Framework for Establishing the Limits of Tabular Aerodynamic Models for Flight Dynamics Analysis

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This paper describes the use of Computational Fluid Dynamics for the generation and testing of tabular aerodynamic models for flight dynamics analysis. Maneuvers for the AGARD Standard Dynamics Model wind tunnel geometry for a generic fighter are considered as a test case. Wind tunnel data is first used to validate the prediction of static and dynamic coefficients at both low and high angles, featuring complex vortical flow, with good agreement obtained at low and moderate angles of attack. Then the generation of aerodynamic tables is described based on an efficient data fusion approach. An optimisation is used to define time optimal maneuvers based on these tables, including level flight trim, pull-ups at constant and varying incidence, and level and 90 degree turns. The maneuver description includes a definition of the aircraft states and also the control deflections to achieve the motion.

The main point of the paper is then to assess the validity of the aerodynamic tables which were used to define the maneuvers. This is done by replaying them, including the control surface motions, through the time accurate CFD code. The resulting forces and moments can be compared with the tabular values to assess the presence of inadequately modeled dynamic or unsteady effects. The agreement between the tables and the replay is demonstrated for slow maneuvers. A study for the pull-up at increasing rates shows discrepancies which are ascribed to vortical flow hysteresis at elevated motion rates.

I. Introduction

The flight mechanics analysis of combat aircraft requires a nonlinear and unsteady aerodynamic model valid for rapid and large amplitude maneuvers.1, 2 The source of nonlinearity and unsteadiness is mainly due to shock waves, separation and vortices at high angles of attack.3 These phenomena impact aircraft performance but developing a model of these flows is difficult due to the complexity of the flow simulation and the limitations of existing dynamic wind-tunnel test facilities.4 Attempts at exploiting CFD for predicting S&c derivatives were reported in references.5–8

The aerodynamic model for flight dynamics analysis considered in this paper is tabular in form. Tables, in contrast to stability derivatives, are not linearized, and are consistent with quasi-steady aerodynamics for a wide range of flight conditions. There are several difficulties associated with aerodynamic tables. First, a large number of table entries must be filled. The brute-force calculation of the entire table using CFD is not feasible due to computational cost. Sampling and data fusion methods have been proposed to overcome this.9 Secondly, the pre-computed nature of tables lacks the ability to describe hysteresis of the aerodynamic phenomena. That flow hysteresis impacts on the maneuvering aircraft was first recognized by Harper and Flanigan10 in 1950. They showed that there is a substantial increase of aircraft lift force if the aircraft is pitched at a rapid rate. It is well known that aerodynamic forces and moments of aircraft responding
to sudden changes of flow around aircraft not only depends on the instantaneous values but also the time history of the motion.\textsuperscript{11–14}

CFD tools are becoming credible for the computation of aerodynamic time history effects. The flight dynamics of a manoeuvring aircraft could be accurately modelled by coupling the Reynolds-Averaged Navier-Stokes (RANS) equations and the dynamic equations governing the aircraft motion. First attempts were limited to two-dimensional test cases,\textsuperscript{15–19} while recently the coupled CFD-flight dynamics of a full aircraft has been studied.\textsuperscript{20} The 6DoF CFD-coupled simulations take substantially longer than when taking the forces and moments from look-up tables raising the open question of when time hysteresis of the aerodynamics is required for flight dynamics analysis.

In the present paper a framework for investigating the limits from flow unsteadiness of aerodynamic tables is investigated. The particular demonstrations of this framework are made for a generic (and sharp leading edge) fighter performing time optimal manoeuvres with the aerodynamics given by the Euler equations. The paper first reviews the flow solver and how the manoeuvres are computed. Then the test case is described and the aerodynamic predictions validated. The generation of the aerodynamic tables and the approach to defining the time optimal manoeuvres is then given. The evaluation of the aerodynamic tables for these manoeuvres is then made by replaying them directly through an unsteady CFD calculation. Finally, conclusions are given.

II. CFD Formulation

A. Flow Code

The flow solver used for this study is the University of Liverpool PMB (Parallel Multi-Block) solver. The Euler and RANS equations are discretised on curvilinear multi-block body conforming grids using a cell-centred finite volume method which converts the partial differential equations into a set of ordinary differential equations. The equations are solved on block structured grids using an implicit solver. A wide variety of unsteady flow problems, including aeroelasticity, cavity flows, aerospike flows, delta wing aerodynamics, rotorcraft problems and transonic buffet have been studied using this code. More details on the flow solver can be found in Badcock et al\textsuperscript{21} and a validation against flight data for the F-16XL aircraft is made in reference.\textsuperscript{22}

B. Motion Replay

The key functionality for the CFD solver in the current application is the ability to move the mesh. Two types of mesh movement are required. First, a rigid rotation and translation is required to follow the motion of the aircraft. Secondly, the control surfaces are deflecting throughout the motion. The control surfaces are blended into the geometry in the current work following the approach given in reference.\textsuperscript{23} After the surface grid point deflections are specified, transfinite interpolation is used to distribute these deflections to the volume grid.

The rigid motion and the control deflections are both specified from a motion input file. For the rigid motion the location of a reference point on the aircraft is specified at each time step. In addition the rotation about this reference point is also defined. Mode shapes are defined for the control surface deflections.\textsuperscript{24} Each mode shape specifies the displacement of the grid points on the aircraft surface for a particular control surface. These are prepared as a preprocessing step using a utility that identifies the points on a control surface, defines the hinge, rotates the points about the hinge and works out their displacements. The motion input file then defines a scaling factor for each mode shape to achieve the desired control surface rotation.

The desired motion to be replayed through the unsteady CFD solution is specified in the motion input file. The aircraft reference point location, rotation angles and control surface scaling factors are needed. The rotation angles are obtained straight from the pitch, yaw and bank angles. The aircraft reference point velocity $v_a$ is then calculated to achieve the required angles of attack and sideslip, and the forward speed. The velocity is then used to calculate the location. The CFD solver was originally written for steady external aerodynamics applications. The wind direction and Mach number are specified for a steady case. If the initial aircraft velocity is denoted as $v_0$, then this vector is used to define the Mach number and the angle of attack for the steady input file. Then the instantaneous aircraft location for the motion file is defined from the relative velocity vector $v_a - v_0$. 
III. Test Case and Validation

The Standard Dynamics Model (SDM) is a generic fighter configuration based on the F-16 planform. The
model includes a slender strake-delta wing, horizontal and vertical stabilizers, ventral fin and a blocked off
inlet section. The three view drawing is shown in figure 1. This geometry has been used for an international
effort to collect wind tunnel data.25–28

In the current paper Euler calculations are used to generate the aerodynamic forces and moments. A
multiblock mesh was generated using the commercial package ICEMCFD. The geometry was slightly
simplified by removing the blocked off intake and the ventral fins. Since the main interest here is on the
impact of vortical flow on the upper lifting surfaces these were considered reasonable simplifications. A fine
Euler mesh was generated with 5.6 million points, and a coarse mesh was obtained with 701 thousand points
by omitting every second point. The calculations described are for the coarse mesh unless otherwise stated.

Mesh block faces were placed on the control surfaces and the mesh points on these faces were deflected
to define the control surface mode shapes. The control surfaces are blended into the airframe as described
above, and views of the surface mesh for the deflected control surfaces are shown in figure 2.

The geometric, mass, inertial and engine properties of SDM are obtained from data in Winchenbach et
al.30 Their free-flight model is 0.1325:1 scale of the studied model in this paper resulting in scaled-up values
of:

\[
S = 27.87 \ m^2 \quad m = 9295.44 \ kg \\
c = 3.45 \ m \quad I_{xx} = 12874.0 \ kgm^2 \\
b = 9.144 \ m \quad I_{yy} = 75673.0 \ kgm^2 \\
T_m = 26.24 \ KN \quad I_{zz} = 85552.0 \ kgm^2
\]

where \(T_m\) is the maximum total thrust force assumed to cross the centre of gravity. The thrust is assumed
to remain unchanged with altitude and flight speed, and to vary linearly with the engine throttle.

The lifting surfaces all have sharp leading edges. This allows the Euler equations to predict the develop-
ment of vortical flow since the separation points are fixed at the leading edge. A complex interaction of the
strake and wing vortices develops as the incidence is increased, as illustrated in figure 3 for low speed and no
sideslip conditions. At ten degrees the vortices form, remain unbroken and do not interact over the airframe.
At fifteen degrees the two vortices wind around each other towards the trailing edge of the wing. At twenty
degrees the wing vortex appears to breakdown quickly after formation, whereas the strake vortex is coherent.
Figure 2. View of Deflected Control Surfaces on the Surface Grid.
for longer. Finally at thirty degrees there is no sign of coherent vortices. Note that these calculations were all run as steady state although vortical interactions and the separated flow field would almost certainly be unsteady. The assumption is that the mean flowfield from unsteady calculations would resemble the steady result shown here.

These vortex dominated flow fields cause nonlinear variations in the aerodynamic forces and moments. Figure 4 compares DATCOM\(^{31}\) and CFD results with available experimental data\(^{29}\) at low speed and zero sideslip conditions. The figures show that there is a good agreement between the Euler predictions and the measurements for angles of attack below 20 degrees. Above 20 degrees the disagreement in the magnitude of the longitudinal quantities increases and the sign of the slope of the lateral quantities is wrong. At these conditions it was seen that the vortical flow is lost, and the lateral forces and moments will be strongly influenced by the separated flow adjacent to the vertical fin. The Euler simulations will not predict this flowfield properly, and Detached Eddy Simulation is a better option. The DATCOM predictions in some respects are adequate (eg for the pitching moment and normal force coefficient), but it should be noted that the empirical database which underlies these predictions is for a conventional aircraft which would not feature vortical flow. This can be seen in the predictions of the normal force coefficient which show a conventional aircraft stall. The lateral coefficients are generally not well predicted by DATCOM in this case.

Figure 3. SDM-Flow Field Visualization. The calculations are for a Mach number of 0.3 and zero degrees sideslip. The streamtraces are coloured by pressure coefficient.

For fast manoeuvres the dynamic contributions to the forces and moments are required. The estimation of the quasi-steady dynamic derivatives is achieved by imposing a forced sinusoidal motion around the aircraft centre of gravity. For the extraction of longitudinal dynamic derivative values from time-histories of the forces and moments, it is assumed that the aerodynamic coefficients are linear functions of the angle of
Figure 4. Static Aerodynamic Predictions- Lift, drag and pitching moments are for zero degrees sideslip, $V_0=100$ m/s. The lateral figures are for five degrees of side-slip and $V_0=100$ m/s. The experimental data are from reference.29
attack, $\alpha$, pitching angular velocity, $q$, and rate, $\dot{q}$.

The increment in the lift coefficient with respect to its mean value during the applied sinusoidal motion is formulated as

$$\Delta C_L = C_{L,\alpha} \Delta \alpha + \frac{l}{V} C_{L,q} \dot{\alpha} + \frac{l}{V} C_{L,q} q + \left( \frac{l}{V} \right)^2 C_{L,q} \dot{q}$$  

(1)

where $V$ is the magnitude of the aircraft velocity vector and $l$ is the characteristic length, here the wing root chord. The pitching moment is treated in an analogous way.

The harmonic motion defines the following relations

$$\Delta \alpha = \alpha_A \sin(\omega t)$$
$$\dot{\alpha} = \alpha_A \omega \cos(\omega t)$$
$$\dot{q} = -\alpha_A \omega^2 \sin(\omega t)$$  

(2)

Equation (1) can be re-written as

$$\Delta C_L = \alpha_A \left( C_{L,\alpha} - k^2 C_{L,q} \right) \sin(\omega t) + \alpha_A k \left( C_{L,\alpha} + C_{L,q} \right) \cos(\omega t)$$  

(3)

where $k = \omega/V$ is the reduced frequency of the applied motion. The in-phase and out-of-phase components of $\Delta C_L$, respectively indicated as $\bar{C}_{L,\alpha}$ and $\bar{C}_{L,q}$, can be defined as

$$\bar{C}_{L,\alpha} = C_{L,\alpha} - k^2 C_{L,q}$$
$$\bar{C}_{L,q} = C_{L,\alpha} + C_{L,q}$$  

(4)

(5)

$$\Delta C_L = \alpha_A \bar{C}_{L,\alpha} \sin(\omega t) + \alpha_A k \bar{C}_{L,q} \cos(\omega t)$$  

(6)

These coefficients can be obtained by taking the first Fourier coefficients of the time history of $C_L$.

The longitudinal dynamic derivatives were computed for the SDM. Periodic motions with five degrees amplitude and a reduced frequency of 0.0493 were applied for six cycles to reach a periodic state.

Figure 5 shows the comparison with wind tunnel and range data from Winchenbach. A good agreement is obtained up to about 17 degrees. DATCOM cannot predict the variation with angle of attack.

IV. Generation of Manoeuvres

A. Aerodynamic Tables

The flight dynamics predictions requires a vector of aerodynamic forces and moments

$$[C_L, C_D, C_m, C_Y, C_t, C_n]^T$$

representing wind axis coefficients of lift, drag, pitching-moment, side-force, rolling moment and yawing moment coefficient, respectively. This vector depends on the aircraft state and control variables. In this paper, this dependency is represented in look-up tables. For a slow motion, the aerodynamic forces and moments are assumed to depend on angle of attack, Mach number, side-slip angle and three control surface deflection angles. The tables are arranged with three parameters in each: Mach number, angle of attack and one other variable. Around 6000 entries are required to define the variations in each table, leading to a large number of calculations required if a brute force method was used.

An alternative approach is used to fill up the tables. The brute-force calculations were used to provide the values with respect to the angle of attack and Mach number (referred to here as the baseline table). The dependence of the longitudinal forces and moments on other parameters is assumed to be an increment of the baseline table and the Co-Kriging data fusion approach\(^9\) is used to include this variation in a computationally efficient way. The baseline table describes the general trends of longitudinal aerodynamic forces and moments. This table consists of 156 conditions, with the angle of attack ranging from -14\(^0\) to 28\(^0\), and the Mach number from 0.1 to 0.4. With just 15 additional samples the variation with the elevator, side-slip angle, aileron and rudder was included in the tables.

For the lateral coefficients a different approach is needed since all the lateral coefficients in the baseline table are zero. Instead the DATCOM lateral coefficients are used as low fidelity data for Co-Kriging and then a few Euler results are used to generate updated tables using co-Kriging.

For the dynamic derivatives, these are assumed to be independent of Mach number, and to vary with angle of attack.
Figure 5. Dynamic Derivative Aerodynamic Predictions. The Mach number in the upper figures is 0.3 and the angle of attack in the lower-left figure is zero. The experimental data is from reference\textsuperscript{30}.
B. Time Optimal Manoeuvres

This paper aims to investigate the validity of aerodynamic tables for the generation of time-optimal manoeuvres. Manoeuvre flight (sometimes named accelerated flight) is the response of the aircraft to the pilot’s applied control inputs starting from one trimmed flight condition to another. The manoeuvre time is the flight period between the initial and final trim points, and this should be minimised for aerial combat manoeuvres.

The optimal control problem finds the optimal controls that transfer a system from the initial state to the final state while minimizing (or maximizing) a specified cost function. The optimal control aims to find a state-control pair \( x^*(t), u^*(t) \) and possibly the final event time \( t_f \) that minimizes the cost function

\[
J[x(t), u(t), t_0, t_f] = E(x(t_0), u(t_f), t_0, t_f) + \int_{t_0}^{t_f} F(x(t), u(t), t)dt
\]

where \( E \) and \( F \) are endpoint cost and Lagrangian (running cost), respectively. For the problem of an aircraft optimal time manoeuvre, the general aircraft equations of motion detailed in Etkin and Stevens and Lewis serve as one of the constraints. Also, the initial and final state parameters are fixed with trimmed flight conditions, but the rest of the manoeuvre is out of trim conditions.

For the solution of the optimal control problem the DIDO code and MATLAB are used. In DIDO, the total time history is divided into \( N \) segments. The boundaries of each time segment are called nodes. The value of \( N \) is normally in the range 5 to 150. The optimal state and control pair are calculate at the nodes.

The optimization uses the solution of the 6DoF equations as a constraint. This solution derives the aerodynamic forces and moments from the look-up tables. Hence, the predictions from these tables need to be realistic for the manoeuvre itself to be valid. The tables can be limited from a number of sources:

- too coarse a representation which misses nonlinear variations
- inadequate representation of the dynamic derivatives
- significant hysteresis in the aerodynamics.

The main contribution of this paper is to consider how to evaluate the limitations by replaying the motion through a time accurate CFD simulation. The predicted forces and moments can then be compared with the tabular model values to evaluate the consistency between the two sets of values.

V. Testing of Manoeuvre Replay

A. Overview

This section shows the comparison between the prediction of the aerodynamic forces from the static tabular model and the motion replay. This is done for slow motions where exact agreement would be expected. The motions used are trimmed level flight, pull-ups with constant and varying angle of attack, wing-over and 90-degree turns. The comparisons test the CFD formulation of the manoeuvre replay, which is done in a time accurate fashion with control surface deflections. Throughout this section the replay predictions are referred to on the figures as Coupled Model.

B. Level Flight

The first manoeuvre is for steady wing-level straight flight at 3,000 ft and 350 fps. To define this manoeuvre the constraints are set for the time rates of angle of attack, pitch angle and flight speed with upper and lower values of \( \pm 1.0 \times 10^{-5} \). The altitude is also set as a constraint. The solution gives a trim angle of attack 7.67° and elevator deflection angle of 1.69°. In order to verify the optimal criteria and validate the solution, the Hamiltonian function and flight altitude during flight are shown in figure 6. During the 85 seconds of flight computed the altitude remains unchanged as required. In addition the Hamiltonian values are nearly constant, showing that the solution is optimum.

It is possible to define the replay calculation through the CFD in different ways. As described in the CFD formulation section the important quantity to be specified is \( \mathbf{v}_a - \mathbf{v}_0 \) where \( \mathbf{v}_0 \) is a fixed vector specified
Figure 6. Level Flight Trim Solution and Replay Test.
through the initial steady state conditions given to the CFD solver and $\mathbf{v}_a$ is the instantaneous velocity vector of the aircraft in an axis system moving with the initial steady velocity. We are free to choose any steady velocity vector. To test the replay calculation the magnitude of this vector was set to three different values, namely $V_0 = 350$, $V_0 = 300$ and $V_0 = 400$ fps, where $V_0$ is the magnitude of the vector $\mathbf{v}_0$. The aircraft velocity was then set to achieve the desired relative velocity and angle of attack. The calculation of the lift coefficient for these three cases is shown in figure 6 and all three cases give identical values as required.

C. Fixed $\alpha$ Pull-Up Manoeuvre

Next two pull-up manoeuvres are considered. The first pulls up faster at 8 deg/sec at a fixed value of $18^\circ$ angle of attack and the second is a slow pitch up case with 2 deg/s pitch rate at a fixed value of $9.5^\circ$ angle of attack. For these manoeuvres, the upper and lower boundary values of the time rates of angle of attack, pitch rate and flight speed are set to $\pm 1.0 \times 10^{-5}$. The boundaries of angle of attack and pitch rate are changed in the code in order to achieve two different manoeuvres. Also, the elevator deflection angles are limited to $\pm 25^\circ$.

The solution for the slow pitch up is shown in figure 7. This shows the trajectory and Hamiltonian. In addition the tabular and replay values for the lift coefficient are shown and are in perfect agreement. Finally the surface pressure contours are shown, with the effect of the weak leading and strake vortices visible. The same plots are shown in figure 8 for the higher rate pull-up. Again the tabular and replay results are in agreement for the lift coefficient, and the footprint of the stronger vortices are apparent in the pressure contour plot.

D. Varying $\alpha$ Pull-Up Manoeuvre

Removing the constraints imposed to time-rate changes of angle of attack and pitch rate for the fixed $\alpha$ pull-up results in a pull-up with varying angle of attack. In order to find a slow pull-up, the upper boundary of pitch rate is set to two degrees per second.

The optimum solution results in a pull-up with varying angle of attack from $4.7^\circ$ to $11.5^\circ$ in 30 seconds. This means that dynamic effects are expected to be small. The elevator varies throughout the motion up to deflections of $5^\circ$. The time variation of the angle of attack and the elevator angle are shown in figure 9. The comparison of the tabular and replay values of the lift and drag coefficients are also shown and are in close agreement.

E. Wing-over

Next manoeuvres featuring lateral motions are considered. A wing-over manoeuvre in 130 seconds is defined. The predicted time-optimal manoeuvre changes the heading of the aircraft from 0 to $180^\circ$, while the final values of the velocity, altitude and latitude ($x$) are fixed with the initial-time ones. The aircraft gains altitude by increasing its angle of attack, while banking at the same time. At its minimum speed, rudder is applied to change the heading and then the aircraft starts to dive.

The start and final speed and altitude of minimum time manoeuvre are set to 300fps and 3,000ft, respectively. The code is free to choose initial and final angles of attack, but they need to be below 20 degrees. Also, throughout time, the angles of attack should not exceed the upper limit of 20 degrees. The elevator, aileron and rudder deflection values are also limited to $\pm 20^\circ$. All the rotation angular speeds are limited to small value, resulting a slow moving manoeuvre.

Figure 10 depicts the predicted manoeuvre. The definition of the angles of attack and sideslip are given in figure 11, together with the control surface deflections. The comparison between the tabular aerodynamics with the replay predictions shows good agreement.

F. Turn 90

The final manoeuvre is a 90 degree turn in 200 seconds, shown in figure 12. The aircraft returns to its original position but with its heading changed by $90^\circ$. In such a manoeuvre the final values of the velocity, altitude, latitude ($x$) and longitude ($y$) are fixed with the initial-time values. The predicted angles of attack and sideslip are shown in figure 12, together with the control surface deflections. The lift, drag and side force coefficients are also shown in figure 13 for the replay and the tables and again are in close agreement.
Figure 7. Steady Pull-Up Manoeuvre at 2 deg/s pitch rate and a fixed angle of attack of 9.5°.
Figure 8. Steady Pull-Up Manoeuvre at 8 deg/s pitch rate and a fixed angle of attack of 18°.
Figure 9. Slow Pull-Up Manoeuvre with varying angle of attack.
VI. Fast Manoeuvres

Having demonstrated the agreement between the static tabular and replay predicted aerodynamics for slow motions, we now consider faster manoeuvres where we would expect significant dynamic and unsteady effects. Increasingly fast pull-up manoeuvres are considered in this section. Dynamic terms are added to the static tabular values. The aim is to evaluate the importance of the dynamic terms, and the presence of unsteady terms that are not represented in the tables.

The pull-up with time varying angle of attack, described above, is considered as the baseline manoeuvre. This is referred to as the Low angle of attack pull-up (Figs 9). Also, a high angle of attack pull-up was defined with the range of angle of attack going from 12.7° to 19.5°. Faster versions of both of these manoeuvres were obtained by reducing the duration from 30 seconds to 3 seconds and then to 0.3 seconds (to test the aerodynamic predictions rather than to assess a realisable manoeuvre).

The time variation of the lift coefficient for the six resulting manoeuvres is shown in figure 14. Looking first to the left column in this figure for the low angle of attack case, agreement between the static tabular predictions and the replay is obtained for the slow 30 sec motion. The dynamic tabular terms are not required at the rates in this motion. Small discrepancies are observed between the static tables and the replay values for the 3 sec motion. The addition of dynamic tabular terms gives agreement with the replay values. Larger differences are observed with the static tabular predictions for the 0.3 sec manoeuvre, but the dynamic terms again bring the tabular predictions into agreement with the replay results.

The story is the same for the high angle case for the 30 sec and 3 sec motions. However, for the 0.3 sec case adding the dynamic corrections do not bring the tabular predictions into line with the replay predictions. To investigate this further the pressure distributions at the time of maximum discrepancy are plotted in figure 15. The time is marked by the red symbols in figure 14. The time accurate replay result is shown on top and the quasi-steady prediction at the conditions at this time is shown below. Note that this calculation...
Figure 11. Wing-Over Manoeuvre definition and replay solution.
includes both the static and dynamic effects in the solution, and the red symbol in figure 14 is the steady state value obtained. This is in perfect agreement with the tabular value calculated from separate estimates of the static and dynamic effects included in this figure as a solid dot. This agreement builds confidence in the static and dynamic tables. The motion replay solution shows a weaker leading edge vortex which leads to the lower value of the lift coefficient. The angle of attack is increasing at this point in the motion and the solution for the replay appears to be lagging the time accurate solution.

VII. Conclusions

This paper describes a framework for generating optimal manoeuvres based on CFD generated tabular aerodynamic models, and then testing the aerodynamic model by replaying the manoeuvre using an unsteady CFD calculation to check the consistency of the aerodynamic forces and moments. The CFD solver uses two types of mesh movement, namely rigid motion, and transfinite interpolation for the control surfaces. Data fusion was used to allow the generation of the aerodynamic tables in a feasible number of static and forced motion CFD calculations.

The test case used was the SDM generic fighter. Sharp leading edges for the lifting surfaces allowed the Euler equations to be used. A validation was made against available experimental data which showed that the predictions fail to predict static lateral forces and moments above 20 degrees when the flowfield is no longer dominated by coherent vortices. The longitudinal dynamic derivatives showed agreement up to 17 degrees. To improve the agreement at higher angles viscous modelling is required, with Detached Eddy Simulation a good candidate. However, the agreement at lower angles means that the Euler equations can be used to demonstrate the replay framework, with credible flow physics and predictions being present in the test case.

A number of slow longitudinal and lateral manoeuvres were used to demonstrate that the tabular and
Figure 13. Turn 90 Manoeuvre definition and replay solution.
Figure 14. Comparison of Pull-ups replay at different rates.
Figure 15. Surface pressure contours for the high angle of attack 0.3 second manoeuvre at the time instant marked by the red symbol on figure 14.
replay aerodynamics agreed closely as expected. Then a number of higher rate manoeuvres for a pull-up were used to investigate the influence of dynamic and unsteady terms. At moderate rates the addition of dynamic tabular terms brought the tabular predictions into agreement with the replay values. However, for a high angle of attack, high rate motion, the dynamic terms were not adequate to achieve agreement. A quasi-steady calculation confirmed that the tabular prediction is correct, and so the disagreement is due to the influence of the history in the unsteady replay manoeuvre. This example shows the usefulness of the framework in investigating the limits of the static and dynamic tables.

Future work will extend this study to include DES flow modelling to try to capture high angle of attack vortical flows more realistically than was achieved with the Euler modelling in this study. The framework will be applied to a fighter trainer aircraft with flight test data to compare with. Finally, the impact of vortical flow history effects will be investigated, and previous work on modelling these effects reconsidered.

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