wings in the Trefftz plane do not intersect or overlap with each other. Although there does not appear to be any significant computational advantage to the present method when compared to that of a constrained-minimization technique,[4] the present method provides a simple closed-form expression for the optimum downwash and, thus, provides additional insight into the aerodynamics of ideal formation flight.

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Validation of a Wall Interference Correction Procedure in Subsonic Flow

G. Lombardi* and M. V. Salvetti†
University of Pisa, 56123 Pisa, Italy
and
M. Morelli‡
CSIR, Pretoria 0001, South Africa

Introduction

ONE of the main sources of error affecting the experimental measurements of the flowfield around a model is the interference effects of wind-tunnel walls. Classical correction criteria are based on theoretical linear models, and thus, their validity is limited. More recently, new correction methods were introduced. (For a general description, see Ref. 1.) These methods are based on more complex procedures that couple measurements, typically pressure and/or velocity on the wall or in the field, with numerical calculations.

A method of correction for the wall interference effects was developed, based on pressure measurements on the wind-tunnel walls coupled with a numerical procedure to evaluate the flow correction, which is described in detail in Refs. 2 and 3.

A preliminary application of the setup methodology to the correction of the aerodynamic coefficients of a complete aircraft model in subsonic conditions was described in Ref. 3. The results were compared with those obtained with a pretest correction method, and a satisfactory agreement was obtained. Clearly, this cannot be considered as a definitive validation of the correction procedure.

In the present paper, the methodology is applied to real experimental data, therefore, as a posttest procedure. Experiments on a complete aircraft configuration have been carried out using two different model sizes. The use of different scale models, operating in a given wind tunnel under identical flow conditions, appears to be the most appropriate procedure to gain information on the validity of the proposed correction procedure. Indeed, this approach gets rid of all differences related to the free-stream flow conditions, and the uncertainty in measurement comparisons is considerably reduced, being limited to the random component (which can be reduced, theoretically, to any desired value) of the measurement procedure.

Subsonic and low-angle-of-attack conditions have been considered. These can be considered the most critical conditions for the correction procedure; indeed, they are characterized by low wall interference effects, and this leads to a great sensitivity to the measurement uncertainty (both for the forces and the wall pressure). Therefore, it is anticipated that the correction procedure is more accurate the more important the wall effects to be corrected are.

Description of the Correction Procedure

The adopted correction methodology is a so-called posttest procedure[4]; in this kind of method, experimental data must be provided on a control surface located near the wind-tunnel walls or directly on them. In particular, a one-array correction procedure has been chosen, in which pressure data are provided at some locations on the wind-tunnel walls. The correction methodology employed and the sensitivity analysis carried out to study the effects of different pressure sensors position and accuracy are described in Ref. 2.

The scheme of the correction procedure, which is based on the method proposed by Sickles,[5] is shown in Fig. 1. Once the model geometry is defined, the experimental tests are carried out, and in addition to the aerodynamic forces acting on the model, the pressure over the wind-tunnel walls is measured at a few selected locations. These measurements are used as boundary conditions in a numerical simulation of the flow around the same geometry (pressure-given simulation). Another numerical simulation is carried out in free-air conditions, that is without the presence of solid walls. The difference between the values of aerodynamic forces obtained in these two simulations is used to correct the experimental data.

For the choice of the flow solver adopted in the numerical simulations, the same criteria used in computational aerodynamics are clearly suitable also in this context. In the present paper subsonic flow conditions at low angles of attack are considered, and thus, a potential flow solver is used.[6] It is based on Morino's formulation, with a wake relaxation procedure and has been extensively validated for aircraft configurations (for instance, see Ref. 7).

Experimental Setup

The experimental setup has been described in detail in Ref. 8. Tests are carried out in the high speed wind tunnel (HSWT) of the CSIR Laboratories. The HSWT is a trisonic, open circuit blow down-type tunnel. Its operational speed ranges from $M = 0.55$ to $M = 4.3$ The test section has a 0.45×0.45 m square section, and the length is 0.9 m.

The Mirage F1 model, a wing–tail configuration with moderate aspect ratio (2.83), was selected because of availablity in different scales: 1:32, and 1:40. The nominal blockage factors, defined as the ratio between the model cross-sectional area and the test section area, at zero angle of attack, are 0.0101 for the 1:40 model and 0.0158 for the 1:32 model, whereas the span/width ratios are 0.210 and 0.263, respectively.

The aerodynamic forces are nondimensionalized with the dynamic pressure and the wing planform area, whereas the reference

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*Associate Professor, Department of Aerospace Engineering. Member AIAA.

†Associate Professor, Department of Aerospace Engineering.

‡Senior Engineer, Medium Speed Wind Tunnel.

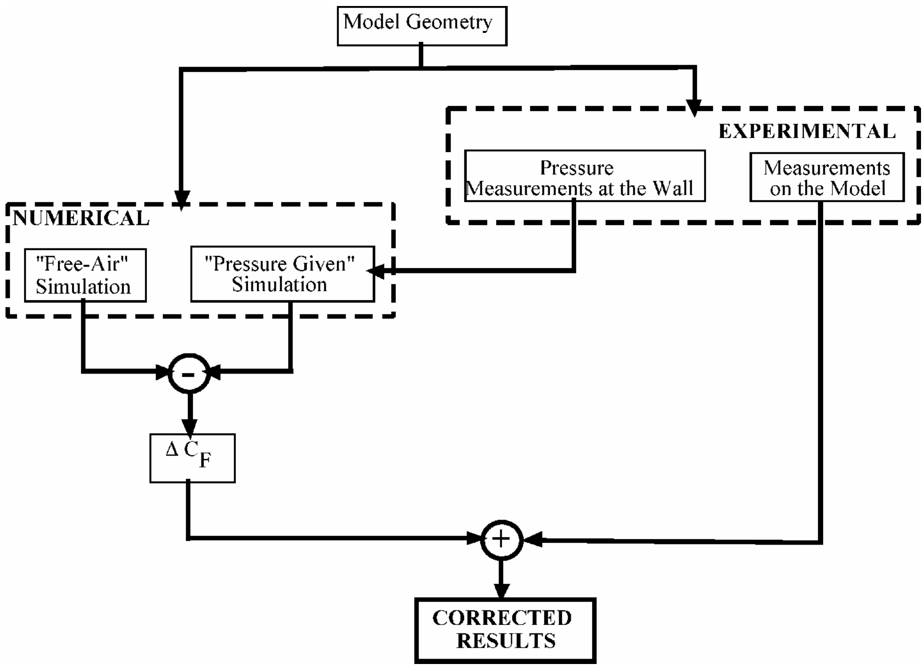

Fig. 1 Scheme of the correction procedure.

Table 1 Distribution of sensors for pressure measurement over wind-tunnel walls

Location	Sensor
Longitudinal, x/L	0.243, 0.351, 0.438, 0.494, 0.532, 0.562, 0.588, 0.611, 0.634, 0.660, 0.686, 0.715, 0.749, 0.792, 0.855, 0.952
Lateral	
Low and up walls y/w	0.083, 0.415
Vertical wall z/h	_0.415, _0.264, _0.083, 0.083, 0.264, 0.415

length for the moment coefficients (referred to the quarter-chord point of the mean aerodynamic chord) is the wing mean aerodynamic chord itself.[8] To reduce the effects of Reynolds number, models were provided with fixed transition stripes, located at 10% of the wing chord. A quality control inspection was carried out on the models; the differences in the geometry section are very small, and the complete analysis is reported in Ref. 8.

The models are supported by means of a sting, and the aerodynamic forces are measured by means of the same internal six-components balance; in this way, the bias component of the error is the same, and therefore, errors in the comparison of the results for the two scale models are reduced. Values are averaged over 5 s, at a sampling rate of 500 Hz. The complete characterization of the balance accuracy, resulting from the calibration procedure, may be found in Ref. 8. The error in the model pitch angle is lower than 0.1 deg, and data are corrected for the sting deflection.

To reduce the number of the required pressure data, it was decided to perform pressure measurements on only half of the wind-tunnel section in the cross direction, that is, the right or the left part. Most of the tests in the considered wind tunnel are carried out at zero yaw angle; if this is not the case, the tests are repeated with an opposite yaw angle to avoid spurious effects of lack of symmetry in the flow or model geometry.

A configuration characterized by 16 and 10 sensors in the longitudinal and lateral directions, distributed as defined in Table 1, was identified as a satisfactory one in previous work,[2] and it is used here. The x axis is the longitudinal direction, y the lateral direction, and z the vertical direction; with the origin in the center of the inlet section; L and h are the length and the height of the test section, respectively.

For the present application, two different runs are performed: A first one is carried out without the model, to obtain the wall pressure distribution in empty conditions and the second one with the model. The wall pressure data are obtained as the difference between the two runs.

Pressure probes (holes) are connected to the Scanivalve through Festo connectors and silicon tubes. Pressures are measured at a sampling rate of 20 Hz. For the present tests, the uncertainty in the pressure measurements was evaluated to 0.03 of the full scale.

Once the pressure data are obtained in the points defined by the procedure, they are linearly interpolated in the longitudinal direction; following the results in Ref. 2, a more accurate interpolation is used for the cross direction, that is, a parabolic law on the upper and lower walls of the cross section and cubic splines on the lateral wall.

Analysis of Results

The total uncertainty in the data can be attributed to instrumentation, reference dimension evaluation (surfaces, lengths, and moment reduction points), data acquisition procedures, and the difference in Reynolds number. However, as already mentioned, because of the use of the wind tunnel under identical flow conditions, the bias uncertainty should not be considered in its entirety when comparing the two models; indeed, it contains a part, dependent on flow measurements, force measurements, and on the evaluation of the model dimensions, that is the same in both cases.

The described tests were carried out at a Mach number of 0.58. The results for an angle of attack of 7.84 deg are summarized in Table 2, in term of lift coefficient, pitching moment coefficient, and estimation of the point of application of the lift $(\_c_M/c_L)$.

As expected, the correction terms, both for lift and pitching moment, increase with the blockage factor. Note that these terms are significant: With the change from 1:40 to 1:32 scale models, the measured lift increases of 4.3%. The difference appears significant also for the pitching moment, however, note that the point of application of the lift practically remains unchanged in the experiments.

After the application of the correction procedure, the lift coefficients difference between the two scale models is reduced to 1.5%. In spite of the significant improvement, this is still far from the desired accuracy (for instance, see Ref. 9), but it can be considered a satisfactory result, when it is taken into account that the difference

Table 2 Results at α = 7.84 deg

	C_L			C_M			$\_C_M/C_L$	
Results	1:40	1:32	Difference	1:40	1:32	Difference	1:40	1:32
Experimental	0.5356	0.5599	0.0243	_0.08057	_0.08492	0.00435	0.1504	0.1517
Correction term	0.0252	0.0418		_0.01870	_0.02200			
Corrected result	0.5104	0.5181	0.0077	_0.06187	_0.06292	0.00105	0.1212	0.1214

Table 3 Results at α = 3.74 deg

	C_L			C_M			$\_C_M/C_L$	
Results	1:40	1:32	Difference	1:40	1:32	Difference	1:40	1:32
Experimental	0.2431	0.2555	0.0124	_0.04714	_0.04817	_0.00103	0.1939	0.1885
Correction term	0.0078	0.0145		_0.00127	_0.00260			
Corrected result	0.2353	0.2410	0.0057	_0.04587	_0.04557	0.00030	0.1949	0.1891

after the correction, as earlier observed, is also related to experimental errors in force measurements and model position.

For the pitching moment, the accuracy of the corrected values appears satisfactory, in that the corrected estimation of the lift point of application is practically the same for the two models.

The results for a lower angle of attack (3.74 deg) are shown in Table 3. This condition is clearly characterized by a lower wall interference effect, and this leads to a greater sensitivity to the measurement uncertainty (both for the forces and the wall pressure). Indeed, when the results are compared with those of the preceding analyzed condition, it is evident that the lift coefficient is characterized by a lower accuracy after the correction procedure, with a difference of 2.4% between the two models. Also, the pitching moment results are less accurate: A difference of about 0.6% of the mean aerodynamic chord remains in the evaluation of the point of application of the lift.

Conclusions

A previously proposed posttest correction procedure has been applied to experimental data in subsonic low angle of attack conditions. It has been shown that the correction procedure effectively reduces the wall interference effects. However, as expected, the correction becomes more accurate when the wall effects to be corrected are important. Therefore, great care must be taken in deciding when to apply the proposed correction procedure: Indeed, for low blockage factors and low angles of attack, when the wall effects are very small, it is possible that measurement errors in the wall pressure evaluation produce errors in the correction procedure greater than the correction term itself.

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Analytical Prediction of Panel Flutter Using Unsteady Potential Flow

N. Soltani* and V. Esfahanian†
University of Tehran, 11365-4563 Tehran, Iran
and
H. Haddadpour‡
Sharif University of Technology, 11365-9567 Tehran, Iran

Introduction

DURING the last 50 years, many investigators have made significant contributions to the study of panel flutter. These authors have considered many aspects of the flutter models and a wide variety of aerodynamic theories. Many structural and aerodynamic issues such as the plate modeling, initial stresses, thermal effects, large deflection,[1–3] piston theory, unsteady potential flow and viscous flow effects,[4] respectively, have been considered. For instability prediction the analytical[1] and computational[5–7] methods have been also developed.

Because of the complexity of the problem, few analytical solutions are available in the literature. All of the analytical solutions use the piston theory for the aerodynamic modeling. This theory have been developed by the application of power series expansion in unsteady potential flow and retention of only the first two terms.[4]

In the present study full unsteady potential flow aerodynamics are applied to predict panel flutter analytically, using the modal analysis technique.

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*Associate Professor, Mechanical Engineering Department; nsoltani@ut.ac.ir.

†Associate Professor, Mechanical Engineering Department; evahid@ut.ac.ir.

‡Assistant Professor, Department of Aerospace Engineering; haddadpour@sharif.edu.