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Comparison of data correction methods for blockage effects in semi-span wing model testing

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Abstract. Wing alone models are usually tested in wind tunnels for aerospace applications like aircraft and hybrid buoyant aircraft. Raw data obtained from such testing is subject to different corrections such as wall interference, blockage, offset in angle of attack, dynamic pressure and free stream velocity etc. Since the flow is constrained by wind tunnel walls, therefore special emphasis is required to deliberate the limitation of correction methods for blockage correction. In the present research work, different aspects of existing correction methods are explored with the help of an example of a straight semi-span wing. Based on the results of analytical relationships of standard methods, it was found that although multiple variables are involved in the standard methods for the estimation of blockage, they are based on linearized flow theory such as source sink method and potential flow assumption etc, which have intrinsic limitations. Based on the computed and estimated experimental results, it is recommended to obtain the corrections by adding the difference in results of solid walls and far-field condition in the wind tunnel data. Computational Fluid Dynamics technique is found to be useful to determine the correction factors for a wing installed at zero spacer height/gap, with and without the tunnel wall.

1 Introduction

One of the major problems associated with the wind tunnel test for a wing is the blockage effects produced by the wind tunnel walls [1]. The presence of tunnel-induced lift increments, as well as the increment in drag count is due to wall bounded conditions [2]. Reliable aerodynamic and static stability coefficients of an aircraft's wing are usually obtained from wind tunnel testing, which needs to be corrected for such affects. The effect of the sidewalls of the tunnel is only applicable to 3-D models where trailing vortices exist. In such cases, the tunnel walls induce a span-wise variation in up-wash, which decreases the normal downwash [3]. Thus, wind tunnel testing on a wing or lifting body will have too little downwash, as the wing appears to have a larger effective aspect ratio than it would have if tested in free air. These effects are small if the span is much less than the tunnel width, but large span models produce serious distortions in lift distribution which also affect stalling characteristics [4]. To alleviate such interference, a correction scheme to cater the blockage effects is always required. But deficiencies in such correction procedure are likely to be most severe when the model is mounted on the tunnel's floor and the interference flow created by such an arrangement has significant non-uniformity in the region of the model [5].

Till now, significant work has been done, to have a better understanding about the blockage corrections due to wall effects applied to the raw data of wind tunnel [5]. Such research efforts are mostly related to the development of analytical relationships for data correction method, especially for blockage correction equations [6], but the validation of the same through experiments is quite limited for half model testing of wing alone models. Furthermore, to the authors' best knowledge; very little effort has been made to correlate the raw data of wing alone half model testing with the numerical simulation techniques in the subsonic regime.

It is well known that wind-tunnel-induced flow distortion around a test model causes drag, lift and pitching moment increments that are absent in free air [6]. Necessary correction must be made before using this data for engineering design. In the present work, results of two data correction schemes were compared with those obtained through CFD. For the proof of concept, qualitative as well as quantitative comparison is done for straight level flight condition.

All the empirical relationships and relative curves used for the estimation of horizontal buoyancy, wake blockage, total blockage and corrections to aerodynamic and static longitudinal stability parameters are taken from Pope and Harper [7] and Herriot [8]. The experimental testing is done on a generic model of a straight semi-span

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wing made up of NACA-65(3)-218 wing and the experimental results are utilized to compare the standard methods of blockage correction. Flow physics related to the creation of horse shoe vortex due to the unavoidable interaction of the tunnel's floor with the base of wing's model is quite matured. Therefore, it has not been discussed here as it a well-known phenomenon in past [9] as well as in recent studies [10].

2 Data Correction Methods

It has been mentioned earlier that the conditions under which a wing's model is tested in a wind tunnel are not the same as those in free air. The walls of a closed test section produces changes in flow patterns, which in turn changes the flow physics. To check this, blockage corrections is one of the prospective options for correcting the raw data obtained in a wind tunnel.

2.1 Blockage

The purpose of the blockage corrections is to yield estimates for the aerodynamic forces and moments experienced by the actual test model in free stream, rather than for the original model [5]. Such a correction falls into the following two categories:

2.1.1 Solid Blockage

The presence of a model in the test section reduces the flow area and thereby increases the velocity of the air through the test section. The increase in velocity due to solid blockage at the scaled down model is usually quite small than that calculated from a direct area reduction since the streamlines near the tunnel walls are displaced more than those near the model [7]. Solid blocking causes an increase in dynamic pressure, which in turn increases all forces and moments [7,11].

2.1.2 Wake Blockage

Any scaled down model tested in a wind tunnel will always have a wake behind it which is basically due to the detachment of the boundary layer from the model. This wake has a mean velocity lower than that of the far field condition, which perhaps keeps the mass flow rate constant from the test section entrance to exit. This results in a lower wake pressure. This effect increases with wake size, hence is highest when large flow separation occurs. The wake blocking also creates an increase of dynamic pressure on the model [7].

2.2 Calculations of Blockage Effects

Available analytical relationships are briefly reviewed are to get first-hand knowledge about the blockage correction techniques for a closed circuit wind tunnel. These relationships are applicable only for the condition when the size of the scaled down model is smaller than the test section size. All such relationships are grouped into two categories, a brief detail of which is given below. Formulas for the first group are taken for incompressible flow [7] and the second group with consideration of the effect of compressibility [8]. Existing *ESDU* standard for half model testing is perhaps based on the potential flow method and some deficiencies in this method have recently been diagnosed by Alvaro [12] . This includes an offset between the angle of attack for wall corrections and the slope of the angle-of-attack correction versus lift coefficient curve [12]. Hence this method has not been discussed and used for the comparison study.

2.2.1 Conventional Method of Correction

Blockage is the sum of wake blockage and horizontal buoyancy. In this way; total correction factor ϵ can be written [7] as follows, Eq. (1):

$$\varepsilon = \varepsilon_{sb} + \varepsilon_{wb} \quad (1)$$

Where, ε_{sb} is the wake blockage and ε_{wb} is the wake blockage. The analytical relationship [7] for solid blockage ε_{sb} is shown below:

$$\varepsilon_{sb} = \frac{k_1 \tau_1 \omega_v}{C^{3/2}} \quad (2)$$

Almost all wind tunnels with closed test sections have a variation in static pressure, even with no model attached with it. This is caused by the thickening of the wall boundary layer, which in turn causes a decrease in the effective test section flow area [7]. This effect can be reduced by having the cross-section area of the tunnel increase towards the test section exit. However, with fixed walls, it is only possible to achieve a uniform axial test section pressure at one flow rate. For all other flows, the test section pressure normally decreases as the end of the test section is approached. This pressure variation over the model produces an additional drag force. Such an effect on a wing's scaled down model can analytically be estimated by using Eq. (3) [7]:

$$\varepsilon_{wb} = \frac{S_{ref}}{4C} C_{DU} \quad (3)$$

Where, C_{D_U} is uncorrected drag coefficient, S_{ref} is reference area of wing and term C is the tunnel's cross sectional area.

The corrected i.e. true angle of attack due to blockage is expressed in Eq. (4) [7]. Similar to the case of a pitot-static tube of an aircraft, we have difference in indicated and true air speed. The same is true for the angle of attack for the wind tunnel testing.

$$\alpha = \alpha_U + \left[\delta \frac{S}{C} C_L (57.3)\right] (1 + \tau_2) \quad (4)$$

In the above equation, terms δ and τ_2 are shape factor of a tunnel and wing respectively. The change in angle of attack causes the lift to increase, the corrected lift is estimated by using Eq. (5)-(8) [7]:

$$\Delta\alpha = \Delta\alpha_1 + \tau_2\Delta\alpha_1 \quad (5)$$

$$\Delta\alpha_1 = \frac{6\sigma}{\pi^2} C_1 \quad (6)$$

$$\sigma = \frac{\pi^2}{48}\left(\frac{\bar{C}}{\sqrt{C}}\right)^2 \quad (7)$$

$$C_L = C_{LU}(1-2\varepsilon) - \tau_2\Delta\alpha.a \quad (8)$$

Similarly, the change in coefficient of drag and the corrected drag coefficient is estimated by using Eq. (9)-(10) [7]:

$$\Delta C_{DW} = \frac{k_1\tau_1\omega_v}{C^{3/2}} c_{DU} \quad (9)$$

$$C_D = C_{DU}(1-2\varepsilon) - \Delta C_{DW} \quad (10)$$

Pitching moment is usually referred at the quarter chord of the wing. This is actually the location of the aerodynamic centre of the wing which may change or may not change in the wing-fuselage case. This is because of the fact that, all analytical methods except the Diehl method state that their results are the location of the wing fuselage aerodynamic centre ahead of the original wing aerodynamic centre, which does not necessary have to be 0.25c, [13]. Since the case under discussion is only of the wing, therefore the coefficient of moment is taken at the quarter chord of the wing, Eq. (11):

$$C_{M1/4} = C_{M1/4U}(1-2\varepsilon) + 0.25\tau_2\Delta\alpha.a \quad (11)$$

It is important to note that all uncorrected coefficients are defined in terms of an uncorrected dynamic pressure q_u; which is the q that would exist if no model was present in the test section [7].

$$C_{L_U} = \frac{Lift}{q_u S} \quad (12)$$

$$C_{D_U} = \frac{Drag - \Delta D_B}{q_u S} \quad (13)$$

$$C_{M_U} = \frac{Moment}{q_u Sc} \quad (14)$$

Where S, c and ΔD_B are reference area of the wing model, its characteristic length and change in drag due to horizontal buoyancy drag respectively.

2.2.2. Correction with constraint of Compressibility and Test Section of Rectangular shape

Herriot's work [8] is based on the estimation of blockage correction for closed-test-section wind tunnels of different cross-sectional shapes. He has also provided the experimental data for blockage estimation of different shapes of wings with tabulation of the appropriate constants. In order to account for the effect of the compressibility of flow, a correction for Mach number 'M' and for the free stream velocity 'U' is suggested in his work:

$$M = M'\left\{1 + \left[1 + 0.2(M')^2\right]K\right\} \quad (15)$$

$$U = U'(1+K) \quad (16)$$

And the corrected dynamic pressure 'q' can be obtained by using Eq. (17), [8]:

$$q = q'\left\{1 + \left[1 + 0.2(M')^2\right] K\right\} \quad (17)$$

$$K = K_\omega + K_{\omega k} \quad (18)$$

In Eq. (18), K_ω and $K_{\omega k}$ have the same meaning as that of ε_{sb} and ε_{wb} as described in Eq. (2) and (3), respectively.

$$K_\omega = \frac{1}{\left[1-(M')^2\right]^{3/2}} \frac{K_1.\tau.\nabla_\omega}{C^{3/2}} \quad (19)$$

Where, value of K_1 is estimated by Eq. (20):

$$K_1 = \frac{\pi^{3/2}}{16} \frac{\Delta}{t/c} \frac{1}{\kappa_1} \quad (20)$$

Here t/c is the thickness to chord ratio of the wing and $K_{\omega k}$ is the wake blockage correction factor, which is given by:

$$K_{\omega k} = \frac{\left[1+0.4(M')^2\right]}{\left[1-(M')^2\right]} \times \frac{C_D' S}{4C} \quad (21)$$

The corrected coefficient of drag is:

$$C_D = C_D'\left\{1 - \left[2-(M')^2\right] K\right\} - \Delta C_{D_W}' \quad (22)$$

Herriot [14] also provided the analytical relationship for an additive wake-induced incremental drag adjustment, shown below in Eq. (23):

$$\Delta C_{D_W}' = \frac{\left[1+0.4(M')^2\right]}{\left[1-(M')^2\right]^{3/2}} \times \frac{K_1.\tau.C_D'.V_\omega}{C^{3/2}} \quad (23)$$

It is important to note that C_{D_U} given in Eq. (13) and C_D' of Eq. (23) have the same meanings. The analytical relationships described in sub section 2.2.1 and 2.2.2 are used to correct the data for the test on the semi-span wing conducted at a free stream velocity equal to 50 m/s. The corrected results are discussed in section 4.2.

3 CFD vis-`a-vis Wind Tunnel Testing

To allow reprocessing of the available wind-tunnel data, with available data correction schemes and comparison of results with CFD, a methodology used earlier for transonic flow [14] is adopted and applied to the available testing data of wing model [15] at a subsonic speed.

3.1 Methodology

The incompatible requirements in scaling down the model, complex interaction of the flow from the walls and its interference with the model raise a question on the reliability of the existing data correction schemes [7]- [8] for half model testing of straight wings. Under such an unpredictable scenario; confidence in the results can be achieved by numerically simulating the flow with and without the walls. In this way, flow field information can easily be obtained. It is well-known that the wall mounted model faces additional interference due to the boundary layer of the tunnel's floor but the assessment of such interference due to the complicated boundary condition can only be achieved by testing the model at different spanner heights/gaps. Moreover, there is no universal school of thought for defining any fixed spanner height/gap. The effect of the walls on the model can be obtained by performing a numerical simulation of the test with no physical gap between the wing and tunnel's floor. Results so obtained can further be compared with those obtained by applying the farfield boundary condition at the outer computational domain Complete details of both the simulations have to be described in section 4.1. The difference so obtained is perhaps a representation of flow interaction due to the wall and can be added to the wind tunnel results to get the data for the free stream condition. This methodology taken from reference [14] is shown in figure 1 for quick reference. Numerical wind tunnel testing for the specified size of the test section has already been in practice in transonic regime for ventilated [16] as well as for solid wall [17] and has been applied earlier for different aerospace applications [14].

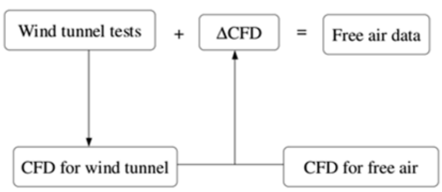


Figure 1. Role of CFD for data correction [14]

In order to validate this numerical approach for subsonic flow, CFD cases are run and the difference so obtained is used later to correct the wind tunnel measurements. These results alongwith those obtained by using the existing analytical formulas are discussed in the following section:

4 Results and Discussion

This section details the analytical and computational calculations used to correct the raw data for a wing tested in a closed test section of *IIUM-LSWT* [15]. Complete details of the experimental setup and flow conditions have been provided in reference [15]. Due to the weight and attachment constraints, no additional gap is considered between the model and the floor of the tunnel. Moreover, no data interpolation technique was applied on the raw data of the wind tunnel which is only corrected for tare and balance affects.

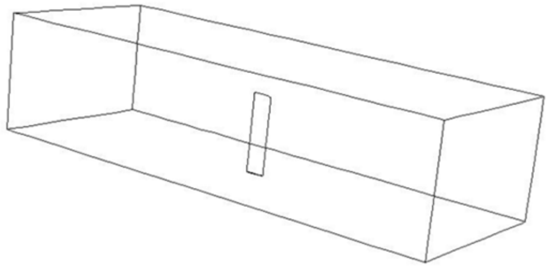
Figure 2. Model of wing tested in IIUM-LSWT [15]

4.1 Numerical Wind Tunnel - with and without wall

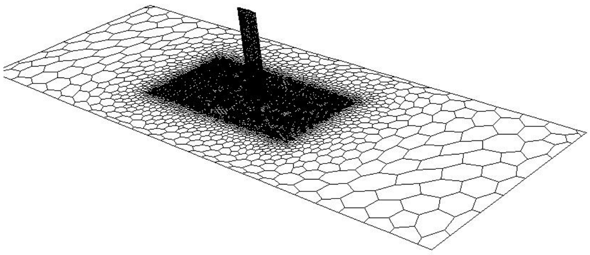
Fluent®, a commercial *CFD* software is used to solve the time averaged Navier-Stokes equations with SST turbulence model. As the scope of the work is limited to find the correction factor by using CFD. Hence, less effort has been put for correlating the numerical results with the raw data of the wind tunnel. Two different cases alongwith the grid generations for CFD analysis are described below to find the correction factor, described earlier in section 3.1.

4.1.1 Numerical Wind Tunnel

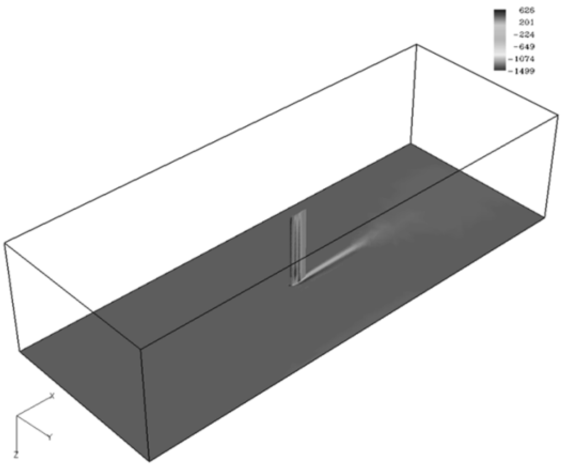
The geometry of the wing alongwith the geometrical details of the test section is modeled using CATIA ® CAD software, figure 3(a). Few necessary simplifications in the model are made for the ease of grid generation. As the flow over the wing is analyzed in the clean configuration therefore it is modeled without modeling the nuts and bolts. These few small external components which are ignored, mainly add to the drag in small amount but their contribution towards the lift and the moments is negligible. Two cases with different grids are used so that flow should be aligned with the grid. In the first case the walls of the tunnel are defined as per the size of the test section and the grid is made keeping in view the flow physics so extensive grid cells are distributed on and near the tunnel's floor where the wing is physically attached with it as shown in figure 3(b). Moreover, velocity inlet condition is defined at the inlet and outflow condition is used at the outlet. The presence of a low pressure zone due to flow separation at the junction of tunnel's floor and the root chord of the wing is obvious from figure 3(c). The quantitative results so obtained alongwith the experimental results for level flight condition are discussed at the end of sub section 4.2.2.



(a) Computational domain for solid wall test



(b) Grid clustering near the junction of wing



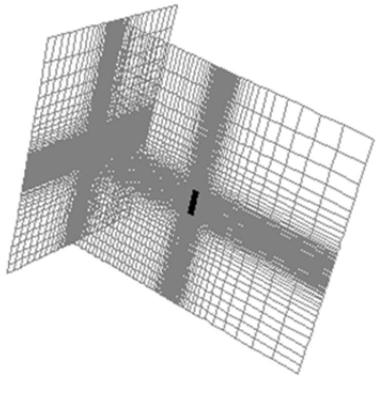
(c) Pressure contour at the tunnel's floor

Figure 3. Results of numerical wind tunnel

4.1.2 Numerical Case of Farfield Condition

For the second case of numerical simulation, pressure farfield boundary condition is used and no-slip wall condition is applied at the wall of the model. The outer domain of farfield boundary is about 30 times the span of the model. This is because of the fact that although the effect of compressibility are quite small for M $\leq$ 0.3. But in nature, the flow is compressible and it is important to run the numerical simulation for the said condition. Free stream Mach number corresponding to the free stream velocity of 50 m/s is taken equal to 0.145. This value of Mach number along with the appropriate value of atmospheric pressure is prescribed in the boundary condition panel. Density is calculated by using the ideal gas law and Sutherland's law is employed to calculate the viscosity in the numerical simulations. For this case, the domain extents are kept at twenty times the length of the body in order to avoid the flow variations near the wing from affecting the boundary conditions. The numbers of grid cells are 0.4 million, out of which 40 % of the cells are on the tunnel's floor and in the vicinity of the test article. Grid is kept coarse in the farfield domain as well as for the sidewalls of the tunnel. The grid clustering can be seen in the sectional plots of grid at the center plane as well as at the inlet, shown in figure 4(a).

Since the temperature problem is not of interest, therefore adiabatic wall conditions with no slip boundary condition is employed on the wing surface. Contours of the pressure distribution on the surface of the wing at α equal to zero are shown in figure 4(b). Grid independent studies are also carried out to obtain grid independent numerical solutions. Values of the estimated aerodynamic coefficients for V=50 m/s are tabulated in Tab. 1.



(a) Sectional planes of computational grid

(b) Contours of total pressure (Pascal) on the model

Figure 4. Grid planes and pressure distribution on the wing for the case of unbounded wall condition

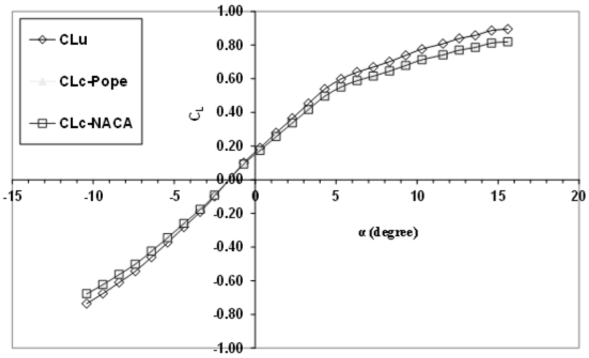
Table 1. Comparison of CFD results for two different cases at $\alpha = 0^0$ and at V=50 m/s

	Farfield	*Solid wall*	*Δ CFD*
C_L	0.175	0.183	0.008
C_D	0.0209	0.032	0.01
C_m	0.0019	0.0026	0.007

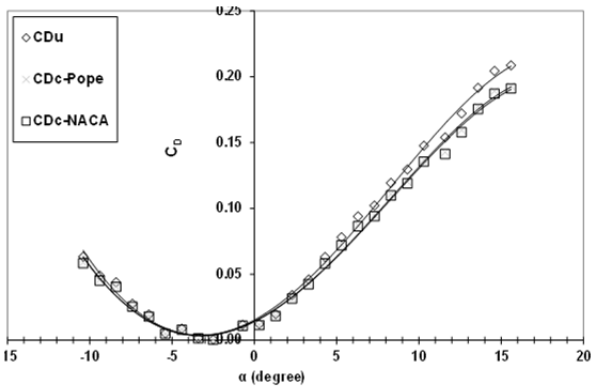
From above mentioned table, the walls' influence on the aerodynamic results in numerical wind tunnel is considerable. This is because of the fact that when the flow is constrained by the wall than it gives blockage effects to the model placed inside the test section. Due to which aerodynamic coefficients increases, when compared to a a farfield condition. For the case under discussion, this increase in C_L, C_D and C_m is 0.008, 0.01 and 0.007 respectively. Although the presented results are only for $\alpha = 0^0$, but it motivates to carry out a parametric study in future, to determine the sensitivity of the numerical results at higher angle of attack as well. It is perceived that the interference flow field will be quite non- uniform due to the massive flow separation at the floor-model junction and *CFD* will provide the interference factor for data correction.

4.2 Comparison of Results

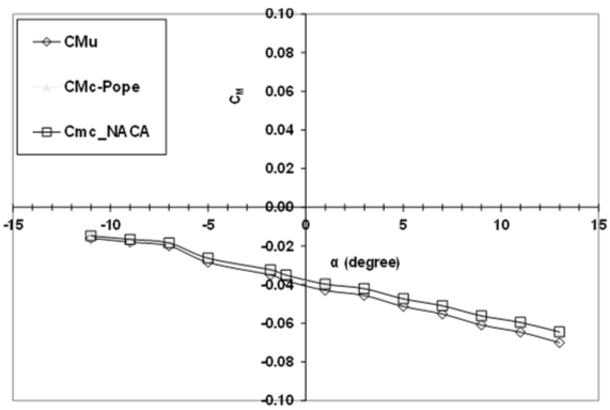
Analytical relationships for blockage correction are applied ow data of the wind tunnel, figure 5. In this figure the corrected coefficients are nomenclatured as per references [7] and [8], which are overlapping each other due to no significant difference in the corresponding values for the defined α sweep. It can be observed from this figure that the difference between the corrected values and the raw data of the wind tunnel increases with increases in the angle of attack (positive as well as negative). For zero degree angle of attack, the impact of data correction scheme is quite less i.e. the values of ΔC_L, ΔC_D and ΔC_M are 0.0016, 0.001 and 0.0001 respectively. Furthermore, both the blockage correction schemes predicted the same results. This is perhaps due to small difference in the estimated values of k_1 and τ_1, which is 0.01 and 0.033 respectively. Moreover, wind tunnel over predicted the stall angle and hence the coefficient of maximum lift, figure 5(a). But there is negligible difference in the value of coefficient of drag at zero lift, figure 5(a). It can be observed from the qualitative comparison of plots of figure 5 that after $\alpha > 6^0$, the effect of data correction method is considerable. Moreover, a comparison of these results with those highlighted in Tab.1 for at $\alpha = 0^0$ condition revealed that the estimated values of ΔC_L, ΔC_D and ΔC_M is more than that predicted by the analytical relationships. *CFD* results for the said condition are not overlapped on figure 5 as the difference in the values will not be obvious due to other larger values at high α.



(a) C_L vs α



(b) C_D vs α



(c) C_m vs α

Figure 5. Comparison of raw and corrected data

at *V=50 m/s*

5 Conclusion

Preliminary *CFD* results obtained from solid-walls and free stream tests showed a difference of values at zero degree angle of attack. Though this difference is quite small but due to strong interaction of the flow with the wall, it will increase with increase in angle of attack. Such an inconsistent offset can be observed between the raw data of wind tunnel and those corrected by standard methods. Compressibility effects on the data correction method are found to be negligible for the defined flow conditions. Moreover, both the analytical schemes for blockage correction decrease the values of aerodynamic coefficient for positive angle of attack and vice versa.

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