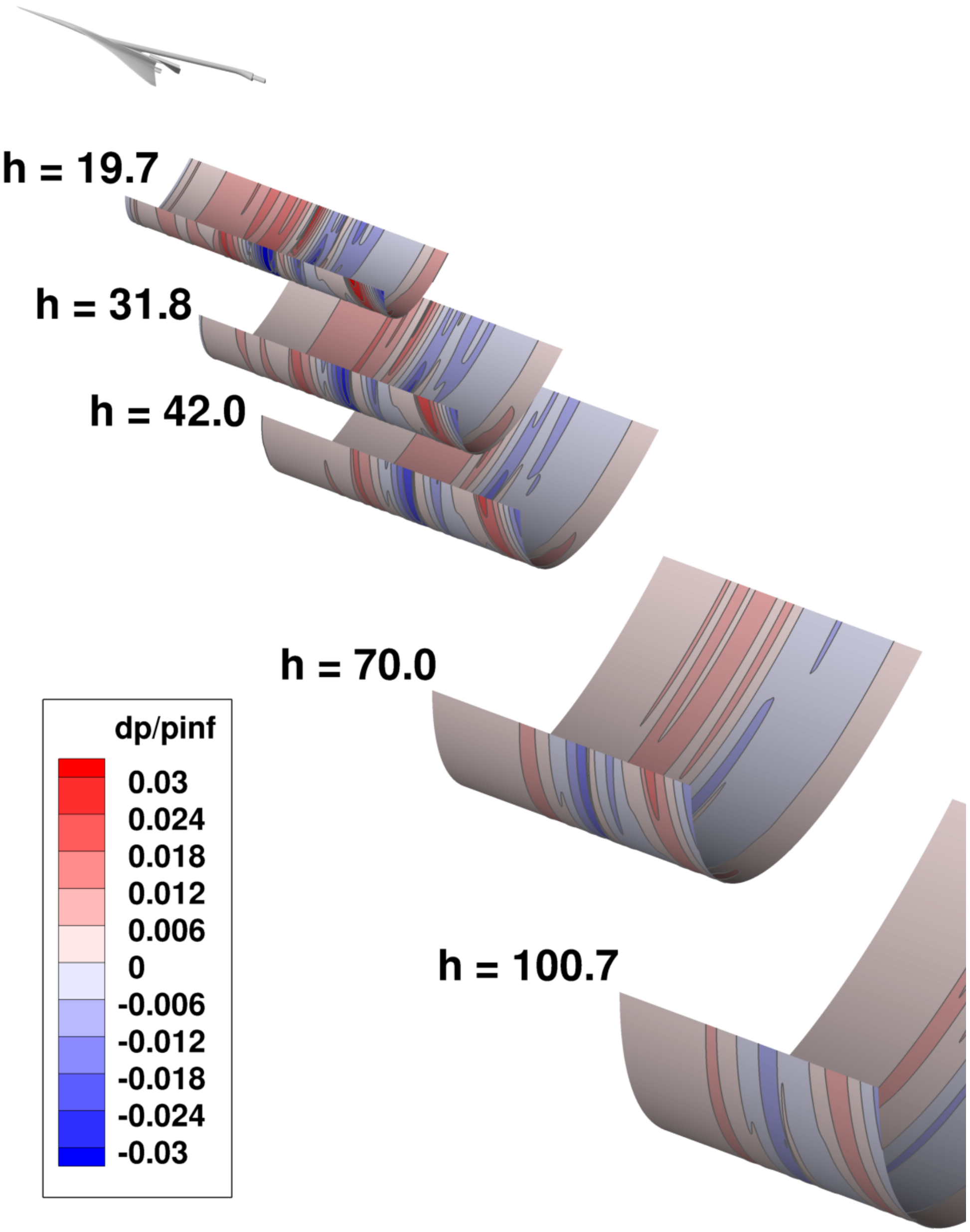
LAVA Simulations for the AIAA Sonic Boom Prediction Workshop

Jeffrey Housman, Emre Sozer, Shayan Moini-Yekta, and Cetin Kiris

Applied Modeling and Simulation Branch
NASA Ames Research Center

32nd AIAA Applied Aerodynamics Conference
June 16-20, 2014
Atlanta, Georgia

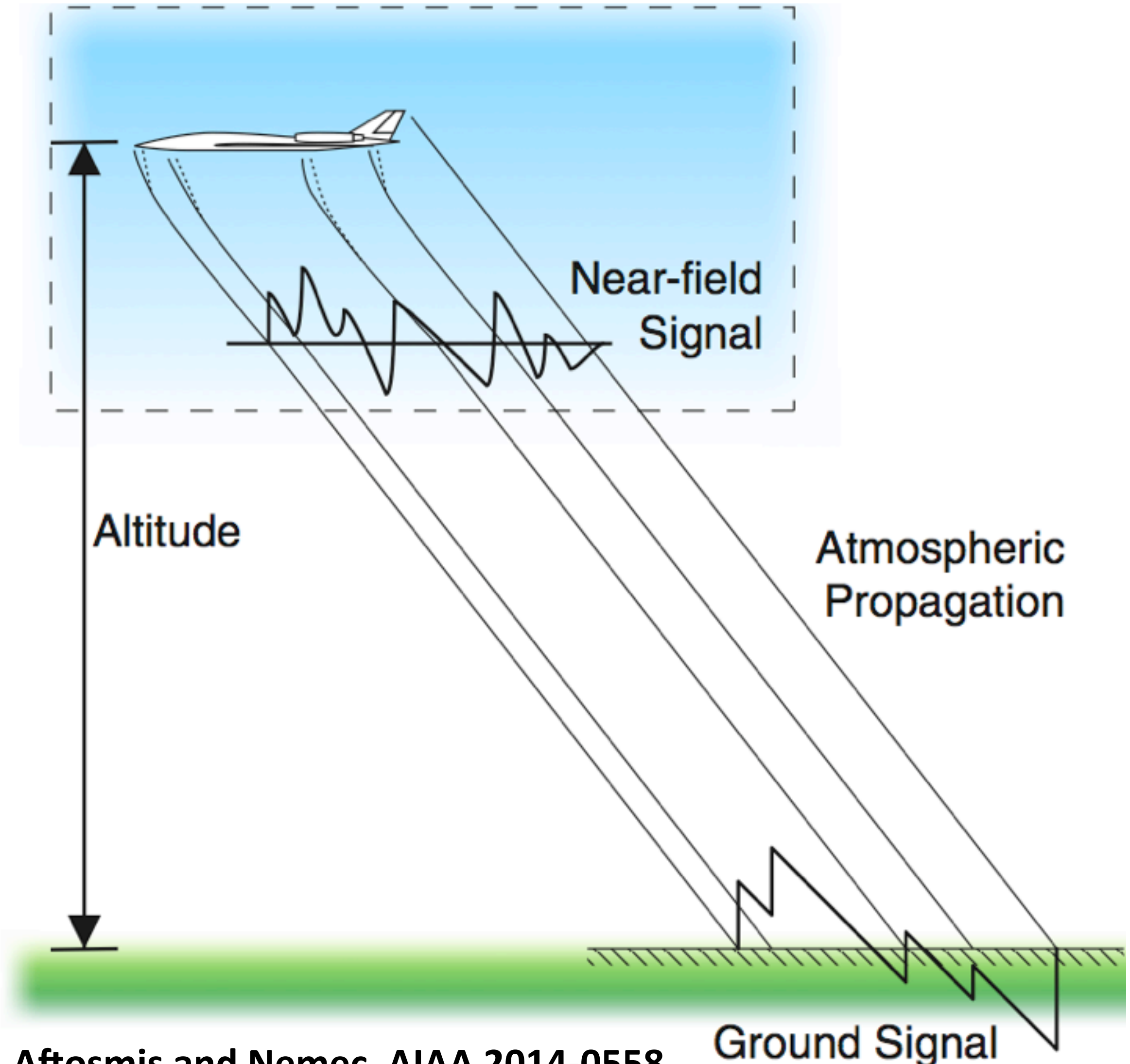
Outline



- Introduction
- Launch Ascent and Vehicle Aerodynamics (LAVA) Framework
- Results
  - Computational Grids
  - Comparison
  - Sensitivity Analysis
- Summary

Introduction

Supersonic Civilian Aircraft



Aftosmis and Nemec, AIAA 2014-0558

- No civilian supersonic aircraft since retirement of Concorde in 2003
- Renewed interest in sonic boom minimization over last decade
- CFD can be a useful tool in the design process
- Accuracy of CFD prediction must be assessed

Introduction

1st AIAA Sonic Boom Prediction Workshop

- Workshop is designed to assess the state-of-the-art in CFD simulation capabilities for sonic boom prediction
- Three models of increasing geometric complexity are included in the study
  - SEEB-ALR
  - 69 Degree Delta-Wing Body
  - Lockheed Martin 1021 model
- LAVA results using structured and unstructured grids have been submitted to the workshop

Lockheed Martin 1021 model

LAVA Framework

*Launch Ascent and Vehicle Aerodynamics Framework**

- Computational Fluid Dynamics (CFD) Solver
  - Cartesian, Curvilinear, and Unstructured Grid Types
  - Overset Grid and Immersed Boundary Methods
  - Reynolds Averaged Navier-Stokes and Detached Eddy Simulation Capabilities

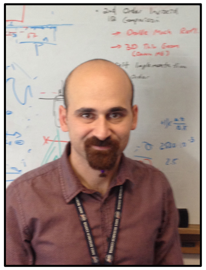
Meshing	Convective Flux Discretization	Turbulence Model	Linear Solvers
Structured Overset	Modified Roe and Central	Spalart-Allmaras	Alt. Line Jacobi
Unstructured Polyhedral	AUSMPW+	Spalart-Allmaras	GMRES

Development Team

Cetin Kiris

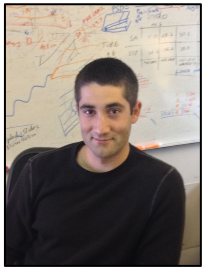
Jeffrey Housman

Michael Barad



Emre Sozer

Christoph Brehm

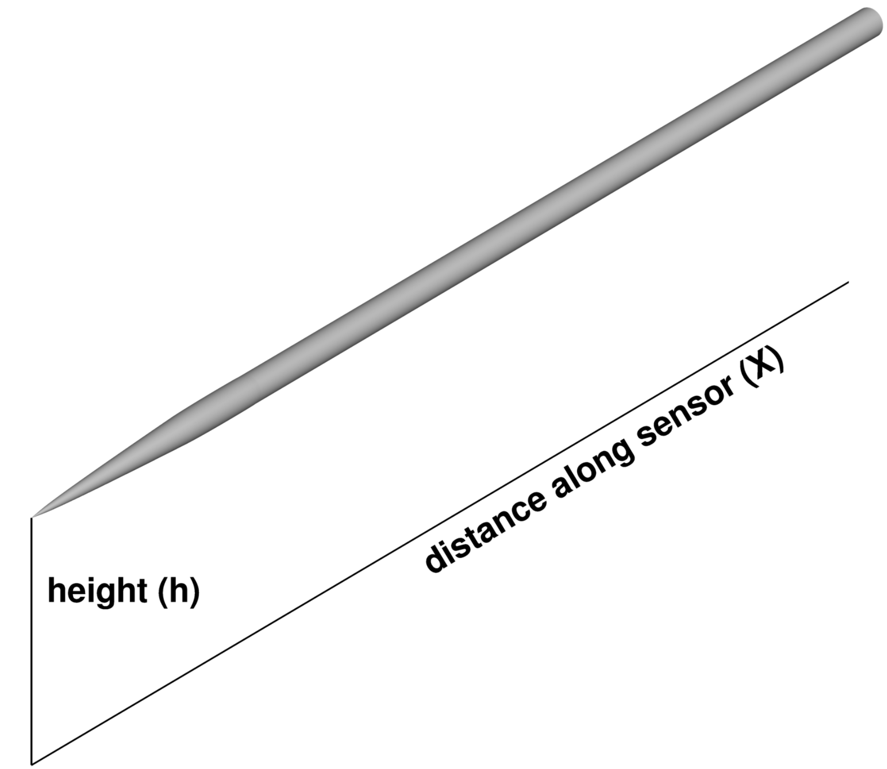


Shayan Moini-Yekta

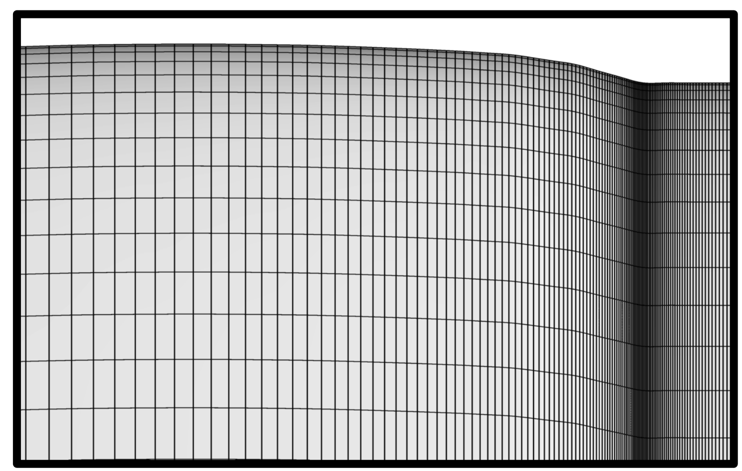
*Kiris et. al. AIAA-2014-0070

Seeb-ALR

Geometric Model



- Axisymmetric model designed for low boom and low drag
- Seeb model modified downstream of shoulder (ALR)
- Model Length: 17.67 inches
- Computation Model: 68.3 inches
- Inviscid Analysis: Mach = 1.6

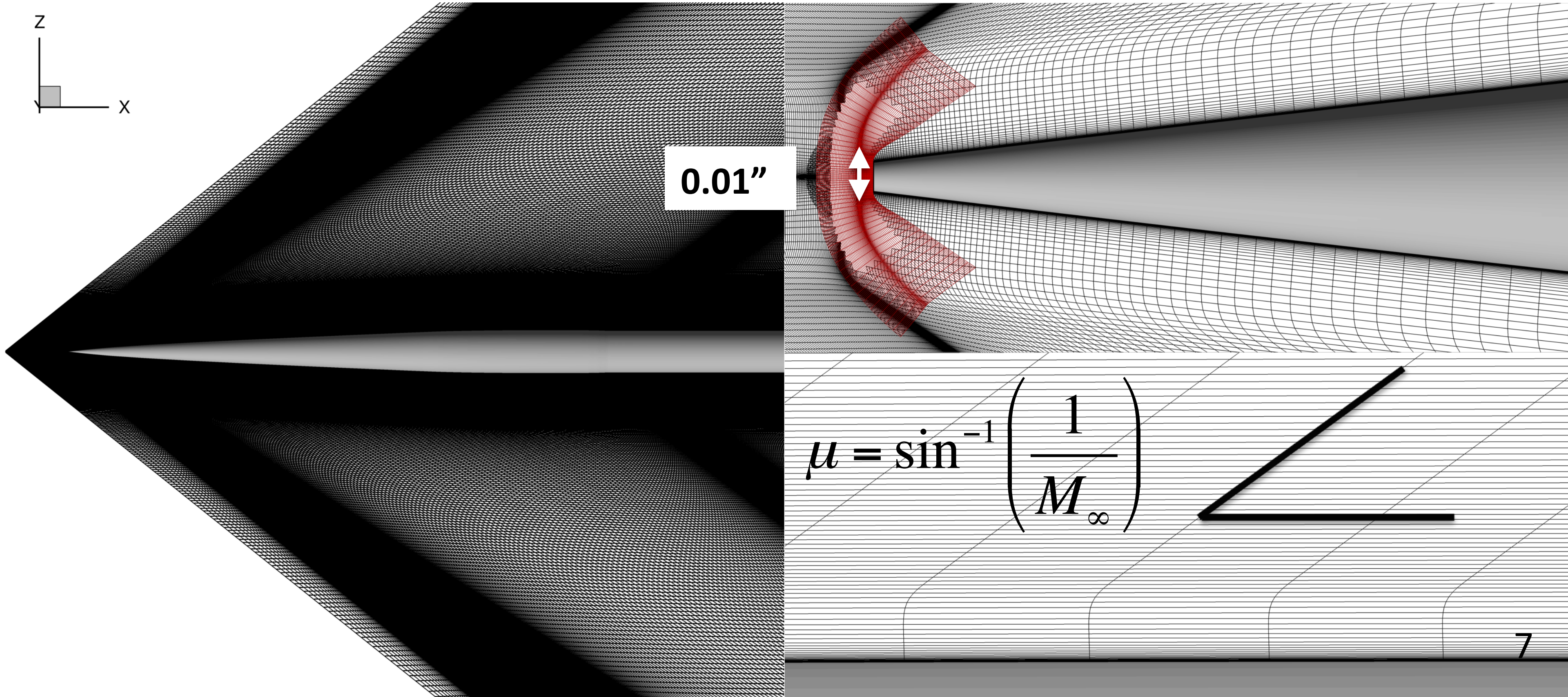


Scaled 10:1 in r

Seeb-ALR

Computational Grid

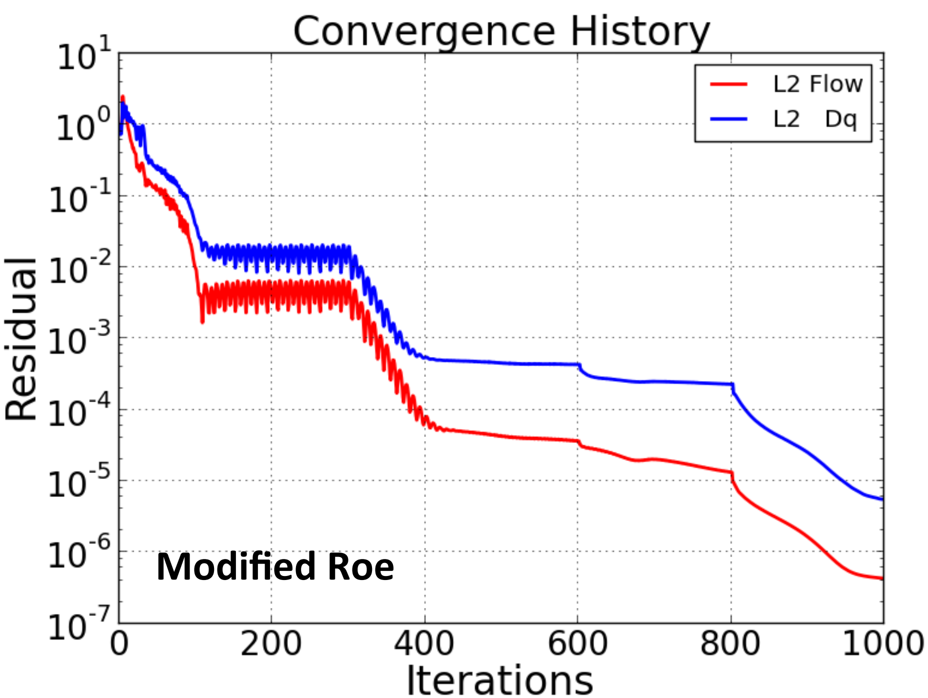
- 4 zones and 21.7 million grid points
- Near-body marched normal to surface then turned to Mach-angle aligned mesh
- Local bow shock capturing grid at near blunt nose

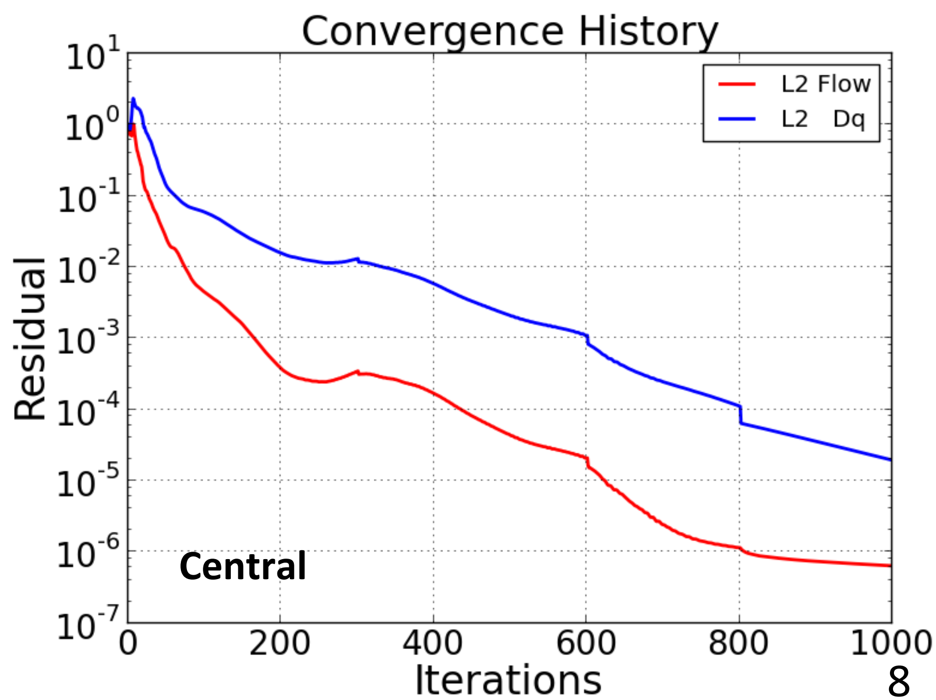


Seeb-ALR

Computational Requirements and Residual Convergence

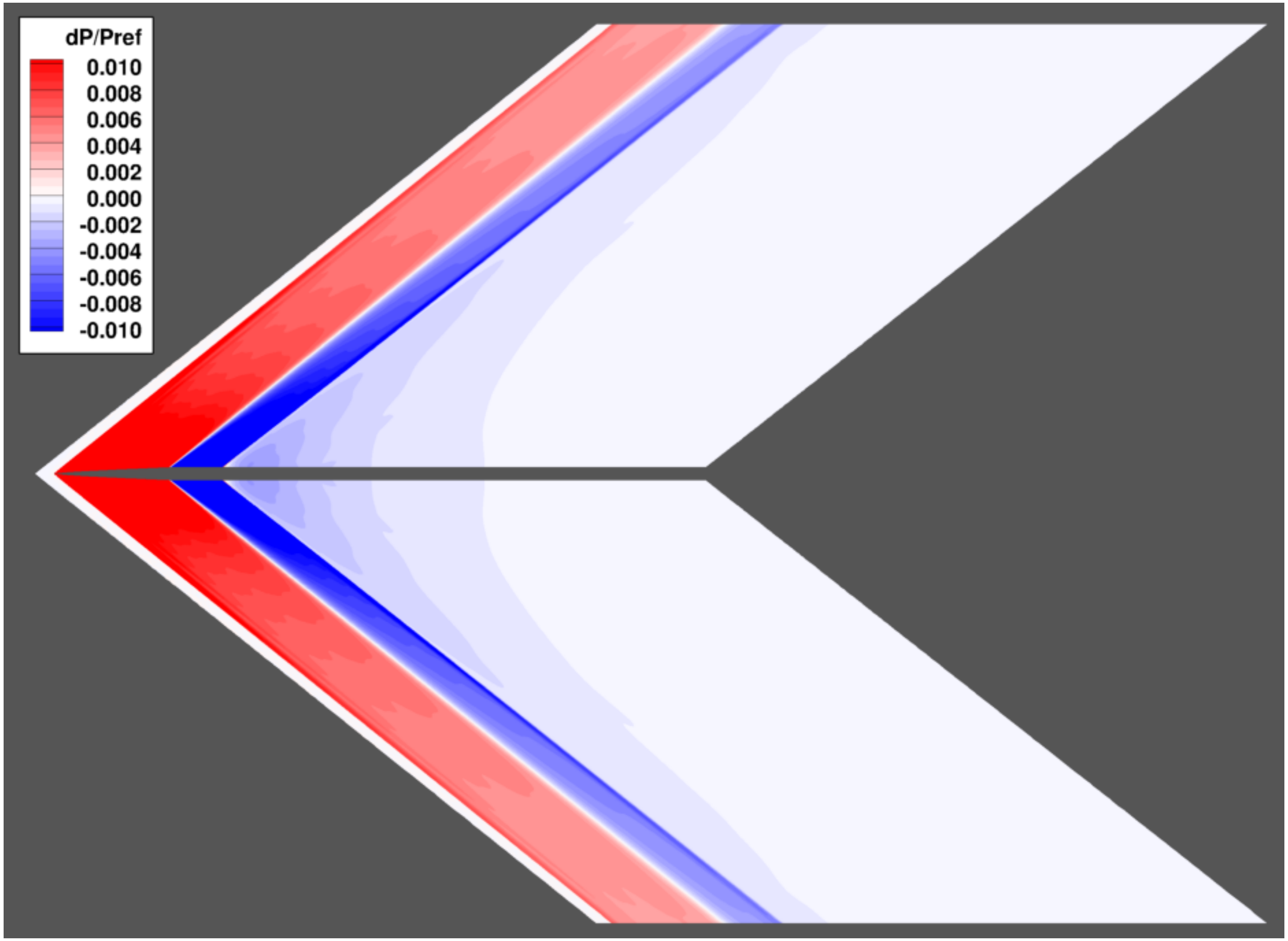
Flux	CPU Type	Cores	Walltime	Total Core Hours
Modified Roe	Westmere	48	1 hr. 30 min.	72.0
Central	Westmere	48	1 hr. 18 min.	62.4





Seeb-ALR

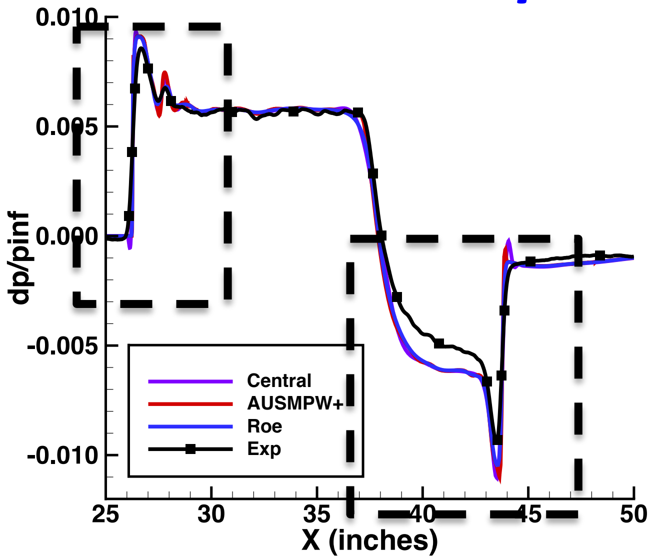
Flow Field Visualization

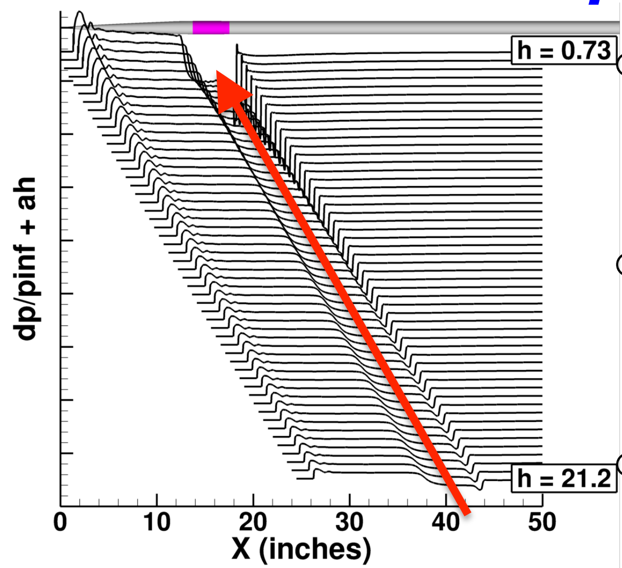


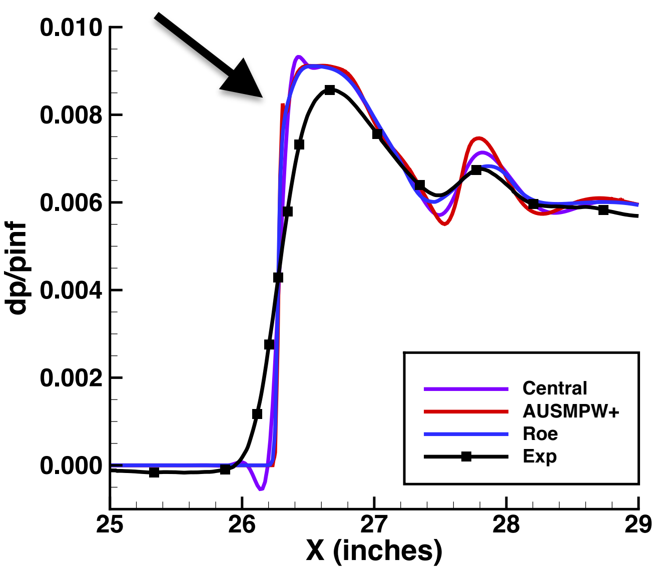
- **Bow shock forms at blunt nose of the model**
- **Secondary shock is generated from small slope change near the nose**
- **Rarefaction wave develops downstream of the shoulder**

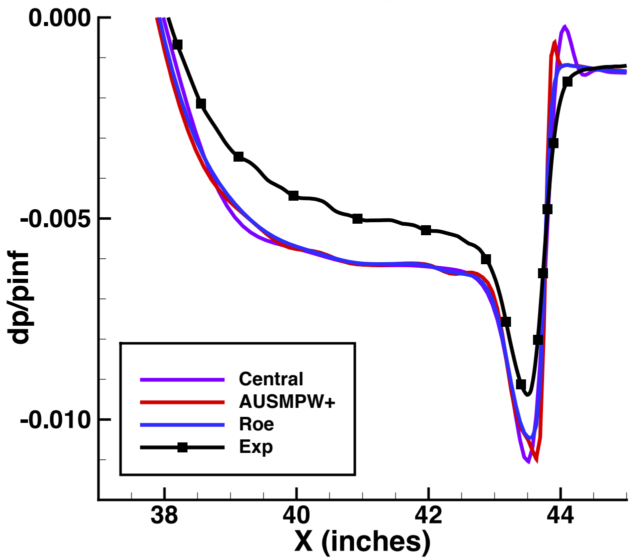
Seeb-ALR

Results and Comparison: h = 21.2 inches ϕ = 0 degrees





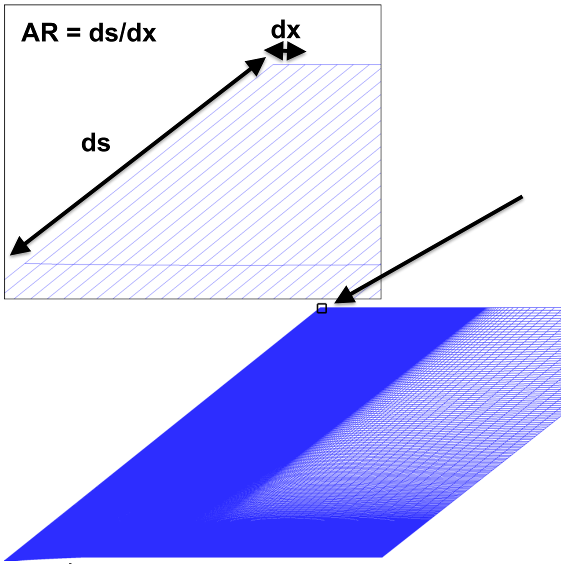


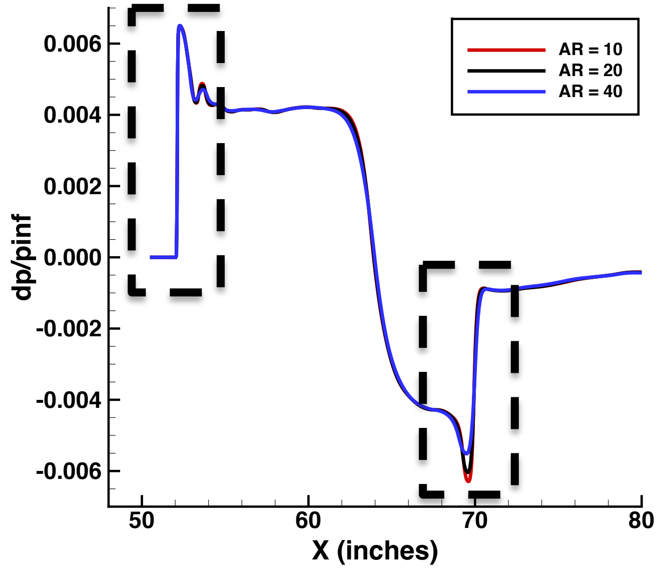


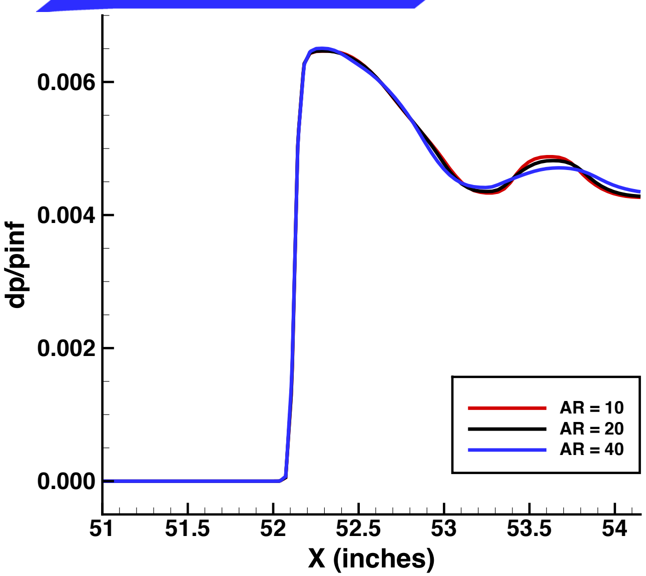
- Good comparison observed over most of the signal
- Bow shock and secondary shock well-captured by CFD
- Experimental result shows a smoother primary shock
- Oscillations observed using Central and AUSMPW+ on bow shock
- CFD over-predicts the pressure decrease over the expansion

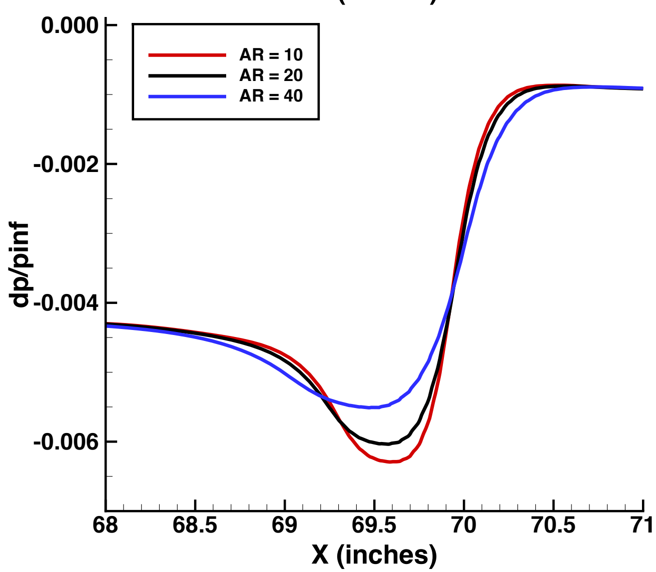
Seeb-ALR

Aspect Ratio Sensitivity Analysis: h = 42 inches ϕ = 0 °





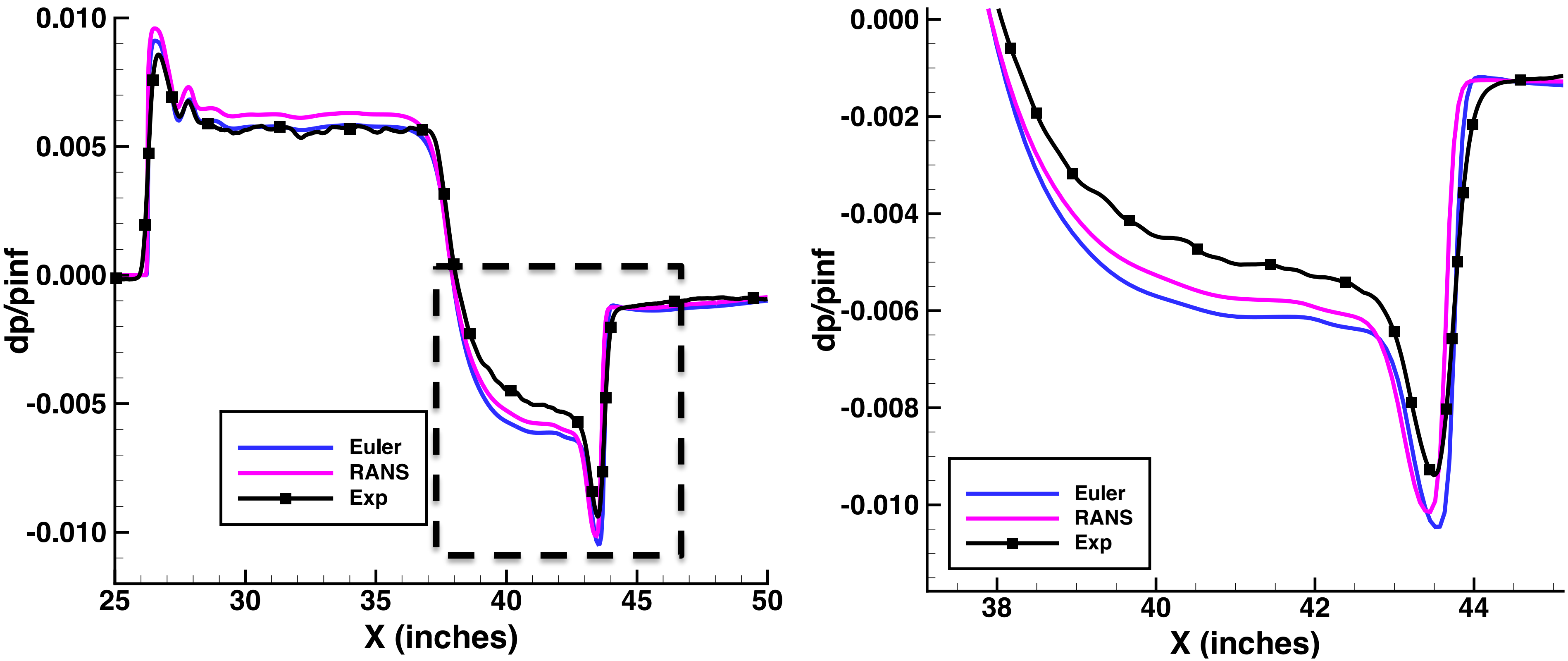




- **Sensitivity to grid Aspect Ratio (AR) at the outer-boundary**
- **No sensitivity observed in bow shock**
- **Secondary peak and pre-recovery peak pressures show some sensitivity to AR**
- **Change in peak decreases with decreasing AR**
- **AR = 20 submitted to the workshop**

Seeb-ALR

Viscous Sensitivity Analysis: h = 21.2 inches $\phi = 0^{o}$

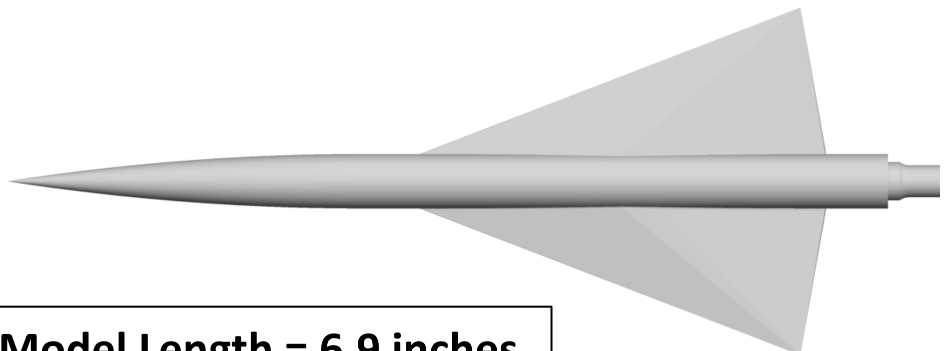


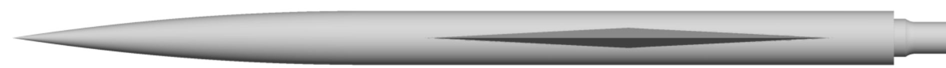
- A delta off-set in pressure signature is observed between the Euler and RANS results at both extraction locations
- Boundary layer appears to make the body effectively thicker
- The same offset was observed using both SA and SST models

69 Degree Delta Wing Body

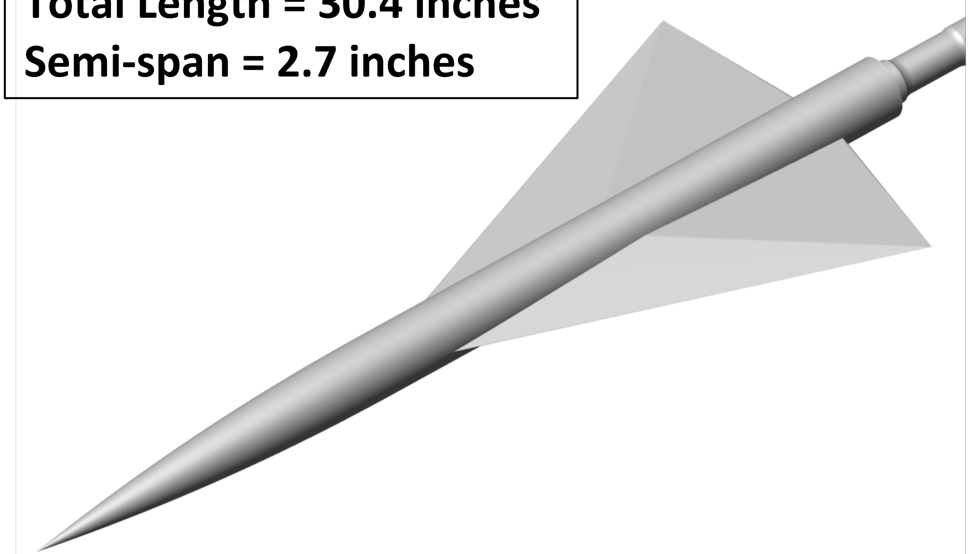
Geometric Model

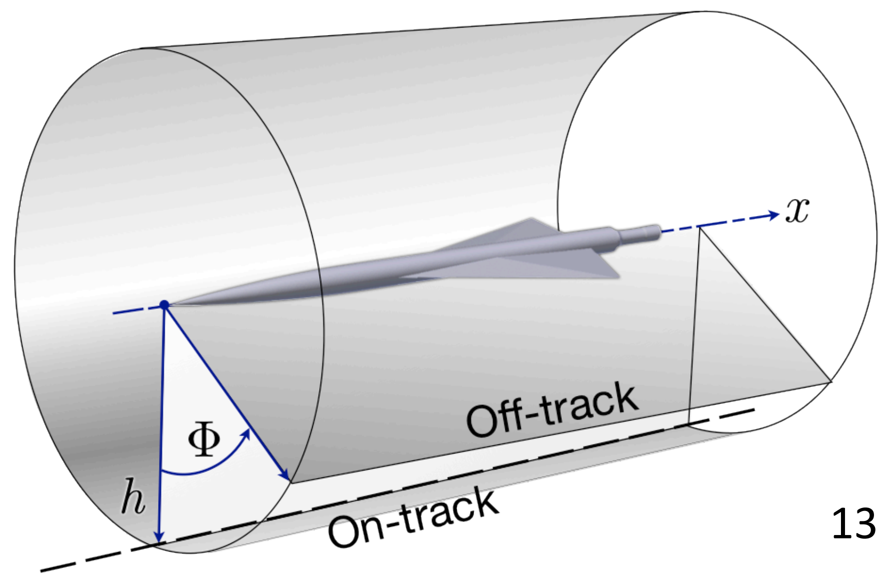
- 69 swept Delta Wing bisecting a cylindrical fuselage attached to an axisymmetric sting
- Mach = 1.7 Reynolds Number = 4.24 M (per ft)





Model Length = 6.9 inches
Total Length = 30.4 inches
Semi-span = 2.7 inches

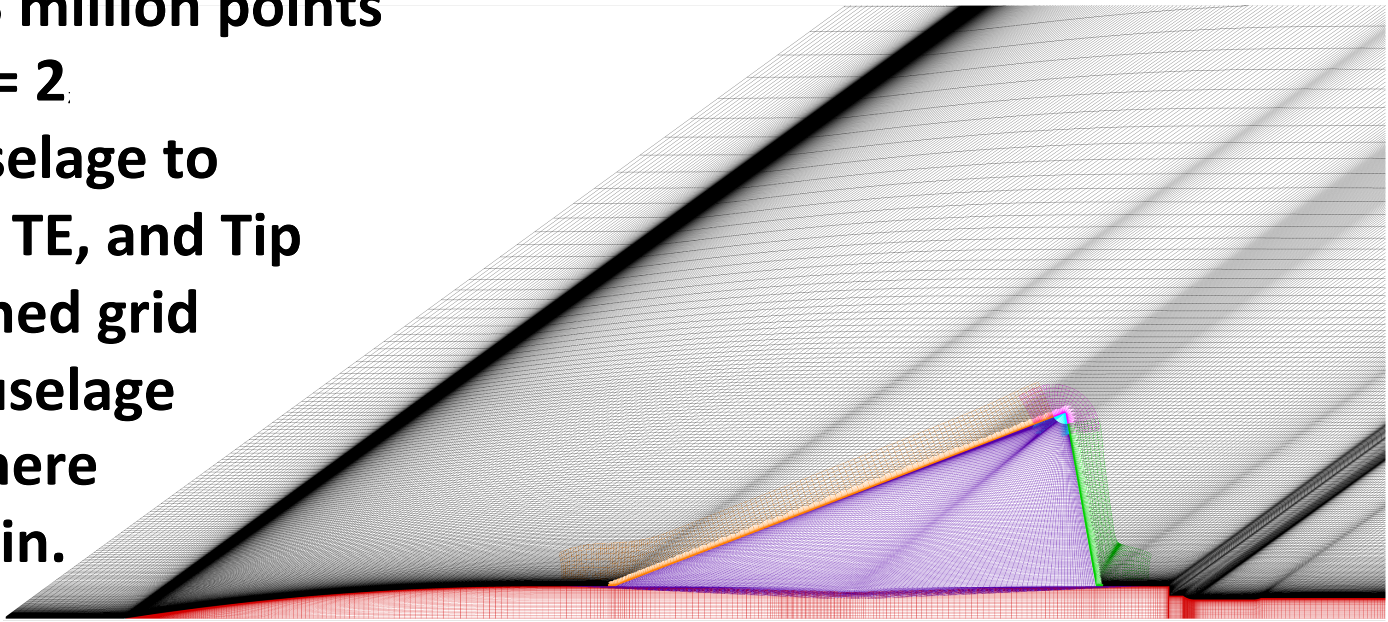


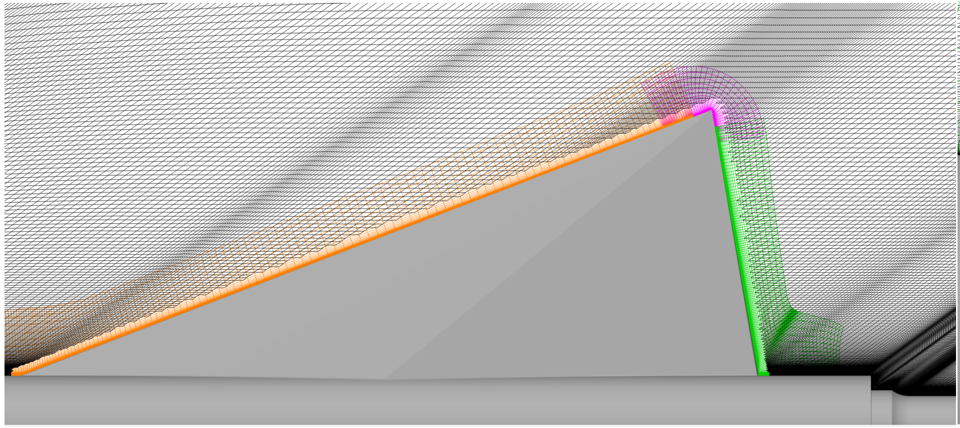


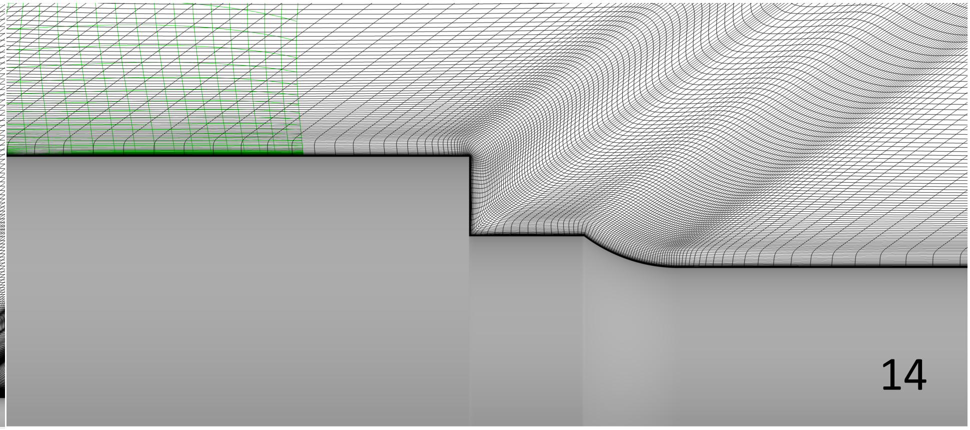
69 Degree Delta Wing Body

Computational Grid: Structured

- 8 zones and 21.3 million points
- Viscous Wall y+ = 2
- Clustering on fuselage to capture wing LE, TE, and Tip
- Mach-angle aligned grid marched from fuselage
- Cores: 48 Westmere
- Walltime: 109 min.
- Core hrs: 87.2



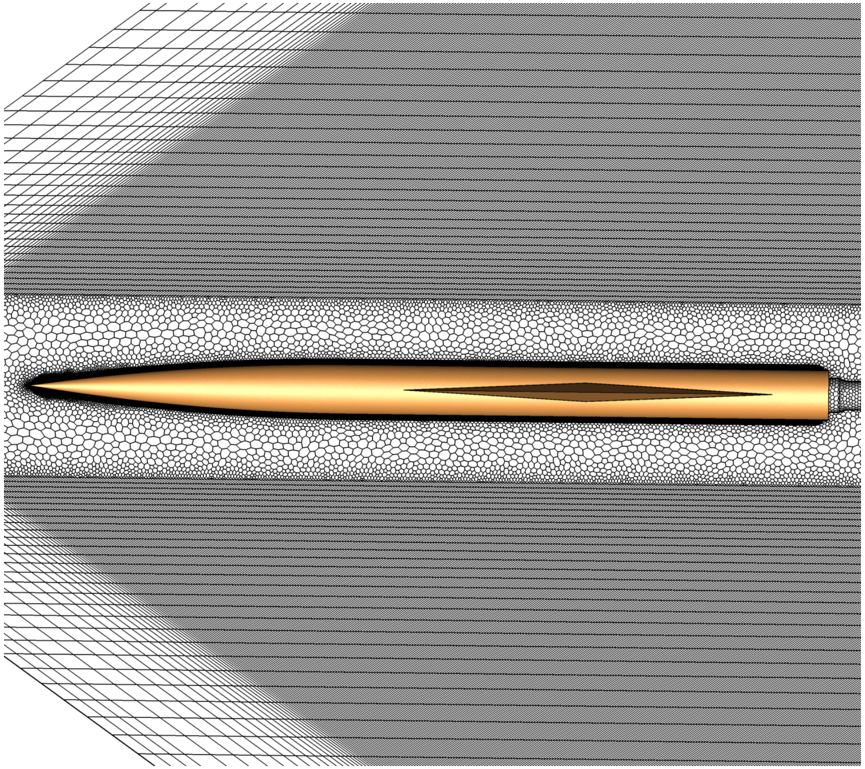


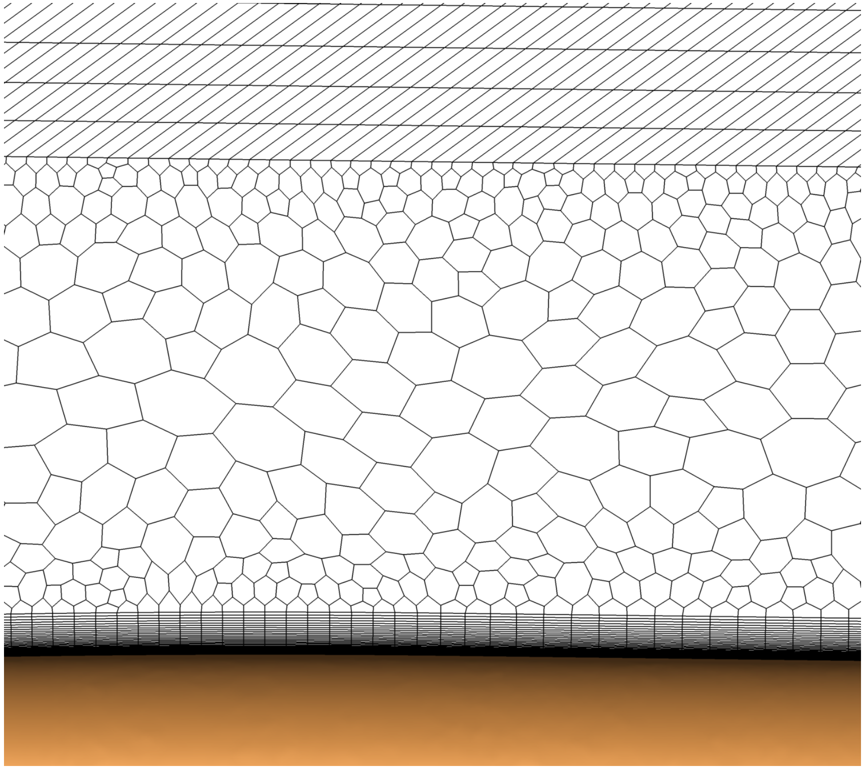


69 Degree Delta Wing Body

Computational Grid: Unstructured

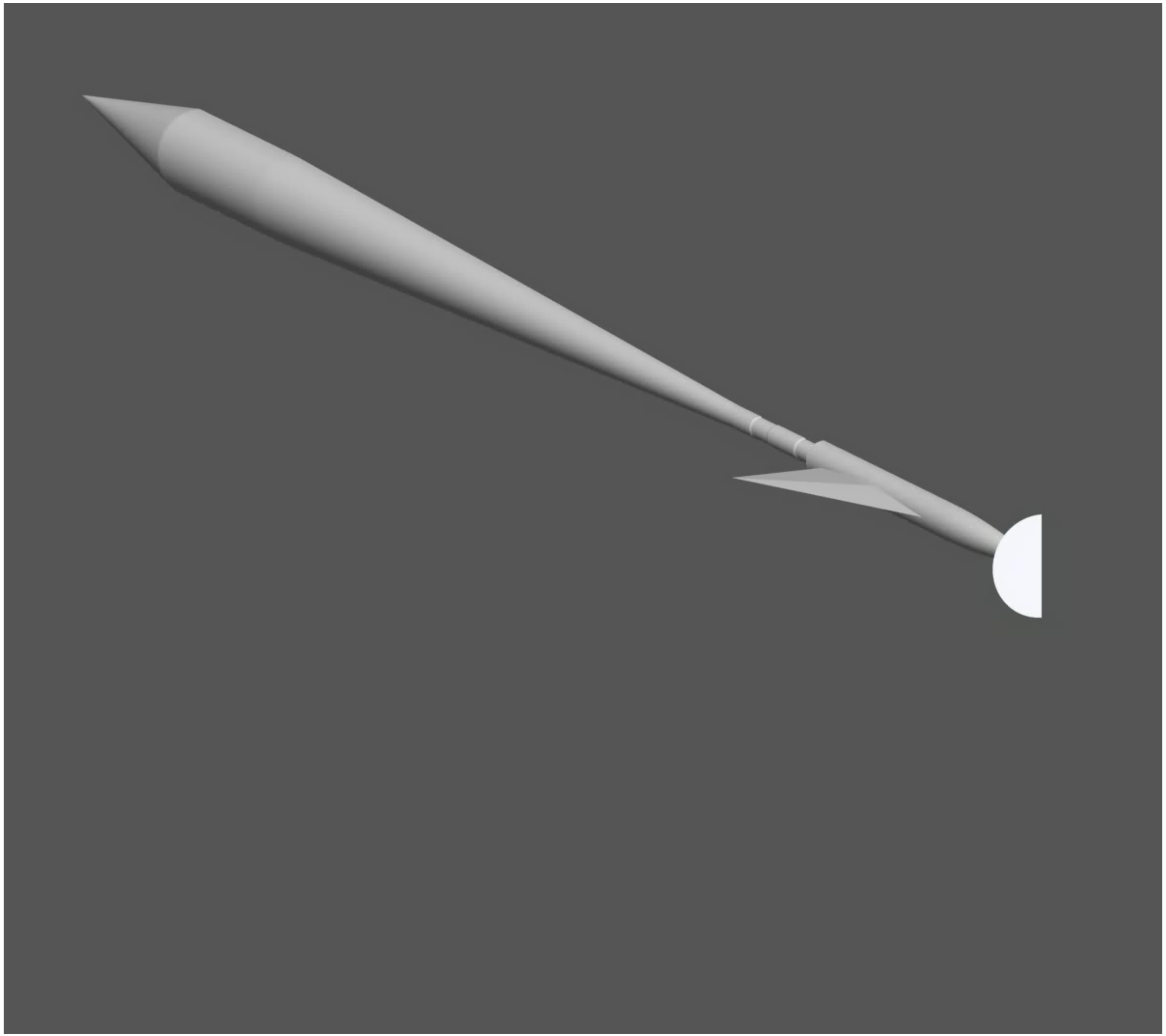
- 12.1 million polyhedral cells
- Core mesh utilizes isotropic cells
- Anisotropic prismatic layers grown from surface
- Mach-angle aligned mesh extruded from outer core boundary
- Cores: 320 Sandy Bridge; Walltime: 25 min.; CPU hrs: 133.3





69 Degree Delta Wing Body

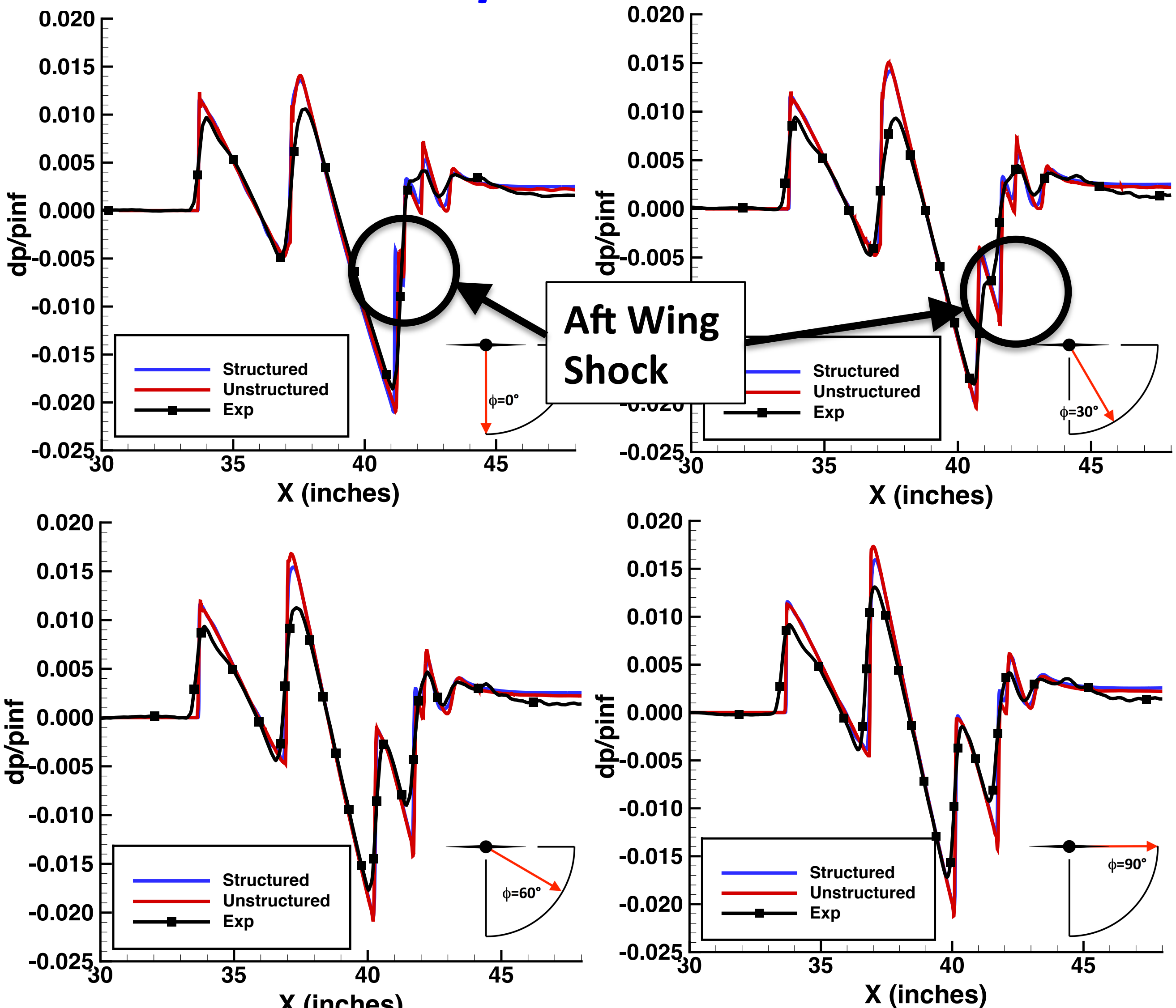
Flow Field Visualization



- Amplitude of pressure waves decay with radial distance (energy)
- Delta wing disturbs the symmetric signal generated by the fuselage
- Signal will eventually regain symmetry with increased radial distance (equivalent area)

69 Degree Delta Wing Body

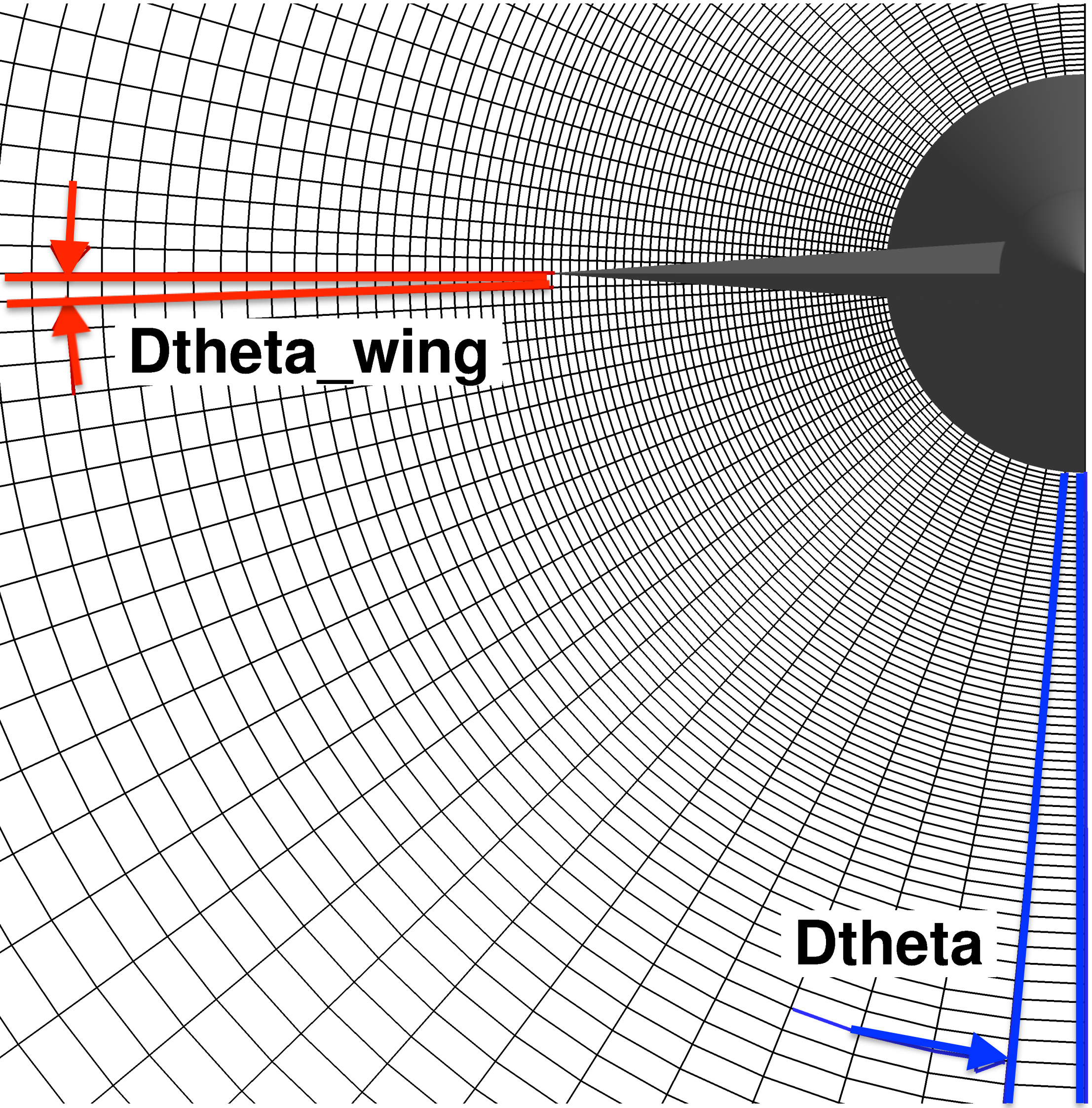
Results and Comparison: h = 24.8 inches

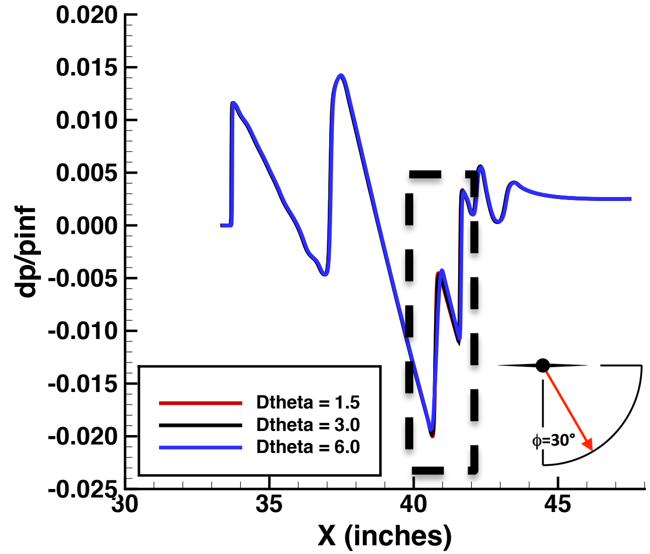


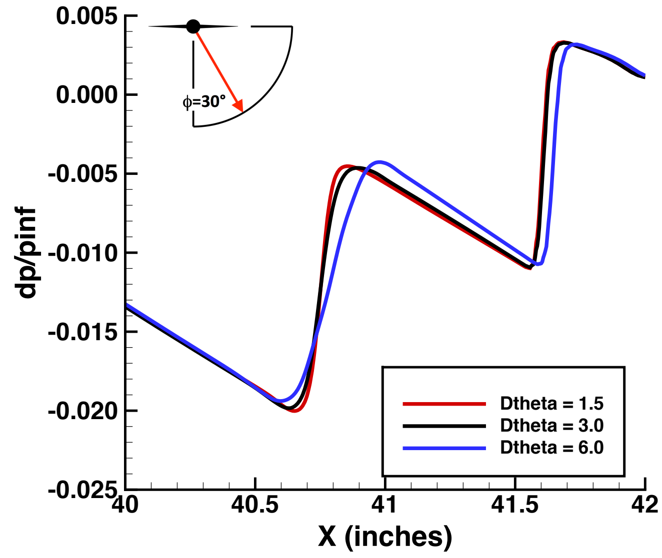
- Pressure peaks are more smooth in Experimental data
- Slope of rarefaction waves match very well
- Stronger aft wing shock predicted by CFD than Experiment at lower Φ angles

69 Degree Delta Wing Body

Circumferential Spacing Sensitivity Analysis: h = 24.8 inches



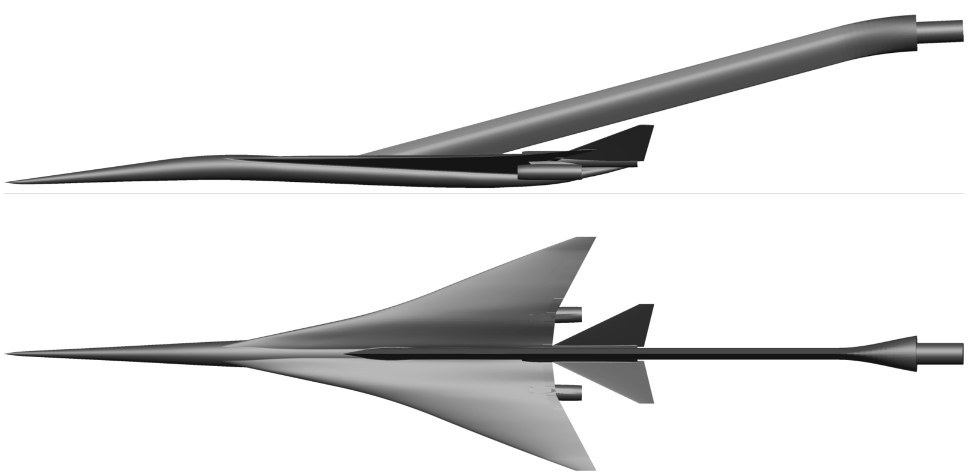


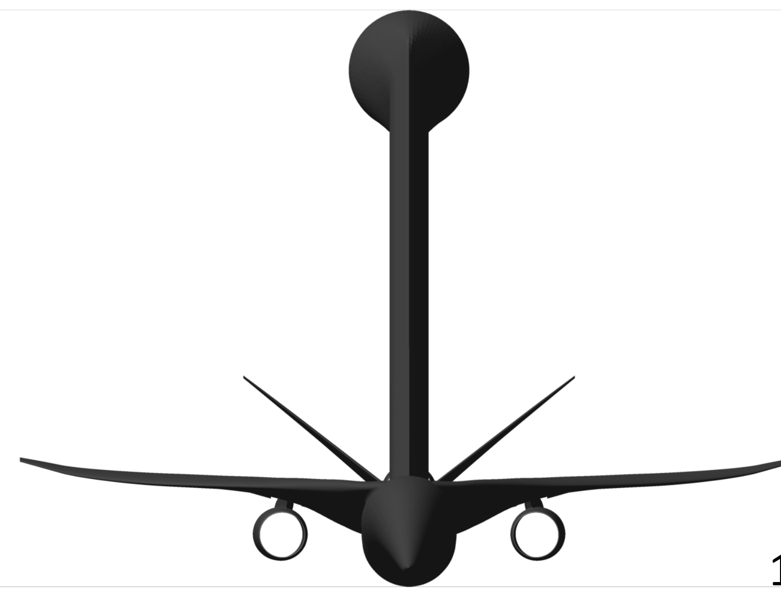


Lockheed Martin 1021

Geometric Model

- Lockheed Martin Phase I low sonic boom model
- Mach = 1.6, Reynolds number 4.36 M (per ft.), $\alpha = 2.1^{o}$
- Designed for low boom on-track and reduced pressure up to 20^{o}
- Model length 22.4 inches representing 0.8 % scale (1:125)
- Swept blade strut designed to minimize interference
- Trip disks added near wing leading edge to force transition

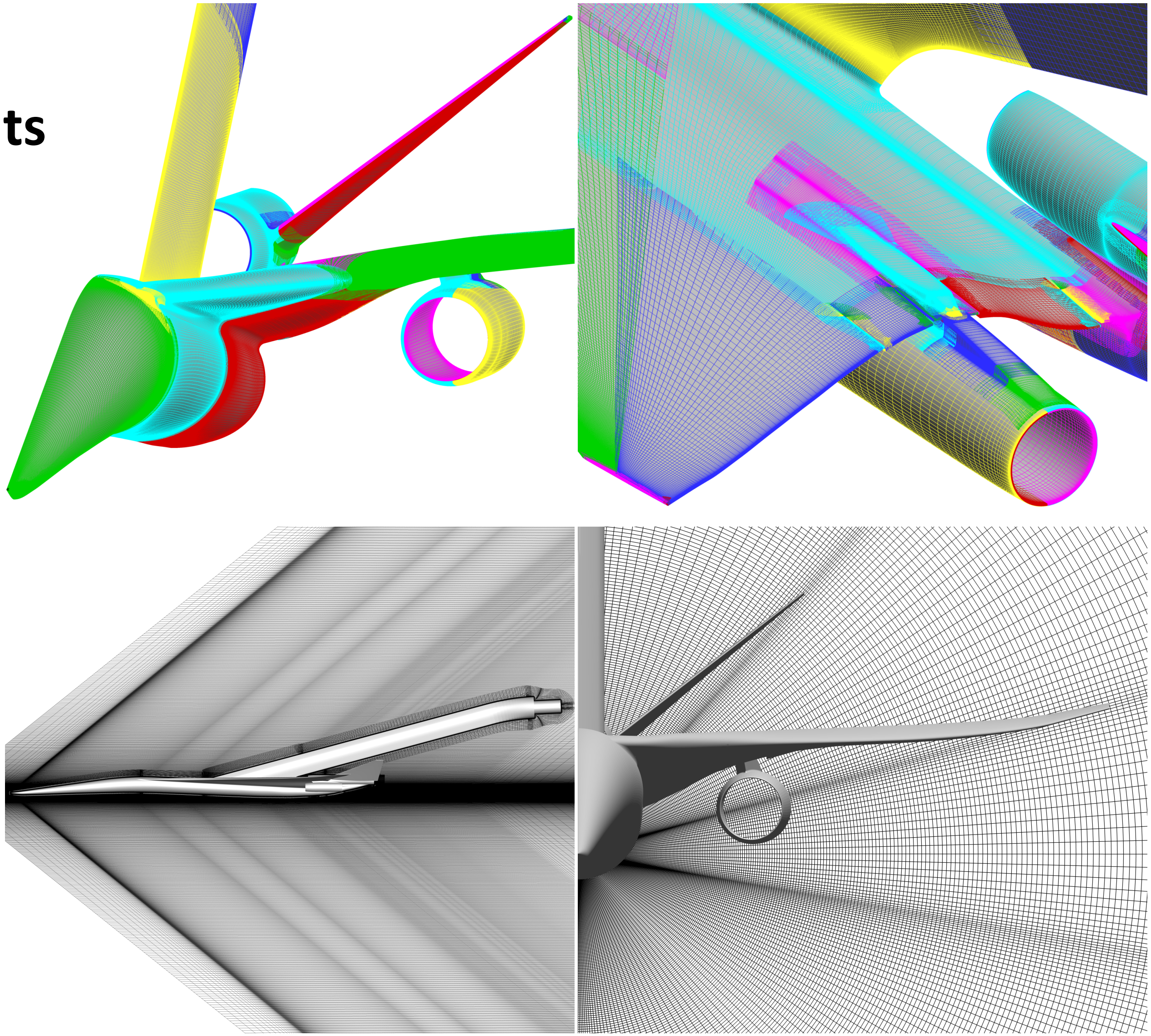




Lockheed Martin 1021

Computational Grid: Structured

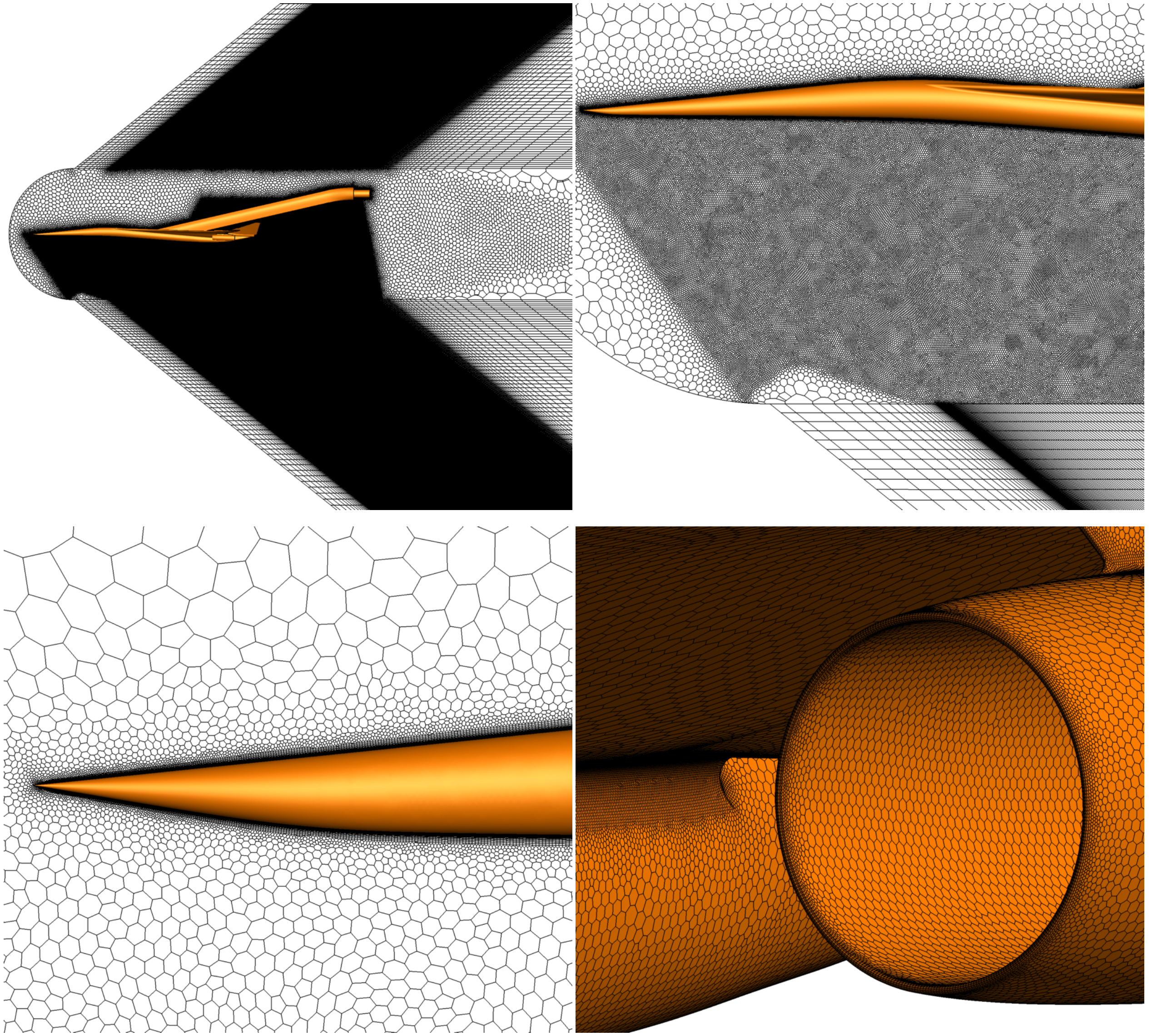
- 97 zones
- 72.7 million grid points
- Wall y+ = 1.2
- Mach-angle aligned grid detached from fuselage
- Clustering in streamwise and circumferential directions
- Cores: 180 Ivy Bridge
- Walltime: 90 min.
- Core hours: 270



Lockheed Martin 1021

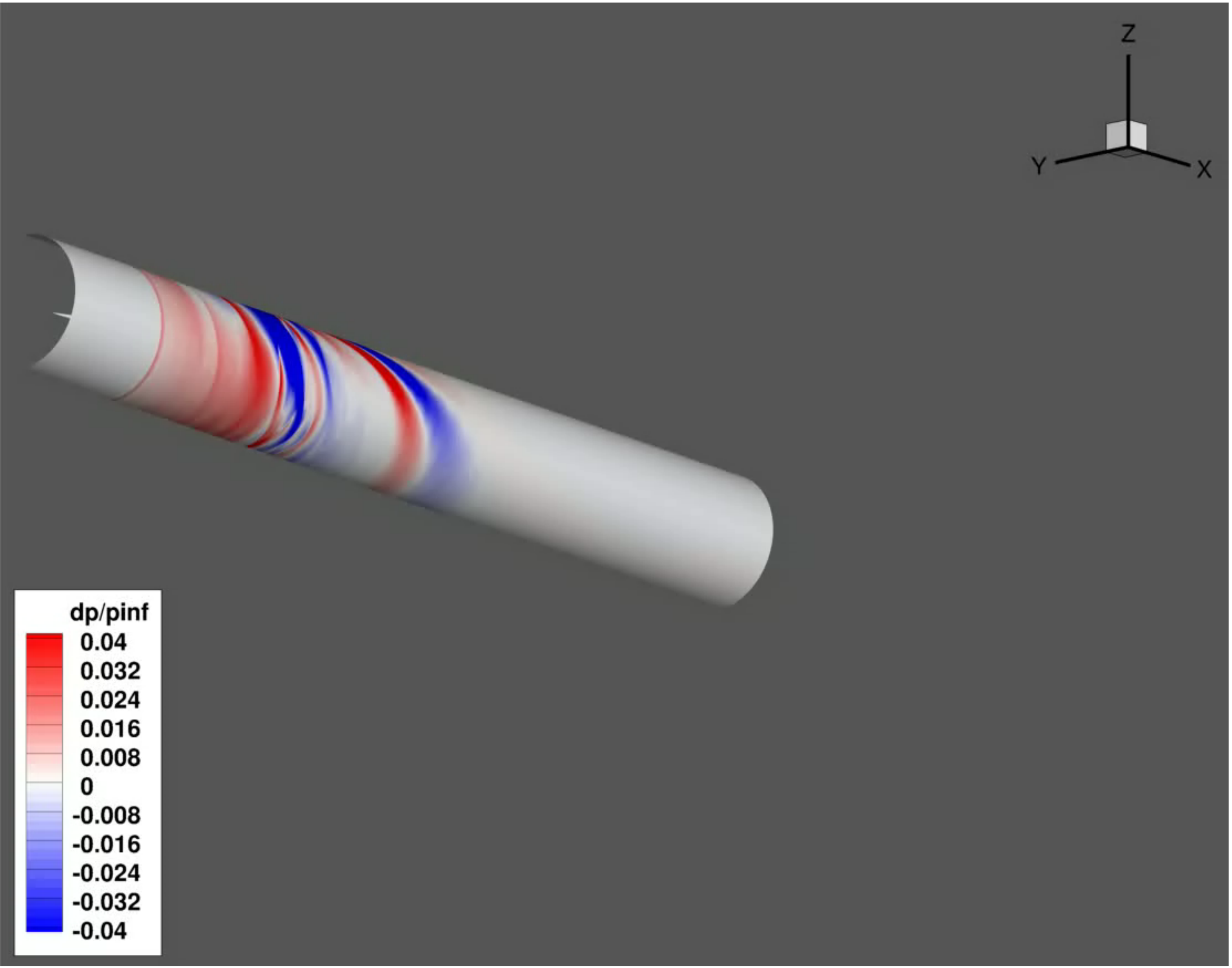
Computational Grid: Unstructured

- 65.5 million polyhedral cells
- Surface cell size specification used on symmetry plane for improved on track accuracy
- Fine surface mesh resolution with clustering near sharp geometric features
- Cores: 2000
- Walltime: 45 min.
- Core hours: 1500



Lockheed Martin 1021

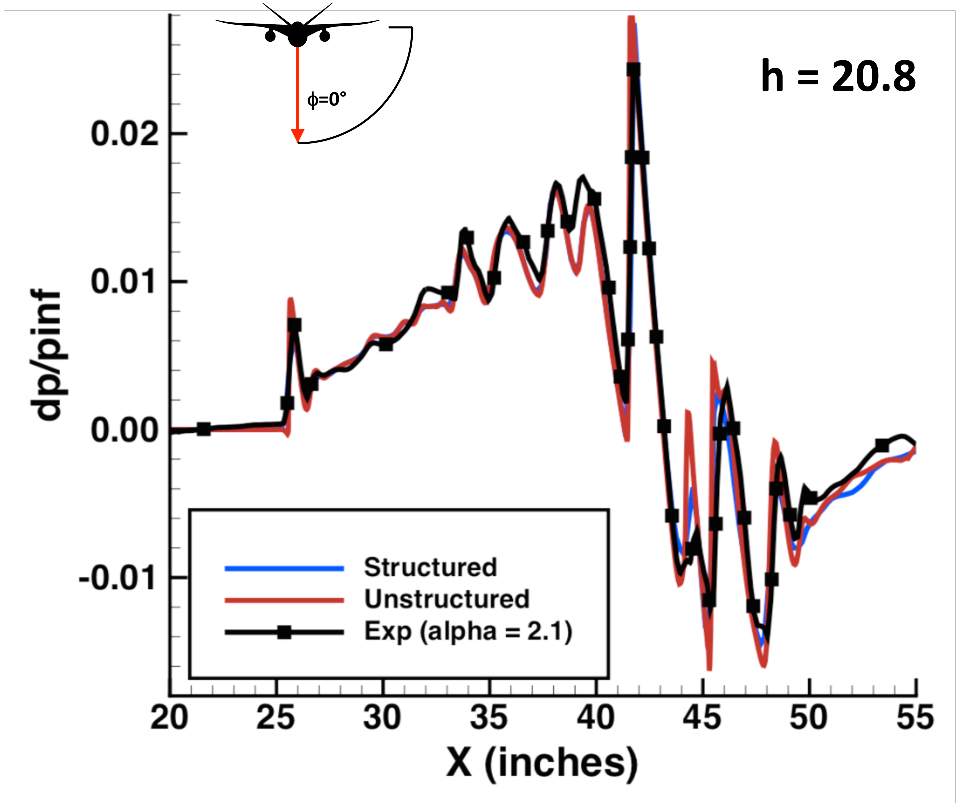
Flow Field Visualization

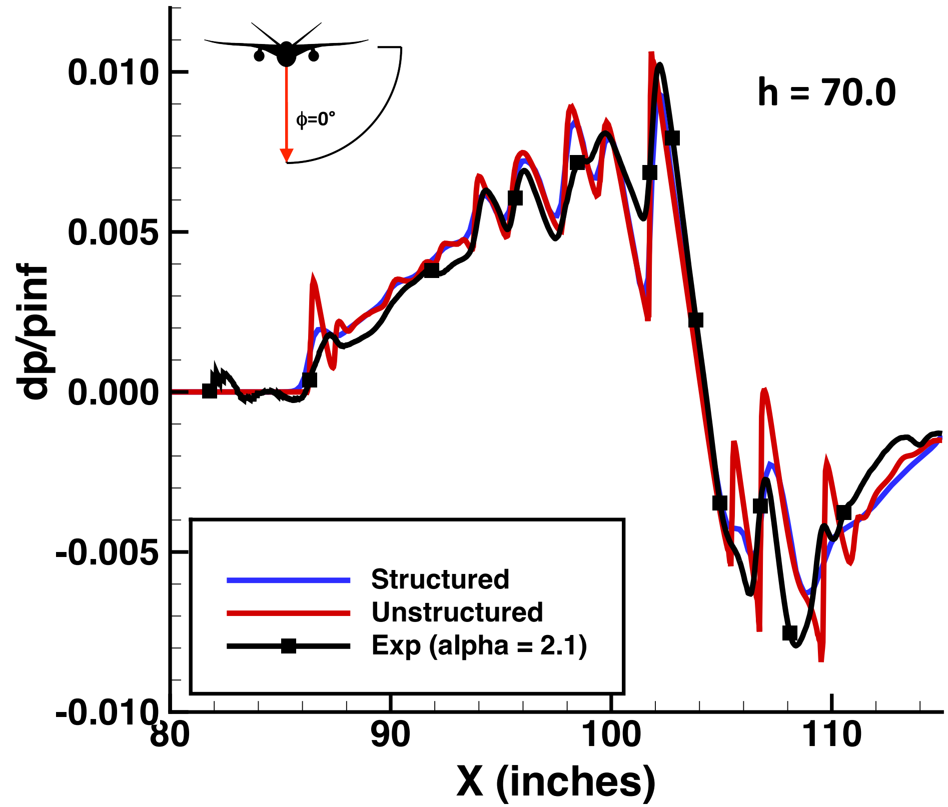


- **Complex wave pattern generated by the model**
- **Magnitude of peaks decay radially**
- **Waves coalesce below the vehicle to help reduce the sonic boom**

Lockheed Martin 1021

Results and Comparison: On-Track

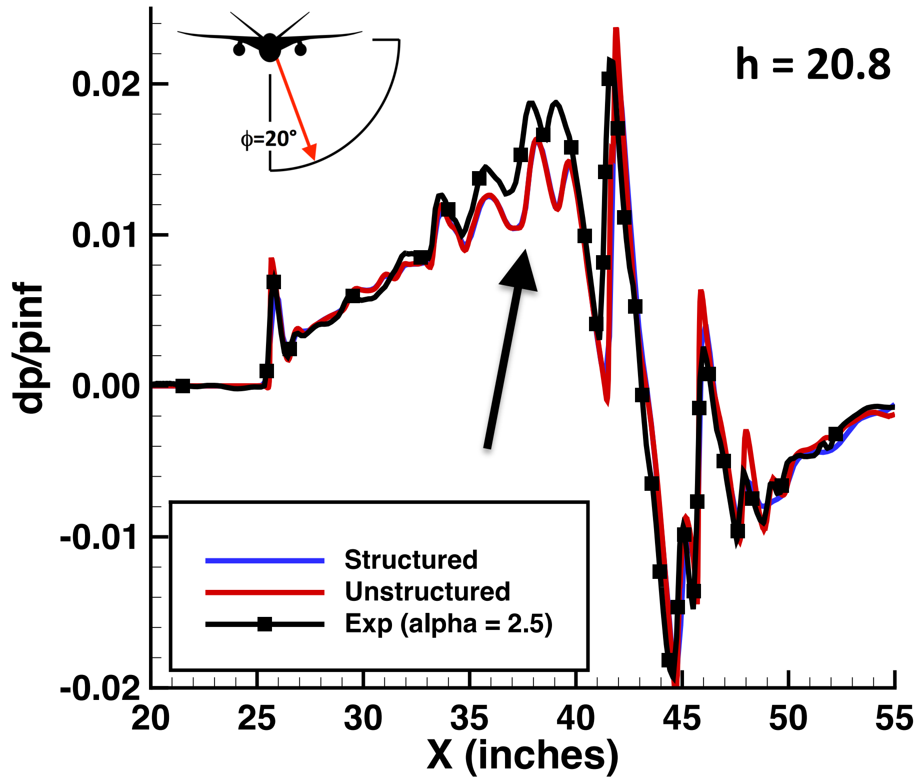


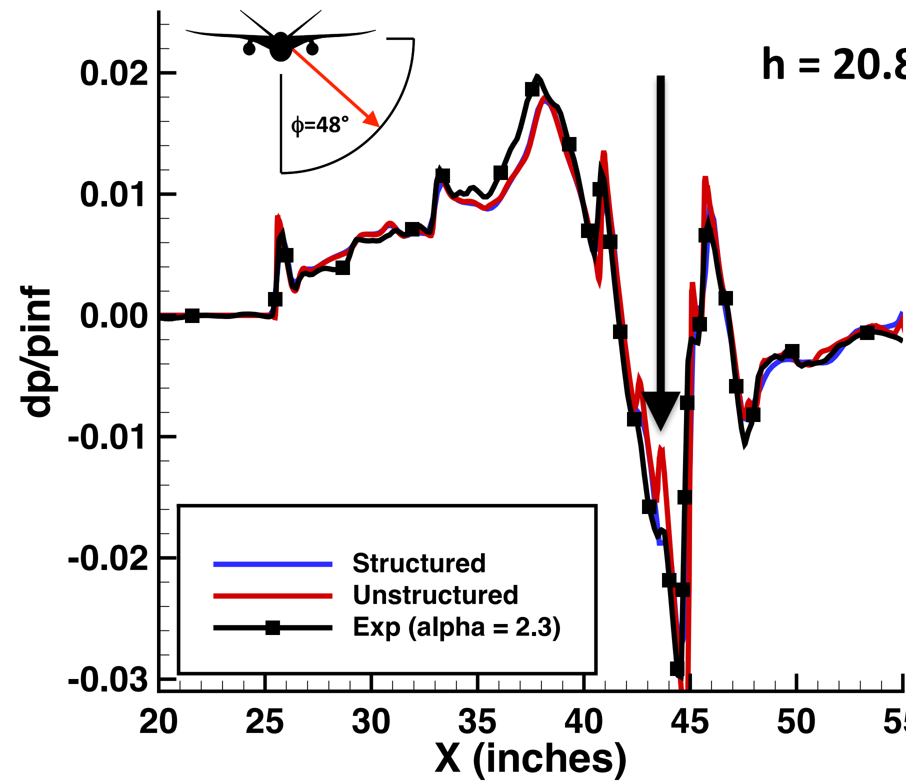


- At h = 20.8 the structured and unstructured solutions match well until X = 43.5
- At h = 70 stronger peaks are observed in the unstructured results (AUSMPW+)
- Both approaches match the experimental data well

Lockheed Martin 1021

Results and Comparison: Off-Track

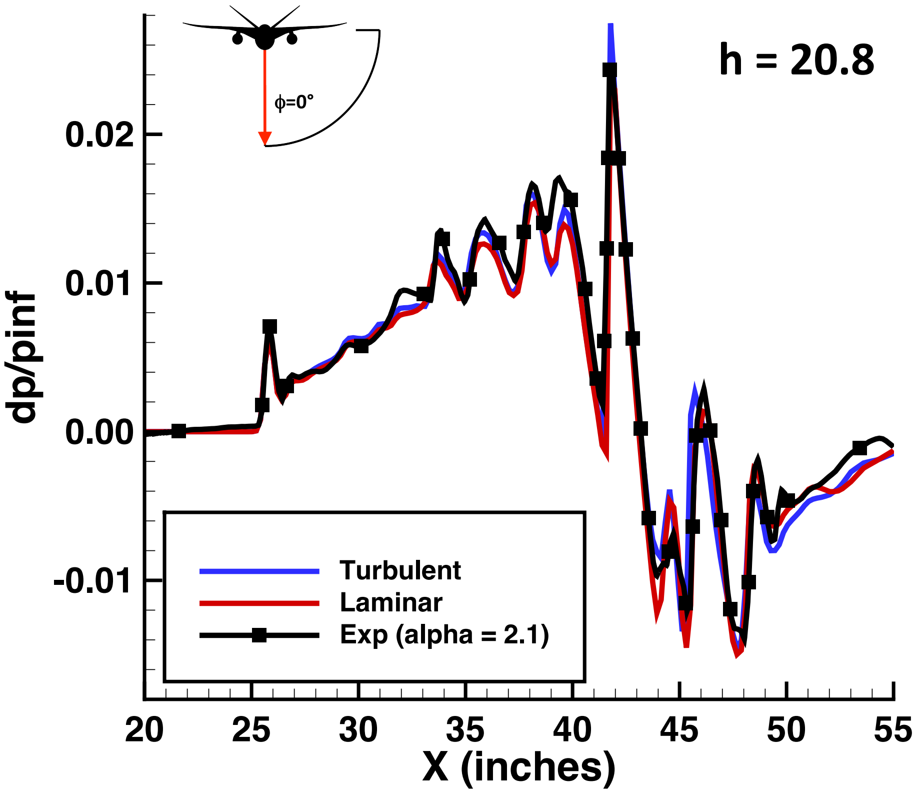


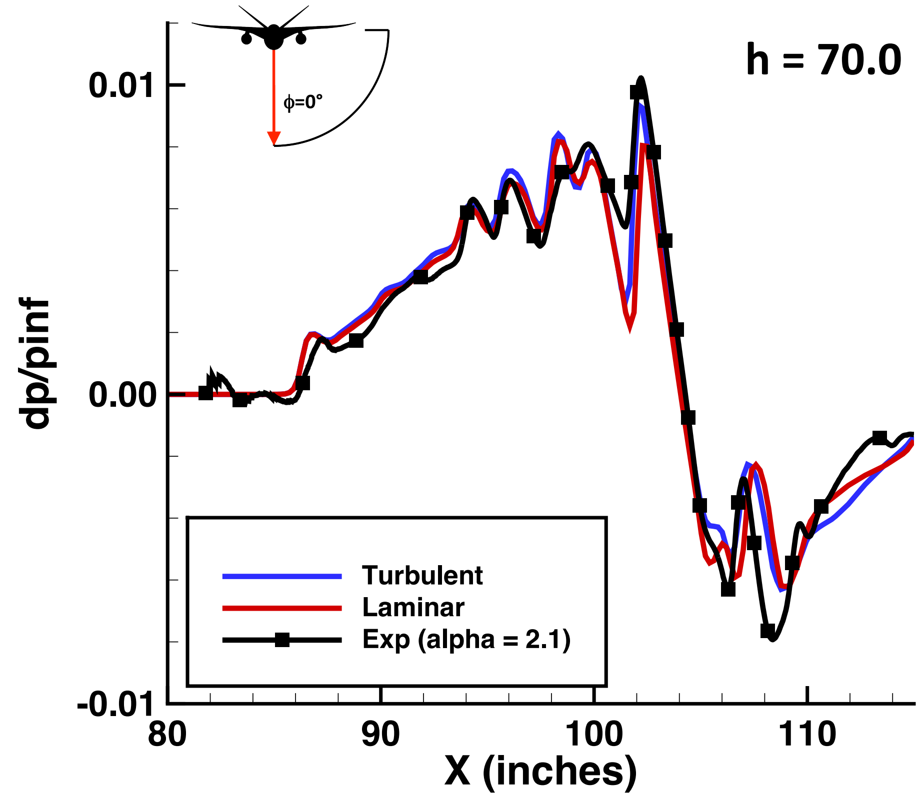


- CFD was solved at α=2.1° while experimental data was only available at near-by α
- At ϕ=20° structured and unstructured solutions match well, discrepancy with experiment from 35 < X < 40 are due to differences in α
- Sharper peaks are generated using the unstructured grid at ϕ=48°

Lockheed Martin 1021

Viscous Sensitivity Analysis: On-Track

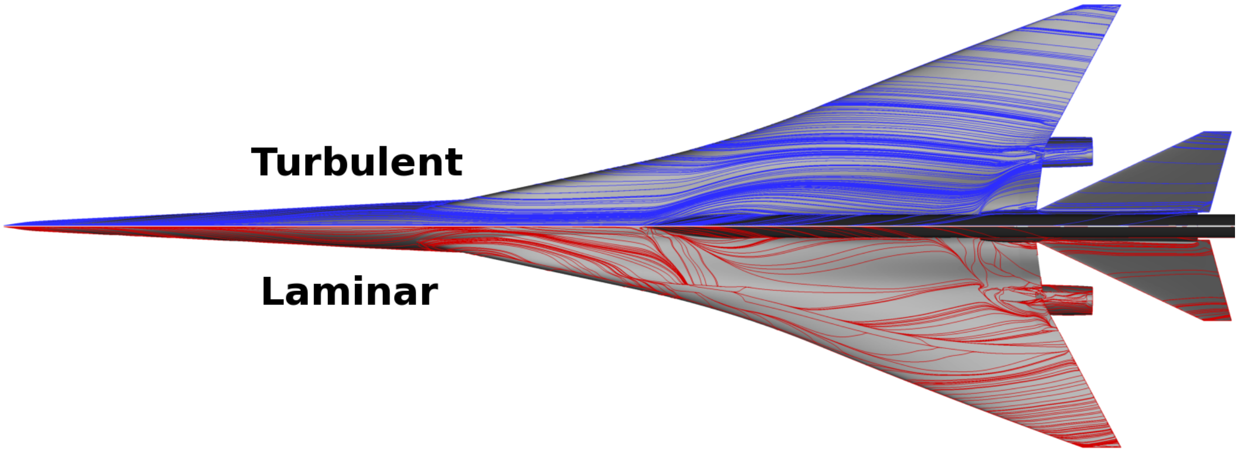




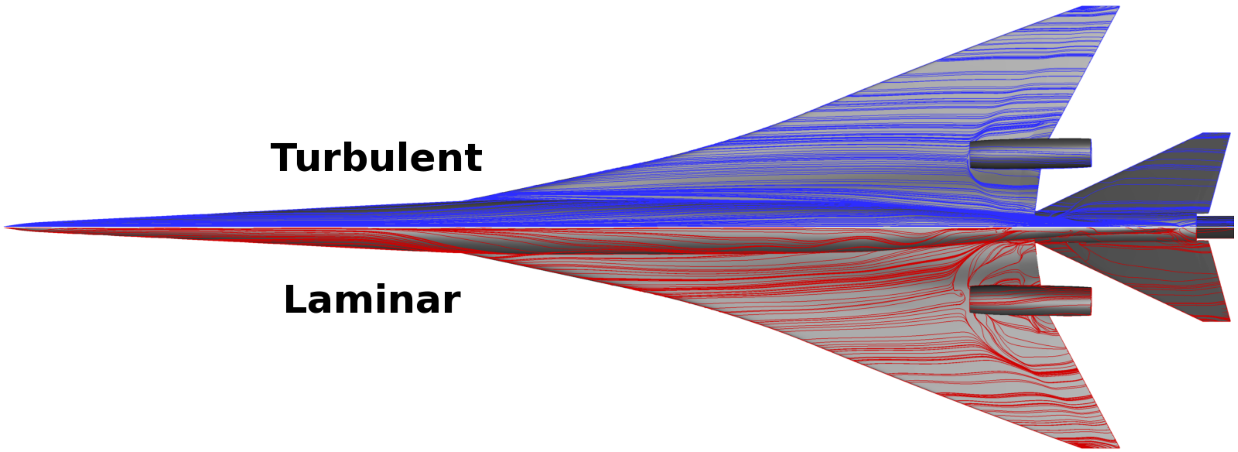
- Laminar flow computations performed to assess the sensitivity to turbulent flow assumption
- Minor differences observed in pressure prediction on the order of structured and unstructured grid differences

Lockheed Martin 1021

Viscous Sensitivity Analysis



Top Surface



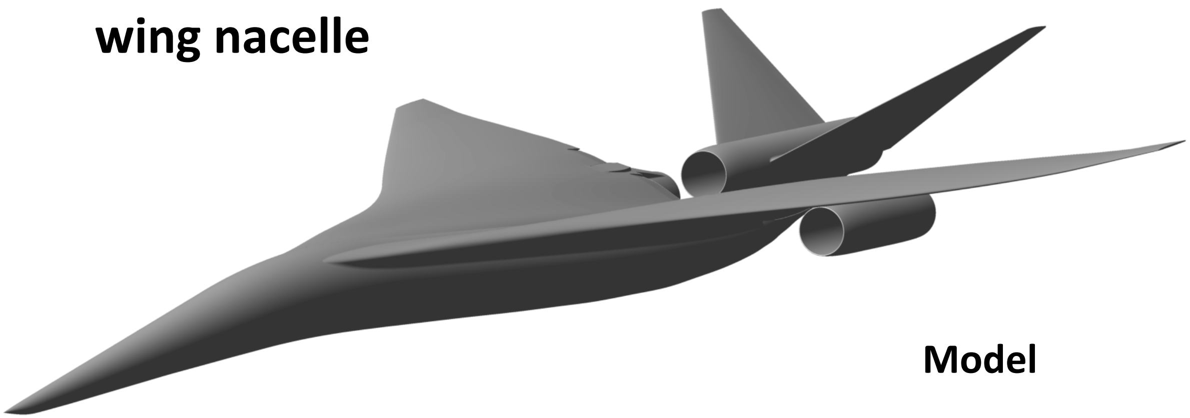
Bottom Surface

- Larger differences observed in surface oil flow
- Shock wave generated by the blade causes laminar flow separation near the leading edge of the top wing surface
- Strength of separation generated from under-wing nacelle is larger using Laminar flow assumption

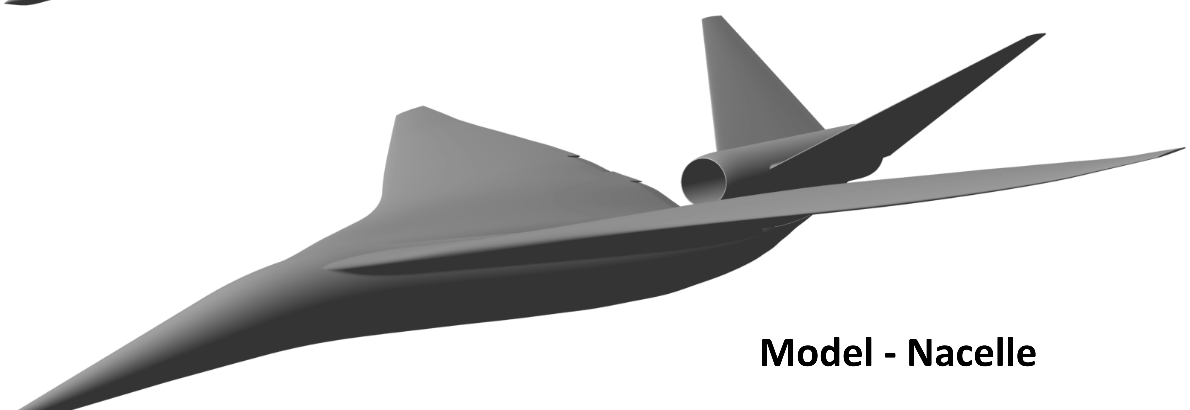
Lockheed Martin 1021

Geometric Sensitivity Analysis

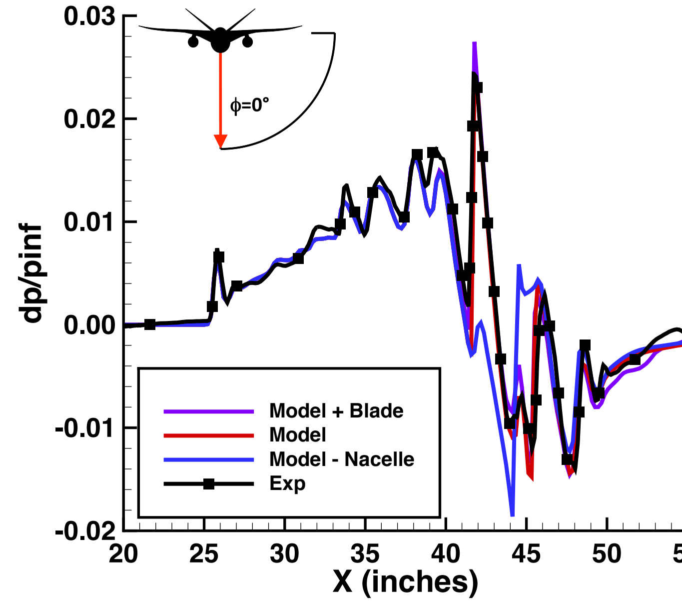
- Two additional configurations performed to assess geometric sensitivity
- Almost no difference excluding the blade
- Largest overpressure attributed to the under-wing nacelle

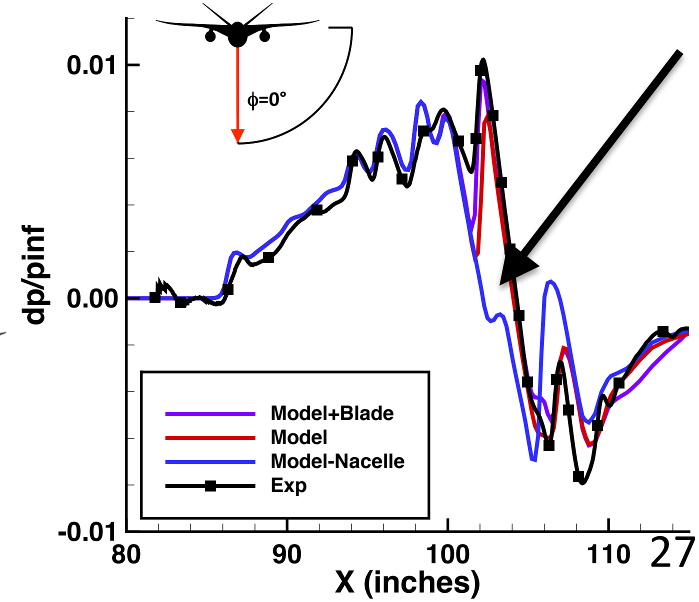


Model



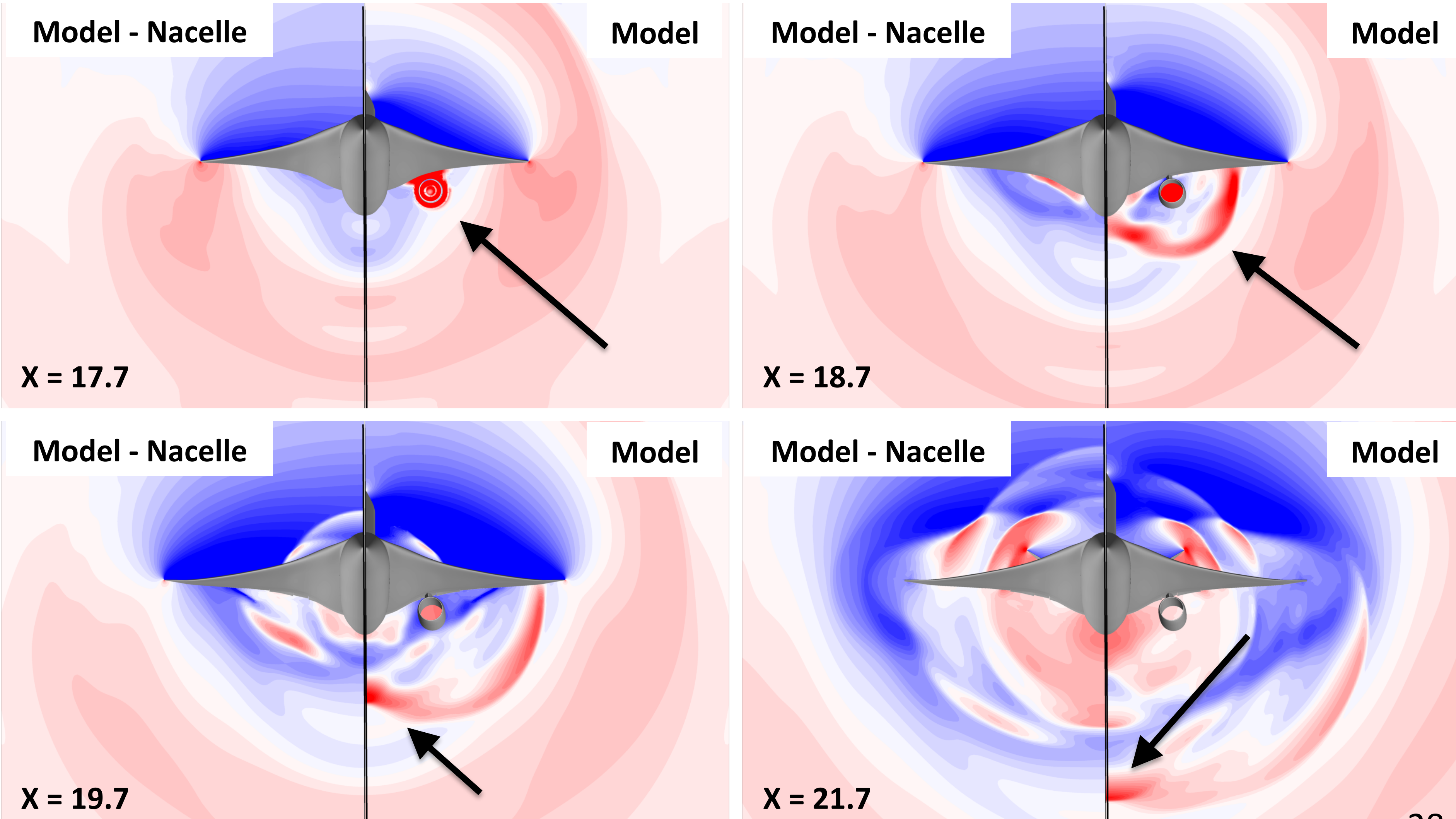
Model - Nacelle





Lockheed Martin 1021

Geometric Sensitivity Analysis



Summary

- LAVA framework has been successfully applied to the Sonic Boom Prediction workshop test cases.
- Both structured and unstructured grid methodologies have been investigated and similar accuracy demonstrated
- Computational resources are approximately 2 – 5.5 times more using the unstructured approach
- The AUSMPW+ and Modified Roe fluxes performed better than central differencing (fewer spurious oscillations)
- Good comparison achieved with experimental data

Acknowledgements

- First AIAA Sonic Boom Prediction Workshop Committee
- NASA Fundamental Aeronautics Program High-Speed Project
- NASA Advanced Supercomputing (NAS) facility at NASA ARC
- ARC collaborators: Susan Cliff, Michael Aftosmis, William Chan, James Jensen, and Donald Durston