CFD Methods at the University of Glasgow
Fundamental Research Supporting Real-World Aerospace Design

George N. Barakos (george.Barakos@Glasgow.ac.uk)
School of Engineering, University of Glasgow, Glasgow G128QQ, Scotland, UK

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R. Steijl, M. Woodgate, A. Jimenez-Garcia, G. Chirico, M. Biava, S. Colonia

Fundamental Work vs Practical Exploitation

1) Adjoint methods and design attempts
   Tilt-rotor blades, ducted fans
2) Work on turbulence and transition models
   Hybrid Air Vehicles, Wind turbine blades
3) Work on high order methods
   Propeller acoustics
4) New computer architectures
   Simulator wakes
5) Work on flight mechanics and coupling of computer codes
   Examples of ADS 33, SHOL
HMB – Helicopter Multi-Block

Finite Volume Method for spatial discretisation
**Parallel** - Shared and Distributed memory (KNL support)
**Multi-block** structured/unstructured grids.
**Fully-Implicit** time marching / **Frequency** domain
**Osher, Roe, LM-Roe, AUSM+/UP** schemes for all Mach numbers.
**MUSCL** schemes for up to 5th order spatial accuracy
Central differences for viscous fluxes
Krylov subspace linear solver with pre-conditioning
**RANS, URANS, LES, DES, SAS**
**Boussinesq turbulence** models, **Explicit Algebraic Stress** models and **transition** models (LCTM)
**Moving/Deforming** grids, **Sliding Planes, Overset** grids
**Validation Database, version control**
Used by Academics and Engineers

HMB3 – Tool structure

- **HFWH** Far-Field Acoustics
- **HFM** Flight Mechanics Method
- **HMBDM** Multi-Body Dynamics Method
- **HMBeam** NASTRAN
- **HSPH method** Ditching and Floating Structures
- **HLBM** Lattice Boltzmann Method
- **Real-Time Wakes**
- **HVPM** Vortex Particle Method
- **Far-Wakes**

Written in C, most of the code differentiated
Introduction – Source Code Differentiation

Challenges in helicopter aeromechanics and design

- Understand the rotor flow in complex manoeuvres
- Aerodynamic optimisation with acceptable turnaround time

A limiting factor for both applications is the expensive in evaluation of flow gradients

- Adjoint methods and Automatic Differentiation are very efficient tools for computing aerodynamic gradients
- The objective is to remove limiting barriers to the use of high fidelity CFD in helicopter flight mechanics and design

Adjoint methods

- Provide efficient way to compute flow solution gradients, regardless the number of independent variables
- The actual implementation for complex CFD models is difficult, but this is eased using source code differentiation

Automatic differentiation

- The adjoint code is obtained directly from the original CFD code by means of source transformation tools
- It produces the exact gradients, even for complex CFD solutions involving advanced turbulence models
Adjoint method theory

Objective: compute the derivative of a functional $I(W(x),x)$ (e.g. lift, drag, a cost function, etc.) subject to $R(W(x),x)=0$ (the flow equations)

$W \Rightarrow$ flow variables, $x \Rightarrow$ input/design variables

The idea is to differentiate formally the functional $I(W(x),x)$, so as to obtain the so called “tangent form” of the sensitivity equation

$$\frac{\partial I}{\partial x} = \frac{\partial I}{\partial W} \frac{\partial W}{\partial x} + \frac{\partial I}{\partial x}$$

The partial derivatives on the RHS are cheap to compute, exception made for the term $\frac{\partial W}{\partial x}$. It could be obtained by differentiating the flow equations $R(W(x),x)=0$, to yield the linear system

$$\frac{\partial R}{\partial W} \frac{\partial W}{\partial x} = \frac{\partial R}{\partial x}$$

Adjoint method theory (cont.)

The dual form of the sensitivity equation is obtained by introducing the “adjoint” variables $\lambda$ as the solution of the linear system

$$\begin{pmatrix} \frac{\partial R}{\partial W} \end{pmatrix}^T \lambda = - \frac{\partial I}{\partial W}$$

Inserting this expression in the original sensitivity equation and after some matrix algebra we obtain the “adjoint form” of the sensitivity equation

$$\frac{dI}{dx} = \frac{\partial I}{\partial x} + \lambda^T \frac{\partial R}{\partial x}$$

Note that:

- the cost for computing derivatives with the tangent form is proportional to the number of input variables
- the cost for computing derivatives with the adjoint form is proportional to the number of functionals
Application to CFD (cont.)

**Example** Computation of $\frac{dC_L}{d\alpha}$ in adjoint mode.

The problem takes the form:

$$\frac{dC_L}{d\alpha} = \frac{\partial C_L}{\partial \alpha} + \lambda^T \frac{\partial R}{\partial \alpha}$$

The partial derivatives are computed as follows:

- $\frac{\partial C_L}{\partial \alpha}$: $\delta\alpha \Rightarrow \text{loads}_d \Rightarrow \delta C_L$
- $\frac{\partial R}{\partial \alpha}$: $\delta\alpha \Rightarrow \text{steady_residual}_d \Rightarrow \delta R$
- $\frac{\partial C_L}{\partial W}$: $\delta C_L \Rightarrow \text{loads}_b \Rightarrow \delta W$

The adjoint variables are obtained via the fixed point iterations

$$\lambda: J^T \Delta \lambda^{n+1} = -\frac{\partial C_L}{\partial W} - J^T \lambda^n, \text{ and } J^T \lambda \text{ is given by } \text{steady_residual}_b$$

---

Sensitivities of Viscous Flows

**NACA0012, M=0.5, Re=2000**

<table>
<thead>
<tr>
<th>Differentiability</th>
<th>Value</th>
<th>Value</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>HMB AD tangent</td>
<td>3.0516083E+00</td>
<td>≈ 0</td>
<td>-6.2441655E-01</td>
</tr>
<tr>
<td>HMB AD adjoint</td>
<td>3.0516083E+00</td>
<td>≈ 0</td>
<td>-6.2441654E-01</td>
</tr>
<tr>
<td>HMB FD</td>
<td>3.1152449E+00</td>
<td>≈ 0</td>
<td>-6.1300505E-01</td>
</tr>
<tr>
<td>Difference [%]</td>
<td>2.09</td>
<td>1.83</td>
<td></td>
</tr>
</tbody>
</table>
Sensitivities of Turbulent Flows

NACA0012, $M=0.5$, $Re=10^6$

<table>
<thead>
<tr>
<th>Method</th>
<th>$\frac{\partial C_L}{\partial \alpha}$</th>
<th>$\frac{\partial C_D}{\partial \alpha}$</th>
<th>$\frac{\partial C_M}{\partial \alpha}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>HMB AD tangent</td>
<td>7.0912685E+00</td>
<td>≈ 0</td>
<td>-1.7213406E+00</td>
</tr>
<tr>
<td>HMB AD adjoint</td>
<td>7.0912685E+00</td>
<td>≈ 0</td>
<td>-1.7213406E+00</td>
</tr>
<tr>
<td>HMB FD</td>
<td>7.0938335E+00</td>
<td>≈ 0</td>
<td>-1.7361087E+00</td>
</tr>
<tr>
<td>Difference [%]</td>
<td>0.04</td>
<td></td>
<td>0.86</td>
</tr>
</tbody>
</table>

Sensitivities of Turbulent Flows with Shocks

NACA0012, $M=0.8$, $Re=10^6$

<table>
<thead>
<tr>
<th>Method</th>
<th>$\frac{\partial C_L}{\partial \alpha}$</th>
<th>$\frac{\partial C_D}{\partial \alpha}$</th>
<th>$\frac{\partial C_M}{\partial \alpha}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>HMB AD tangent</td>
<td>1.3220458E+01</td>
<td>≈ 0</td>
<td>-4.2402488E+00</td>
</tr>
<tr>
<td>HMB AD adjoint</td>
<td>1.3220459E+01</td>
<td>≈ 0</td>
<td>-4.2402489E+00</td>
</tr>
<tr>
<td>HMB FD</td>
<td>1.3182143E+01</td>
<td>≈ 0</td>
<td>-4.2244556E+00</td>
</tr>
<tr>
<td>Difference [%]</td>
<td>0.29</td>
<td></td>
<td>0.37</td>
</tr>
</tbody>
</table>
Sensitivities of 3D Flows

M6 wing, $\alpha=3.06^\circ$, $M=0.8395$

Previous Optimisation Framework

USER

Parameterisation Technique

CFD Database

Training

Metamodel

Performance Prediction

Optimiser

A-posteriori validation

Pareto Front check

Objectives
Gradient Based Optimisation

Inverse Problem

Initial value of design variables

Baseline design

Volume grid deformer

HMB CFD Solver

CFD solver

Adjoint solver

Updated shape

Optimiser

Gradient based optimiser

Nonlinear constraints

Problem specific!!!

Mesh Deformation

Mesh deformation is based on **Inverse Distance Weighting**

\[
u(x) = \begin{cases} 
\frac{\sum_{i=1}^{N} w_i(x) u_i}{\sum_{i=1}^{N} w_i(x)} & \text{if } d(x, x_i) \neq 0 \text{ for all } i \\
\frac{\max(0, R - d(x, x_i))}{d(x, x_i)} & \text{if } d(x, x_i) = 0 \text{ for some } i 
\end{cases}
\]

where

\[
w_i(x) = \left(\frac{\max(0, R - d(x, x_i))}{d(x, x_i)}\right)^2
\]

Only the *k* nearest points in the ball with radius *R* are considered and an **Alternating Digital Tree** is used for point search.

This results in a **faster** and more **robust** algorithm.
Mesh Deformation

High Re Transitional Flow

- Investigate transition prediction methods
- Find alternatives to e^N
- Compare with test data at high Re

- EERA Project on large Wind Turbines
- Blade design for 10MW machine
- Traditional methods vs CFD
The k-ω SST-γ model [1]

- The starting point for the model is the k-ω SST-γ-Reθ model [2] widely used in CFD.
- The formulation is reduced to only the γ-equation for the transition prediction.
- The correlations have been simplified and tunable coefficients are provide to match the required application.
- A strong characteristic of the LCTM concept is its flexibility and relatively straightforward implementation into practical CFD simulations.

The k-ω SST-γ model \([1]\)

\[
\frac{\partial (\rho \gamma)}{\partial t} + \frac{\partial (\rho U_j \gamma)}{\partial x_j} = P_\gamma - E_\gamma + \frac{\partial}{\partial x_j} \left[ \left( \mu + \frac{\mu_t}{\sigma_\gamma} \right) \frac{\partial \gamma}{\partial x_j} \right]
\]

\[
P_\gamma = F_{\text{length}} \rho S \gamma (1 - \gamma) \quad F_{\text{onset}}
\]

\[
E_\gamma = c_{a2} \rho \Omega \gamma F_{\text{turb}} (c_{e2} \gamma - 1)
\]

\[
F_{\text{onset1}} = \frac{R_{\text{e}}}{2.2 R_{\theta c}}, \quad F_{\text{onset2}} = \min \left( F_{\text{onset1}}, 2.0 \right)
\]

\[
F_{\text{onset3}} = \max \left( 1 - \left( \frac{R_T}{3.5} \right)^3, 0 \right), \quad F_{\text{onset}} = \max \left( F_{\text{onset2}} - F_{\text{onset3}}, 0 \right)
\]

\[
F_{\text{turb}} = e^{-\left( \frac{T_T}{T} \right)^{\gamma}}, \quad R_T = \frac{\rho k}{\mu}, \quad R_{\text{e}} = \frac{\rho d_f^2 S}{\mu}, \quad R_{\theta c} = f \left( T u_L, \lambda_{\theta L} \right)
\]

when \(F_{\text{onset}} > 1\) the production of \(\gamma\) is activated.

The k-ω SST-γ model

\[
R_{\theta c} (T u_L, \lambda_{\theta L}) = C_{TU1} + C_{TU2} \exp \left[ -C_{TU3} T u_L \ F_{\text{PG}} (\lambda_{\theta L}) \right]
\]

\[
C_{TU1} = 100.0, \quad C_{TU2} = 1000.0, \quad C_{TU3} = 1.0
\]

- \(C_{TU1}\) defines the minimal value of the critical \(R_{\theta c}\) number.
- \(C_{TU1} + C_{TU2}\) defines its maximum value.
- \(C_{TU3}\) controls how fast \(R_{\theta c}\) decreases as the turbulence intensity increases.
- \(F_{\text{PG}} (\lambda_{\theta L})\) is used to include the stream-wise pressure gradient in the transition onset and it is purely empirical.
Model Calibration

Only natural transition, i.e. $Tu < 1\%$, is considered.

The first modification proposed consists in

$$C_{TU1} = 100; C_{TU2} = 1000 \quad \rightarrow \quad C_{TU1} = 163; C_{TU2} = 1002.25$$

so that equation $Re_{th}(Tu, \lambda)$ exactly matches the Abu-Ghannam and Shaw correlation for zero pressure gradient, i.e. $\lambda = 0$, where the minimum value of 163.0 for the critical is in accordance with the Tollmien-Schlichting limit of stability \[1\],


Modifying $C_{TU1}$ and $C_{TU2}$ is not enough
Model Calibration

Previous works in the literature for the $\gamma$-$Re_\theta$ model have observed that the constant employed in the ratio

$$F_{\text{onset}} = \frac{Re_\theta}{C_{\text{onset}}1 Re_\theta}$$

needs to be increased at higher Re$^{[1-2]}$.


An optimal $C_{\text{Onset}}$ can be found at various $Re$. A simple logarithmic curve fitting can be given to define $C_{\text{Onset}}$ as function of the $Re$ can be given.

$M = 0.1$
$Tu = 0.0816 \%$

The log. fit. leads to good agreement.
Model Calibration

Repeated at higher Tu levels.

Aerofoil | Mach | Re  | Tu  |
---------|------|-----|-----|
DU00-w-240 | 0.075 | 3 mil | 0.0864 |
DU00-w-240 | 0.080 | 15 mil | 0.3346 |

M = 0.075
Re = 3 mil
Tu = 0.2582 %
Results- Comparison with $e^N$ Methods

$M = 0.075$
$Re = 3 \text{ mil}$
$Tu = 0.2582 \%$

$M = 0.080$
$Re = 15 \text{ mil}$
$Tu = 0.3346 \%$
Results - Comparison with \( e^N \) Methods

\[
\begin{align*}
M = 0.080 \\
Re = 15 \text{ mil} \\
Tu = 0.3346 \% 
\end{align*}
\]

6:1 Prolate Spheroid

Re: 4.2 million (based
\( M_\infty = 0.15 \)
Grid \approx 40 \text{ million cells}
Pitch of 20 deg.

Study the effect of the turbulence models:

- Fully turbulent flow: \textbf{\( k-\omega \) SST}
- Fully turbulent flow: \textbf{\textit{EARSM}} (Explicit Algebraic Stress Model)
- Transitional flow: \textit{\( k-\omega-Y-Re_\theta \)}
Algebraic Reynolds Stress Model

Modified linear eddy-viscosity model
Implemented as a correction to existing models in the solver

\[
P = \tau_{ij} \frac{\partial u_i}{\partial x_j}
\]

\[
\tau_{ij} = 2\mu \left( \frac{1}{2} \left( \frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right) - \frac{1}{3} \frac{\partial u_k}{\partial x_k} \delta_{ij} \right) - \frac{2}{3} \rho k \delta_{ij} - \rho \frac{\partial \omega}{\partial x_j}
\]

\[
\mu_l = \frac{C_{\mu} \rho k}{\omega^2}
\]

\[
a^{(ex)}_{ij} = \beta_l \left( \Omega_{ij} \Omega_{ij}^* - \frac{1}{3} I_{ij} \Omega_{ii}^* \right) + \beta_4 \left( S_{ii} \Omega_{ij}^* - \Omega_{ij}^* S_{ii} \right) + \beta_6 \left( S_{ii} \Omega_{ij}^* + \Omega_{ij}^* S_{ii} - \frac{1}{3} I_{ij} \right)
\]

\[
+ \beta_8 \left( \Omega_{ij}^* S_{ii} + \Omega_{ij}^* S_{ii}^* - \Omega_{ij}^* S_{ii}^* \right)
\]

\[
C_{\mu} = -\frac{1}{2} \left( \beta_1 + I_{ij} \beta_6 \right)
\]

Study of Turbulence Models (I)

Experiments:
Study of Turbulence Models (ii)

Experiments:

Study of Turbulence Models (iii)

Contours of $Q$-criterion

K-ω SST

EARSM

Effect of Transition
Higher Order Schemes

- Huge volume of work on high order schemes published in the literature.
- Theoretical developments stemming from FD methods, spectral methods (limitations on the grid), DG methods (FE methods).
- Good but somehow fewer works on finite volume methods.
  - Review of J. A. Ekaterinaris (PIAS, vol 41 2005) is a good starting point
  - Huge volume of earlier work within the rotorcraft community
    - G. Wang, L.N. Sankar, Low Dispersion FV schemes, 2000
    - D. Ghosh D., J. Baeder, CRWENO schemes, 2013
  - Too many options available each with pros/cons

Selection of Scheme

- In order of priority:
  - Finite volume method
  - High-order scheme with low dissipation flux-difference splitting
  - High-order spatial accuracy (up to 4th-order) achieved using high-order correction terms through a successive differentiation
  - Applicable to any kind of mesh
  - Loss of robustness
  - In parallel, minimum information exchanged between processors to a minimum – but something must change!
  - Some CPU and memory overhead could be tolerated
- In conclusion:
High-Order Formulation in HMB3

➢ To obtain an upwind scheme, numerical fluxes on the faces of each control volume are computed using e.g. Roe’s or Osher’s approximate Riemann solvers.

\[ F_{j+1/2} = f(F_{j+1}, F_{j+1/2}) \]

➢ The variable-extrapolation formulation MUSCL (Monotone Upstream-Centred Scheme for Conservation Laws) is used to compute L and R of each face.

<table>
<thead>
<tr>
<th>(k)</th>
<th>Scheme</th>
<th>(F_{j+1/2}^L = F_j + \psi \left[ k \frac{F_{j+1} - F_j}{2} + (1 - k)\nabla F_j \cdot r_j \right] )</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>2\textsuperscript{nd} upwind</td>
<td>(F_{j+1} = F_{j+1/2} - \psi \left[ k \frac{F_{j} - F_{j+1}}{2} + (1 - k)\nabla F_{j+1} \cdot r_{j+1} \right] )</td>
</tr>
<tr>
<td>1/3</td>
<td>3\textsuperscript{rd} upwind</td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>2\textsuperscript{nd} central</td>
<td></td>
</tr>
</tbody>
</table>

High-Order Formulation in HMB3

➢ Following Yang et al. \[5\], the proposed 4\textsuperscript{th} order MUSCL scheme is written in a similar fashion:
High-Order Formulation in HMB3

➢ To compute the first and second derivatives, one can use the Green’s theorem or the least-squares approach.

➢ For curvilinear meshes, the least-square method is not stable. Therefore, we use the Green’s theorem to compute derivatives.

\[ \frac{\partial^2 Q}{\partial x^2} = \frac{1}{\Omega} \int_{\Omega} \frac{\partial Q}{\partial x} \, dx; \quad \frac{\partial^2 Q}{\partial x \partial y} = \frac{1}{\Omega} \int_{\Omega} \frac{\partial Q}{\partial x} \, dy; \quad \frac{\partial^2 Q}{\partial y^2} = \frac{1}{\Omega} \int_{\Omega} \frac{\partial Q}{\partial y} \, dy \]

Cost

- CPU overhead

➢ The order-of-accuracy of the proposed scheme can be derived for 1-D uniform grid.

➢ We can approximate the derivatives at the cell centre as:

\[ \int_{x-\frac{1}{2}}^{x+\frac{1}{2}} \frac{\partial F}{\partial x} \, dx \approx F_{x+\frac{1}{2}} - F_{x-\frac{1}{2}} \]

\[ = \frac{1 + k_1 F_{j+1} + \frac{7 + 8k_1 - 3k_2}{32} F_{j+1} + \frac{11 - 12k_1 + k_2}{16} F_j}{16} + \frac{-19 + 12k_1 + k_2}{32} F_{j-1} + \frac{9 - 8k_1 - 3k_2}{32} F_{j-2} + \frac{-1 + k_2}{32} F_{j-3} \]

\[ = F_j \Delta x + \frac{1 + 6k_1}{24} F_j \Delta x^3 + \frac{1 - 2k_1 + k_2}{16} F_j \Delta x^4 + O(\Delta x^5) \]

Expansion

➢ By setting \( k_1 = -1/6 \) and \( k_2 = -4/3 \), the interpolation is fourth-order accurate.

➢ A small amount of dissipation \( \delta \) can be introduced to reduce spurious oscillations and at the same time maintain the high-order accuracy by setting \( k_2 \) to \( k_2 + \delta \).
Continuing with the formal analysis of the scheme the dispersion-dissipation properties had to be looked at.

Dissipation and dispersion errors are quantified as function of the grid wavenumber $\omega \Delta x$ for the optimal values of $\kappa_1$ and $\kappa_2$.

If the flux function is assumed to be a periodic sinusoidal of the grid wavenumber $\omega \Delta x$, the Fourier transformation of the equation presented in the previous slide is expressed as:

$$\omega \Delta x = \frac{45 - 16k_1 - 5k_2}{32} \sin(\omega \Delta x) + \frac{-8 + 8k_1 + 4k_2}{32} \sin(2\omega \Delta x) + \frac{1 - k_2}{32} \sin(3\omega \Delta x)$$

Fundamental equation in the frequency domain

This technique allows us to quantify the dissipation and dispersion properties.
S-76 Helicopter Blade in Hover

Medium/Coarse ~ 50%

- MUSCL-2 (foreground)
- MUSCL-4 (background)

RANS equations
- $k - \omega$ SST

$M_{\text{tip}}$ 0.65
$Re_{\text{tip}}$ $1.18 \times 10^6$
$
\theta_{75}$ 6.5°, 7.5°, 9.5°

Structured grid
7 million cells per blade

Unstructured grid
61.3 million cells per blade
This section demonstrates the performance of the MUSCL-4 scheme when used with chimera grids for a three-dimensional tiltrotor flow.

- This highly loaded rotor produces strong wakes, thus the resolution of which may benefit from a high-order scheme.
Experiments were carried out by Felker [12], Light [13] and Betzina [14].

Experiments show a maximum discrepancy of 4 counts of FoM.

The key conclusion is that that difference between schemes is smaller than what can be measured in the WT XV-15 Tiltrotor Blade

<table>
<thead>
<tr>
<th>Case</th>
<th>FoM</th>
<th>ΔFoM [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Experiment [15]</td>
<td>0.760</td>
<td>-</td>
</tr>
<tr>
<td>Coarse Mesh (MUSCL-2)</td>
<td>0.775</td>
<td>1.97%</td>
</tr>
<tr>
<td>Medium Mesh (MUSCL-2)</td>
<td>0.768</td>
<td>1.05%</td>
</tr>
<tr>
<td>Coarse Mesh (MUSCL-4)</td>
<td>0.765</td>
<td>0.65%</td>
</tr>
</tbody>
</table>

It is clear that a good agreement is found between HMB3 and experimental data on the prediction of the FoM with modest computer resources.
MUSCL-4 scheme preserves much better the helical vortex filaments that trail from each of the tip blade, and the wake sheets trailed along the trailing edge of XV-15 Tiltrotor Blade.

MUSCL-2 solution  MUSCL-4 solution

UH-60A Rotor in Forward Flight
Vorticity contours at the plane x/c=1.

MUSCL-2 solution

MUSCL-4 solution

Vorticity contours at the plane x/c=2.

MUSCL-2 solution

MUSCL-4 solution
Examples of Hybrid Air Vehicles

Application

Airlander 10, Cardington, August 2016
CFD Analysis of the HAV Propulsor

- Validation with experimental data
- Analysis of the duct effect
- Analysis of the blade twist effect
- Optimisation of the blade twist and of the duct

<table>
<thead>
<tr>
<th>Flow equations</th>
<th>RANS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Turbulence model</td>
<td>$k$-$\omega$ SST</td>
</tr>
<tr>
<td>Blade angle</td>
<td>$14.1^\circ$</td>
</tr>
<tr>
<td>RPM</td>
<td>470-1880</td>
</tr>
<tr>
<td>Re (based on $V_{tip}$ and blade chord)</td>
<td>1-3 million</td>
</tr>
<tr>
<td>Advance ratio $J=V_{inf}/(nD)$</td>
<td>0 (static test)</td>
</tr>
</tbody>
</table>

Validation of the CFD model (cont.)

1171 blocks
34M cells

Axial velocity distribution
Validation of the CFD model (cont.)

Axial velocity distribution

921 blocks
31M cells

Validation of the CFD model (cont.)

Test rig for Static Tests
*Hybrid Air Vehicles Ltd*

Static tests – CFD vs Exp.
Effect of the Duct and Twist Distribution

<table>
<thead>
<tr>
<th>Flow equations</th>
<th>RANS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Turbulence model</td>
<td>k-ω SST</td>
</tr>
<tr>
<td>Blade angle</td>
<td>14.1-21.2º</td>
</tr>
<tr>
<td>RPM</td>
<td>1880</td>
</tr>
<tr>
<td>Re (based on ( V_{\text{tip}} ) and blade chord)</td>
<td>3 million</td>
</tr>
<tr>
<td>Advance ratio ( J=V_{\text{inf}}/(nD) )</td>
<td>0 – 0.136 – 0.540</td>
</tr>
<tr>
<td>( V_{\text{inf}} )</td>
<td>0 – 20 – 80 knots</td>
</tr>
</tbody>
</table>

Two sets of blades | Low-twist / High-twist

Effect of the Duct

Ducted propeller
- 324 blocks
- 9.8M cells

Free propeller
- 256 blocks
- 9.3M cells
Effect of the Duct (cont.)

Ducted propeller +40% efficiency
Free propeller

Cruise speed (20 knots, J=0.136)

Effect of the Duct (cont.)

Ducted propeller Same efficiency
Free propeller

Dash speed (80 knots, J=0.540)
Effect of the Blade Twist

High-twist blade

<table>
<thead>
<tr>
<th>J</th>
<th>β</th>
<th>C_T</th>
<th>C_P</th>
<th>η</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.136</td>
<td>17.2°</td>
<td>0.190</td>
<td>0.075</td>
<td>0.345</td>
</tr>
<tr>
<td>0.540</td>
<td>17.2°</td>
<td>0.048</td>
<td>0.030</td>
<td>0.850</td>
</tr>
</tbody>
</table>

Low-twist blade

<table>
<thead>
<tr>
<th>J</th>
<th>β</th>
<th>C_T</th>
<th>C_P</th>
<th>η</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.136</td>
<td>17.2°</td>
<td>0.190</td>
<td>0.077</td>
<td>0.330 ( -4% )</td>
</tr>
<tr>
<td>0.540</td>
<td>17.2°</td>
<td>0.036</td>
<td>0.029</td>
<td>0.680 ( -20% )</td>
</tr>
</tbody>
</table>

Blade Twist Optimisation

Flow equations | RANS
Turbulence model | k-ω SST
Blade angle | 17.2°
RPM | 1880
Re (based on V_{tip} and blade chord) | 3 million
Advance ratio J=V_{inf}/(nD) | 0.136
V_{inf} | 20 knots

Baseline

<table>
<thead>
<tr>
<th>C_T</th>
<th>0.190</th>
</tr>
</thead>
<tbody>
<tr>
<td>C_P</td>
<td>0.075</td>
</tr>
<tr>
<td>η = J C_T / C_P</td>
<td>0.345</td>
</tr>
</tbody>
</table>
Blade Twist Optimisation (cont.)

Objective

Maximise $C_T$ subject to:
$C_P < 0.075$

Parametrisation:

- Bernstein polynomial base (7 coefficients) for the twist perturbation of the blade
- Twist variation limited to $\pm 5^\circ$ at every blade section

+1% efficiency

Blade and Duct Optimisation

Objective

Maximise $C_T$ subject to:
$C_P < 0.075$

Parametrisation:

- Bernstein polynomial base (7 coefficients) for the twist perturbation of the blade
- Bernstein polynomial base (16 coefficients) for the diffuser shape
- Length and exit radius of the duct
Blade and Duct Optimisation (cont.)

Baseline

Optimised

+2% efficiency

Model scale
Re: 3 million (based on L)
\( U_\infty = 40 \text{m/s} \)

Configurations:
- HAV(1): Hull
- HAV(2): + Fins
- HAV(3): + LERX
- HAV(4): + Strakes
- HAV(5): + Skids, gondola, bowthruster and duct engines

Grids ≈ 10 million cells

Study the effect of:
- Changes in pitch
- Changes in yaw
- Combined pitch and yaw

Hybrid Air Vehicle
Model scale
Re: 3 million (based on L)
$U_\infty = 40\text{m/s}$

Configurations:
- HAV(1): Hull
- HAV(2): + Fins
- HAV(3): + LERX
- HAV(4): + Strakes
- HAV(5): + Skids, gondola, bowthruster and duct engines

Grids ≈ 10 million cells

Study the effect of:
- Changes in pitch
- Changes in yaw
- Combined pitch and yaw

Comparison with Experiments (ii)

$C_{L0}$: $C_L$ of bare hull at $\alpha = 0\text{deg}$.
$C_{D0}$: $C_D$ of bare hull at $\alpha = 0\text{deg}$. 
Comparison with Experiments (ii)

**CL\textsubscript{0}**: CL of bare hull at $\alpha = 0$ deg.
**CD\textsubscript{0}**: CD of bare hull at $\alpha = 0$ deg.

Comparison with Experiments (iii)

**CL\textsubscript{0}**: CL of bare hull at $\alpha = 0$ deg.
**CD\textsubscript{0}**: CD of bare hull at $\alpha = 0$ deg.
Effect of Pitch Angle – Flow Vis. (iii)

\[ \alpha = 24 \text{ deg.} \]
\[ \beta = 0 \text{ deg.} \]

Effect of Yaw Angle – Flow Vis. (i)

\[ \alpha = 0 \text{ deg.} \]
\[ \beta = 10 \text{ deg.} \]
Effect of Yaw Angle – Flow Vis. (ii)

\( \alpha = 0 \) deg.
\( \beta = 20 \) deg.

Effect of Combined Pitch-Yaw – Flow Vis. (i)

\( \alpha = 10 \) deg.
\( \beta = 10 \) deg.
Effect of Combined Pitch-Yaw – Flow Vis. (ii)

Visible Differences!

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XV-15 Rotor Geometry

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Number of blades</td>
<td>3</td>
</tr>
<tr>
<td>Rotor radius</td>
<td>150 inches</td>
</tr>
<tr>
<td>Reference blade chord</td>
<td>14 inches</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>10.71</td>
</tr>
<tr>
<td>Rotor solidity</td>
<td>0.089</td>
</tr>
<tr>
<td>Linear twist angle</td>
<td>-40.25 degrees</td>
</tr>
</tbody>
</table>

Grid Size | Background Size | Blade Size | Wall distance |
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Coarse</td>
<td>6.2 M</td>
<td>2.6 M</td>
<td>3.6 M</td>
</tr>
<tr>
<td>Medium</td>
<td>9.6 M</td>
<td>6.0 M</td>
<td>3.6 M</td>
</tr>
</tbody>
</table>

Topology of the body-fitted mesh:
- C-H topology.

Two CFD grids were generated for a grid resolution study:
- The blade grid had 3.6 million cells, to ensure good resolution of the tip vortex.
- Refinement of the background mesh to resolve the rotor wake
- Coarse and medium grids had 6.2 million and 9.6 million cells, respectively.

Cell ratio Coarse/Medium ~ 50 %
Test and Computation Conditions [1/2]

<table>
<thead>
<tr>
<th></th>
<th>Helicopter Mode</th>
<th>Aeroplane Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td>Blade-tip Mach number, $M_{tip}$</td>
<td>0.69</td>
<td>0.54</td>
</tr>
<tr>
<td>Freestream Mach number, $M_{\infty}$</td>
<td>0</td>
<td>0.18</td>
</tr>
<tr>
<td>Climb ratio, $\mu$</td>
<td>0</td>
<td>0.337</td>
</tr>
<tr>
<td>Reynolds number, $Re$ (*)</td>
<td>$4.95 \cdot 10^6$</td>
<td>$4.50 \cdot 10^6$</td>
</tr>
<tr>
<td>Blade pitch angle, $\theta_{75}$</td>
<td>3°, 5°, 7°, 10°, and 13°</td>
<td>26°, 27°, 28°, and 28.8°</td>
</tr>
<tr>
<td>Grid employed</td>
<td>Coarse and Medium</td>
<td>Coarse and Medium</td>
</tr>
<tr>
<td>Turbulence model</td>
<td>$\kappa$-$\omega$ SST$^{[12]}$</td>
<td>$\kappa$-$\omega$ SST</td>
</tr>
</tbody>
</table>

(*) The Reynolds number was based on the reference blade chord of 14 inches and on the blade-tip speed.

XV-15 Hover Mode [1/6]

- Experiments show a maximum scatter of 4 counts of FoM.
- CFD results present an excellent agreement with the test data of Betzina$^{[3]}$.
- The scatter between experiments is larger than between CFD data.
XV-15 Aeroplane mode [1/3]

➢ The indicator of the rotor efficiency is the propeller propulsive efficiency:

\[ \eta = \frac{C_T V_a}{C_Q V_{tip}} \]

➢ The experimental data was performed on a propeller test rig in the NASA 40-by-80 Foot Wind Tunnel\(^{[17]}\).

➢ The effect of the mesh density is noticeable for all collective pitch angles.

XV-15 Aeroplane mode [1/3]

➢ Flowfield visualisation of the rotor wake for the full-scale XV-15 in aeroplane mode using the \( Q \) criterion\(^{[18]}\).

➢ First and second passage of the vortex are captured.
XV-15 Transition Model [1/5]

➢ The effect transitional flow is investigated in predicting the FoM and $C_F$. In this regard, we use the $k$-$\omega$ SST-$\gamma$ transition model.[19]

➢ Comparison with the solution obtained with the fully-turbulent $k$-$\omega$ SST model and experimental data of Wadcock[4] is presented.

➢ Experiments revealed the presence of natural transition on the blade surface for all collective pitch angles.

➢ Skin friction coefficients were measured at five radial stations along the span of the blade.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Blade-tip Mach number, $M_{tip}$</td>
<td>0.69</td>
</tr>
<tr>
<td>RPM</td>
<td>587.37</td>
</tr>
<tr>
<td>Collective pitch angles, $\theta_{75}$</td>
<td>$3^\circ$ and $10^\circ$</td>
</tr>
</tbody>
</table>

XV-15 Transition Model [2/5]

➢ The transition onset is indicated with a peak on the $C_F$.

➢ The $k$-$\omega$ SST-$\gamma$ captures very well the transition onset and $C_F$.

➢ After the transitional onset, FT solutions correctly predict the $C_F$.

$\theta_{75} = 3^\circ$
XV-15 Transition Model [3/5]

- The onset locations move closer to the LE.
- The \( \kappa-\omega \) SST-\( \gamma \) show a good agreement with the experiments.

\[ \theta_{75} = 10^\circ \]

XV-15 Transition Model [4/5]

- \( \kappa-\omega \) SST, \( \theta_{75} = 3^\circ \)
- \( \kappa-\omega \) SST-\( \gamma \), \( \theta_{75} = 3^\circ \)

Overview of the \( C_f \)

- \( \kappa-\omega \) SST, \( \theta_{75} = 10^\circ \)
- \( \kappa-\omega \) SST-\( \gamma \), \( \theta_{75} = 10^\circ \)
XV-15 Aerodynamic Optimisation

1) The blade twist was optimised first:
   - Helicopter Mode
   - Aeroplane Mode

2) The twist, chord, and sweep distributions were optimized in:
   - Aeroplane Mode

3) Multi-point aerodynamic optimisation is attempted for the twist

<table>
<thead>
<tr>
<th>Helicopter</th>
<th>Propeller</th>
</tr>
</thead>
<tbody>
<tr>
<td>M_{tip}</td>
<td>0.69</td>
</tr>
<tr>
<td>M_{∞}</td>
<td>0</td>
</tr>
<tr>
<td>θ_{75} (deg)</td>
<td>10°</td>
</tr>
</tbody>
</table>

Implicit adjoint method

Effect of the Twist – Helicopter Mode

➢ The twist was optimised at two conditions, at lower thrust and within the operational range.

<table>
<thead>
<tr>
<th>θ_{75} (deg)</th>
<th>ΔFoM (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>+3.08</td>
</tr>
<tr>
<td>10</td>
<td>+2.19</td>
</tr>
</tbody>
</table>
Effect of the Twist – Helicopter Mode

- Local AoA is decreased inboard.
- Non-linear behaviour is found.
- Maximum reduction of the AoA at the tip region.

Effect of the Twist – Aeroplane Mode

- Linear behaviour is found.
- Maximum reduction of the AoA at the tip.
Effect of the Twist – Aeroplane Mode

<table>
<thead>
<tr>
<th></th>
<th>Baseline</th>
<th>Optimised</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_T$</td>
<td>0.00305</td>
<td>0.00307</td>
</tr>
<tr>
<td>$C_Q$</td>
<td>0.00283</td>
<td>0.00267</td>
</tr>
<tr>
<td>$\eta$</td>
<td>0.819</td>
<td>0.873</td>
</tr>
</tbody>
</table>

Propeller efficiency is increased by 6.61%
Effect of the Twist, Chord, Sweep – Aeroplane Mode

<table>
<thead>
<tr>
<th>Baseline</th>
<th>Optimised</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_T$</td>
<td>0.00305</td>
</tr>
<tr>
<td>$C_Q$</td>
<td>0.00283</td>
</tr>
<tr>
<td>$\eta$</td>
<td>0.819</td>
</tr>
</tbody>
</table>

Propeller efficiency is increased by 8.05%

Breakdown of the efficiency

Multi-Point Aerodynamic Optimisation

Formally, the problem reads:

$$\text{Minimise } \sum_{i=1}^{N} w_i C_{P_i} \text{ subject to } C_{T_i} = C_{T_{ip}}, \quad i = 1, \ldots, N$$

$w_1 = 1/3$ Helicopter mode

$w_2 = 2/3$ Aeroplane mode

<table>
<thead>
<tr>
<th></th>
<th>Helicopter</th>
<th>Propeller</th>
</tr>
</thead>
<tbody>
<tr>
<td>$M_{ip}$</td>
<td>0.69</td>
<td>0.60</td>
</tr>
<tr>
<td>$M_\infty$</td>
<td>0</td>
<td>0.45</td>
</tr>
<tr>
<td>$\Theta_{75}$ (deg)</td>
<td>10</td>
<td>47</td>
</tr>
</tbody>
</table>
Multi-Point Aerodynamic Optimisation

<table>
<thead>
<tr>
<th></th>
<th>Baseline</th>
<th>Optimised</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_T$</td>
<td>0.00909</td>
<td>0.00908</td>
</tr>
<tr>
<td>$C_Q$</td>
<td>0.000791</td>
<td>0.000783</td>
</tr>
<tr>
<td>$\eta$</td>
<td>0.775</td>
<td>0.780</td>
</tr>
</tbody>
</table>

Propeller efficiency is increased by 2.68%

Motivation and Objectives

- Aviation main target is to develop aircraft with lower environmental impact\(^1\), regarding fuel burnt as well as acoustic emissions
- Turboprops are the best choice for short and medium range flights in terms of fuel efficiency but
  a) their noise emissions are still higher than those required from future certification\(^1\)
  b) propeller noise characteristics perceived as annoying from passengers

Aim: Reducing and/or modifying the acoustic spectra generated from the whole propulsion system of a turboprop aircraft
- Propeller blades and hubs design
- Options for propeller installation on the aircraft


\(^*\) from “Handbook of Engineering Acoustics”, Muller and Moser (2012)
1. CO: co-rotating propellers (both propellers clockwise as viewed from the rear, as conventional twin-engined civil aircraft)

2. CNTI: counter-rotating propellers top-in (left engine clockwise and right counter-clockwise as viewed from the rear)

3. CNTO: counter-rotating propellers top-out (right engine clockwise and left counter-clockwise as viewed from the rear, aerodynamic benefic layout – eg. V22)

Reference cases

A. CLEAN aircraft (Steady state computation applying symmetry boundary conditions)

B. ISOLATED propeller in axial flight conditions (single blade simulation)
Computational Grids

- **Grid Strategy**
  - Matched multi-block structured grid for the airplane (nacelle, wing, fuselage)
  - Propellers inserted in the full grid using sliding plane technique
  - Cartesian background grid extended until the far-field added with the chimera method
  - Preparation of half of the airplane mesh and one-blade mesh so to have a grid perfectly symmetric
  - Numerical probes included in the simulations to record pressure-time history

- **Dimensions**
  - Propeller: 2 x ~16.5 million cells
  - Aircraft: ~132 million cells
  - ~170 million cells in total (13326 blocks)

- **Computational geometry**
  - Twin-engined turboprop with standard commercial high-wing design and a capacity of 70-80 passengers, similar to the ATR72 and Bombardier Dash 8 series
  - No tail included

- **URANS**
  - Computations closed by the k-ω SST turbulence model with 360 steps resolved per propeller revolution

Aerodynamic Results: Flow-Field Visualisations

Instantaneous (6th propeller revolution, $\zeta = 0.25$) wake visualisation via iso-surfaces of Q-criteria (non-dimensional value of 0.005), coloured by axial velocity

- Propeller wake resolved by the adopted mesh resolution up to about the aircraft tail, allowing to capture the interaction with the airframe
- Different nature of interaction in the case of top-in or top-out rotating propellers visible
- Vortices generated by the back of the nacelles and the aft tail inclination also represented
- Flow-field in the case of counter-rotating propellers symmetric as expected
➢ Wing loaded in different ways depending on the propeller rotational direction
➢ Large depression area on the inboard part of the wing extended up to the fuselage junction in the case of inboard-up rotating propeller
➢ The expected symmetry of the counter-rotating options yields to natural equilibrium of the forces in the lateral direction and the rolling moment, whereas a significant unbalance is registered for the CO layout.
Aerodynamic Results: Wing Loads

Averaged normal wing span-wise loading distribution (6th propeller revolution, 90 deg)

- Propeller effect on the wing loading captured by HMB3
- Differences up to mid-span for same propeller rotation but different configuration

Clean aircraft pressure distribution (RANS results) – loading distortion due to the nacelle

- Top-OUT scenario
- Top-IN scenario

- Averaged total aircraft lift w.r.t. CO layout: -1.16% for CNTI, +1.19% for CNTO
- The CNTO installation option shows the best aerodynamic efficiency (higher L/D ratio)

Aerodynamic Results: Propeller Loading

Thrust loading for the port propeller in the case of CO layout

- Thrust and torque coefficients of the reference blade as a function of the blade azimuthal position over a revolution: CO configuration – results scaled w.r.t. the isolated axial flight propeller values

- The interaction propeller-airframe has a mutual character and also the propeller is affected by the presence of wing and nacelle
- Thrust and torque blade coefficients vary during a propeller revolution, showing the maximum deviations from the axial flight values in correspondence of the blade passage in front of the wing
- Loading conditions and efficiency of the propeller are not significantly different in the cases of inboard-up and inboard-down propeller rotational direction

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Acoustic Results: Acoustic Field Visualisations

Instantaneous (6th propeller revolution, $\zeta = 0.25$) unsteady pressure visualisation: transversal plane 1

radius behind the propeller plane

- Acoustic field symmetric as expected for counter-rotating propellers
- Pressure perturbations generated by the interaction of the blade tip sound waves and the wing significantly larger on the up-stroking blade side, where the wing is also more loaded
Acoustic Results: Acoustic Field Visualisations

Instantaneous (6th propeller revolution, $\zeta = 0.25$) unsteady pressure visualisation: longitudinal plane at spinner height

- Pressure perturbations of propeller blades tip preserved down-stream up to the aircraft tail
- Sound waves propagation physics well represented

Acoustic Results: Acoustic Field Visualisations

- Pressure perturbations in the area between inboard propeller blades, fuselage and wing significantly larger in the case of inboard-up propeller rotational direction
- In the CO case a second front of up-stream propagating waves visible on the starboard aircraft side

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Again, the symmetry in the case of counter-rotating propellers installation can be seen.

On the aircraft fuselage significant pressure perturbations are seen in proximity of the propeller plane, from about one propeller radius up-stream up to the wing trailing edge station.

Pressure perturbations due to the impact of the propeller wake on the wing clearly visible, with differences between up-wash and down-wash propeller sides.

Pressure fluctuations due to the root blade vortices also observed.

Acoustic Results: Acoustic Field Visualisations

Instantaneous unsteady pressure visualisation on the aircraft fuselage (6th propeller revolution, \( \zeta = 0 \))

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Acoustic Results: External Noise Estimation

OSPL distribution along the fuselage azimuth at the propeller plane (data from numerical probes over the last full propeller revolution run)

- Differences between isolated and installed propeller predictions are important for both trends and sound levels
- Interferences between sound waves are seen in the unsteady pressure signals
Acoustic Results: Inboard-Up vs Inboard-Down Propellers

Unsteady pressure visualisation for the co-rotating layout: why is the starboard side (i.e. propeller rotating inboard-up) noisier than the port side (i.e. propeller rotating inboard-down)?

➢ Feedback loop mechanism!

Acoustic Results: Interior Noise Estimation

Unsteady pressure signals, input to the TF, are taken from the numerical probes included in the simulation at azimuthal locations corresponding to the available DLR TF measurements on the Fokker 50 aircraft.
Acoustic Results: Interior Noise Estimation

Unsteady pressure amplitude maps at the BPF, outside and inside the fuselage shell
(data from numerical probes over the last full propeller revolution run)

➢ Modifications of the pressure field through the fuselage structure evident
➢ Transmission losses non-uniform

Unsteady pressure amplitude maps at the BPF, outside and inside the fuselage shell
(data from numerical probes over the last full propeller revolution run)

➢ Amplitude of pressure fluctuations inside considerably smaller
➢ The co-rotating layout is the noisiest at this position, while the counter-rotating top-in configuration the quietest
➢ Small differences audible inside

0.4 m up-stream of the propeller plane
Motivation

Sea King landing on the deck of HMS Dumbarton Castle – Falkland Islands, 2007

Initialise Flight Mechanics

- Create Multi-Body Model of the Helicopter
- Ensure Consistency between CFD and Flight Mechanics
- Set-up problem to solve Differential Equations using Euler or 4th Order Runge-Kutta integration method in time
Calculate 6-DOF Linear Model

- Calculate state and control matrices based on 12 parameters and 4 pilot inputs.

\[ \dot{x} = f(x, u) \implies \delta \dot{x} = A \delta x + B \delta u \]

\[ x = (u, v, w, p, q, r, x_c, y_c, z_c, \Phi, \Theta, \Psi) \]

\[ u = (\Omega_0^{\text{main}}, \Omega_{1s}^{\text{main}}, \Omega_{1c}^{\text{main}}, \Omega_0^{\text{out}}) \]

\[ A = \left( \begin{array}{c} \frac{\partial \delta \dot{x}}{\partial x_j} \\ \frac{\partial \delta \dot{x}}{\partial u_j} \end{array} \right)_{1<j<12} \]

\[ B = \left( \begin{array}{c} \frac{\partial \delta \dot{x}}{\partial u_j} \\ \frac{\partial \delta \dot{x}}{\partial u_j} \end{array} \right)_{1<j<4} \]

- Centred Finite Differences is used to calculate partial derivatives. Source code differentiation

- System describes the Response of the helicopter state to changes in Attitude and Pilot Inputs

Grid Motion/Deformation

- Classic forward-flight simulations use a Wind-Tunnel Frame Of Reference

- Combination of Rigid-Body Motion and Grid Deformation for articulated rotor assembly

- Natural “earth-fixed” frame of reference

- The local grid velocity is computed using Finite Differences
Grid Motion/Deformation

- Classic forward-flight simulations use a Wind-Tunnel Frame Of Reference
- Combination of Rigid-Body Motion and Grid Deformation for articulated rotor assembly
- The local grid velocity is computed using Finite Differences

Wind-Tunnel Frame of Reference

Earth-fixed Frame of Reference
CFD vs. HFM Solutions

- Aircraft is **Trimmed** using HFM before starting the CFD calculation
- The **LQR pilot model** is used for piloted simulations, using the linear models generated with HFM.
- Maintain **10 m/s Forward-Flight**
  - HFM and CFD **Load predictions** agree well overall
  - Discrepancies in terms of **Rotor Torque**
  - Variation of **Pitching Moment** partially due to the influence of the fuselage
- Simulations performed:
  1. Fixed aircraft in free air and above ship
  2. Free response to a pilot input
  3. LQR piloted manoeuvre using HFM
  4. LQR piloted manoeuvre in free air using HFM+HMB
  5. LQR piloted manoeuvre using HFM+HMB in the vicinity of the ship
Ship-Helicopter Model

- Sea-King fuselage with 5-bladed main and tail rotors is used as test case.
- Two positions over the deck are tested.
- Chimera method is used to interface the ship and helicopter grids.

Coupled Calculations

- Isolated Sea King in forward flight
- Shipborne Sea King approaching the deck
- Shipborne Sea King before touchdown
Simulation of the Coupled Landing

Helicopter Response to a piloted descent using coupled HMB/HFM method. Target trajectory is a 10m descent in 4 seconds. Results are within standard acceptable error (ADS33). Ship wake included.
Simulation of the Coupled Landing

To look at:
• CPU-months
• Data storage
• Coarse mesh

Simulation of ADS-33 Manoeuvres

• Automatic differentiation has been applied to the flight mechanic HFM and HMB, to compute the helicopter linearised state space model and aerodynamics
• The state space model is used to implement a LQR + PI based autopilot
• Results for the slalom manoeuvre shown below
Summary

• Spending time to improve CFD tools using simple, canonical cases is a good way of de-risking industrial application of CFD

• Working between academia and industry is challenging and stimulating at the same time
  • Sets objectives for what methods should do and how accurate results should be

• There is great progress, but there is still mileage to go with:
  • Optimisation methods - passing the tool to practicing engineers
  • Turbulence – is it right to always use DES?
  • High order schemes – stability and efficiency
  • Direct noise prediction – better broadband estimates

• New challenges appear if manoeuvring flight is to be simulated with CFD
  • Software complexity and new computing platforms
  • Storage of solutions
  • Validation data!

• These are exciting times for CFD/CAA/CSD/FM/MDO and rotorcraft!

Thank you for your attention!