasonably accurate propagation of the pressure signature can only be obtained with small computational mesh spacings that would render the analysis of the problem intractable for even the largest parallel computers. An approach that has been used successfully in the past is the use of near to far field extrapolation of pressure signatures based on principles of geometrical acoustics and non-linear wave propagation. These methods are based on the solutions of simple ordinary differential equations for the propagation of the near field pressure signature to the ground.

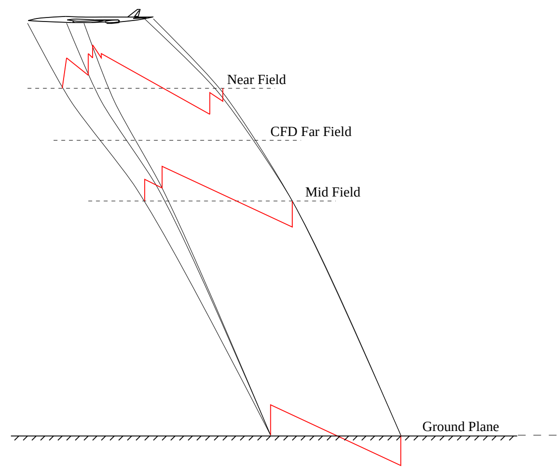

Figure 1: Schematic of Sonic Boom Minimization

Figure 1 is a schematic of the sonic boom minimization problem. 'CFD Far Field' indicates the far field boundary of the mesh. At a pre-specified distance below the aircraft and still within the CFD mesh, the location of a near field plane can be seen. This plane is the effective interface between the CFD solution and the wave propagation program. The lower portion of the domain between the CFD near field and the ground plane is where the pressure signature propagation method will be active. Given the flow field conditions, w_o, the propagation altitude, and the altitude dependent atmospheric properties $\rho(z), p(z), T(z)$, the propagation method produces a flow solution at the ground plane we are interested in, which can be used to determine any of a variety of measures of sonic boom impact such as overpressures, rise time, impulse, etc.

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The Remote Inverse Design Problem using Control Theory

The aerodynamic properties that define the cost function are functions of the flow-field variables, w, and the physical location of the boundary, which may be represented by the function S.

Suppose that the performance is measured by a cost function

$$I = \varpi_1 \int_{\mathcal{B}_W} \mathcal{M}(w, S)\, d\mathcal{B}_\xi + \varpi_2 \int_{\mathcal{B}_{NF}} \mathcal{N}(w, S)\, d\mathcal{B}_\xi, \quad (1)$$

containing both wall boundary ($\mathcal{B}_W$) and near field boundary ($\mathcal{B}_{NF}$) contributions, where $d\mathcal{B}_\xi$ includes the surface and near field elements in the computational domain, while ϖ_1 and ϖ_2 are the weighting coefficients. The design problem is now treated as a control problem where the boundary shape represents the control function, which is chosen to minimize I subject to the constraints defined by the flow equations. A shape change produces a variation in the flow solution δw and the metrics δS which in turn produce a variation in the cost function

The weak form of the Euler equations for steady flow is

$$\int_{\mathcal{D}} \frac{\partial \psi^T}{\partial \xi_i} \delta F_i d\mathcal{D} = \int_{\mathcal{B}} n_i \psi^T \delta F_i d\mathcal{B}. \quad (2)$$

where the test vector ψ is an arbitrary differentiable function and n_i is the outward normal at the boundary. The domain can then be split into two parts: First, the near field domain ($\mathcal{D}_1$) whose boundaries are the wing surface and the near field boundary plane. Second, the far field domain ($\mathcal{D}_2$) which borders the near field domain along the near field boundary plane and the far field boundary. Since equation (2) equals zero, it may be subtracted from the variation in the cost function (1) to give

$$\begin{aligned} \delta I &= \int_{\mathcal{B}_W} \left[\varpi_1 \delta \mathcal{M} - n_i \psi^T \delta F_i \right] d\mathcal{B}_\xi \\ &+ \int_{\mathcal{B}_{NF}} \left[\varpi_2 \delta \mathcal{N} - n_i \left(\psi^+ - \psi^- \right)^T \delta F_i \right] d\mathcal{B}_\xi \\ &+ \int_{\mathcal{D}_1} \frac{\partial \psi^T}{\partial \xi_i} \delta F_i d\mathcal{D}_\xi + \int_{\mathcal{D}_2} \frac{\partial \psi^T}{\partial \xi_i} \delta F_i d\mathcal{D}_\xi. \end{aligned} \quad (3)$$

where ψ^+ and ψ^- are the values of ψ above and below the boundary. Now, since ψ is an arbitrary differentiable function, it may be chosen in such a way that δI no longer depends explicitly on the variation of the state vector δw. The gradient of the cost function can then be evaluated directly from the metric variations without having to re-compute the variation δw resulting from the perturbation of each design variable.

Figure 2 illustrates the baseline and optimized airfoil section at span station $z = 0.0957$. Figure 3 shows the initial and final near field pressure distribution after 50 design cycles. The final peak pressure has been reduced to almost 40% its original value at the root section and 25% at the mid-span section. A closer observation of the optimized shape revealed that in order to reduce the farfield pressure peak, a large expansion region is required to weaken the leading edge compression wave. Therefore an airfoil with a slightly larger leading edge angle but with a shorter compression than expansion region was produced.

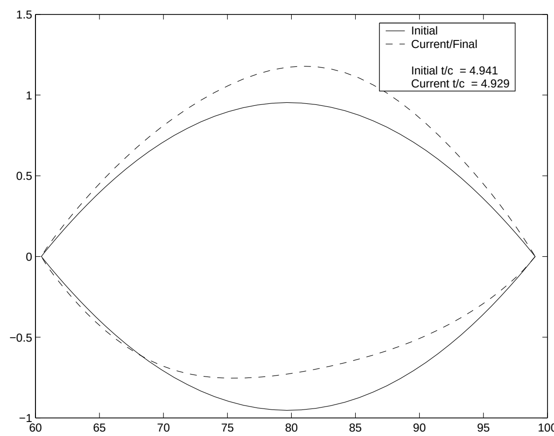

Figure 2: Sonic Boom Minimization: Initial and Final Airfoil Shape After 50 Design Cycles.

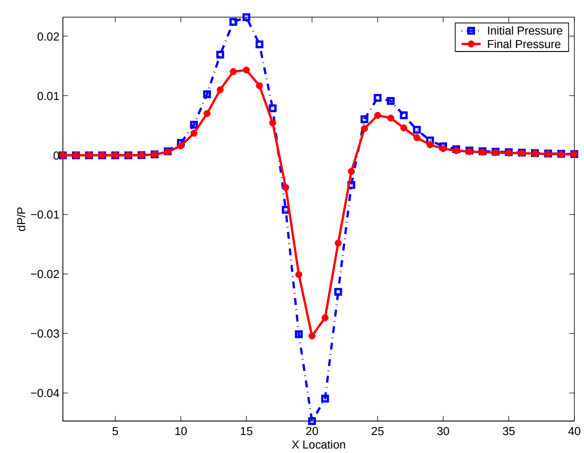

Figure 3: Sonic Boom Minimization: Target, Initial, and Final Near Field Pressure Distribution After 50 Design Cycles. Mach = 1.5, α = 1.75 deg, Fixed Lift Coefficient = 0.05, Fixed Thickness Ratio