Development of Automatic CFD Grid Generation for the Design of the Preliminary Wing Sweep Schedule for the Gulfstream Quiet Supersonic Jet

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Development of Automatic CFD Grid Generation for the Design of the Preliminary Wing Sweep Schedule for the Gulfstream Quiet Supersonic Jet

By

Ali Kazeneh

Thesis

Presented to the Faculty of the Graduate School of Embry-Riddle Aeronautical University In Partial Fulfillment of the Requirements for the Degree of Master of Science in Aerospace Engineering

Embry-Riddle Aeronautical University
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This thesis was prepared under the direction of the candidate’s thesis committee chairman, Dr. Eric R. Perrell, Department of Aerospace Engineering, and has been approved by the members of his thesis committee. It was submitted to the Aerospace Engineering Department and was accepted in partial fulfillment of the requirements for the degree of Master of Science in Aerospace Engineering.

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Abstract

The wing sweep schedule for the Gulfstream Quiet Supersonic Jet (wing and body) was investigated using CFD analysis methods. The use of automating the gridding procedure allowed for numerous cases to be investigated. The examination of the drag polar for different Mach cases indicated the optimum corresponding wing sweep position. The study results revealed that for the Mach 0.85 case, the wings should remain at the forward most pivoted position ($\Lambda=25^\circ$). The Mach 1.1 case indicted that the wings should be in the fully swept back position ($\Lambda=60^\circ$). The majority of the wing sweep transition occurs from Mach 0.85 to 0.90 where the optimum position was found to be for a sweep of $40^\circ$. 
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Nomenclature

**English Symbols**

- $c$: speed of sound
- CFD: computational fluid dynamics
- $C_D$: Drag coefficients
- $C_f$: Force coefficients
- $C_L$: Lift coefficients
- $C_M$: Moment coefficients
- $C_P$: Pressure coefficients
- $\cdot m$: mass flow rate
- $M$: Mach number
- $n$: local unit
- $p$: pressure
- $s$: surface
- $S$: wing area
- $U$: flow speed

**Greek Symbols**

- $\alpha$: angle of attack
- $\beta$: sideslip angle
- $\delta$: pressure ($P/P_{\text{Sea Level}}$)
- $\gamma$: specific heat ratio
- $\Lambda$: sweep angle
- $\rho$: density
- $\sigma$: stress tensor

**Subscripts**

- $\text{ref}$: reference
- $\infty$: freestream
1 Introduction

1.1 Present Problem

The Federal Aviation Administration (FAA) has recently solicited studies to reduce supersonic aircraft noise, in order to relax the overland flight restrictions imposed by 14 CFR 91 [1]. In response, beginning of 2004, Gulfstream Aerospace Corporation (GAC) a company well known for their business jets, generated and analyzed aero-designs by applying Computational Fluid Dynamics (CFD) to features thought to be important to reducing supersonic aircraft noise [2, 3 & 4]. During the same time frame, GAC studies of the market for supersonic business jets have been promising. Subsequent flight tests conducted in conjunction with NASA instilled confidence that CFD is a sufficiently accurate tool for providing design data to justify its further application to the Quiet Supersonic Jet (QSJ) development.

This project develops and demonstrates capabilities for streamlining the CFD process in the preliminary design of the QSJ aircraft. As design is an iterative process, and CFD mesh generation is highly labor intensive, an automated procedure for complete vehicle grid generation is developed. In particular, transonic flight requires the wing of the QSJ to sweep from a forward, subsonic position, to a highly aft-swept position at its supersonic design Mach number. Kress [5] explains a number of reasons why variable sweep is desirable, including takeoff and landing speed reduction, drag reduction and wing loading considerations in the different Mach number regimes, and flight stability. The work presented here explores the application of automatic CFD grid generation to the design of the preliminary wing sweep schedule for the QSJ. The objective is to perform a “test matrix” of CFD calculations in order to determine optimum wing sweep position for different Mach numbers, to minimize drag, for a specified lift.
1.2 Approach

The computational tools, and the general procedure in which they were used in the wing sweep schedule analysis are shown in Figure 1. The aerodynamic configuration is executed, using GAC Fortran utilities, as a set of vehicle components – Quiet Spike, fuselage, wing, wing collar, empennage, engines – according to the product design requirements, and cross-sectional area constraints based upon Whitcomb’s area rule. The aircraft component surface geometries are computed, and output as Plot 3D format “database files” which constitute the starting point for this study.

The component database files are imported into Gridgen, an industry-standard mesh generation tool. Body-fitted three-dimensional mesh blocks are created in Gridgen using Tcl macro scripts, or “glyphs”. The glyph files allow the user to specify certain design parameters, such as relative sizes, positions, and orientations of the components, as well as mesh block descriptors like boundary positions and mesh density. The mesh automation procedure is detailed in Chapter 3.

A multiple-mesh domain decomposition method called ‘Chimera’ is used to allow complex aerodynamic configurations to be modeled by a system of individually generated overset meshes. The chimera approach is implemented through the use of two NASA computer codes, PEGSUS and OVERFLOW. PEGSUS determines the overlap regions between mesh blocks, blanks cells within these regions, and assigns the cell-to-cell connectivity between the blocks. OVERFLOW is the companion flow solver to PEGSUS. Chapter 4 explains further the use of PEGSUS and OVERFLOW.
The forces and moments for each component as well as the overall configuration are calculated by a utility called FOMOCO. FOMOCO is a software package for computing flow coefficients such as force, moment, and mass flow rate on a collection of overset surfaces with accurate accounting of the overlapped regions. FOMOCO is further discussed in Chapter 5.
2 Background

2.1 Sonic Boom Signatures

Shockwaves develop around aircraft as they near the speed of sound. When an aircraft travels faster than the speed of sound, the shockwaves can produce a "sonic boom," capable of shattering windows below. A supersonic aircraft produces a unique pressure signature, the general form being an ‘N’ wave resulting from the bow shock, and recompression at the tail [6] (Figure 2).

The challenge is to reduce the overpressures to a level tolerable by people on the ground. In theory this can be done by reducing the turning angles of the flow over the vehicle, such that a series of weak shocks, rather than two strong shocks, is produced. Also, Howe [4] reports that persons on the ground can be shielded somewhat from the engines’ sonic signature by placing the engines above the wings.
2.2 Sonic Boom Reduction with the Extendable Nose Spike

In 2003 a Quiet Spike [7] Project was formed through the partnership between Gulfstream Aerospace and NASA's Dryden Flight Research Center to investigate the suppression of sonic booms [8]. The project was to create a spike which would be mounted on the nose of NASA Dryden's F-15B research testbed aircraft (Figure 3).

Figure 3: NASA’s F-15B in flight with Quiet Spike fully extended, Sept 27, 2006

The Quiet Spike [3] was developed as a means of controlling and reducing the sonic boom caused by an aircraft breaking the sound barrier. In the NASA flight tests, a chase plane measured pressures in the near field behind the F-15 research aircraft. Figure 4 illustrates the probe readings mapped over the F-15.
Figure 4: Pressure Signature mapped over NASA’s F-15B #386 with Quiet Spike fully extended

From this it is possible to notice that the Quiet Spike creates three small pressure waves. The CFD image in Figure 5, from a calculation done prior to the flight, illustrates three small shock waves caused by the Quiet Spike, that travel parallel to each other all the way to the ground.
Figure 5: CFD predicted pressures of NASA's F-15B with Quiet Spike fully extended

Figure 6 depicts the Quiet Spike pressure signature and shows a comparison of the probe readings to the CFD results and shows excellent agreement on the position and magnitude of the shock waves. [9]
The Quiet Spike is the initial phase in changing the N-wave sonic boom into a smooth and more rounded pressure wave. The NASA tests were intended to validate CFD design calculations for the Quiet Spike only. Note in Figure 4 that the F-15 still exhibits a multiple N-wave pressure signature. The next step is to design a complete vehicle profile, using CFD, so that the strengths of these shocks are also minimized.

2.3 Transonic Drag Reduction

In the early 1950's, supersonic flight was still largely an enigma to manufacturers. Designers attempted to overcome the high transonic drag rise by providing aircraft with more powerful engines. The F-102's designers, for example, chose an engine they believed would provide enough thrust to reach a maximum speed of about Mach 1.2. However, initial flight tests of the YF-102 prototype indicated that the aircraft could not even reach Mach 1. The engineers were puzzled by this lack of performance until a NACA researcher named Dr. Richard Whitcomb [10] developed the area rule. Designers had found that the drag on these aircraft increased substantially when the planes traveled near Mach 1. This increase in drag is due to the formation of shock waves over portions of the aircraft, typically from approximately Mach 0.8.

Figure 7: Increase in wave drag at transonic Mach numbers
Whitcomb’s area rule is an important concept as it is related to the drag on an aircraft or other body in transonic and supersonic flight. The ‘area rule’ states that the transonic wave drag of an aircraft is fundamentally the same as the wave drag of an equivalent body of revolution having the same cross-sectional area distribution as the aircraft. Whitcomb experimented with several different axisymmetric bodies and wing-body combinations in a transonic wind tunnel. What he found was that the drag created on these shapes was directly related to the change in cross-sectional area of the vehicle from the nose to the tail. The shape itself was not as critical in the creation of drag, but the rate of change in that shape had the most significant effect.

Figure 8: Whitcomb area rule test models: (a) cylindrical fuselage, (b) fuselage with wings, (c) bulged fuselage, (d) waisted fuselage with wings

Figure 8 illustrates four of Whitcomb's experimental models, representing a simple cylindrical fuselage, the same fuselage with wings attached, a bulged fuselage, and a "pinched" fuselage with wings. The experimental results conducted by Whitcomb showed that the addition of wings to the basic cylinder produced twice as much drag as the cylinder alone. It was discovered that the drag rose by the same amount if a simple bulge were added to the cylinder, the bulge being of equivalent volume as the wings.
Whitcomb’s most significant discovery was that the reduced cross-sectional area of the fuselage over the region where the wings were attached (Figure 8, model D) had approximately the same total drag as that of the cylinder alone.

The conclusion of this research was that shaping the vehicle to create a smooth cross-sectional area distribution from the nose to the tail could drastically reduce the drag on an aircraft. The area rule tells us that the volume of the body should be reduced in the presence of a wing, tail surface, or other projection so that there are no discontinuities in the cross-sectional area distribution of the aircraft. Figure 9 shows the waisted shape of the QSJ fuselage due to the fundamental concepts developed by Whitcomb.
3 Gridgen

3.1 Gridgen Description

Gridgen [11] is a meshing software that is used by engineers and scientists worldwide to reliably generate high quality grids for engineering analysis. Gridgen can import surface geometry definition files in several formats, produced from CAD or 3-D rendering programs. This capability was not exploited in the present study, however. Instead, Plot3d format surface meshes were generated using GAC in-house utilities. The data in these Plot3d files became the Gridgen “database entities” upon which the meshes were built. The basic Gridgen vocabulary including connectors, domains and blocks is further discussed in Appendix A.

3.2 Glyph Process

Glyph [11] is a means for automating repetitive processes. The commands are written to a Tcl script file for playback and files can be edited as necessary. The fuselage, collar and wing have been ‘glyphed’ which means that the whole gridding process is automated. In this section the process for gridding each component will be explained.

3.2.1 Fuselage and Retracted Spike Process

The fuselage.dba file must first be imported before executing any glyphs. Figure 10 depicts the fuselage database entity with the spike fully retracted. The fuselage process begins by running the “db_extract.glf” which simply extracts the fuselage surface edges to make connectors.

Figure 10: Fuselage database entity
The next phase of the fuselage process is to execute ‘mesh_upper.glf’ and ‘mesh_lower.glf’ scripts which are created with a Mach line spreadsheet program by D. Howe (Figure 11). The glyphs also create connectors for the missing Mach lines where they are redimensioned and redistributed.

Figure 11: Imported meshup.glf and meshlo.glf that track the Mach lines:

(a) \( M = 1.6 \)  
(b) \( M \leq 1.0 \)

The next phase is to execute the ‘fuselage_surface.glf’ which creates connectors that are constrained to the fuselage entity (between the upper and lower mach lines). This is followed by executing the ‘outer_boundary_edge.glf’ which simply constructs the edges of the block (Figure 12).

Figure 12: Connectors defining the outer boundaries of the fuselage grid
The final steps are to execute the ‘domains.glf’ and ‘blocks.glf’ which creates the domains and creates two sub-blocks (Figure 13). The sub-blocks however are not joined in Gridgen as the files take up too much memory to perform this action. Instead a utility called grided concatenates the two blocks in batch mode.

![Figure 13: Fuselage block (31,182,437 cells)](image)

### 3.2.2 Collar Process

The automated grid generation for the collar was developed by E. Perrell. The process begins by running ‘wg_databaser.inp’ which creates a PLOT3D representation of the upper and lower collar surfaces. The ‘wg_up_surf.glf’ and ‘wg_lo_surf.glf’ creates a b-spline fit to trim the upper and lower collar surfaces. The next phase of the collar process is to run all the ‘wg_db_lines.glf’ which simply creates 1-D database entities on the collar inboard section, outboard section and spanwise section. The ‘wg_connectors.glf’ creates connectors on the database entities as well as redimensioning and redistributing those connectors (Figure 14).
The closing phase is to execute ‘wg_domains.gif’ and ‘wg_blocks.gif’ which creates the domains and generates the sub-blocks (Figure 15). The reason for creating the sub-blocks is to allow the user to examine sections of the block individually, which uses up less memory. The final step is to execute the ‘wg_final.gif’ which joins the sub-blocks into a single block.
3.2.3 Wing Process

Since this study requires the wing to sweep, the glyph produced for the wing will be very valuable as multiple sweep angles are assessed (Figure 16).

The wing.dba file (Figure 17) must first be imported before running any of the glyphs. The first glyphs that need to be executed are ‘extract.gif’, ‘aero_split.gif’ and ‘mod.gif’. This set of glyphs extracts all edges, creates intersections in the wing to control the shape of the wing block, and finally creates connectors on the database entities.

Figure 16: Wing database entities for multiple sweep configurations

Figure 17: Wing database entity
The next set of glyphs ‘vorticy_line.gif’, ‘dimension.gif’ and ‘redistribute.gif’ creates the wake plane edges where they are redimensioned and redistributed.

Figure 18: Wing c-mesh creation

The next set of glyphs ‘c_mesh.gif’, ‘c_mesh_copy.gif’ and ‘outer_domain_edge.gif’ create c-mesh grids, and the outer edge domain, as illustrated in Figure 18.

Figure 19: Wing block (6,269,031 cells)

The final step is to execute ‘domains.gif’ and ‘block.gif’ which create the remaining domains, and then create three separate blocks which are joined by the final glyph ‘block_join.gif’ (Figure 19). There is an optional glyph called ‘wing_sweep.gif’, which allows the user to control the location, and orientation of the pivot axis as well as the sweep angle.
3.2.4 Link Process

The final and most challenging block that was created is known as the link block. The wing was purposely made smaller than the collar to allow the wing to sweep without breaking the collar surface boundaries. The area in green from the image below (Figure 20) is the area that links the collar to the wing. This area is neither part of the collar block, nor the wing block. Therefore a separate block needed to be created so that OVERFLOW could solve the flow at the break line.

![Figure 20: Wing and collar database entities](image)

The link process begins by running the `prep.glf` which imports the wing and collar database entities. Next is the `intersection.glf` which creates intersections through the wing and collar, where the translation distance can be controlled by the user. The `wake_line.glf` simply creates two points which are then translated by a given distance which forms the trailing edge of the wake line (Figure 21). The `connector.glf` creates the connectors on the database entities, which are then redimensioned and redistributed.
The next step is to execute ‘c-mesh.gif’ which creates a c-mesh at the inboard and outboard sections of the link block (Figure 22). The reason for having such a large distance from the break-line is to allow for more overlap between the wing and collar blocks. The final two glyphs that need to be executed are the ‘domain.gif’ and the ‘block.gif’ which create the remaining domains, and a single block.
Every variation in sweep angle requires a modified link block. As the wing pivots from a sweep of 60 degrees to 25 degrees, the leading and trailing edge of the wing gets further away from the leading and trailing edge of the collar (Figure 23). The opening of the wing/collar opening increases, this requires a separate link block for each sweep angle and again the glyph process saves an immense amount of time for any user.

There are many variables that need to be changed when running for different sweep angles. A variable `glyph` was created to assign values to all the variables used by link process scripts, rather than having them scattered in different script files.
Figure 24: Sample of the c-mesh.gif script

Figure 24 represent a sample of the 'c-mesh.gif' script. Within the script the value of the geometric growth rate was replaced by `$Geometric_growth_1`. Setting a path for the c-mesh glyph to access the variable glyph allows the user to set all variables with a unique name, where they can be accessed from the variable.gif file. A sample from this file, showing assignment of the value 1.125 to the geometric growth rate, is shown in Figure 25.

Figure 25: Sample of the variable.gif
3.3 Glyph Summary

Each process has the option of running all the glyphs automatically by executing the ‘Execute_all.gif’

- The fuselage Execute_all.gif takes approximately 161 seconds for completion.
- The collar Execute_all.gif takes approximately 637 seconds for completion
- The wing Execute_all.gif takes approximately 53 seconds for completion
- The link Execute_all.gif takes approximately 87 seconds for completion

To create these grids manually takes an experienced Gridgen user four to five days. The automated grid process could run all the processes in minutes rather than days. Without the glyph procedure this project would not have been completed in the time frame required. The total number of cells for this configuration is approximately 39 million.
4 Flow Solver Description

4.1 Computational Fluid Dynamics (CFD)

The governing equations for this study are Euler's equations for inviscid, compressible flow: the conservation of mass, momentum, and energy. CFD is the science of determining a numerical solution of these governing equations using high-powered computers. CFD provides a cost effective alternative to "cut and try" physical testing.

4.2 Chimera Approach

Many aerodynamic configurations are too complex to be represented by a single computational mesh. A multiple-mesh domain decomposition method called chimera allows complex aerodynamic configurations to be modeled by a system of individually generated meshes, each representing a component of the overall configuration.

The chimera approach requires the use of two computer codes, PEGSUS and OVERFLOW. These codes are NASA research codes, and are only available within the United States.

Figure 26: Chimera Scheme
4.2.1 PEGSUS Grid Joining Code

PEGSUS [12] is fully three-dimensional code that processes individual component grids, cutting holes and finding interpolation stencils to create an overset grid file and an interpolation file. In essence the code prepares the overset volume grids for the flow solver by computing the domain connectivity database, and blanking out grid points which are contained inside a solid body.

The general concept behind chimera is illustrated in Figure 27, which depicts two independently generated meshes representing a flapped airfoil. The flap mesh is embedded within the airfoil mesh. The flap mesh outer boundary receives flow-field information interpolated from appropriate mesh points (interpolation stencils) of the
airfoil mesh. However the airfoil mesh must receive flow-field information from the flap mesh. An artificial “hole creation boundary” is defined within the airfoil mesh, since certain points are interior to the flap. These interior points are excluded from the computational domain of the airfoil mesh. In Chimera terminology they are ‘blanked’ points or ‘hole’ points. Figure 28 shows the blanked points (dark blue region) within the fuselage mesh.

Figure 28: Imported Plot3D file of the QSJ displaying blanked points in TECPLOT
4.2.2 OVERFLOW Flow Solver

OVERFLOW [13] is a Navier-Stokes flow solver for structured grids. It can use single block grids or Chimera overset (structured) grid systems. The Right-hand side option used was central differencing with Roe upwinding [14] and the Left-hand side option used was LU-SGS scheme [15]. Low-Mach number preconditioning is available for accuracy in computing low-speed steady-state flows.

First-order implicit time advance is used. A time-accurate mode is available, or local time-step scaling can be selected for acceleration to steady state. Grid sequencing and multigrid was also implemented for convergence acceleration. A total of 120 nodes out of 364 nodes were used for each case. Approximately 150 cases were executed with an average run time of 2 hours.
5.0 Forces and Moments Computation Tools (FOMOCO)

5.1 Introduction and Overview

FOMOCO [16] is a software package for numerically integrating pressures on a collection of overset surfaces, with accurate accounting of the overlapped regions such as the wing-body interface as shown in Figure 29 to obtain the forces and moments.

Figure 29: Overset surface grids with domain connectivity IBLANKS for wing/body configuration

FOMOCO utilities can be used in stand-alone mode or in conjunction with the flow solver OVERFLOW. The FOMOCO software package consists of two modules corresponding to a two-step procedure: (1) hybrid surface grid generation (MIXSUR module) and (2) flow quantities integration (OVERINT module).
Figure 30: surface grids with integration IBLANKS

Figure 30 shows the surface grid after integration iblank array has been constructed by MIXSUR. Gaps are created between neighboring grids while the remaining unblanked points belong to quadrilaterals that do not overlap. The hybrid grid in Figure 31 is generated based on the default blanking priority (this means that points from the coarser of the two overlapping subsets are blanked).

Figure 31: Hybrid composite surface grid with non-overlapping quadrilaterals and triangles
The OVERINT module is used to compute and report the force and moment coefficients on the hybrid composite grid generated by the MIXSUR module. Flow coefficients at the time step level for the given solution file are reported. Figure 32 shows a general overview of the FOMOCO utilities and modules.
5.2 Surface Integrals of Flow Quantities

The OVERINT code computes projected areas, force, moment and mass flow rate coefficients for each integration surface and component defined. The data reported includes:

1. Projected areas (in the X, Y and Z directions) and the total area.
2. X, Y and Z force coefficients (pressure, viscous, momentum flux and total)
3. Lift, drag and side force coefficients (pressure, viscous, momentum flux and total)
4. Moment coefficients about X (roll), Y (pitch) and Z (yaw) axes centered at (XMC, YMC, ZMC).
5. Mass flow rate coefficient.

The equations used to compute forces, moments and mass flow rate for a surface will be described in this section. In general, the forces and moments acting on a solid or field surface consist of contributions from the pressure, viscous stress and momentum flux. The force $F_i$ acting on a surface $S$ due to a moving fluid is given by

$$F_i = \int_S \sigma_{ij} n_j dS$$

Where $\sigma_{ij}$ is the stress tensor and $n_i$ is the local unit normal to the surface. The force $\Theta_i$ due to momentum flux through a field surface is given by

$$\Theta_i = \int_S \rho u_i u_j n_j dS$$

Where $\rho$ is the density and $u_i$ is the velocity. The mass flow rate $\dot{m}$ through a field surface is given by

$$\dot{m} = \int_S \rho u_i n_j dS$$
The force coefficients $C_f$, moment coefficients $C_m$ and mass flow coefficient $C_p$ are obtained as

$$C_f = \frac{\hat{F}}{Q_\infty A_{ref}}$$

$$C_m = \frac{\hat{M}}{Q_\infty L_{ref} A_{ref}}$$

$$C_p = \frac{\hat{m}}{\rho_\infty \hat{V}_\infty A_{ref}}$$

Where

$$\hat{Q}_\infty = \frac{1}{2} \rho_\infty (\hat{V}_\infty)^2 = \frac{1}{2} \rho_\infty M_\infty^2 (c_\infty)^2$$

and ($\hat{\cdot}$) denotes non-dimensional variables; the subscript $\infty$ is used to denote free stream quantities; $\rho$, $V$, $M$ and $c$ are the density, flow speed, Mach number and speed of sound, respectively; $L_{ref}$ and $A_{ref}$ are the reference length and area, respectively.

The lift, drag and side forces ($C_L$, $C_D$, $C_S$) are computed from the $X$, $Y$ and $Z$ components of forces ($C_x$, $C_y$, $C_z$) as follows.

$$C_L = -C_x \sin \alpha + C_z \cos \alpha$$

$$C_D = (C_x \cos \alpha + C_z \sin \alpha) \cos \beta - C_y \sin \beta$$

$$C_S = (C_x \cos \alpha + C_z \sin \alpha) \sin \beta + C_y \cos \beta$$
6.0 Results

6.1 Run Matrix

The table below (Figure 33) illustrates the initial run matrix. It was decided to run for mach numbers of 0.80, 0.90, 0.95, 1.1, 1.2, 1.4 and 1.6. For the mach cases of 0.80 and 1.60 all the sweep angles would be investigated. To minimize the number of computational runs a linear prediction was made, the ‘x’ marks representing the predicted wing sweep angle that would be optimal.

![Figure 33: Run matrix](image)

The coefficient of Lift \([17, 18]\) \(C_L\) is defined as

\[
C_L = \frac{W}{\frac{1}{2}\rho U^2 S}
\]

Velocity \(U\) is a function of Mach number

\[
M = \frac{U}{a} \quad \rightarrow \quad U = M \cdot a
\]

\[
a^2 = \frac{\gamma p}{\rho} \quad \rightarrow \quad a = \sqrt{\frac{\gamma p}{\rho}}
\]

\[
U = M \sqrt{\frac{\gamma p}{\rho}} \quad \rightarrow \quad U^2 = M^2 \left(\frac{\gamma p}{\rho}\right)
\]

\(C_L\) may be defined in terms of the Mach number by substituting \(U^2\) into the \(C_L\) equation
\[ C_L = \frac{W}{\left(\gamma P M^2 \right) S} \]

For air at standard atmospheric conditions at 10,000ft, \( \gamma = 1.4 \) and \( P = 1455.34 \text{lbf/ft}^2 \). By substituting the value of gamma (\( \gamma \)) and pressure (\( P \)) the coefficient of lift can be further simplified where \( \delta \) is the ratio of local and sea level pressure.

\[ C_L = \frac{W}{(\delta 1481 M^2) S} \]

\( C_L \) is a function of Mach number since the weight (\( W = 30,000 \text{lb} \)), wing area (\( S = 600 \text{ft}^2 \)) and altitude (10,000ft) for this study remain constant. Figure 34 (plot of \( C_L \) vs Mach) illustrates this variation. The low speed (\( M = 0.25 \)) analysis yields a \( C_L \) of 0.785481 based on an altitude of 10,000ft.

The tabulated results for the lift and drag coefficients for all the Mach cases can be found in Appendix E. The results in this report will be presented in the order that the results were obtained. A low speed case (\( M = 0.25 \)) is presented first, in order to gain confidence in the computational method. Then Mach cases of 0.80 and 1.60 were investigated to establish the optimum wing sweep angle for the lowest and highest spectrum of the Mach cases in this project. Finally, a set of intermediate Mach numbers is studied, to complete the sweep schedule.
6.2 CFD Predicted Pressures: Low Speed, Mach = 0.25

The contours from Figure 35 show the pressure coefficients for sweep angles of 25 and 60 degrees.

![Figure 35: CFD predicted pressures (low speed), M=0.25](image)

The plot from Figure 36 illustrates how the angle of attack influences the $C_L$ for each given sweep angle. In general the maximum $C_L$ is achieved for low sweep angles. As the wing sweeps forward from 60 to 25 degrees, for an alpha of 6 degrees, the $C_L$ increases substantially (Figure 36).

![Figure 36: CFD predicted lift curve ($C_L$ vs $\alpha$), M=0.25](image)
Figure 37 displays the drag polar for the Mach 0.25 case. The optimum wing sweep angle for the low speed case would be 25° as this would produce the lowest drag compared to the other sweep angles.

Figure 37: CFD predicted drag polar (C_L vs C_D), M=0.25
6.3 CFD Predicted Pressures: Mach = 0.8

The contours from Figure 38 below show the pressure coefficients for sweep angles of 25 and 45 degrees.

Figure 38: CFD predicted pressures (low speed), M=0.80

Figure 39 shows the drag polar for the Mach 0.80 case. From the Lift equation the required $C_L$ for the Mach 0.80 case was 0.1776. By intersecting each wing sweep line it is possible to obtain the corresponding drag coefficient.
Figure 39: CFD predicted drag polar ($C_L$ vs $C_D$), $M = 0.80$

Figure 40 illustrates the drag coefficient vs sweep angle. The optimum wing sweep angle for the Mach 0.80 case would be 25° as this would produce the minimum drag compared to the other sweep angles (Figure 40).

Figure 40: CFD predicted minimum drag ($C_D$ vs $\Lambda$), $M=0.80$ and 0.85
6.4 CFD Predicted Pressures: $M = 1.6$

The contours from Figure 41 show the pressure coefficients for sweep angles of 60 degrees. The figure illustrates how the angle of attack influences the $C_p$ across the top section of the wing. As the angle of attack increases from -1 to 1, the velocity on the upper surface increases.

![CFD predicted pressures (upper wing), M=1.60](image)

The plot from Figure 42 shows the drag polar for the Mach 1.60 case. From the Lift equation the required $C_L$ for the Mach 1.60 case was 0.0444. By comparing the drag polars for the Mach 0.8 to the Mach 1.6 case it is clear that there is a shift with the $\Lambda$ lines. By intersecting the drag polar by the required $C_L$ it is possible to generate another plot illustrating drag coefficient vs sweep angle.
The drag coefficient vs sweep angle for the Mach 1.60 case is shown in Figure 43. From this plot the optimum wing sweep angle for the Mach 1.60 case would be 60°.
6.5 CFD Predicted Pressures: Mach = 0.90, 0.95

Since the optimum wing sweep angles which happen to be the fully swept forward and back have been obtained for the Mach cases of 0.80 and 1.60, the schedules between these Mach cases need to be determined. The Mach cases that were investigated next were 0.90 and 0.95. The contours from the Figure 44 below show the pressure coefficients for a sweep angle of 50 degrees. The formation of normal shock waves is clearly visible from the figure below.

Figure 44: CFD predicted pressures (upper wing and side profile view), M=0.95

Figure 45 shows the drag polar for the Mach 0.90 case. The required $C_L$ for the Mach 0.90 case was 0.14035. By comparing the drag polar for the mach 0.8 and 1.6 case to the 0.90 case it is clear that the 0.90 case illustrates the transition in the shift of the wing sweep lines. By intersecting the drag polar by the required $C_L$ it is possible to generate the drag coefficient vs sweep angle plot. Inspection of the drag polar for the 0.90 case
indicates that the optimum wing sweep position is either 35 or 40 degrees. The plot of $C_D$ vs $\Lambda$ for $M = 0.90$ will best indicate the optimum position.

Figure 45: CFD predicted drag polar ($C_L$ vs $C_D$), $M=0.90$

Figure 46 illustrates the drag coefficient vs sweep angle for the Mach 0.90 case. The optimum wing sweep angle for the Mach 0.90 case would be 40° as this generates the minimum drag (Figure 46).

Figure 46: CFD predicted minimum drag ($C_D$ vs $\Lambda$), $M=0.90$
The plot above (Figure 47) shows the drag polar for the Mach 0.95 case where the required \( C_L \) is 0.12597. The results of the drag polar are similar to the 0.90 case and the optimum wing sweep position is 45 degrees. The plot of \( C_D \) vs \( \Lambda \) (Figure 48) confirms this.

Figure 47: CFD predicted drag polar (\( C_L \) vs \( C_D \)), \( M = 0.95 \)

Figure 48: CFD predicted minimum drag (\( C_D \) vs \( \Lambda \)), \( M=0.95 \)
6.6 CFD Predicted Pressures: Mach = 1.10

The contours from Figure 49 show the pressure coefficients for sweep angles of 45 and 60 degrees. The formation of oblique shock waves is clearly visible from the figure below. From the side profile view of Figure 49 the shock wave produced at the leading edge of the wing for a sweep of 45 degrees is stronger than that compared to a sweep of 60 degrees. It can therefore be said that pressure signature is influenced by the wing sweep.

![Strong leading edge shock (\(\Lambda=45^\circ\))](image1)

![Weaker leading edge shock (\(\Lambda=60^\circ\)) compared to \(\Lambda=45^\circ\)](image2)

Figure 49: CFD predicted pressures, M=1.10

The plot from Figure 50 shows the drag polar for the Mach 1.10 case. The required \(C_L\) for the Mach 1.10 case was 0.0939. There is a clear similarity between the drag polar of the Mach 1.1 and 1.6 cases. The ordering of the sweep lines is exactly the same. The results of the drag polar are similar to the 1.10 case and the optimum wing sweep position is 60 degrees. The plot of \(C_D\) vs \(\Lambda\) (Figure 50) confirms this.
The optimum wing sweep angle for the Mach 1.10 case is 60°, the plot of $C_D$ vs $\Lambda$ (Figure 51) confirms this since this sweep generates the minimum drag. Since the optimum sweep position for the Mach 1.1 case was 60 degrees the Mach cases of 1.2 and 1.4 were neglected.
6.7 Result Summary

The relevant Mach cases with the wings swept at the optimum angle are depicted in Figure 52. It is possible to distinguish the formation of the shock waves as the Mach number increases from 0.8 to 1.6.

![Figure 52: CFD predicted pressure summary](image)

(a) $M=0.80$, $\Lambda$ (optimum) = 25°  
(b) $M = 0.95$, $\Lambda$ (optimum) = 45°  
(c) $M=1.10$, $\Lambda$ (optimum) = 60°  
(d) $M=1.60$, $\Lambda$ (optimum) = 60°

The plot from Figure 53 shows the optimal wing sweep schedule ($\Lambda$ vs $M$). As the largest jump for $\Lambda$ occurs between $M=0.80$ and $M=0.90$, it was decided to run the $M=0.85$ case. Upon interpolating the results for $M=0.80$ and 0.90, it was anticipated that the optimum sweep angle would be between 30 and 35 degrees. However, the results indicated that the optimum position for the $M=0.85$ case was also 25 degrees (Figure 40). The sweep schedule (Figure 53) illustrates that the entire subsonic-supersonic transition
occurs between Mach 0.85 to 1.10. The results would also prove valid for the supersonic-
subsonic transition.

![Figure 53: CFD predicted sweep schedule (\(\Lambda\) vs M)](image)

### 6.8 Conclusion and Future Improvements

The aim of this project was to design the preliminary wing sweep schedule for the
QSJ. Most of the wing sweep occurs in the transonic region. The optimum wing sweep
angle is 25 degrees up until Mach 0.85 and fully swept back from Mach 1.10 and above.
Subsequently the total transition occurs between Mach 0.85 to 1.10.

The configuration that was evaluated in this task was the wing and body. The next
step would be to run a full configuration that would include specifically the empennage
and engine nacelles. The addition of the horizontal tail would provide additional accuracy
as the resultant lift and drag would be influenced. There would be major benefits to
adding the engine nacelles as the inlet shock would impinge the wing which would again
modify the results. For the low speed analysis the incorporation of flaps would also
provide realistic results. The low speed results indicated that a high enough \(C_L\) could not
be achieved from alphas of up to 6 degrees, even though the wing is in the fully swept forward position. Therefore flaps would be essential in low speed performance analysis.

The results also indicated that wing sweep can influence the center of pressure location. Thus, there may be some advantage to studying other pivot point locations, in conjunction with computations including the empennage. Also, by modifying the pivot point location the spacing of the wing and collar leading edges can be reduced as the wing is swept forward. This spacing is evident in Figure 35. There may be aerodynamic benefits in placing the pivot point closer to the leading edge that would reduce the break-line spacing at the leading edge.

It could also be beneficial in running more Mach cases especially between Mach 0.85 and 0.90 cases to better define the sweep schedule. Here the wing sweeps were incremented by five degrees, a somewhat course discretization given the small Mach number range for the complete transition. Finally, the wing block (specifically the wing tip location of the block) may need to be improved to reduce the number of orphan points, which can alter the results, especially for the high sweep angles.
Appendix A: Gridgen Terminology

Databases refer to the collection of geometry information that can be used to define the shape to which the grid is to be built and typically come from computer aided design (CAD) software. An entity is any individual element that makes up part of a grid or geometry. Examples of Gridgen entities are connectors, domains and blocks.

Connectors are Gridgen’s lowest level (one dimensional) grid element. Connectors are curve entities that define all grid edges where nodes mark the connector end points. Connectors include three attributes; Shape – defined by one or more segments (control points can define the segment shape), Dimension – the number of grid points on a connector and Distribution – describes how the grid point are placed (distributed).

Domains are Gridgen’s mid-level (two dimensional) grid element. They are surface grids. Two types of domains are supported in Gridgen: structured and unstructured. A structured domain consists exclusively of quadrilateral cells and is defined by the four edges on its perimeter. An unstructured domain consists of triangular cells and is defined by one perimeter edge and possibly several interior edges enclosing holes in the domain.

Blocks are Gridgen’s highest level (three dimensional) grid element. They are volume grids in 3D and surface grids in 2D. A structured block consists entirely of hexahedral cells that are bounded by exactly six faces where as an unstructured block consists of tetrahedral, pyramidal and/or prismatic cells.

A grid is composed of volume and/or surface grid blocks. There are three main grids;

- Structured Grids – contain only quadrilateral and hexahedral elements
- Unstructured Grids – contain only triangular and tetrahedral elements
- Hybrid Grids – contain both structured and unstructured grid elements

Glyph, Gridgen’s scripting language is based on the Tcl programming language and provides text based, procedural interface to Gridgen’s feature.
Appendix B: PEGSUS input file

$GLOBAL
QUALITY = 0.8,0.1,-1.,
EPS = 0.005,
FRINGE = 2,

END

$SMESH NAME = 'FUSELAGE',
    LINK = 'WING','COLLAR','LINK',
    KINCLUDE = 1,-1,
    LINCLUDE = 2,-1,

END

$SMESH NAME = 'WING',
    LINK = 'COLLAR','FUSELAGE','LINK',
    KINCLUDE = 1,-1,
    LINCLUDE = 2,-1,

END

$SMESH NAME = 'COLLAR',
    LINK = 'WING','FUSELAGE','LINK',
    KINCLUDE = 2,-1,
    LINCLUDE = 2,-1,

END

$SMESH NAME = 'LINK',
    LINK = 'COLLAR','WING','FUSELAGE',
    KINCLUDE = 2,-1,
    LINCLUDE = 2,-1,

END

C
C...The wing cuts into the fuselage and vertical also - lower wake
C

$BOUNDARY NAME = 'LOWER WING HOLE',
ISPARTOF = 'WING',
MHOLEIN = 'FUSELAGE',

END

$SURFACE ISPARTOF = 'LOWER WING HOLE',
    JRANGE= 10,10, KRANGE= 5,-10, LRANGE= 1,-31, NVOUT= '-J',

END

$SURFACE ISPARTOF = 'LOWER WING HOLE',
    JRANGE= 61,61, KRANGE= 5,-10, LRANGE= 1,-31, NVOUT= '+J',

END

$SURFACE ISPARTOF = 'LOWER WING HOLE',
    JRANGE= 10,61, KRANGE= 5,5, LRANGE= 1,-31, NVOUT= '-K',

END

$SURFACE ISPARTOF = 'LOWER WING HOLE',
    JRANGE= 10,61, KRANGE= -10,-10, LRANGE= 1,1, NVOUT= '+K',

END

$SURFACE ISPARTOF = 'LOWER WING HOLE',
    JRANGE= 10,61, KRANGE= 5,-10, LRANGE= -31,-31, NVOUT= '+L',

END

C
C...The wing cuts into the fuselage and vertical also - surface
C

$BOUNDARY NAME = 'LEADING WING HOLE',
ISPARTOF = 'WING',
MHOLEIN = 'FUSELAGE',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',
    JRANGE= 61,61, KRANGE= 5,-10, LRANGE= 1,-41, NVOUT= '-J',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',
    JRANGE= -61,-61, KRANGE= 5,-10, LRANGE= 1,-41, NVOUT= '+J',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',
    JRANGE= 61,-61, KRANGE= 5,5, LRANGE= 1,-41, NVOUT= '-K',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',
    JRANGE= 61,-61, KRANGE= -10,-10, LRANGE= 1,1, NVOUT= '+K',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',
    JRANGE= 61,-61, KRANGE= 5,-10, LRANGE= -31,-31, NVOUT= '+L',

END

$SURFACE ISPARTOF = 'LEADING WING HOLE',

48
The wing cuts into the fuselage and vertical also - lower wake

$BOUNDARY NAME = 'UPPER WING HOLE',
ISPARTOF = 'WING',
MHOLEIN = 'FUSELAGE',
$END
$SURFACE ISPARTOF = 'UPPER WING HOLE',
JRANGE = 10,10, KRANGE = 1, 1, LRANGE = 1, -28, NVOUT = '+L',
$END
$SURFACE ISPARTOF = 'UPPER WING HOLE',
JRANGE = -10, -10, KRANGE = 1, 1, LRANGE = -28, -28, NVOUT = '+L',
$END
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ISPARTOF = 'COLLAR',
MHOLEIN = 'WING', 'FUSELAGE',
$END
$SURFACE ISPARTOF = 'COLLAR HOLE',
JRANGE = 10, 10, KRANGE = 1, 1, LRANGE = 1, -28, NVOUT = '+J',
$END
$SURFACE ISPARTOF = 'COLLAR HOLE',
JRANGE = 10, 10, KRANGE = -1, -1, LRANGE = 1, -28, NVOUT = '+J',
$END
$BOUNDARY NAME = 'LINK HOLE',
ISPARTOF = 'LINK',
MHOLEIN = 'COLLAR', 'WING',
$END
$SURFACE ISPARTOF = 'LINK HOLE',
JRANGE = 10, 10, KRANGE = 8, 8, LRANGE = 1, -27, NVOUT = '+J',
$END
$SURFACE ISPARTOF = 'LINK HOLE',
JRANGE = -10, -10, KRANGE = 8, 8, LRANGE = 1, -27, NVOUT = '+J',
$END
$SURFACE ISPARTOF = 'LINK HOLE',
JRANGE = 10, 10, KRANGE = 8, 8, LRANGE = 1, -27, NVOUT = '+K',
$END
$SURFACE ISPARTOF = 'LINK HOLE',
JRANGE = 10, 10, KRANGE = -8, -8, LRANGE = 1, -27, NVOUT = '+K',
$END
$SURFACE ISPARTOF = 'LINK HOLE',

JRANGE= 10,-10, KRANGE= 8,-8, LRANGE = -27,-27, NVOUT='+L',
SEND

$BOUNDARY NAME = 'INBDWING1 HOLE',
  MHOLEIN = 'WING',
SEND
$BOX ISPARTOF = 'INBDWING1 HOLE',
  XRANGE= 330.0, 800.0,
  YRANGE=-150.0, 55.0,
  ZRANGE= -80.0, 140.0,
SEND

C $SURFACE ISPARTOF = 'INBDWING HOLE',
C  JRANGE= 1,-1, KRANGE= 2,2, LRANGE= 1,-1, NVOUT='+K',
C SEND
C Block outer surfaces

$BOUNDARY NAME = 'WING OUTER',
  ISPARTOF = 'WING',
SEND
$SURFACE ISPARTOF = 'WING OUTER',
  JRANGE = 1,-1, KRANGE = 1,-1, LRANGE = -1,-2,
SEND
$SURFACE ISPARTOF = 'WING OUTER',
  JRANGE = 1,2, KRANGE = 1,-1, LRANGE = 1,-1,
SEND
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SEND
$SURFACE ISPARTOF = 'WING OUTER',
  JRANGE = 1,-1, KRANGE = -1,-2, LRANGE = 1,-1,
SEND

$BOUNDARY NAME = 'COLLAR OUTER',
  ISPARTOF = 'COLLAR',
SEND
$SURFACE ISPARTOF = 'COLLAR OUTER',
  JRANGE = 1,-1, KRANGE = 1,-1, LRANGE = -1,-2,
SEND
$SURFACE ISPARTOF = 'COLLAR OUTER',
  JRANGE = 1,2, KRANGE = 1,-1, LRANGE = 1,-1,
SEND
$SURFACE ISPARTOF = 'COLLAR OUTER',
  JRANGE = -1,-2, KRANGE = 1,-1, LRANGE = 1,-1,
SEND
$SURFACE ISPARTOF = 'COLLAR OUTER',
  JRANGE = 1,-1, KRANGE = -1,-2, LRANGE = 1,-1,
SEND

$BOUNDARY NAME = 'LINK OUTER',
  ISPARTOF = 'LINK',
SEND
$SURFACE ISPARTOF = 'LINK OUTER',
  JRANGE = 1,-1, KRANGE = 1,-1, LRANGE = -1,-2,
SEND
$SURFACE ISPARTOF = 'LINK OUTER',
  JRANGE = 1,2, KRANGE = 1,-1, LRANGE = 1,-1,
SEND
$SURFACE ISPARTOF = 'LINK OUTER',
  JRANGE = -1,-2, KRANGE = 1,-1, LRANGE = 1,-1,
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SEND
$SURFACE ISPARTOF = 'LINK OUTER',
  JRANGE = 1,-1, KRANGE = -1,-2, LRANGE = 1,-1,
Appendix C: OVERFLOW input file

WBVAS Lofted WT Geometry, Segmented fwd, Aft Closed, 6 Balks - cf27x03b
C Dist. 007t Based
C
$GLOBAL
CHIMRA = T, NSTEPS = 1000, RESTRT = F, NSAVE = 500,
MULTIG = T, FMG = T NGLVL = 3, FMGCYC = 500, 500,
INCORE = T, NFOMO = 10, NQT = 0,
$END
C
$FLOINP
ALPHA = 1.00, BETA = 0.00, FSMACH = 0.80,
GAMINF = 140, REY = 200133, PR = 0.72, TINF = 390.0,
$END
$VARGAM
$END
C
Inputs repeated for each block

$GRDNAM
NAME = 'FUSELAGE',
$END
C
$NITERS
ITER = 1,
$END
C
Method arc3d diagonal scheme, tlns3d smoothing

$METPRM
IRHS = 0, ILHS = 2, IDISS = 3,
$END
C
Time stepping local timestep, first order accurate

$TIMACU
ITIME = 1, DT = 0.30, TFOSO = 1.00, CFLMIN = 3.0,
$END
C
Smoothing

$SMOACU
ISPECJ = 2, DIS2J = 2.0, DIS4J = 0.04,
ISPECK = 2, DIS2K = 2.0, DIS4K = 0.04,
ISPEC1 = 2, DIS2L = 2.0, DIS4L = 0.04, SMOO = 1.00,
$END
C
Viscous inputs

$VISINP
VISCI = F, VISCK = F, VISCL = F, CFLT = 5,
NTURB = 0,
$END
C
Boundary Condition inputs

$BCINP
NBC = 8,
IBTYP = 1, 15, 17, 17, 17, 47, 32, 30
IBDIR = 3, 3, 3, 2, -2, 1, -1, -3,
JBCS = 281, 1, -200, 1, 1, 1, -1, 1,
JBC E = -200, 281, -1, -1, 1, 1, -1, -1,
KBCE = 1, 1, 1, 1, -1, 1, 1, 1,
KBCE = -1, -1, -1, -1, -1, 1, -1, -1,
LBCE = 1, 1, 1, 1, 1, 1, 1, -1,
LBCE = 1, 1, 1, -1, -1, -1, -1, -1,
$END
C
surf axis axis rsym rsym

$SCINP
$END
C
Inputs repeated for each block

$GRDNAM
NAME = 'WING',
$END
C
$NITERS
ITER = 1,
$END
C
C Method arc3d diagonal scheme, tlns3d smoothing
$SMETPRM
IRHS = 0, ILHS = 2, IDISS = 3,
$END
C
C Time stepping local timestep, first order accurate
$STIMACU
ITIME = 1, DT = 0.3, TFOSO = 1 00, CFLMIN = 3 0,
$END
C
C Smoothmg
$SSMOACU
ISPECJ = 2, DIS2J = 2 0, DIS4J = 0 04,
ISPECK = 2, DIS2K = 2 0, DIS4K = 0 04,
ISPECL = 2, DIS2L = 2 0, DIS4L = 0 04, SMOO = 1 00,
$END
C
C Viscous inputs
$SVISINP
VISCJ = F, VISCK = F, VISCL = F, CFLT = 5,
NTURB = 0,
$END
C
C Boundary Condition inputs
$SBCINP
NBC = 3,
IBTYP = 1, 51, 51,
IBDIR = 3, 3, 3,
JBCS = 61, 1, 1,
JBCE = -61, 61, 301,
KBCS = 1, 1, 171,
KBCE = 171, 171, -1,
LBCS = 1, 1, 1,
LBCE = 1, 1, 1,
$END
C
BCs wing wake wake
C
$S$CEINP
$END
C
Inputs repeated for each block BLOCK 3, w/b collar
C
$GRDNAM
NAME = 'COLLAR',
$END
$N$ITERS
ITER = 1,
$END
C
Method arc3d diagonal scheme, tlns3d smoothing
$SMETPRM
IRHS = 0, ILHS = 2, IDISS = 3,
$END
C
C Time stepping local timestep, first order accurate
$STIMACU
ITIME = 1, DT = 0.3, TFOSO = 1 00, CFLMIN = 3 0,
$END
C
C Smoothmg
$SSMOACU
ISPECJ = 2, DIS2J = 2 0, DIS4J = 0 04,
ISPECK = 2, DIS2K = 2 0, DIS4K = 0 04,
ISPECL = 2, DIS2L = 2 0, DIS4L = 0 04, SMOO = 1 00,
$END
C
C Viscous inputs
$SVISINP
VISCJ = F, VISCK = F, VISCL = F, CFLT = 5,
NTURB = 0,
$END
C Boundary Condition inputs

SBCINP
NBC = 3,
IBTYP = 1, 1, 51,
IBDIR = 3, 2, 3,
JBCS = 31, 1, 1,
JBCF = -31, -1, 31,
KBCS = 1, 1, 1,
KBCEF = -1, 1, -1,
LBCS = 1, 1, 1,
LBCEF = 1, -1, 1,
$END

C BCs wmg fuse wake

SSCEINP
$END

C

C Inputs repeated for each block BLOCK 4, link

C

SGRDNAM
NAME = 'LINK',
$END

C

$NITERS
ITER = 1,
$END

C Method arc3d diagonal scheme, tlns3d smoothing

SMETPRM
IRHS = 0, ILHS = 2, IDISS = 3,
$END

C Time stepping local timestep, first order accurate

$TIMACU
ITIME = 1, DT = 0.3, TFOSO = 1 00, CFLMIN = 3 0,
$END

C Smoothing

$SMOACU
ISPECJ = 2, DIS2J = 2 0, DIS4J = 0 04,
ISPECK = 2, DIS2K = 2 0, DIS4K = 0 04,
ISPECL = 2, DIS2L = 2 0, DIS4L = 0 04, SMOO = 1 00,
$END

C Viscous inputs

$VISINP
VISCJ = F, VISCK = F, VISCL = F, CFLT = 5,
NTURB = 0,
$END

C Boundary Condition inputs

SBCINP
NBC = 3,
IBTYP = 1, 1, 51,
IBDIR = 3, 2, 3,
JBCS = 20, 1, 1,
JBCF = -20, -1, 20,
KBCS = 1, 1, 1,
KBCEF = -1, 1, -1,
LBCS = 1, 1, 1,
LBCEF = 1, -1, 1,
$END

C BCs wmg fuse wake

SSCEINP
$END

53
Appendix D: FOMOCO input File

0.80, 1.0, 0.0, 1.67E5, 1.4, 390.0
1
210.326, 43200.0, 691.114, 0.0, 0.0
3

2, 1
1, 3, 281, -200, 1, -1, 1, 1
3, 2, 1, -1, 1, 1, 1, 41
0

3, 1
4, 3, 260, 15, 27, 1, 1
3, 3, 253, 1, -1, 1, 1
2, 3, 61, 301, 1, 131, 1, 1
0

3, 1
4, 3, 260, -20, 15, 27, 1, 1
3, 3, 253, -31, 1, -1, 1, 1
2, 3, 301, -61, 1, 131, 1, 1
0

3 NCOMP (next line is component name)
WING
2, 1 NUMBER OF SURFACES, REFERENCE SET #
2, 3 SURFACE NUMBERS
BODY
1, 1 NUMBER OF SURFACES, REFERENCE SET #
1, SURFACE NUMBERS
WB
3, 1 NUMBER OF SURFACES, REFERENCE SET #
1,2,3 SURFACE NUMBERS
Appendix E: Tabulated Results, M = 0.25, 0.80, 0.85

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<tr>
<th>M=0.25</th>
<th>C&lt;sub&gt;f&lt;/sub&gt; @ M=0.25</th>
<th>C&lt;sub&gt;p&lt;/sub&gt; @ M=0.25</th>
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</thead>
<tbody>
<tr>
<td>Λ =25</td>
<td>0.103141 0.230661 0.359021 0.486431</td>
<td>0.001667 0.001739 0.001710 0.001585</td>
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<tr>
<td>Λ =30</td>
<td>0.108373 0.228755 0.350353 0.469338</td>
<td>0.001739 0.001808 0.001780 0.001655</td>
</tr>
<tr>
<td>Λ =35</td>
<td>0.110298 0.222673 0.336074 0.446479</td>
<td>0.001808 0.001878 0.001850 0.001725</td>
</tr>
<tr>
<td>Λ =40</td>
<td>0.108908 0.214322 0.320762 0.424298</td>
<td>0.001878 0.001948 0.001920 0.001795</td>
</tr>
<tr>
<td>Λ =45</td>
<td>0.102750 0.197890 0.293773 0.386357</td>
<td>0.001948 0.002018 0.001990 0.001865</td>
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<tr>
<td>Λ =50</td>
<td>0.095435 0.181027 0.267483 0.353707</td>
<td>0.002018 0.002088 0.002060 0.001935</td>
</tr>
<tr>
<td>Λ =55</td>
<td>0.091202 0.171071 0.253591 0.343817</td>
<td>0.002088 0.002158 0.002130 0.002005</td>
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<tr>
<td>Λ =60</td>
<td>0.081941 0.154731 0.233080 0.316830</td>
<td>0.002158 0.002228 0.002200 0.002070</td>
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<table>
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<th>C&lt;sub&gt;p&lt;/sub&gt; @ M=0.8</th>
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<td>0.067527 0.165108 0.263725</td>
<td>0.000022 0.000120 0.000188</td>
</tr>
<tr>
<td>Λ =30</td>
<td>0.076182 0.166209 0.257141</td>
<td>0.000120 0.000221 0.000309</td>
</tr>
<tr>
<td>Λ =35</td>
<td>0.082244 0.162979 0.244241</td>
<td>0.000221 0.000330 0.000429</td>
</tr>
<tr>
<td>Λ =40</td>
<td>0.083189 0.159595 0.229247</td>
<td>0.000330 0.000439 0.000538</td>
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<tr>
<td>Λ =45</td>
<td>0.078562 0.142900 0.207667</td>
<td>0.000439 0.000548 0.000647</td>
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<tr>
<td>Λ =50</td>
<td>0.071955 0.128905 0.186283</td>
<td>0.000548 0.000657 0.000756</td>
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<tr>
<td>Λ =55</td>
<td>0.066450 0.117902 0.169444</td>
<td>0.000657 0.000766 0.000865</td>
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<tr>
<td>Λ =60</td>
<td>0.058589 0.102822 0.147119</td>
<td>0.000766 0.000875 0.000974</td>
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<table>
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<tr>
<th>M=0.8</th>
<th>C&lt;sub&gt;f&lt;/sub&gt; @ M=0.8</th>
<th>C&lt;sub&gt;p&lt;/sub&gt; @ M=0.8</th>
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<td>0.077029 0.186724 0.299571</td>
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<td>Λ =30</td>
<td>0.085147 0.182701 0.282449</td>
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<td>Λ =35</td>
<td>0.089783 0.176000 0.263611</td>
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<td>0.089344 0.166104 0.244011</td>
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<td>Λ =45</td>
<td>N/A       N/A       N/A</td>
<td>0.000439 0.000548 0.000647</td>
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<tr>
<td>Λ =50</td>
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<td>0.000548 0.000657 0.000756</td>
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<tr>
<td>Λ =55</td>
<td>N/A       N/A       N/A</td>
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<tr>
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<td>N/A       N/A       N/A</td>
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<th>M=0.8</th>
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<th>C&lt;sub&gt;p&lt;/sub&gt; @ M=0.8</th>
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<td>0.000022 0.000120 0.000188</td>
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(Appendix E: Cont), $M = 0.90, 0.95, 1.1, 1.6$

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<td>0.090917</td>
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<table>
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<td>0.003355</td>
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<table>
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<tr>
<td>$\Lambda = 60$</td>
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<td>0.004940</td>
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</tbody>
</table>
References


[9] In-Flight Quiet Spike Near-field Pressure Measurements with Computational Fluid Dynamics Comparisons, NASA_FAP_NO, Oct 2007


