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Verification, Validation and Application of Shear Stress Transport Transitional Model to a R/C Aircraft

Jon Willems

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VERIFICATION, VALIDATION AND APPLICATION OF SHEAR STRESS
TRANSPORT TRANSITIONAL MODEL TO A R/C AIRCRAFT

A Thesis

Submitted to the Faculty

of

Embry-Riddle Aeronautical University

by

Jon Willems

In Partial Fulfillment of the

Requirements for the Degree

of

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VERIFICATION, VALIDATION AND APPLICATION OF SHEAR
STRESS TRANSPORT TRANSITIONAL MODEL TO A R/C AIRCRAFT

by

Jon Willems

A Thesis prepared under the direction of the candidate's committee chairman, Dr. William Engblom, Department of Aerospace Engineering, and has been approved by the members of the thesis committee. It was submitted to the School of Graduate Studies and Research and was accepted in partial fulfillment of the requirements for the degree of Master of Science in Aerospace Engineering.

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SYMBOLS

α	Angle of attack
I	Turbulence intensity
ω	Turbulence frequency
x/c	Axial (x) location normalized by chord length
y^+	Non-dimensional distance from the wall

ABBREVIATIONS

AoA	Angle of Attack
CFD	Computational Fluid Dynamics
CPU	Central Processing Unit
L/D	Lift to Drag Ratio
LES	Large Eddy Simulation
LSB	Laminar Separation Bubble
PIV	Particle Image Velocimetry
RANS	Reynolds Averaged Navier Stokes
Re	Reynolds Number
R/C	Remote Control
RMS	Root Mean Square
S-L	Sea Level
SST	Shear Stress Transport
UAV	Unmanned Aerial Vehicle
VLM	Vortex-Lattice Method

ABSTRACT

Willems, Jon MSAE, Embry-Riddle Aeronautical University, October 2017.

Verification, Validation and Application of Shear Stress Transport Transitional Model to a R/C Aircraft.

Accurate numerical prediction of transition onset as well as transition extent is crucial when evaluating the performance of many remotely controlled and autonomous aircraft. The four-equation Menter-Langtry Transitional viscous model in ANSYS Fluent is applied to a number of 2D and 3D airfoils and wings operating within the transitional flow regime, to permit comparison to relevant experimental and numerical results for transition onset, extent, and overall aerodynamic performance. The sensitivity to grid topology and fineness is also examined. With sufficient confidence in the numerical approach, the performance of a well-known high performance R/C glider is examined and compared to results obtained from a vortex-lattice method based approach.

1. Introduction

1.1. Background

Accurate aerodynamics assessments of remotely controlled and autonomously controlled aircraft (i.e., unmanned aerial vehicles, UAVs) are highly desirable. However, these aircraft tend to operate within the transitional flow regime (i.e., mixed laminar/turbulent flow regime) due to their small characteristic geometric scales and low speeds. The common numerical assumptions in computational fluid dynamics (CFD) of entirely laminar flow or entirely turbulent flow can result in major prediction errors for aerodynamic performance. For example, a conventional Reynolds-Averaged Navier-Stokes (RANS) model often results in huge overestimation of wing skin friction drag. While less sophisticated methods like vortex-lattice methods (VLM) and panel methods are not equipped to accurately model transition for complex 3D flows. These other numerical methods also are not well equipped to capture the laminar separation bubbles that develop in these low Reynolds number transitional flows.

Large eddy simulation has also been applied to resolve the transition phenomenon for airfoils. A study conducted by Skarolek and Miyaji have used 3rd and 4th order flux reconstruction schemes to simulate transitional flow over the SD7003 wing under a low Re of 60,000 (Sharolek & Miyaji, 2014). Their FR scheme results exhibit good agreement with other similar LES efforts, including from Catron de Wiart et al (de Wiart & Hillewaert, 2012) and Galbraith et al (Galbraith & Visbal, 2010). This study also highlights the phenomena of laminar separation bubbles that occur in low Reynolds transitional flows. Although large-eddy simulation (LES) promises to enable high fidelity prediction of the

transition phenomenon, the computational expense is currently excessive when addressing an entire aircraft.

Other groups of researchers have successfully attempted to accurately predict transitional flows within a RANS framework. Basha and Galy (Basha & Ghaly, 2007) have combined the Spallart-Allmaras (SA) turbulence model with the transition onset prediction method developed by Cebeci and Smith (Cebeci & Smith, 1974) within ANSYS Fluent. Excellent agreement to experimental data is shown when applied to the NLF(1)-0416 airfoil operating at a Mach of less than 0.1, and at Re of 2 and 4 million. Another study conducted by Wang and Sheng (Wang & Sheng, 2015) implemented the local correlation-based transition model developed by Menter and Langtry into the unstructured CFD solver U2NCLE. They coupled this transition model with the one equation SA model as well as the SST turbulence model. For a number of flat plate simulations for Reynolds numbers ranging from 50,000 to 3 million, they are able to accurately predict transition onset and the extent of transition. They also validated their model against three separate airfoils showing very good agreement. They then went onto apply their derived model to two industry rotors, the Bell-Boeing J VX as well as the Sikorsky S-76. Halila, Bigarella, and Azevedo (Halila, Bigarella, & Azevedo, 2016) validated the Langtry-Menter 4 equation shear stress transport transitional model (Langtry & Menter, 2009) using the CFD++ solver for a flat plate, an airfoil, a multi-sectional airfoil, and a 3D wing body, showing excellent agreement with experiment.

Other relevant studies include one conducted by DeMauro et al (DeMauro, Dell'Orso, Zaremski, Leong, & Amitay, 2015). They have utilized experimental data from Particle Image Velocimetry (PIV) to visualize laminar separation bubbles (LSB). Once

they had the LSB characterized they were able to use active flow control methods to reduce the size and effect of the LSB. Selig and Guglielmo (Selig & Guglielmo, 1997) evaluated a number of airfoils at low Reynolds numbers of 20,000 – 30,000 in order to analyze the design of high lift airfoils. They produced excellent experimental data for all the airfoils through use of a wind tunnel.

A final noteworthy study was done using the SST Transitional model with 3D wings to analyze the effectiveness of slot span on wing performance. This study showcased validation efforts with excellent agreement to experiment (Granizo, Gudmundsson, & Engblom, 2017).

1.2. Objectives

The main objective behind this study is to provide accurate aerodynamic performance data for the Dual Aircraft Project (DAP) (Engblom & Decker, 2016). The DAP's novel controls strategy requires accurate aerodynamic data to determine appropriate flight conditions. This data could be generated through a number of approaches. One approach is to perform wind tunnel testing, but it was determined to be too expensive and would require outsourcing from ERAU. Another option is to estimate aerodynamics from flight testing, and an effort to do so is currently underway, but unavailable at this time. Another option is to apply an industry standard VLM-XFOIL method. This option was exercised, performed by a consultant, who provided the results used in comparisons made in this thesis document. A problem with relying on this VLM-XFOIL data alone is that there are simplifying assumptions that may lead to significant error within the solutions. For example, this method is not equipped to discover laminar separation bubbles. Since the

DAP will operate in a transitional Reynolds Number it is important that all these effects are accurately modeled, especially in the vicinity of flight Re of 200,000 to 300,000.

A CFD approach which was selected for this thesis effort which makes use of transitional RANS. More specifically, for the present study, we chose to use the Langtry-Menter 4 equation shear stress transport transitional model within Ansys Fluent v17. This model offers the potential to accurately address transitional flow effects within an expedient RANS framework. In order to gain confidence in the transitional RANS framework, the Menter-Langtry SST Transitional Model is verified and validated using a number of 2D airfoil cases covering a range of Reynolds numbers. Then, this model is applied to the well-known R/C glider aircraft the MAXA Pro4m. These full aircraft performance results are to be compared with results obtained from a more traditional VLM-based approach. The impetus for this effort was to refine the aerodynamics assessment of the MAXA Pro4M for use in flight simulations.

2. Numerical Methodology

2.1 ANSYS Fluent Methodology

Six airfoils have been analyzed, a NACA 0009, an Eppler 473, an Eppler 387, an Eppler 397, a NLF(1)-0416 and a SD7003 for verification and validation efforts using the 4 equation SST Transitional model within ANSYS Fluent. These airfoils were selected either to represent typical symmetric and high camber airfoils that may be incorporated within a low-speed gliders (i.e., on tail and wing surfaces, respectively), or for the fact that relevant experimental data existed. Additional results are obtained using the standard 2 equation SST turbulence model to see the relative impact of the transition model.

In order to address the sensitivity to grid topology, C grid and an O grid topologies are constructed (see Figure 2.1.1). C and O grids are applied to the Eppler 473, Eppler 397, and NLF(1)-0416. Once it was observed that there was minimal grid topology dependency, remaining airfoils (NACA 0009, Eppler 387, and SD7003) were evaluated only using an O-grid type topology. For the C grids the airfoils have an infinitely thin trailing edge, while for the O grids the trailing edge has a radius of 1 mm. Both grids ensure the domain to be 25 plus chord lengths in all directions.

The sensitivity to grid fineness is also addressed for both C grids and O grids. The Coarse C grid contains 18,000 cells with a max y^+ value of 7. The medium C grid consists of 72,000 cells with a maximum y^+ value of 3.75. The fine C grid consists of 288,000 cells with a maximum y^+ value of 1.9. Note that the grid are systematically increased such that the cell count doubles in size in each direction for each successive fineness level.

A single fine O-grid was created for each airfoil, with a cell count of approximately 30,000 cells with a maximum y^+ value less than 5 for the entire surface. An additional Medium O-grid was created solely for the NLF(1)-0416.

Figure 2.1.1 NLF(1)-0416 airfoil (O grid on left; medium C grid on right)

The geometry of each airfoil is scaled to match the Reynolds number for each study. The NACA 0009 is evaluated at Mach 0.00631 and using a 1 meter chord results in a Reynolds number of 20,000 to match the PIV experiment study. While the SD7003 is evaluated with a Mach number of 0.1 and the chord length is 0.0264 meters in to obtain Re of 60,000, to match the LES study. The Eppler 387 was ran with a Mach number of 0.00875 with a 1 meter chord length resulting in a Reynolds number of 200,000 the high lift airfoil design study. The Eppler 473 and 397 airfoils are evaluated with a Mach number of 0.0216, which with a 1 meter chord length resulting in a Re of 500,000. Finally, the NLF(1)-0416 airfoil is evaluated with a Mach number of 0.1 and the chord length of 0.873 meters to match the Re of 2 million from experiment (Somers, 1981).

The SST Transitional model has also been applied to the MAXA Pro4M glider main wing with and without flap deflection. Each grid was created using Pointwise's TRex feature, resulting in multi-element unstructured grids of roughly 7.5 million cells. The grid for this wing configuration achieved a maximum y^+ value of 12 at the leading edge, with a majority of the wing less than 5. The 24° flap deflection grid is depicted below in Figure 2.1.2.

Figure 2.1.2 Mesh for MAXA Pro4M 3D wing with 24-degree flap deflection
(Left shows symmetry plane; Right shows entire wing surface)

The final application of the SST Transitional model was to the full MAXA Pro4M glider, with and without various control surface deflections. The full glider was ran at a Mach number of 0.03 with a chord length of 0.3m resulting in a Reynolds number of roughly 200,000. Control surface deflections analyzed are as follows: Rudder 5° deflected, Elevator 5 and 10° deflected, Flaps 5 and 10° deflected, and Aileron 5 and 10° deflected. The pertinent dimensions of these control surfaces were measured from the actual glider and then implemented into pointwise. All grids were constructed in pointwise with the TRex parameters shown in Table 2.1.1 below. These were the same parameters that were used for the wing only grid. All full glider grids resulted in roughly 22 million cells. The max y^+ value along the aircraft surfaces is 8, with most of the surface mesh being 5 or below. Images of the created grids are also provided below. Moments are taken about the center of gravity, which is assumed to lie 108 mm back from the root wing point. Positive pitching moment is nose upward with z axis out the right wing. Positive roll moment is in

the x axis towards the tail. Positive yawing moment is axis oriented downwards in the y direction.

Convergence of all force and moment coefficients is evaluated based on a FORTRAN 90 utility code created by the author. This utility computes the root mean square (RMS) and max/min values of the coefficients for a specified iteration interval. If both max and min values are within 5% difference of the RMS value over 1500 iterations, convergence is considered to be satisfied. Typically, for the main glider, this convergence criteria for all grids was achieved within 6500 iterations on 72-84 CPUs taking roughly 24 hours each case. The governing equation residuals typically decrease by three or more orders of magnitude from initialization.

Table 2.1.1 Full Glider T-Rex Parameters

Max Layers	0
Full Layers	60
Growth Rate	1.25
T-Rex initial spacing (in)	0.006
Collision Buffer	2.0
Skew Criteria, Max Angle	160

Figure 2.1.3 Baseline no deflection glider surface mesh

Figure 2.1.4 Various control surface deflection grids: A) Elevator 10° ,
B) Rudder 5° , C) Aileron 10° (right wing) and D) Flap 10°

2.2. VLM-XFOIL Methodology

The MAXA Pro4M aerodynamics (i.e., static force and moment coefficients) is also evaluated at the same transitional Re of 200,000 using a combination of AVL and XFOIL analysis, to provide a direct comparison to the RANS-based evaluations previously described. The approach, outlined below, is applied to the same matrix of flight conditions computed using the RANS-based approach. This work was conducted by a consultant of the DAP project, Joe Wurts (private communication, 2017).

A pseudo-viscous solution for the entire aircraft, with a control surface deflection, is generated using a combination of AVL (3-D, inviscid vortex-lattice method) and XFOIL (2-D, viscous panel-method). For a full range of AoA , XFOIL is used to evaluate the performance of each 2-D aircraft surface (e.g., outer wing section with aileron deflection) under both inviscid and viscous boundary layer assumptions. The ratio of the local lift coefficients from the viscous and inviscid assumptions is typically less than unity (i.e., a reduction in lift due to viscous boundary layer effects). This ratio is used to approximate the “effective deflection” as produced by a purely inviscid VLM analysis. Thus, AVL may be used to evaluate the 3-D performance of the aircraft using this effective deflection in place of the actual deflection. The final correction is to use the two XFOIL 2-D solutions to isolate and add the viscous drag effect to the VLM drag coefficient, as well as any related effects on the moment coefficients. The main simplification in this method is to introduce the viscous effects in a local 2-D sense.

For reference, the XFOIL data was run for an α range of -10° to $+20^\circ$ every 0.2° , with 13 deflections (-30° to $+30^\circ$ every 5°). The wing is composed of multiple airfoils, so this data was collected for 3 airfoils. The AVL data was run for four different control

surfaces, with an α range of -10° to $+18^\circ$ with an α step of 2° , with control surface deflections of -30° to $+30^\circ$ every 5° . The AVL data collected includes the local lift coefficient distribution in addition to the force and moment coefficients.

3. Results

3.1 Verification using 2D airfoils at Re of 500,000

The first two Eppler airfoils, a symmetric (473) and cambered (397) airfoil, are analyzed at transitional Reynolds number of 500,000 to verify model performance for transitional flows similar to those expected for the MAXA Pro4M aircraft to be evaluated later. ANSYS Fluent results for L/D using the SST Transitional models for both airfoils are compared with XFOIL results in Figures 3.1.1 and 3.1.2. It is known that XFOIL is not necessarily accurate. These comparisons are simply to understand the sensitivities the model possess and to see if expected trends arise. This evaluation is important considering the complexity of the SST Transitional model which contains a number of blending function and highly nonlinear partial differential equations (Langtry & Menter, 2009). The medium and fine C grid solutions do agree well over the entire angle of attack range, indicating that grid independence is achieved when using the SST Transitional model. Also, the O-grid results compare favorably with the C-grid showing that the grid topology sensitivity is minor. The differences in the L/D values shown here are found to be driven much more by differences in the predicted drag coefficients than the lift coefficients.

Figure 3.1.1 L/D predictions for Eppler 473 using SST Transitional model for various grid treatments

Figure 3.1.2 L/D predictions for Eppler 397 using SST Transitional model for various grid treatments

Figures 3.1.3 and 3.1.4 show the L/D results from using the fully turbulent assumption (i.e. using the SST model) compared to XFOIL for the Eppler 473 and 397 airfoils, respectively. Figures 3.1.3 and 3.1.4 results are computed from the same inflow conditions used in Figures 3.1.1 and 3.1.2. The fully turbulent drag coefficients are nearly double when compared to predictions with the SST Transitional model in Figures 3.1.1 and 3.1.2, and result in a correspondingly low L/D. The C and O grid results strongly agree, suggesting reduced sensitivity to grid fineness and grid topology when using the fully turbulent SST model. It is easy to see for these low Reynolds number cases that a fully turbulent assumption will result in inaccurate values of drag.

Figure 3.1.3 L/D predictions for Eppler 473 using SST Turbulence model

Figure 3.1.4 L/D predictions for Eppler 397 using SST Turbulence model

Figures 3.1.5 and 3.1.6 show the variation of the lift coefficient over the AoA range for all grid fineness levels and topologies using the Transitional SST model, for both airfoils. Weak sensitivity to the C-grid fineness and grid topology is observed.

Figure 3.1.5 Lift coefficients for the Eppler 473 symmetric airfoil

Figure 3.1.6 Lift coefficients for the Eppler 397 cambered airfoil

The results of this verification exercise suggest that the transitional SST model produces vastly superior predictions for aerodynamic performance compared to a fully turbulent approach for the Reynolds number regime of relevance to our R/C aircraft. Also the weak sensitivity to both grid fineness and grid topology will provide guidance on grid development for subsequent airfoils and the R/C aircraft.

3.2 Verification using SD7003 2D airfoil

Another interesting comparison is generated by applying the SST Transitional model to a LES case studied using a higher order Flux Reconstruction scheme (Sharolek & Miyaji, 2014). LES requires orders-of-magnitude more computational effort due to less reliance on turbulence modeling. A simulation using the RANS SST Transitional model was completed for the SD7003 to directly compare with LES results by Sharolek and Miyaji (Sharolek & Miyaji, 2014). In their paper they have results for the evolution of lift and drag with time, which involve 3rd and 4th order spatial accuracy for a low transitional Reynolds number of 60,000. Excellent agreement is obtained for both coefficients when comparing the LES results to the present SST Transitional model results. Table 3.2.1 shows percentage differences of roughly 1% and 5% for lift and drag, respectively.

Table 3.2.1 Comparisons of LES study to RANS study at $\alpha = 4^\circ$

<i>Measure</i>	<i>LES</i>	<i>SST Trans</i>	<i>% difference</i>
cd	0.0207	0.0219	5.49
cl	0.5887	0.5806	1.38

Another key aspect showcased in the LES study is the presence of a laminar separation bubble, which is an important feature of low Reynolds number flows. Below in

Figure 3.2.1 is a comparison of flow visualization between the LES study and the SST Transitional Model. Both plots have contours of normalized stream wise velocity in order to display where the reversed flow occurs resulting in a laminar separation bubble.

Figure 3.2.1 Comparisons of normalized streamwise velocity on SD7003
A) LES (Sharolek & Miyaji, 2014), B) SST Transitional

It can be seen that the SST Transition model still does a fairly good job of capturing this complicated phenomenon. The SST Transitional model predicts the onset of the laminar separation bubble well when compared to the LES results. The reattachment of the separation bubble is not easily identifiable in 3.2.1 for the SST Transitional model, so the

supplementary image 3.2.2 is shown with streamlines for the SST Transition model to illustrate where the flow first reattaches. Now it can be seen that the reattachment (note recirculation is evident from streamlines) of the laminar separation bubble compares favorably to the LES visualization.

Figure 3.2.2 Normalized streamwise velocity for SST Transitional model on SD7003 airfoil.

3.3 Validation using NACA 0009 at Re of 20,000

Considering that laminar separation bubbles are a dominating feature of low Reynolds number transitional flows, another visualization comparison was conducted. This time however, the SST Transitional model was compared to experiment in the form of Particle Image Velocimetry (PIV). The experimental results are from the study conducted by DeMauro et al (DeMauro, Dell'Orso, Zaremski, Leong, & Amitay, 2015) for a Reynolds number of 20,000. The comparisons are shown in Figure 3.3.1, and 3.3.2, and show contours of normalized streamwise velocity along with velocity vectors on the NACA 0009 airfoil geometry. Visualization of results from the fully turbulent SST model shown along with Figure 3.3.2.

Figure 3.3.1 Contours of normalized streamwise velocity with velocity vectors for an AoA of 3.5° for NACA 0009. A) PIV Experiment (DeMauro et al, 2015), B) SST Transitional Model

From Figure 3.3.1 one can observe that a laminar separation bubble is not present in either case for an angle of attack of 3.5° . When we increase the angle of attack to 5° a laminar separation bubble forms as seen in Figure 3.3.2 below.

Figure 3.3.2 Contours of normalized streamwise velocity with velocity vectors for an AoA of 5°. A) PIV Experiment (DeMauro et al, 2015), B) SST Transitional, C) SST Turbulence

It is seen from Figure 3.3.2 that the SST Transition model has excellent agreement in the prediction of the onset of the bubble, at around 20 percent of the chord, where the SST Turbulence model (3.3.2 part C) has no evidence that a LSB has formed. These comparisons further support the validity of the SST Transitional model and continue to showcase the limitation of using a fully turbulent for these Reynolds numbers. These

visualizations are important because LSB have a significant effect on the aerodynamic performance at sufficiently low Re.

Another important model sensitivity came out of the analysis of the NACA 0009. The SST model displays a sensitivity to the freestream turbulence intensity percentage. This is a known characteristic of the $k-\omega$ Wilcox model (Menter, 1992) which is the underlying turbulence model for the SST Transition model.

Turbulence intensity is defined using the equations below (Russo & Basse, 2016).

$$I \equiv \frac{u'}{U}$$

Where u' is defined as the root-mean-square of the turbulent velocity fluctuations and U is the mean velocity. If the turbulent kinetic energy is known u' can be computed shown below.

$$u' \equiv \sqrt{\frac{1}{3}(u_x'^2 + u_y'^2 + u_z'^2)} = \sqrt{\frac{2}{3}k}$$

Also U can be computed from the three mean velocity components shown below.

$$U \equiv \sqrt{U_x^2 + U_y^2 + U_z^2}$$

Figure 3.3.3 Lift coefficient as a function of freestream turbulence intensity percentage

Figure 3.3.4 Drag coefficient as a function of freestream turbulence intensity percentage

From Figure 3.3.3 it is evident that there is a strong sensitivity to the freestream turbulent intensity percentage. The lift coefficient changes nearly 15 percent when increasing the turbulence intensity from 1 to 5 percent. Where the drag sees a much less pronounced effect in the change of the freestream turbulence intensity, drag only changes by roughly 1.5 percent from 1 to 5 percent turbulence intensity. It is speculated that the reason for this sensitivity is due to an invigorated boundary layer, thus delaying separation, and causing a rise in the lift coefficient.

3.4 Validation using Eppler 387 2D airfoil at Re of 200,000

The ultimate goal of the validation and verification of this SST Transitional model is to apply it to a real full-scale glider. This glider will be flying in a Reynolds regime of roughly 200,000. In order to establish even more confidence in the present SST Transitional model, a 200,000 Reynolds number validation case is performed using the Eppler 387 airfoil. Figure 3.4.1 shows the comparisons of drag polars produced by the SST Transitional model, experimental data, and XFOIL. The experimental drag polars in Figure 3.4.1 are from the study conducted by Selig and Guglielmo (Selig & Guglielmo, 1997).

Figure 3.4.1 Drag polar comparisons for Eppler 387

As can be seen from Figure 3.4.1 there is excellent agreement throughout a wide range of angles of attack between the SST Transitional Model and the experiment. This helps showcase that the choice to use this particular model for our full glider application is appropriate.

3.5 Validation using NLF 2D Airfoil at Re of 2,000,000

The NLF(1)-0416 airfoil was evaluated using the SST Transitional model to predict transition onset location and extent as well as aerodynamic performance for direct comparison to experimental data from the study done by Basha and Ghaly (Basha & Ghaly, 2007). As before, C-grids of various fineness and O-grid topologies are developed and

utilized to evaluate grid sensitivity. Figures 3.5.1 and 3.5.2 indicate good agreement with experiment for both Fine O and C grid type topologies as well as the Medium C grid.

Figure 3.5.1 Drag coefficient vs angle of attack comparisons for NLF(1)-0416

Figure 3.5.2 L/D vs angle of attack comparisons for NLF(1)-0146

In order to pick out the transition location for the SST Transition model plots similar to Figure 3.5.3 were created for each angle of attack. The transition location was taken to be the point where the turbulent kinetic energy (TKE) has sharply increased. This is a good estimate of the transition location because a laminar flow will have no TKE.

Figure 3.5.4 illustrates the transition onset predicted by the SST Transitional model for the O grid topology versus experimental measurements. The transition onset location obtained from the CFD simulations follow the same trends of the experiment data versus AoA and also closely agrees with the experimentally obtained onset location. As the angle of attack is increased, the onset location moves upstream, closer to the leading edge, for the upper surface, and moves aft, towards the trailing edge for the lower surface.

Figure 3.5.3 Turbulent kinetic energy vs x/c for the NLF(1)-0146 airfoil

In Figure 3.5.3, we can see the two sharp increases in TKE from 0 at x/c locations of roughly 0.4 and 0.6 for the upper and lower surfaces respectively. This process was applied for all the angles of attack and was compiled in Figure 3.5.4 below was created to compare to experiment.

Figure 3.5.4 Transition location vs angle of attack for CFD and Experiment

3.6 Application to MAXA Wing at Re of 200,000

The SST Transitional model was applied to a MAXA Pro4M wing only geometry, and analyzed with no flap deflection, as well as 24° flap deflection. This study was conducted to calibrate an air data probe that will be used on the actual MAXA Pro glider in flight tests. The SST Transitional Model solutions were used to predict the local u , v , and w flow velocity components at a location consistent with the sensing inlet of the air data probe. Once the local α was computed for both the no flap deflection case as well as the 24° flap deflection case, and then directly compared to predictions using the VLM-XFOIL method described in Section 2.2. These comparisons are shown below in Figures 3.6.1 and 3.6.2.

Figure 3.6.1 Comparison of probe sensed local α for given freestream α

Figure 3.6.2 Comparison of probe sensed local α for given freestream α

Excellent agreement is obtained between the VLM-XFOIL and SST Transitional models.

Note that the difference between the local α and the freestream α is the upwash at the probe inlet, which is positioned roughly 12 inches forward of the leading edge at the inboard/outboard wing junction.

An image of the turbulent kinetic energy on the surface of the wing, as well as along a slice across along the wing span, is provided in Figure 3.6.3. This image indicates a transition line near the LE of the suction surface of the wing.

Figure 3.6.3 TKE on the surface of MAXA wing with 24° flap deflection, as well as a slice located at $y = 1.27$ meters from root.

It is important to note that transition was not found to be occurring on the underside of the wing even for low angles of attack of $4^\circ \alpha$.

3.7 Application to MAXA Full Glider at Re of 200,000

The main goal of this study is to perform a credible, high fidelity numerical analysis of the MAXA Pro4M Glider's aerodynamics. The first half of the paper was devoted to the verification and validation of the transitional model. The second portion of the study is dedicated to the application of the model. Fig. 3.7.1 provides visualization of the solution for the full glider at 8° angle-of-attack. Note that wing tip flow separation is predicted with this model.

Figure 3.7.1 Full glider solution with streamlines and pressure contour at $\alpha = 8^\circ$

3.8 Comparisons of SST Transitional Results with VLM-XFOIL Results

The SST Transitional model was applied to a wide range of grids with and without control surface deflections for various α . Below is the comprehensive run matrix describing how the model was applied.

Table 3.8.1 Run matrix for full glider with SST Transitional Model application

Grid	α (°)	Grid	α (°)
Baseline no deflections	-6	5° Flap Deflection	0
	-4		4
	-2		8
	0	10° Flap Deflection	0
	2		8
	4		
	6		
	8	5° Rudder Deflection	0
	10		8
5° Elevator Deflection	0	5° Aileron Deflection	0
	4		8
	8		
10° Elevator Deflection	0	10° Aileron Deflection	0
	8		8

Once the solutions for all the above cases were computed, they were compared to the corresponding runs from the VLM-XFOIL methodology. Comparison with no control surface deflections are shown below in Figures 3.8.1 – 3.8.3.

Figure 3.8.1 C_l vs α for SST Transitional and VLM-XFOIL methods

Figure 3.8.2 C_d vs α for SST Transitional and VLM-XFOIL methods

Figure 3.8.3 C_m vs α for SST Transitional and VLM-XFOIL

There is good agreement between the CFD and VLM throughout α range for c_l , c_d , and c_m . However at the higher α s near stall there is growing disagreement. In order to demonstrate why at 8° AOA and above the agreement falls apart, Figure 3.8.4 displays the massive wing flow separation that is occurring at 10° AOA. This is a phenomena that the VLM-XFOIL method is not well equipped to discover. However, wind tunnel or flight testing is needed to verify the development of flow separation at this flight condition.

Figure 3.8.4 Constant Z slices down the length of the wing at $10^\circ \alpha$. A) Slice at $Z \sim \text{root}$, B) Slice at $Z \sim \text{mid span}$, C) Slice at $Z \sim 3/4 \text{ wing}$, D) Slice at $Z \sim \text{near tip}$

Tables 3.8.2 – 3.8.15 are shown below to disseminate the predicted changes in static force and moment coefficients due to control surface deflections. They are presented in the form of delta coefficients. These deltas are defined to be the magnitude and direction of the change from the baseline results when a control surface is deflected. When comparing the tables between the VLM-XFOIL results to the SST Transitional method there is good agreement in the magnitudes and directions of the deltas. The expected trends for all of the various control surfaces are obtained. For example, a downward deflection of the elevator

creates a large negative (nose down) change in the pitching moment coefficient from the baseline c_m . Also with the flaps deflected there is a positive change in the lift and drag from the baseline. With the rudder deflected to the left a negative change in yawing moment is obtained as expected. Finally, aileron deflections with the left down and right up creates a negative as expected.

It is worthwhile to discuss the control surface effectiveness while the aircraft is at high angle of attack. Note that some of key results for comparison at high angle of attack are in bold face. The SST Transition model predicts large scale flow separation at high angles of attack, as described early in the paper, which apparently significantly reduce the effectiveness of the control surfaces. Note that the delta changes in the coefficients in the tables below drop significantly at α of 8deg versus α of 4deg. When looking at the VLM-XFOIL results you can observe that the deltas remain constant from the baseline results even at the high angle of attack. This is likely due to the lack of flow separation predicted by the VLM-XFOIL method.

Table 3.8.2 Elevator deflected 5° downward CFD

α (°)	delta c_l	delta c_y	delta c_d	delta c_m	delta roll	delta yaw
0	0.0256	0.0001	0.0006	-0.1126	0.0001	0.0000
4	0.0256	-0.0002	0.0017	-0.1140	0.0000	0.0000
8	0.0164	0.0001	0.0030	-0.1004	0.0002	0.0002

Table 3.8.3 Elevator deflected 5° downward VLM-XFOIL

α (°)	delta c_l	delta c_y	delta c_d	delta c_m	delta roll	delta yaw
0	0.0287	0.0000	0.0005	-0.1240	0.0000	0.0000
4	0.0284	0.0000	0.0016	-0.1228	0.0000	0.0000
8	0.0276	0.0000	0.0024	-0.1196	0.0000	0.0000

Table 3.8.4 Elevator deflected 10° downward CFD

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.0510	0.0001	0.0022	-0.2239	0.0002	0.0000
8	0.0507	0.0002	0.0088	-0.1884	0.0012	0.0001

Table 3.8.5 Elevator deflected 10° downward VLM-XFOIL

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.0572	0.0000	0.0019	-0.2470	0.0000	0.0000
8	0.0540	0.0000	0.0053	-0.2342	0.0000	0.0000

Table 3.8.6 Flap deflected 5° downward CFD

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.1113	0.0001	0.0038	0.0367	0.0003	0.0000
4	0.0822	-0.0002	0.0059	0.0397	0.0000	0.0000
8	0.0506	0.0002	0.0078	0.0212	0.0015	0.0001

Table 3.8.7 Flap deflected 5° downward VLM-XFOIL

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.1104	0.0000	0.0030	0.0284	0.0000	0.0000
4	0.1054	0.0000	0.0051	0.0270	0.0000	0.0000
8	0.0931	0.0000	0.0103	0.0280	0.0000	0.0000

Table 3.8.8 Flap deflected 10° downward CFD

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.2268	0.0000	0.0104	0.0652	-0.0001	0.0000
8	0.0763	0.0007	0.0163	0.0419	0.0031	0.0001

Table 3.8.9 Flap deflected 10° downward VLM-XFOIL

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.2100	0.0000	0.0078	0.0564	0.0000	0.0000
8	0.1680	0.0000	0.0218	0.0536	0.0000	0.0000

Table 3.8.10 Aileron deflected 5° CFD (left down, right up)

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	-0.0024	0.0146	0.0018	0.0016	-0.0455	-0.0013
8	-0.0116	0.0068	0.0026	0.0029	-0.0184	-0.0035

Table 3.8.11 Aileron deflected 5° VLM-XFOIL (left down, right up)

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	-0.0019	0.0131	0.0012	0.0002	-0.0399	0.0004
8	-0.0036	0.0116	0.0010	0.0010	-0.0340	-0.0079

Table 3.8.12 Aileron deflected 10° CFD (left down, right up)

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	-0.0081	0.0289	0.0074	0.0052	-0.0902	-0.0026
8	-0.0731	0.0157	0.0081	0.0059	-0.0470	-0.0067

Table 3.8.13 Aileron deflected 10° VLM-XFOIL (left down, right up)

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	-0.0097	0.0257	0.0045	0.0003	-0.0779	0.0008
8	-0.0175	0.0226	0.0031	0.0014	-0.0661	-0.0155

Table 3.8.14 Rudder deflected 5° leftward CFD

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	-0.0003	-0.0142	0.0005	0.0017	0.0006	-0.0043
8	0.0111	-0.0149	-0.0002	0.0029	0.0081	-0.0024

Table 3.8.15 Rudder deflected 5° leftward VLM-XFOIL

α (°)	delta cl	delta cy	delta cd	delta cm	delta roll	delta yaw
0	0.0001	-0.0136	0.0003	-0.0001	0.0004	-0.0043
8	0.0000	-0.0138	0.0005	-0.0001	0.0010	-0.0043

4. Conclusion

The Menter-Langtry SST Transitional Model in ANSYS fluent was verified and validated using a number of airfoils at a wide range of Reynolds numbers from 20,000 up to 2,000,000. This SST Transition model displayed excellent agreement to experiment for both aerodynamic coefficients as well as transition location for a number of geometries and flow conditions. The model's sensitivities were also analyzed through the use of different levels of grid fineness as well as grid topologies. It was found that this model showcases a consistent sensitivity to grid fineness (near grid independence is obtained) and weak sensitivity to grid topology. However, it is found that this model has a moderate sensitivity to the prescribed freestream turbulence intensity percentage. This model was also validated through visual comparisons to both LES and experiment. These comparisons showcase that the SST Transition model is capturing the flow features present in these low Reynolds number flows, including laminar separation bubbles.

All these validation cases establish sufficient confidence in the model to believe that its application to a full glider will yield accurate results. Furthermore the results of the SST-Trans model to the MAXA Pro compare well to an industry like VLM-XFOIL method for angles of attack of 6° and less. Where the SST Transition model does disagree with the VLM-XFOIL method (i.e. above 6°) appears to be due to flow separation at high angles of attack that the VLM-XFOIL method does not predict. However, wind tunnel testing or flight testing would be needed for further validation. Consequently, the aerodynamic performance data from this investigation will be used by the DAP flight control software but its validity is assumed limited to angles of attack of less than 6° .

5. Recommendations for Future Work

The Author concludes with recommendations for future work on this project:

1. Extend study to include the effects of sideslip to see how the SST Transitional model handles cross flow.
2. Increase the number of angles of attack in the matrix used to evaluate control surface effectiveness for a more complete comparison with VLM-XFOIL method.
3. Conduct flight testing and/or conduct wind tunnel testing in order to produce accurate aerodynamic performance data to validate the SST Transitional model results for the full glider.

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A. Convergence Checking FORTRAN 95 Code

```

program Convergence_Check_new
  implicit none

  ! Begin the defining of variable types
  integer :: i, tol, Nstart, Niter, dum

  real :: avg_cl, avg_cd, rms_cl, rms_cd, cl_max_diff, cd_max_diff, cl_min_diff, cd_min_diff
  real :: avg_cm, rms_cm, cm_max_diff, cm_min_diff, avg_cy, avg_roll, avg_yaw
  real :: max_cl, max_cd, min_cl, min_cd, rms_cy, rms_roll, rms_yaw
  real :: max_cm, min_cm, max_cy, min_cy, max_roll, min_roll, max_yaw, min_yaw
  real :: cy_max_diff, cy_min_diff, roll_max_diff, roll_min_diff, yaw_min_diff, yaw_max_diff
  real, allocatable :: cd(:), cl(:), sq_cl(:), sq_cd(:), it(:), cd_new(:), cl_new(:)
  real, allocatable :: cm(:), sq_cm(:), cm_new(:), cy(:), sq_cy(:), cy_new(:)
  real, allocatable :: roll(:), sq_roll(:), roll_new(:), yaw(:), sq_yaw(:), yaw_new(:)
  character (len=*), parameter :: Header='it    cl    cd    cm    cy    roll    yaw'

  ! read in the cl, cd cm cy roll and yaw data from fluent
  open(unit=10, file='inputfile_cl.dat')
  open(unit=12, file='inputfile_cd.dat')
  open(unit=15, file='inputfile_cm.dat')
  open(unit=20, file='inputfile_cy.dat')
  open(unit=22, file='inputfile_roll.dat')
  open(unit=25, file='inputfile_yaw.dat')

  open(unit=17, file='input.dat')
  read(17,*)Niter,Nstart,tol

  allocate(it(Niter))
  it = 0
  allocate(cd(Niter))
  cd = 0
  allocate(cl(Niter))
  cl = 0
  allocate(cm(Niter))
  cm = 0
  allocate(cy(Niter))
  cy = 0
  allocate(roll(Niter))
  roll = 0
  allocate(yaw(Niter))
  yaw = 0
  allocate(sq_cl(Niter-Nstart))
  sq_cl = 0
  allocate(sq_cd(Niter-Nstart))
  sq_cd = 0
  allocate(sq_cm(Niter-Nstart))
  sq_cm = 0
  allocate(sq_cy(Niter-Nstart))
  sq_cy = 0
  allocate(sq_roll(Niter-Nstart))
  sq_roll = 0

```

```

allocate(sq_yaw(Niter-Nstart))
sq_yaw = 0
allocate(cd_new(Niter-Nstart))
cd_new = 0
allocate(cl_new(Niter-Nstart))
cl_new = 0
allocate(cm_new(Niter-Nstart))
cm_new = 0
allocate(cy_new(Niter-Nstart))
cy_new = 0
allocate(roll_new(Niter-Nstart))
roll_new = 0
allocate(yaw_new(Niter-Nstart))
yaw_new = 0

do i=1,Niter
    read(12,*) dum,  cd(i)
    read(10,*) it(i), cl(i)
    read(15,*) dum,  cm(i)
    read(20,*) dum,  cy(i)
    read(22,*) dum,  roll(i)
    read(25,*) dum,  yaw(i)
end do

open(unit=11, file='cl.dat')
open(unit=13, file='cd.dat')
open(unit=14, file='cm.dat')
open(unit=30, file='cy.dat')
open(unit=31, file='roll.dat')
open(unit=32, file='yaw.dat')

do i =Nstart, Niter
    write(11,*) it(i),cl(i)
    write(13,*) it(i),cd(i)
    write(14,*) it(i),cm(i)
    write(30,*) it(i),cy(i)
    write(31,*) it(i),roll(i)
    write(32,*) it(i),yaw(i)
end do

! use the cl and cd that is need for computations
do i = 1,Niter-Nstart
    cl_new(i) = cl(i+Nstart)
    cd_new(i) = cd(i+Nstart)
    cm_new(i) = cm(i+Nstart)
    cy_new(i) = cy(i+Nstart)
    roll_new(i) = roll(i+Nstart)
    yaw_new(i) = yaw(i+Nstart)
end do

sq_cl = cl_new**2
sq_cd = cd_new**2
sq_cm = cm_new**2
sq_cy = cy_new**2
sq_roll = roll_new**2

```

```

sq_yaw = yaw_new**2

    avg_cl = sum(sq_cl)/size(sq_cl)
    avg_cd = sum(sq_cd)/size(sq_cd)
    avg_cm = sum(sq_cm)/size(sq_cm)
    avg_cy = sum(sq_cy)/size(sq_cy)
    avg_roll = sum(sq_roll)/size(sq_roll)
    avg_yaw = sum(sq_yaw)/size(sq_yaw)
print *, avg_cl
print *, avg_cy

    rms_cl = avg_cl**0.5
    rms_cd = avg_cd**0.5
    rms_cm = avg_cm**0.5
    rms_cy = avg_cy**0.5
    rms_roll = avg_roll**0.5
    rms_yaw = avg_yaw**0.5

print *, rms_cl

! find the max and min values of the period being analyzed
    max_cl = maxval(cl_new)
    max_cd = maxval(cd_new)
    max_cm = maxval(cm_new)
    max_cy = maxval(cy_new)
    max_roll = maxval(roll_new)
    max_yaw = maxval(yaw_new)

    min_cl = minval(cl_new)
    min_cd = minval(cd_new)
    min_cm = minval(cm_new)
    min_cy = minval(cy_new)
    min_roll = minval(roll_new)
    min_yaw = minval(yaw_new)

! calculate the percentage differences between the max and min values from the RMS of the
oscillation period
    cl_max_diff = (abs(rms_cl - abs(max_cl))/(abs(rms_cl + abs(max_cl))/2))*100;
    cl_min_diff = (abs(rms_cl - abs(min_cl))/(abs(rms_cl + abs(min_cl))/2))*100;
    cd_max_diff = (abs(rms_cd - abs(max_cd))/(abs(rms_cd + abs(max_cd))/2))*100;
    cd_min_diff = (abs(rms_cd - abs(min_cd))/(abs(rms_cd + abs(min_cd))/2))*100;
    cm_max_diff = (abs(rms_cm - abs(max_cm))/(abs(rms_cm + abs(max_cm))/2))*100;
    cy_max_diff = (abs(rms_cy - abs(max_cy))/(abs(rms_cy + abs(max_cy))/2))*100;
    roll_max_diff = (abs(rms_roll - abs(max_roll))/(abs(rms_roll + abs(max_roll))/2))*100;
    yaw_max_diff = (abs(rms_yaw - abs(max_yaw))/(abs(rms_yaw +
abs(max_yaw))/2))*100;
    cm_min_diff = (abs(rms_cm - abs(min_cm))/(abs(rms_cm + abs(min_cm))/2))*100;
    cy_min_diff = (abs(rms_cy - abs(min_cy))/(abs(rms_cy + abs(min_cy))/2))*100;
    roll_min_diff = (abs(rms_roll - abs(min_roll))/(abs(rms_roll + abs(min_roll))/2))*100;
    yaw_min_diff = (abs(rms_yaw - abs(min_yaw))/(abs(rms_yaw +
abs(min_yaw))/2))*100;

! sign check
    if ( sum(cl_new) < 0) then
print *, 'cl is negative'

```

```

                                rms_cl = -rms_cl
end if

if ( sum(cm_new) < 0) then
    print *, 'cm is negative'
    rms_cm = -rms_cm
end if

if ( sum(cy_new) < 0) then
    print *, 'cy is negative'
    rms_cy = -rms_cy
end if

if ( sum(roll_new) < 0) then
    print *, 'roll is negative'
    rms_roll = -rms_roll
end if

if ( sum(yaw_new) < 0) then
    print *, 'yaw is negative'
    rms_yaw = -rms_yaw
end if

print *, rms_cl

! use the specified tolerance to see if good convergence was achieved
    if (cl_max_diff > tol) then
        print *, 'Error difference between maximum cl and RMS is', cl_max_diff,
'percent'

    else if (cl_min_diff > tol) then
        print *, 'Error difference between minimum cl and RMS is', cl_min_diff,
'percent'

    else if (cd_max_diff > tol) then
        print *, 'Error difference between maximum cd and RMS is', cd_max_diff,
'percent'

    else if (cd_min_diff > tol) then
        print *, 'Error difference between minimum cd and RMS is', cd_min_diff,
'percent'

    else if (cm_max_diff > tol) then
        print *, 'Error difference between maximum cm and RMS is', cm_max_diff,
'percent'

    else if (cm_min_diff > tol) then
        print *, 'Error difference between minimum cm and RMS is', cm_min_diff,
'percent'

    else
        print *, 'None of the oscillations are outside of your tolerance'
    end if

! report the final values
    print *, 'L/d is ', rms_cl/rms_cd
print *, 'cl is ', rms_cl
print *, 'cd is ', rms_cd
print *, 'cm is ', rms_cm
print *, 'cy is ', rms_cy
print *, 'roll is ', rms_roll
print *, 'yaw is ', rms_yaw
print *, "cl_max_diff: ", cl_max_diff

```



```
print *, "cl_min_diff: ",cl_min_diff
print *, "cd_max_diff: ",cd_max_diff
print *, "cd_min_diff: ",cd_min_diff
print *, "cm_max_diff: ",cm_max_diff
print *, "cm_min_diff: ",cm_min_diff
print *, "cy_max_diff: ",cy_max_diff
print *, "cy_min_diff: ",cy_min_diff
print *, "roll_max_diff: ",roll_max_diff
print *, "roll_min_diff: ",roll_min_diff
print *, "yaw_max_diff: ",yaw_max_diff
print *, "yaw_min_diff: ",yaw_min_diff
```

```
end program Convergence_Check_new
```

B. Bash Script for setting up convergence checking code

```
#!/bin/bash

# remove the first two lines of the files
ReRun=1 #This determines if have re ran these files
if [ "$ReRun" -eq 0 ]; then

    echo 1000,5 > start_tol.dat

    # Move the data files to more generic names
    mv cl-* inputfile_cl.dat
    mv cd-* inputfile_cd.dat
    mv cm_* inputfile_cm.dat
    mv cy-* inputfile_cy.dat
    mv cm2* inputfile_roll.dat
    mv cm3* inputfile_yaw.dat

    # Remove the header info from the files
    sed -i 1,2d inputfile_cd.dat
    sed -i 1,2d inputfile_cl.dat
    sed -i 1,2d inputfile_cm.dat
    sed -i 1,2d inputfile_cy.dat
    sed -i 1,2d inputfile_roll.dat
    sed -i 1,2d inputfile_yaw.dat
fi

# Remove old pictures from the directory
rm *.png
#grab the amount of iterations ran
X=`tail -1 inputfile_cl.dat | awk '{print $1}'`
Y=`cat start_tol.dat`

echo $X,$Y > input.dat
Convergence_Check_new > Results

gnuplot ~/bin/it_vs_cl_cd
#python Plot_All_Coeffs.py
```

C. Example input file for Fluent

```

rc Manta_FINAL.cas
/mesh/scale
0.0254
0.0254
0.0254
/define/models/viscous/transition-sst?
y
define/models/viscous/trans-sst-roughness-correlation
y
n
1e-06
/define/models/viscous/curvature-correction?
y
/define/boundary-conditions/velocity-inlet
,
n
yes
,
,
,
,
,
,
10
,
0
n
0
n
y
n
1
5
10
/report/reference-values/area
0.82
/report/reference-values/compute/velocity-inlet
,
/solve/set/p-v-coupling
21
/solve/set/discretization-scheme/mom
0
/solve/set/discretization-scheme/intermit
0

```

```

/solve/set/discretization-scheme/k
0
/solve/set/discretization-scheme/retheta
0
/solve/set/discretization-scheme/omega
0
/solve/set/under-relaxation/body-force
0.25
/solve/set/under-relaxation/density
0.25
/solve/set/under-relaxation/k
0.2
/solve/set/under-relaxation/mom
0.175
/solve/set/under-relaxation/omega
0.2
/solve/set/under-relaxation/pressure
0.075
/solve/set/under-relaxation/retheta
0.2
/solve/set/under-relaxation/turb-viscosity
0.25
/solve/set/under-relaxation/intermit
0.2
/solve/monitors/force/set-drag-monitor
cd-1
y
glider
,
y
y
cd-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta
y
,
n
1
0
0
/solve/monitors/force/set-lift-monitor
cl-1
y
glider
,
y
y
cl-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta

```

```

y
,
n
0
1
0
/solve/monitors/force/set-lift-monitor
cy-1
y
glider
,
y
y
cy-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta
y
,
n
0.0000000
0.0000000
-1.0000000
/solve/monitors/force/set-moment-monitor
cm-1_pitch
y
glider
,
y
y
cm_pitch_moment-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta
y
,
n
0.108097
0.0356211
0.00000111506
0
0
1
/solve/monitors/force/set-moment-monitor
cm-2_roll
y
glider
,
y
y
cm2_roll_moment-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta
y

```

```

,
n
0.108097
0.0356211
0.00000111506
-1
0
0
/solve/monitors/force/set-moment-monitor
cm-3_yaw
y
glider
,
y
y
cm3_yaw_moment-MAXA-Full-0-deg-Baseline-6- $\alpha$ -0-beta
y
,
n
0.108097
0.0356211
0.00000111506
0
-1
0
/solve/monitors/residual/check-convergence
n
n
n
n
n
n
n
n
n
file/auto-save/data-frequency
1000
/solve/initialize/compute-defaults/velocity-inlet
inlet
/solve/initialize/initialize-flow
it 500
/solve/set/discretization-scheme/density
1
/solve/set/discretization-scheme/mom
1
/solve/set/discretization-scheme/intermit
1

```

```
/solve/set/discretization-scheme/k
1
/solve/set/discretization-scheme/retheta
1
/solve/set/discretization-scheme/temperature
1
/solve/set/discretization-scheme/omega
1
it 8000
wc Manta_Maxa_Full_Glider_Analysis_6_A_0_Beta.cas
wd Manta_Maxa_Full_Glider_Analysis_6_A_0_Beta.dat
```