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Viscous Drag Predictions

Coordinating Responsibility: The Boeing Company

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High Speed Research II – Configuration Aerodynamics

WBS 4.3.1.1: Viscous Drag Predictions

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1.0 Objectives

Recent CFD validation studies have shown significant variations in viscous drag predictions between the various methods used by the NASA and industry HSCT organizations. The prediction methods include Navier Stokes CFD codes in which the viscous forces are part of the solutions, and predictions obtained from the different fully turbulent flow flat plate skin friction drag equations used by the various organizations. The Navier stokes codes are typically used for detailed configuration analyses as well as parametric design optimization studies. The flat plate skin friction analyses are used for preliminary design (PD) development and trade studies, performance calculations and as the initial viscous drag estimates for optimized configurations. Figures 1 and 2 show viscous drag predictions for the TCA wing / body and wing / body /nacelle configurations. Significant differences in the various Navier Stokes predictions as well as the skin friction calculations are evident in these Figures.

The primary objectives of this study are to resolve the differences in the viscous drag predictions of the different Navier-Stokes codes employed in the HSR program and to select a common flat plate skin friction drag prediction method.

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2.0 Approach

A systematic study consisting of five sequential tasks has been defined to achieve the study objectives. These include:

- 1. Provide an experimental database of fully turbulent flow skin friction measurements on flat plate adiabatic surfaces at subsonic through supersonic Mach numbers and for a wide range of Reynolds numbers.
- Conduct CFD calculations of fully turbulent flat plate skin friction for comparison to the accumulated database. The turbulence models and grid techniques that resulted in the best agreement of the CFD predictions with the flat plate test data were to be selected for the subsequent validation activities.
- Define, build and test a symmetric version of the TCA at zero degrees of angle of attack to obtain total drag measurements on a realistic configuration at conditions where viscous drag is a major contributor to the total drag, the flow is attached, and aeroelastic effects are eliminated.
- 4. Conduct CFD drag predictions of the symmetric TCA configuration for comparison with the corresponding test data.
- 5. Calculate the drag of the TCA at angles of attack near 1 g climb and cruise conditions using the selected turbulence models and grid techniques. This is complicated since that the flow may no longer be attached everywhere, viscous drag is not the dominate contributor to total drag, and aeroelastic effects may become significant.

3.0 Accomplishments

3.1 Flat Plate Skin Friction Database

Overview

The objective of this activity was to provide an experimental database of fully turbulent flow skin friction measurements on flat plate adiabatic surfaces at subsonic through supersonic Mach numbers and for a wide range of Reynolds numbers.

The database was originally assembled in 1960 from selected experiments conducted prior to that time period(Ref 1). The criteria used to select the appropriate test data are described in the reference. Data were also found on turbulent boundary layer velocity profiles and it was therefore possible to analyze other boundary layer properties such as shape factor, displacement thickness and boundary layer thickness.

Statistical analyses were made between the test data and the corresponding predictions by various fully turbulent flat plate skin friction prediction methods. An

improved method of predicting compressible turbulent skin friction drag was developed.

Boundary layer profile data measurements are also included along with a new method for predicting boundary layer growth characteristics. These include approximate velocity profile representation, boundary displacement thickness, and boundary layer thickness. The detailed results of this database development study are presented in Reference 2.

Skin Friction Calculation Methods

The current turbulent flow skin friction theories have been developed by assuming that compressible turbulent skin friction drag could be obtained using well known incompressible skin friction equations by evaluating all of the fluid properties that appear in the incompressible equations at some appropriate reference temperature, T*. This assumption parallels the analytical transformation methods used in laminar boundary compressible flow analyses. In the current study, the reference temperatures selected for evaluation included: the Monagham mean enthalpy equation and the Sommer / Short equation. Previous studies have shown both to provide accurate assessments of compressible skin friction. The Sommer / Short equation is currently used in Boeing Seattle HSCT PD methods.

Experimentally, it is much easier to obtain force measurements of local skin friction drag than measurements of average skin friction drag. Consequently, the initial step in the current evaluation process was to compare incompressible local skin friction data with the most generally accepted incompressible skin friction equations. Data from many different sources were used. The selected reference temperatures were then used to transform measured compressible local skin friction data to equivalent incompressible Cf and Reynolds numbers. Statistical analyses of the transformed compressible friction data were compared with the incompressible predictions, to assess the adequacy of the selected reference temperatures to account for the compressibility effects.

In Figure 3 comparisons are made between measured incompressible local skin friction data and the predictions obtained by a modified Schultz - Grunow equation which is simple representation of the most widely accepted in compressible local skin friction equation, the Karmen - Schoenherr equation. The test data appears to scatter about the theoretical predictions for the entire Reynolds number range of the test data

Statistical analysis of the differences between the test data and corresponding Cf predictions shows that the mean of the differences is Δ Cf = -.000000671 which corresponds to an average difference of 0.13% .The standard deviation of data about the mean is approximately 0.7 counts of drag (Δ Cf = 0.000067) which corresponds to 2.8% of the corresponding predicted value. Statistical analyses of the differences between the flat plate theory and the test data have been used to

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establish both the consistency of the test data and the adequacy of the theoretical predictions. This will allow more effective use of the data in CFD viscous drag prediction validation studies.

Figure 4 includes comparisons of the predicted effects of Mach number on the ratio of compressible skin friction to incompressible skin friction at the same Reynolds numbers. The experimental data are from thirteen independent experiments. The sources of the test data are given in Reference 1. The test data correspond to Reynolds numbers between 10⁶ and 10⁷. The theoretical predictions shown in the Figure were obtained using the Monaghan T* equation, and Sommer / Short T* equation. The predictions appear to match the Mach number trends quite well. The Sommer-Short T* equation results in compressible skin friction values consistently higher than those predicted using the Monaghan method.

Statistical analyses of the differences between Cf predictions and the corresponding test data are shown in Figure 5. The theoretical predictions were obtained using three different T* equations. The "scatter" in the test - theory increments are essentially equal. The mean of the differences between the test and theory, however differs between the predictions obtained using the different T* equations.

The "mean" of the theory - test differences obtained using the Monaghan T* equation is approximately 1% low. The "mean" of the theory - test differences obtained using the Sommer-Short T* equation is approximately 1% high. The constant for the Kulfan T* equation was therefor chosen to be the average of the Sommer-Short and the Monaghan constants. This essentially resulted in a mean error between the test data and the theoretical predictions of zero for the new prediction method.

The test data scatter about the mean has a standard deviation of about 4.5%. This large scatter is in part due to the variations of Reynolds number of the test data. The Reynolds numbers for the test data vary from 10⁶ to 10⁷.

Figure 6 contains comparisons of theoretical predictions of Cf with test data for three Mach numbers from 0.0 to approximately 3.0. The theory in this Figure used the Kulfan T* equation.

Figure 6 contains comparisons of theoretical predictions of Cf with test data for three Mach numbers from 0.0 to approximately 3.0. The theory in this Figure used the Kulfan T* equation.

In order to assess the accuracy of the Cf predictions to account for compressibility or Mach number effects, the test data were converted to equivalent incompressible values of Cfi and Reynolds number. This

transformation procedure, as shown in Figure 7, "collapses" all of the test data about the incompressible skin friction curve. The experimental data includes six different sets of test data obtained at Mach numbers from 1.7 to 2.95. This approach provides a convenient means to assess the accuracy of the theoretical methods to account for compressibility effects simultaneously over a range of Mach numbers and Reynolds numbers.

The Figure includes the statistically determined differences between the transformed equivalent incompressible skin friction and the modified Schultz-Grunow theoretical Cf predictions. The Kulfan T* equation was used for the transformation process. The "mean" of the differences between the transformed skin friction data and the incompressible Cf predictions is essentially zero. The " scatter" of the test has a standard deviation of about 1 drag count (DCf ~ 0.0001). This corresponds to about a 3.8% scatter of the test data about the theoretical Cf predictions over the entire Reynolds number range and Mach number conditions represented by the test data.

The table below shows the results obtained with different T* equations. On the average, the Monaghan predictions tend to underestimate the test data by about 0.3 counts or 1.2 %, and the Sommer-Short predictions are about 0.3 counts high or about 1.0%. The Kulfan T* method provides the best estimate of the compressibility effects.

	Monaghan T* Eqn.		Sommer-Short T* Eqn		Kulfan T* Eqn.	
	∆Cf (counts)	∆Cf /Cf (%)	∆Cf (counts)	∆Cf /Cf (%)	∆Cf (counts)	∆Cf /Cf (%)
Mean	301	-1.2	.308	1.00	.00071	.085
σ	1.022	3.8	.985	3.0	1.066	4.0

The "scatter" in the compressible theoretical - experimental transformed skin friction increments are only slightly higher than the scatter in the incompressible data shown in Figure 3. (0.7 counts versus 1 count).

Based on the results of the current study, it is recommended that the Kulfan method be adapted as the official HSCT flat plate skin friction calculation method.

Boundary Layer Growth

In the current HSCT studies estimates of the boundary layer height are used to specify the height of the boundary layer diverter to keep the inlet from ingesting portions of the boundary layer. During the course of the investigation described in Reference 1, experimental measurements of velocity profiles were found. It was also then possible to study the growth characteristics of a turbulent boundary layer over a flat plate. A method was developed to predict the growth of a turbulent boundary layer on a flat plate. This method has been revised in the current study.

Often in boundary layer studies, it is convenient to represent the velocity profile by a power law relation of the form:

$$\frac{u}{U_{\infty}} = \left(\frac{y}{d}\right)^{\frac{1}{N}}$$

This approximate form of the turbulent boundary velocity profile has been used to develop a process for predicting the boundary layer thickness. The boundary layer thickness is defined as the height at which the velocity is essentially equal to the freestream velocity.

Incompressible velocity profile data from a number of independent sources were used to determine "appropriate" values of N to represent a turbulent boundary layer. The results as shown in Figure 8, indicate that the value of "N" is strongly dependent on Reynolds number. The equation shown in the Figure was developed in the current study to represent the effect of Reynolds number on "N" as determined from the experimental data.

Experimental values of "N" were also determined from compressible boundary layer measurements for a number of Mach numbers from 1.5 to 4.2. The compressible values of "N" appeared to scatter about the empirical equation that was developed from the incompressible velocity profile data. Thus is appears that the shape of a turbulent depends on Reynolds number but is independent of Mach number.

Calculations of the variation of incompressible flat plate boundary layer thickness are compared with test data in Figure 9. The theoretical predictions obtained using this method presented in Reference 2 closely match the test data.

Compressible boundary layer thickness predictions are compared with test data in Figure 10 for Mach numbers of 1.7, 2.0 and 3.0. Although there is quite a bit of data scatter, the data appears to validate the boundary layer thickness predictions. The incompressible data from the Figure 9 when combined with the three sets of compressible data, appear to substantiate the conclusion that the thickness of a turbulent boundary layer is indeed relatively insensitive to Mach number.

3.2 Flat Plate Viscous Drag Predictions

Overview

It is felt that the first step in validating the viscous drag predictions of any Navier Stokes code is to make sure that predictions of the local and average fully turbulent flow skin friction drag and boundary layer growth must match the "simple" flat plate measured fully turbulent flow test data over the range of Mach numbers and Reynolds for which the codes will be used. This process will help to evaluate the applicability of the various turbulence models and grid techniques. The results of the Boeing Long Beach flat plate studies with CFL3D, and Boeing Seattle flat plate calculations with OVERFLOW are summarized below.

CFL3D Calculations

CFL3D was used to perform Navier-Stokes computations for subsonic, transonic, and supersonic flows over a flat plate with no pressure gradient. The solutions were obtained with turbulence models of different levels of complexity: the algebraic Baldwin-Lomax model, the one-equation Spalart-Allmaras model, and the two-equation shear-stress transport (SST) model by Menter.

Two sets of computations were performed. The first set was used to compare with the empirical skin-friction correlations by Sommer & Short, Kulfan, and Monaghan. The flow conditions ($M = 0.50, 1.50, 2.25, and 2.50 at Re = 1x10^{6}, 5x10^{6}, 10x10^{6}, 50x10^{6}, and 100x10^{6}$) were chosen such that they bracketed the transonic and supersonic cruise Mach numbers of an HSCT aircraft.

Figure 11 shows the variation of local and average skin friction with Reynolds number at $M_{=} 2.50$. For this Mach number, the local skin-friction values obtained with the Baldwin-Lomax and SST turbulence models show an acceptable agreement with the empirical data over the entire range of Reynolds numbers considered. The Spalart-Allmaras model compares reasonably well with the empirically-determined local skin friction at Re = 1×10^6 , but slightly overpredicts the empirical values for higher Reynolds numbers.

The second plot on Figure 11 shows that the Baldwin-Lomax results predict considerably higher average skin friction than any of the empirical methods for $\text{Re} < 10 \times 10^6$. This trend is opposite to that observed for the Spalart-Allmaras predictions. The average skin-friction values predicted by Menter's SST model are in good agreement with the empirical data over the full range of Reynolds numbers considered in this study. The results for the other Mach numbers indicate that, in general, the Baldwin-Lomax and SST models both agree reasonably well with the empirical average skin-friction values at M_{_} = 0.50, 1.50 for all Reynolds numbers considered. As the Mach number increases to M_{_} = 2.25 and 2.50, Menter's SST model shows the best correlation with the empirical data.

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The Spalart-Allmaras turbulence model, on the other hand, significantly underpredicts the empirical average skin-friction data for low Reynolds numbers ($Re < 5x10^6$) at all four Mach numbers. The agreement with empirical results improves for higher Reynolds numbers, but the Spalart-Allmaras computations tend to overpredict the empirical data as the Reynolds numbers increases. The Spalart-Allmaras turbulence model has a built-in transition model that simulates a laminar run with reduced local skin friction immediately downstream of the leading edge. The extent of this region of laminar flow decreases as the freestream Reynolds number increases. This explains why the correlation between the Spalart-Allmaras and the empirical results is poor at low Reynolds number.

The variation of local and average skin-friction values at $Re = 10 \times 10^6$ is shown in Figure 12. Here, it can be seen that for the low Mach numbers ($M_{\bullet} = 0.5$ and 1.5), the Spalart-Allmaras model agrees better with the empirical local skin-friction data than the other two turbulence models. As the Mach number increases, however, the SST model seems to be in better agreement with the empirical data. In terms of average skin friction, all three models display almost the same agreement with the empirical values (except at $M_{\bullet} = 0.5$, where the Baldwin-Lomax prediction is slightly lower than the other two Navier-Stokes results).

The second set of flat-plate computations were performed to compare the Navier-Stokes boundary-layer profiles with measurements by Smith and Walker for $M_{\infty} = 0.31$ at $Re = 6.78 \times 10^6$. Figure 13 shows that the agreement between the measured and the computed boundary-layer velocity profiles improves with the complexity of the turbulence model. That is, the algebraic Baldwin-Lomax turbulence model shows the worst correlation with experiment, and the two-equation SST model the best, with the one-equation Spalart-Allmaras model falling in between.

OVERFLOW Calculations

OVERFLOW fully turbulent flow flat plate calculations were obtained, with various turbulence models, at Mach numbers of 0.9 and 2.4 for both wind tunnel and full scale Reynolds numbers. Calculations were also made for Mach 0.2 at wind tunnel Reynolds number.

The models included the Baldwin-Lomax (BL) model (algebraic), the oneequation turbulence models of Baldwin-Barth (BB) and Spalart-Allmaras (SA), the two equation k-e and the k- ω turbulence models, and Menter's two equation SST (Shear Stress Transport) model.

The algebraic model of Baldwin-Barth expresses the turbulent viscosity via algebraic correlation's based on the universal law of the wall and the law to the wake. The one-equation models solve for the turbulence Reynolds "ret" which is

a measure of the ratio of turbulent viscosity to laminar viscosity. The two equation Menter SST model is a blend of the k- ϵ and k- ω models. The k- ω model is applied over the inner region of the of the boundary layer (up to roughly one half of the boundary layer thickness) and gradually changes to the k- ϵ model in the outer wake region.

The two equation model k- ω model solves for the turbulent kinetic energy k, and the specific dissipation rate of turbulence w with the turbulent viscosity given by $u_t = a^*pk/\omega$. $a^* = 1$ for the high Reynolds number model, and $a^* = f(ret)$ for the low Reynolds number model.

The turbulence kinetic energy is: $k = k'/{u_{\infty}}^2$, where k' is the turbulence kinetic energy per unit mass. The specific dissipation rate of turbulence energy is: $\omega = \omega' L/u_{\infty}$, where the ω' is the specific dissipation rate of turbulence energy, L is the characteristic length, and u_{∞} is the freestream velocity. The value of w is a measure of the average frequency of the turbulence. By definition, $\omega' = \epsilon'/k'$, where ϵ' is the dissipation rate of turbulence energy per unit mass. The length scale of turbulence is $\lambda = k^{3/2}/\omega$. The freestream kinetic energy k_{∞} is related to the freestream turbulence level or intensity Tu by the equation: Tu = $(2k_{\infty}/3)^{1/2}$. Internal to the code, the reference velocity is taken as the velocity of sound in the freestream may be interpreted as the average frequency of the freestream turbulence. In all the calculations, the turbulent viscosity μ_t is taken such that the $\mu t/\mu = Re_t = (k/\omega)Re < 0.1$, where Re_t is the turbulence Reynolds number, and μ is the laminar viscosity.

The original k-w model in OVERFLOW did not properly account for low Reynolds effects in the wall region. A low Reynolds number version of the k-w model was developed by Wilcox (1981). This k- ω low Reynolds model was implemented into OVERFLOW.

Results of the OVERFLOW calculations at full scale Reynolds number are shown in Figures 14 and 15. The Cf calculations are seen to be very dependent on the turbulence model. At Mach 0.9, the Cf calculations obtained with all of the various turbulence models over predict the Sommer-Short calculations which in turn are slightly higher than the mean of the flat plate skin friction test data. At Mach 2.4, The calculations using the Baldwin-Lomax turbulence model most closely match the Sommer-Short predictions and hence the test data.

The corresponding comparisons are shown in figures 16 and 17 for Mach 0.9 and 2.4 respectively for wind tunnel Reynolds numbers. At Mach = 0.9, the CFD predictions obtained with the various turbulence models seem to bracket the flat plate calculations . The dip in the calculated Cf distribution obtained using the Spalart-Allmarus turbulence model is similar to partly laminar flow with transition occurring just aft of the minimum Cf station. The Mach 2.4 calculations are

LIMITED EXCLUSIVE RIGHTS NOTICE These data are subject to limited exclusive rights under Government Contract NAS1-20220. shown in Figure 17. For this Mach number, The BL and SST models result in fully turbulent flow Cf distributions. The SA, BB and k- ω models have Cf distributions similar to partly laminar flow calculations but with different pseudo-transition locations.

Local skin friction distributions obtained with the high Reynolds and the low Reynolds version of the $k=\omega$ turbulence models are compared with the SA model, and flat plate calculations in Figures 18 and 19. The high Reynolds number K- ω calculations agree quite well with the fully turbulent flow flat plate skin friction calculations.

Sensitivity studies were also made to determine the variation of the k - ω pseudo transition Cf calculations to freestream specific dissipation rate and to freestream turbulence kinetic energy. These results are shown in Figures 20 through 23. The pseudo transition location is very dependent on the fore mentioned freestream turbulence parameter. By proper selection it might therefore be possible to calculate the viscous drag for specified transition locations. This could be very helpful for correcting wind tunnel data obtained with partly laminar flow on the model.

Symmetric TCA Model Definition and Test Program

Overview

The objective of the Symmetric Model test program is to acquire accurate transonic and supersonic data at high Reynolds Number in order to support validation of CFD viscous drag predictions. Accurate force and moment data will be obtained by testing the model primarily at zero degrees angle of attack which eliminates aeroelastic effects and drag-due-to-lift. Measurements with and without the outboard wing panel will also be made to remove the impact of potential partially laminar flow on the sharp supersonic outboard leading edge section.

The symmetric model geometry has been ϵ derived from TCA configuration definition. An isometric view of the model is shown in Figure 24. Figure 25 shows the differences between the TCA configuration and the proposed Symmetric Model. The model will have removable wing tips to permit testing without the outboard supersonic leading edge wing panel. The overall dimensions of a 1.675-percent model are shown in Figure 26. The actual model scale may change if an increased body diameter is necessary to achieve the desired test Reynolds number range. In addition to force balance measurements, some pressures will be measured on the wing in order that flow visualization techniques such as PSP can be used as enhance understanding surface flow characteristics. Preston tube measurements will be made to provide local

turbulent skin friction data at several locations on the model. The extent and location of this instrumentation are shown in Figure 27.

Testing is planned for the Boeing Supersonic Wind Tunnel and the NTF facility at Langley during 1999. Tests will be conducted at Mach numbers from 0.6 to 2. The planned Reynold's number range in NTF is 10 to 60 million/ft and 4 to 16 million/ft in the industry supersonic tunnels.

Symmetric TCA Viscous Drag Predictions

CFL3D ANALYSES

As the second stage of the viscous drag validation studies, flows with pressure gradient were investigated by analyzing the TCA symmetric wing/body model. First, the Baldwin-Lomax and the k- ω SST turbulence models were used to compute the flow over the TCA symmetric model at zero angle-of-attack. Figure 28 shows the variation of drag with freestream Mach number at a wind-tunnel Reynolds number of $Re_c = 6.36 \times 10^6$. The first plot on Figure 28 shows the viscous drag, predicted by four different methods. The CFL3D SST solutions predict the least amount of viscous drag throughout the Mach number range. The CFL3D Baldwin-Lomax results show higher drag (approximately two counts throughout the entire range of Mach numbers), but the shape of the curve follows the SST predictions. The equivalent flat-plate methods predict higher drag. At low Mach numbers, the flat-plate estimates as expected, agree very well with each other. However, as the Mach number increases, the flat-plate methods start to deviate from each other. For supersonic Mach numbers, the Sommer & Short predictions show close agreement with the Baldwin-Lomax solutions. This is in part due to the assumptions made in the Baldwin-Lomax formulation; namely, that the boundary-layer flow is self-similar. These assumptions are compatible with the equivalent flat-plate correlations, which apply to boundarylayer flows without pressure gradients. Also, it can be seen that both Navier-Stokes curves show a discontinuity in the vicinity of Mach 1.0 that the flat-plate results do not display. This discontinuity is probably due to the shock/boundary layer interaction that flat-plate theory does not capture.

The pressure drag for the TCA symmetric model at zero angle-of-attack is also shown in Figure 28 as a function of the Mach number. The two Navier-Stokes solutions predict very similar pressure drag. Also, at the subsonic Mach numbers, the Navier-Stokes computations clearly show the effect of the boundary-layer displacement by predicting higher pressure drag than the Euler solutions, as expected. At $M_{\infty} = 0.5$, for instance, the pressure drag increases from 0.4 counts (Euler) to 2.4 counts (Navier-Stokes). It is evident that even at subsonic speeds there is a pressure drag component that must be accounted for.

For high Mach numbers, the Euler results agree fairly well with the Navier-Stokes solutions.

At the flight Reynolds numbers (not shown) the Baldwin-Lomax turbulence model also predicts higher viscous drag than the SST model (about 1 to 1.5 counts higher throughout the Mach range considered). The equivalent flat-plate estimates fall in between the two Navier-Stokes predictions for M_{∞}^2 1.5. For higher Mach numbers, Sommer & Short calculations match the Baldwin-Lomax solutions, while the van Driest II method predicts the highest drag. The variation of pressure drag with Mach number at the flight Reynolds numbers (not shown) follows the same trend as for the wind-tunnel Reynolds number.

In addition to the results mentioned above, Navier-Stokes calculations have been performed for the TCA symmetric W/B model at $M_{\infty} = 2.4$ and $\text{Re}_c = 10 \times 10^6/\text{ft}$, for angles-of-attack ranging from 0 to 6 degrees. The solutions were obtained using several turbulence models available in CFL3D: the algebraic Baldwin-Lomax model (with and without the Degani-Schiff modification), the one-equation Spalart-Allmaras model, and the two-equation SST model by Menter. The agreement between the predicted lift (not shown) from all models is very good. However, there are significant differences in the drag predictions, shown in Figure 29. At zero lift, for instance, the predicted drag values differ by 5.5 counts. At $\alpha = 6^{\circ}$, the difference in predicted drag values is slightly less than five counts. The Baldwin-Lomax turbulence model with the Degani-Schiff modification predicts the lowest drag, while the Spalart-Allmaras model predicts the highest. There are also substantial differences in the pitching moment predictions (not shown).

Figure 30 shows the pressure distributions at six different span stations for $\alpha = 0^{\circ}$. For this angle-of-attack, the pressures from all turbulence models shown (Baldwin-Lomax, Spalart-Allmaras, and Menter's SST) agree quite well with each other. However, as the angle-of-attack is increased, the agreement is seen to deteriorate, especially on the upper surface of the wing between 30 and 70% semi-span (not shown). In spite of these differences in pressures, the total C_L and the pressure component of drag are very similar for all turbulence models.

Surface pressure distributions and streamlines from the Baldwin-Lomax solutions at a = 0, 2, 4, and 6 degrees are shown in Figure 31. At $\alpha = 0^{\circ}$, the flow is attached and well behaved; the streamlines are mostly chordwise. However, at the other angles-of-attack, a leading-edge vortex is present, and there is significant spanwise flow on the upper surface of the inboard wing. The flow on the wing lower surface (not shown here) is attached at all four angles-of-attack considered.

OVERFLOW Analyses

OVERFLOW Navier-Stokes solutions were obtained to determine the viscous drag for the symmetric TCA wing/body configuration (angle of attack alpha=0 deg) at wind tunnel Reynolds number (Mach 0.7 through 2.4) and at flight Reynolds number (Mach 0.9 through 2.4). Detailed local and chordwise averaged skin friction coefficients were computed at two selected Mach numbers (M=0.9 and 2.4). The predicted viscous drag, local and averaged skin friction coefficients were compared with the flat plate theories of Sommer-Short and Frankl-Voishel. Skin friction coefficients were evaluated for the baseline TCA configuration at M=0.9 and M=2.4 at both flight and wind tunnel Reynolds number and corresponding to cruise angles of attack.

OVERFLOW solutions with the Spalart-Allmaras (SA) one-equation turbulence model have been obtained for the symmetric TCA skin-friction drag at flight Reynolds number for freestream Mach numbers of 0.9,0.95,1.1,1.2,1.8, and 2.4. In order to understand the sensitivity of the viscous drag with respect to the turbulence model considered, some additional solutions have been obtained with the Baldwin-Barth (BB) one-equation turbulence model at two selective Mach numbers (M=0.9 and 2.4).

Figure 32 shows a comparison of the skin-friction drag from OVERFLOW and the flat plate theory (Sommer-Short) at flight Reynolds number. At flight conditions, the OVERFLOW solution with the SA model agrees well with the flat plate theory over the entire Mach number range (except at M=0.95). The difference in the component viscous drag from the two methods is within one count in the entire Mach number range, except at M=0.95 where the departure is about two counts. In general except at M=0.95, the body viscous drag from OVERFLOW is about one count higher than that from the flat plate theory, and the wing viscous drag one count less than that from the flat plate theory. The total viscous drag for the wing-body combination from the two methods is also within one count except at M=0.95. The total viscous drag from the BB model is seen to match that from the SA model at M=2.4, but is considerably below that from the SA model (by 4 cts) and the flat plate theory (by 3 cts) at M=0.9. The difference at M=0.9 is primarily due to departures in the predicted viscous drag for the wing

Symmetric TCA viscous flow solutions from OVERFLOW with Spalart-Allmaras (SA) one-equation turbulence model at the wind tunnel Reynolds number of 6E6 have been obtained. These solutions correspond to various Mach numbers in the range of M=0.7 through 2.4 (M=0.7, 0.9, 0.95, 1.07, 1.2, 1.5, 1.8, 2.1 and 2.4), and complement those at flight Reynolds number. In order to understand the sensitivity of the viscous drag with respect to the turbulence model considered, some additional solutions have been obtained with the Baldwin-Barth ~B) one-equation turbulence model at two selective Mach numbers ~=0.9 and 2.4).

Figure 33 shows a comparison of the OVERFLOW solutions and the flat plate theory at the wind tunnel Reynolds number. The total wing/body viscous drag from the OVERFLOW code with the SA model is about three to four counts below the flat plate theory over the entire Mach number range studied (M=0.7 to 2.4). The solution with the BB model for the total viscous drag is seen to be close to that with the SA model at M=2.4, but at M=0.9 is about four counts below that from the SA model, and six counts below the flat plate theory. The component comparisons suggest that the SA model yields wing viscous drag that is about four to five counts below the flat plate theory, and the body viscous drag about 1.5 counts above that from the flat plate theory. On the other hand, the BB model predicts total viscous drag close to the SA model at M=2.4, but at M=0.9 are primarily due to departure in the predicted body viscous drag.

Figure 34 displays the variation of skin-friction drag CDV, pressure drag CDP and the total drag CD predicted from OVERFLOW with the SA model as a function of freestream Mach number. The zero-lift pressure drag from the SA model shows a peak value in the vicinity of M=1 .1. The zero-lift pressure drag computed from OVERFLOW with the SA model increases from about one count at M=0.9 to about 19 counts at M=2.4. A comparison of the predicted zero-lift pressure drag from the two turbulence models shows that the zero-lift pressure drag obtained from the two models is close to each other at both M=0.9 and M=2.4 (less than half a count).

From the comparison of the pressure drag from the two turbulence models in Figure 35, it is evident that the zero-lift pressure drag from the two models is less than half a count at both M=0.7 and 2.4, just as in the case of flight Reynolds number. The zero-lift pressure drag computed by the SA model shows a peak in the vicinity of M=1.07.

The observed discrepancy between the flat plate theory and the OVERFLOW viscous solution (SA model) at the wind tunnel conditions merits further exploration. Also the relative accuracy of the flat plate theories at the wind tunnel conditions (low Reynolds number) needs further examination. Referring to the turbulence models, there is some speculation that at low Reynolds number the Spalart-Allmaras turbulence model may predict relaminarization of an initially turbulent boundary layer. Reversion to laminar flow is known to occur when a turbulent boundary layer is highly accelerated under a strong favorable pressure gradient. A more detailed study of the local skin-friction coefficient distribution on the wing surface is planned to provide further understanding in this area. The comparisons of the two turbulence models with the flat plate theory show that the Baldwin-Barth model significantly underpredicts the skin-friction at transonic conditions at both flight and wind tunnel conditions

Remaining Tasks

The remaining tasks that remain to be done in the plan to resolve the differences in the viscous drag predictions using various CFD codes, different turbulence models include:

- Conduct the symmetric model wind tunnel test programs
- Evaluate symmetric model test versus theory results, resolve differences, select most appropriate turbulence model(s)

Apply the selected CFD codes/turbulence model(s) to the prediction of the TCA drag at selected Mach numbers, Reynolds Numbers, and angles of attack

References

- 1. Kulfan, Robert M., "Turbulent Boundary Layer Flow Past a Smooth Adiabatic Flat Plate", Boeing Document D6-7161, May 1961
- Kulfan, Robert M. "Historic Background on Flat Plate Turbulent Flow Skin Friction and Boundary Layer Growth", HSR Airframe Technical Review, Feb 1998

Comparison of Predicted Viscous Drags From Various CFD Codes for the TCA Wing/Body Configuration

 $M_{\infty} = 2.4$, $C_{L} = 0.1$, $R_{e} = 6.36 \times 10^{6}$





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Comparison of Predicted Viscous Drags From Various CFD Codes for the TCA Wing/Body/Nacelle/Diverter Configuration

 $M_{\infty} = 2.4$, $C_{L} = 0.1$, $R_{e} = 6.36 \times 10^{6}$



Figure 2

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Evaluation of Reference Temperature Equations

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Conversion of Compressible Cf Data to Equivalent Incompressible (Cfi)_{eq} Data



Kulfan T* Method

Modified Schultz-Grunow Cf Equation

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Conversion of Compressible Cf Data to Equivalent Incompressible (Cfi)_{eq} Data

- Kulfan T* Method
- Modified Shultz-Grunow Cf Equation





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Incompressible Velocity Profile "N" Factor u/Uo = (y/δ)^{1/N}



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Compressibility Effects on Boundary Layer Thickness

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Skin Friction for Flat Plate with No Pressure Gradient

High Speed Aerodynamics, Long Beach

CFL3D, Navier-Stokes, 129 x 193 Fine Grid, Re = 10 x 10⁶



Figure 12

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Flat Plate (M=0.9, Re=164E6, Tinf=390 R)

Comparison of the local skin friction coefficient for a flat plate at M=0.9 and Re=164E6 from OVERFLOW with various turbulence models, and flat plate correlations.

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Flat Plate (M=2.4, Re=197E6, Tinf=390 R)



Comparison of the local skin friction coefficient for a flat plate at M=2.4 and Re=197E6 from OVERFLOW with various turbulence models, and flat plate correlations.



Flat Plate (M=0.9, Re=6.04E6, Tinf=511.8 R)

Comparison of the local skin friction coefficient for a flat plate at M=0.9 and Re=6.04E6 from OVERFLOW with various turbulence models, and flat plate correlations.

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Flat Plate (M=2.4, Re=6.362E6, Tinf=283.5 R)

Comparison of the local skin friction coefficient for a flat plate at M=2.4 and Re=6.362E6 from OVERFLOW with various turbulence models, and flat plate correlations.

4

cf





Comparison of the local skin friction coefficient for a flat plate at M=0.9 and Re= 6.04×10^6 with high and low Reynolds number k- ω models, and the Spalart-Allmaras model.

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Flat Plate (M=2.4, Re=6.362E6, Tinf=283.5 R)



Comparison of the local skin friction coefficient for a flat plate at M=2.4 and Re= 6.36×10^6 with high and low Reynolds number k- ω models, and the Spalart-Allmaras model.

Flat Plate (M=0.9, Re=1.E7, Tinf=511.8 R)



Dependence of the transition location on freestream specific dissipation rate ω_{∞} for a flat plate at M=0.9 and Re=10⁷ with the low Reynolds number k- ω model.

Flat Plate (M=0.9, Re=1.E7, Tinf=511.8 R)





Flat Plate (M=2.4, Re=1.0E7, Tinf=283.5 R)



Dependence of the transition location on freestream specific dissipation rate ω_{∞} for a flat plate at M=2.4 and Re=10⁷ with the low Reynolds number k- ω model.

Figure 22

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Flat Plate (M=2.4, Re=1.0E7, Tinf=283.5 R)







Isometric sketch of Symmetric model.

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Approximate size of 1.675% Symmetric model. (All dimensions in inches.)

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Locations of various types instrumentation. (All dimensions in inches. Not to scale)igure 27

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Comparison of Skin Friction Drag at Flight Reynolds Number

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Comparison of Skin Friction Drag at Wind Tunnel Reynolds Number

Figure 33

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Drag Predictions from OVERFLOW at Flight Reynolds Number

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Drag Predictions from OVERFLOW at Wind Tunnel Reynolds Number

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