

Generation of a Reduced-Order Model of an Unmanned Combat Air Vehicle using Indicial Response Functions

Michel P.C. van Rooij

NLR – Royal Netherlands Aerospace Centre
Dept. of Flight Physics and Loads
Amsterdam, The Netherlands

Neal T. Frink and Brett R. Hiller

NASA Langley Research Center
Hampton, Virginia, 23681, USA

Mehdi Ghoreyshi

High Performance Computing Research Center, U.S. Air Force Academy,
Colorado, 80840, USA

Mark Voskuil

Netherlands Defence Academy, Faculty of Military Sciences
Den Helder, The Netherlands

SUMMARY

Within the NATO STO AVT-251 Task Group, a generic Unmanned Combat Air Vehicle (UCAV) planform was redesigned based on requirements derived from parts of the flight envelope of the defined mission. Because of the lambda-shape planform and associated flow phenomena including shocks and vortices, the aerodynamic design process relies heavily on high-fidelity predictions from computational fluid dynamics (CFD). These simulations enable accurate flight dynamics predictions, allowing for the identification of any potential performance issues early in the design process. Potentially, a complete flight dynamics model can be derived from CFD calculations provided that the underlying reduced order model (ROM) captures the underlying physics contained within the CFD results. This work investigates the creation of a nonlinear ROM using indicial response functions. The response functions are obtained using a grid motion approach that separates the effects of angle of attack and pitch rate. This approach is demonstrated using three different CFD codes from various organizations: ENSOLV at Royal Netherlands Aerospace Centre, USM3D at NASA Langley Research Center, and Kestrel at the United States Air Force Academy. ROM predictions were generated for a manoeuvre resembling a low-speed pull-up. The predictions are found to be sensitive to the quality of the steady state solutions from which step response calculations were started, as well as the convergence of the step responses themselves. Nevertheless, the indicial response method is shown to provide aerodynamic predictions with acceptable accuracy including transient effects, and as such is a promising method for computationally efficient flight dynamics predictions.

NOMENCLATURE

C_L	=	lift coefficient, $\equiv \text{Lift}/(q_{ref}S_{ref})$
C_N	=	normal force coefficient, $\equiv \text{normal force}/(q_{ref}S_{ref})$
$C_{m,p}$	=	pitching moment coefficient, $\equiv M_p/(q_{ref}S_{ref}L_{ref})$
c_{ref}	=	reference chord, m
M	=	Mach number
q	=	pitch rate, rad/s
q_{ref}	=	reference dynamic pressure, Pa

L_{ref}	=	reference length, m
Re	=	Reynolds number
s_{ref}	=	reference span, m
S_{ref}	=	reference area, m ²
$t, \Delta t$	=	time, incremental time step, s
Δt^*	=	characteristic time step, $\Delta t(V_\infty/c_{ref})$
V_∞	=	freestream velocity, m/s
α	=	angle of attack, deg
γ	=	climb angle, deg
θ	=	pitch angle, deg
ASG	=	Aerodynamic Shaping Group
AVT	=	Applied Vehicle Technology
CFD	=	Computational Fluid Dynamics
DLR	=	German Aerospace Center
LaRC	=	NASA Langley Research Center
MRP	=	Moment Reference Point
NASA	=	National Aeronautics and Space Administration
NATO	=	North Atlantic Treaty Organization
NLDA	=	Netherlands Defence Academy
NLR	=	Royal Netherlands Aerospace Centre
ROM	=	Reduced-Order Model(ing)
RTG	=	Research Task Group
STO	=	Science and Technology Organization
UCAV	=	Unmanned Combat Air Vehicle
UQ	=	Uncertainty Quantification
USAF	=	United States Air Force Academy

I. INTRODUCTION

The NATO STO AVT-251 Task Group on “MultiDisciplinary Design and Performance Assessment of Effective, Agile NATO Air Vehicles” was created in order to apply advanced numerical tools and collaborative approaches to the design of an Unmanned Combat Air Vehicle (UCAV) for a specific mission [1]. The Task Group was subdivided into several subgroups addressing a specific aspect of the design: Design Specification and Assessment Group (DSAG), Aerodynamic Shaping Group (ASG), Engine Integration Group (EIG), Control Concept Group (CCG), and Structural Concept Group (SCG).

The Aerodynamic Shaping Group was tasked with the application and development of advanced numerical methods for aerodynamic design and performance analysis of an agile UCAV [2]. Several studies were conducted by ASG members, encompassing various design approaches [3-10], as well as investigations into control devices [11], propulsion-airframe integration effects [12], uncertainty quantification [13], and reduced-order modeling (ROM).

Several motivations exist for the pursuit of reduced-order model (ROM) development: creation of a flight

mechanics model suitable for agile aircraft undergoing rapid manoeuvres, expanding the validity of the flight mechanics model to include loss of control, flight envelope prediction, and comprehensive evaluation of design iterations. These are typically areas for which the accuracy of high-fidelity CFD is required, but its cost is too high for practical use. The purpose of a ROM is to provide an accurate description of system behaviour with drastically reduced computational costs compared to the full-order model. Two general categories of ROM methods can be distinguished: parametric and nonparametric methods. Parametric methods assume a certain system modeling form for which the coefficients of the model are to be found. A well-known conventional ROM method is the stability-derivative method, which approximates aircraft flight dynamics using a Taylor series expansion of the dependent vehicle-state parameters. When unsteady, nonlinear aerodynamic problems are considered, many of the arising phenomena are analytically unresolved, resulting in inaccurate aerodynamic predictions. On the other hand, nonparametric methods pursue mappings between system inputs and outputs without explicit functional forms. This allows for prediction of responses for arbitrary inputs (within the model's boundaries) and inclusion of time-history effects. Examples are given by Ballhaus and Goorjian [5], Tobak et al. [6], Hall et al. [7], Goman et al. [8], and Zhou et al. [9].

Four approaches were investigated by ASG members: indicial response functions, system identification [14], multifidelity modeling [15] and neural networks. This paper presents the results obtained using indicial response functions. Four organizations participated in this study. The tested codes include ENSOLV at NLR, Kestrel at USAFA, and USM3D at NASA LaRC. The research into ROM development at NLR is supported by NLDA. Since the methodology and its basic applicability has been demonstrated in earlier studies [17-21], the purpose of the study within the group was to assess whether the formulation was general enough to be used with each participant's best practices in terms of CFD method and computational grid, and assess whether different approaches impacted the obtained model. It therefore resembles more of a "blind test" of ROM creation using indicial response functions.

The remainder of this paper is organized as follows: Section II introduces the reduced-order model formulation. Section III presents the aircraft configuration. Section IV defines the test manoeuvre of interest. Section V reviews the various computational grids and flow solvers as part of the collaborative effort. Finally, the modeling and simulation results are presented in Section VI.

II. INDICIAL RESPONSE ROM FORMULATION

An indicial response is the transient response of a system to a unit step change of a system's input. An example is the change in lift coefficient, C_L , due to a step change in angle of attack α , or pitch rate q . Given indicial response functions describing the change in lift due to step changes in α and q , Duhamel's superposition integral is used to obtain the unsteady lift coefficient at time t :

$$C_L(t) = C_{L0} + \frac{d}{dt} \left(\int_0^t \alpha(\tau) C_{L\alpha}(t - \tau) d\tau \right) + \frac{d}{dt} \left(\int_0^t q(\tau) C_{Lq}(t - \tau) d\tau \right) \quad (1)$$

C_{L0} denotes the zero-angle of attack lift coefficient that is obtained from a steady-state simulation. Equation (1) is limited to linear flow behavior as the indicial response functions with respect to angle of attack and pitch rate do not depend on flow conditions. Tobak [16] extended this model to approximate non-linear flow behavior. This is done by calculating indicial response functions at different angles of attack and implementing them in Duhamel's integrals. The resulting equation then becomes:

$$C_L(t) = C_{L0} + \frac{d}{dt} \left(\int_0^t \alpha(\tau) C_{L\alpha}(t - \tau, \alpha) d\tau \right) + \frac{d}{dt} \left(\int_0^t q(\tau) C_{Lq}(t - \tau, \alpha) d\tau \right) \quad (2)$$

Similarly, this formulation can be expanded further by including dependencies on Mach number, side slip and roll and yaw rates. Here, only longitudinal motions at constant Mach number are considered. The pitching moment $C_{m,p}$ prediction is given by the following:

$$C_{m,p}(t) = C_{m,p0} + \frac{d}{dt} \left(\int_0^t \alpha(\tau) C_{m,p\alpha}(t - \tau, \alpha) d\tau \right) + \frac{d}{dt} \left(\int_0^t q(\tau) C_{m,pq}(t - \tau, \alpha) d\tau \right) \quad (3)$$

Equations 2 and 3 form the basis for calculation of the ROM predictions. The underlying physics of the full-order model is captured by the indicial step response functions. These functions are then used to create a ROM capable of predicting the unsteady aerodynamic force and moment coefficients of an aircraft during flight.

The indicial response functions are obtained directly from CFD using a grid motion technique proposed by Ghoreyshi et al. [17]. This technique not only allows for a straightforward means to calculate step responses, but also makes possible the uncoupling of input parameters, e.g. the angle of attack and pitch rate. For example, a positive step change in angle of attack is obtained by translating the grid downwards and aft at a constant rate. A unit step change in pitch rate is achieved by simultaneously moving and rotating the grid. This approach has been used successfully to calculate step responses of several aircraft configurations [17-21]. Analytical and computational indicial responses have been used to predict dynamic derivatives and the maximum lift coefficient following an encounter with a gust [22]. In the remainder of this paper, the normal force coefficient C_N will be used instead of the lift coefficient.

III. MULDICON CONFIGURATION

The MULDICON (MULtiDisciplinary Configuration) platform was created at the start of the task group. It is based on the SACCON [23-26], with the same span and leading-edge sweep angle, but with the trailing edge sweep angle reduced to 30° in order to improve efficacy of trailing edge control devices and increase the surface area for lift production (Figure 1). The geometry under consideration here is designated as Design 3. It is the result of a design study using an inverse design method [3]. It is the final design release of ASG and meets the specified design requirements specified in [1,2].

IV. MULDICON TEST MANOEUVER

A test manoeuvre was selected for assessing the ROM prediction against the full CFD solution. It was created using a rudimentary flight dynamics model containing steady-state values of lift and pitching moment of the baseline MULDICON, along with estimated control surface effects. Resembling a pull-up manoeuvre, it starts with steady horizontal flight at $\alpha = 15^\circ$, after which it rotates at the maximum required pitch rate of $q = 20$ deg/s until $\alpha = 22^\circ$ at the pitch angle $\theta = 67.7^\circ$. It then rotates back with $q = -20$ deg/s until the initial angle of attack of $\alpha = 15^\circ$ is reached. The velocity is maintained at Mach 0.20 throughout the manoeuvre, with a Reynolds number $Re = 27.95 \cdot 10^6$. The variation in angle of attack and pitch rate is shown in Figure 2, along with the flight path. The climb angle γ varies linearly with pitch angle, from $\gamma=0^\circ$ at $\theta=15^\circ$ to $\gamma=44.7^\circ$ at $\theta=66.7^\circ$. During the final 0.248 s, the aircraft is in a steady descent with a climb angle of $\gamma=-5.9^\circ$ with $\theta=9.1^\circ$.

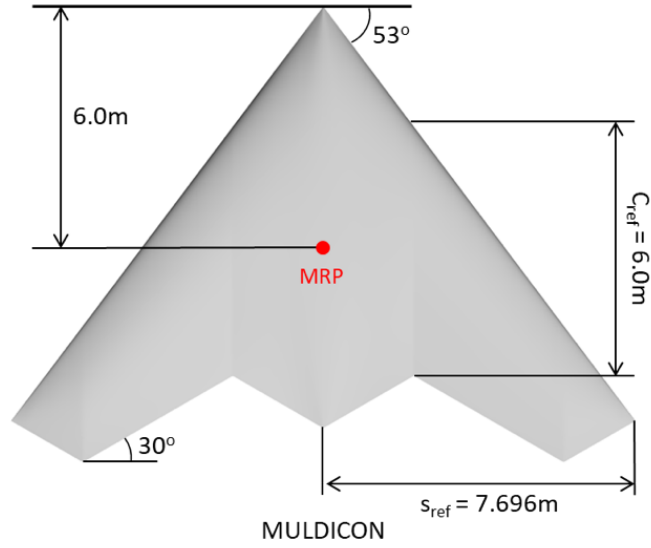


Figure 1: Planform of the MULDICON wing.

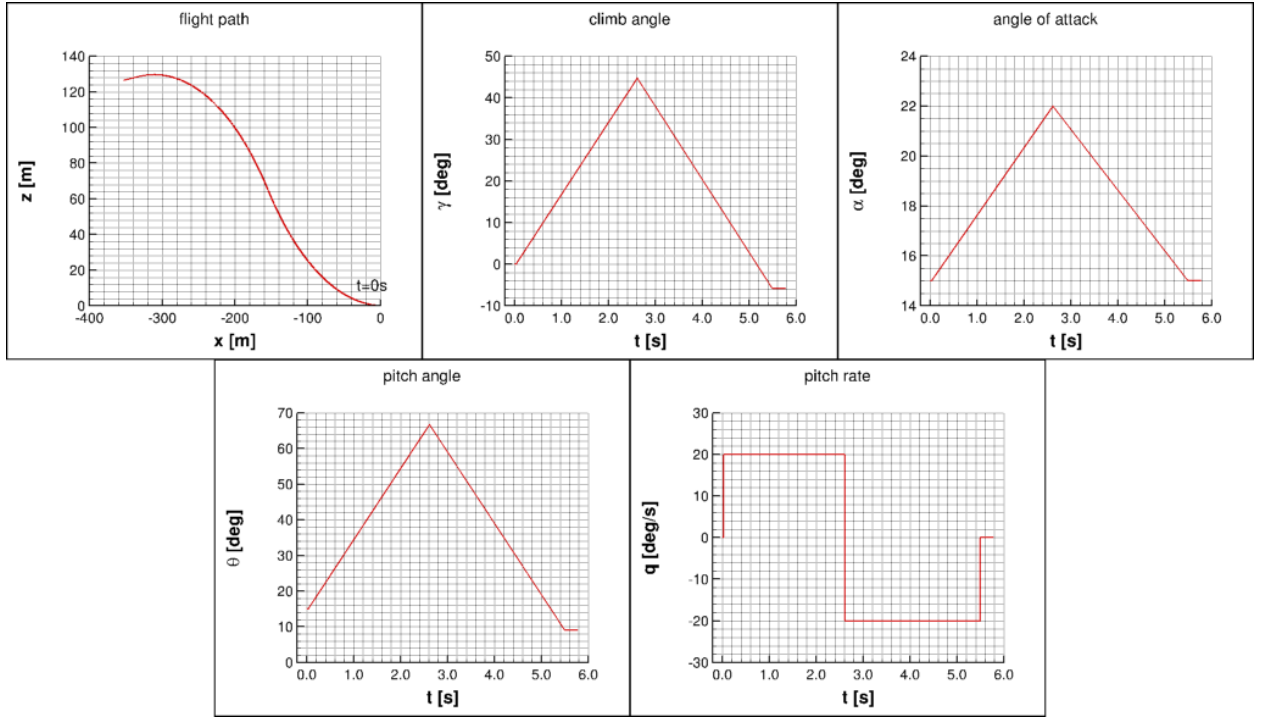


Figure 2: Pull-up manoeuvre of the MULDICON used to assess ROM predictions. Top: flight path, climb angle and angle of attack. Bottom: pitch angle and pitch rate.

V. CFD METHODS

A. NLR

Grid Generation

A multiblock structured grid was created on a half-span of the MULDICON Design 3 geometry with NLR's in-house tools for domain modeling and grid generation. A semiautomatic Cartesian grid mapping technique was used with which a multiblock domain decomposition around a complex aircraft configuration is created automatically by exploiting Cartesian space [27-30]. This enables a fast and efficient

construction of three-dimensional multiblock domain topology and a high-quality structured grid. An O-type mesh envelopes the aircraft geometry. The grid used in the computations described here contained 12 million cells, with 112 cells distributed along the chord, 128 cells along the span, and 32 across the trailing edge and wing tip. The average y^+ in the first cell was 0.75 ($\alpha=0^\circ$). An impression of the grid can be obtained from Figure 3.

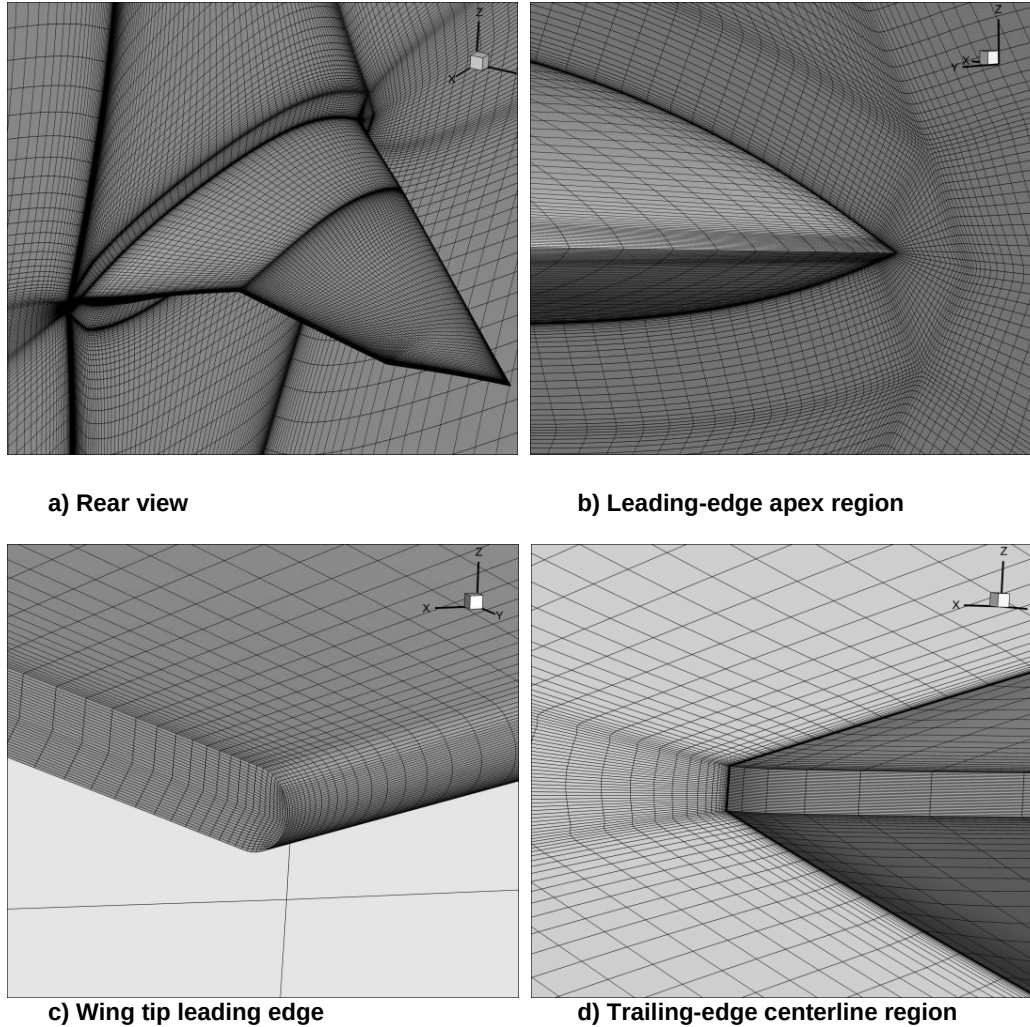


Figure 3: Details of half-span MULTICON Design 3 structured grid on wing surface and symmetry plane used for ENSOLV computations.

Flow Solver

NLR's in-house flow solver ENSOLV is an advanced CFD code with which three-dimensional steady or time-dependent, incompressible or compressible flows around complex aerodynamic configurations can be computed [31-33]. These configurations can be either fixed or moving relative to an inertial reference frame, they can be either rigid or flexible, and they can consist of multiple bodies moving relative to one another. In the case of flexible configurations, the required elastomechanical input data for carrying out analysis involving fluid/structure interaction are derived from finite element models in the form of mode shapes. ENFLOW has been applied extensively to aerodynamic, aeroelastic and

aeroacoustic applications. The Euler/Reynolds-averaged Navier-Stokes equations are discretized following a cell-centered finite-volume method. The numerical scheme is fourth-order accurate on curvilinear meshes with zero dissipation and low dispersion errors. Time-accurate integration is carried out using a dual-time stepping scheme employing the

Runge-Kutta method for the relaxation in pseudotime. Convergence acceleration is provided by a full approximation storage (FAS) multigrid method and a line relaxation method in the boundary layer regions. The flow solver has been optimized for the HPC facilities at NLR and applies a hybrid MPI/OpenMP parallelization for efficient parallelization on multicore processors. The computations were performed using the Menter Shear Stress Transport (SST) two-equation model. The standard 1994 model [34] is used, with an adaption from SST-2003 that uses the strain invariant rather than magnitude of vorticity in the definition of turbulent viscosity [35].

B. NASA

Grid Generation

NASA grids were generated using Heldenmesh[™], which is a software system developed by the Helden Aerospace Corporation for the rapid generation of high quality, three-dimensional, unstructured meshes about arbitrary complex configurations [36]. This capability was developed to streamline the application of CFD in a demanding production environment by transforming the mesh generation process from one currently measured in days-per-mesh to one measured in meshes-per-day. The Heldenmesh[™] software system consists of two primary tools, Heldenpatch[™] and Heldenmesh[™]. The former is the interface between the underlying CAD geometry definition and the Heldenmesh[™] grid generator, and includes geometry clean-up, management, and geometric transformation capabilities. The Heldenpatch[™] will typically provide around 95% of the surface patching automatically, leaving only a few minutes of handwork to complete. An input-based spatial sourcing file will typically produce adequate spatial control of the cell size and distribution with only a few parameters with no additional handwork needed. Heldenmesh was designed for compatibility with the NASA developed Tetrahedral Unstructured Software System (TetrUSS) [37,38] in general and the VGRID mesh generation process in particular [39]. An important component of this system is the GridTool application, which is an interactive program for geometry processing. The GridTool application is currently utilized for interactive geometry processing in the Heldenmesh[™] software system. A new integrated graphic user interface is planned for future releases.

A hybrid unstructured half-span grid was generated for USM3D computations on the MULDICON Design 3 using Heldenmesh[™] as depicted in Figure 4. The total number of cells is 5,565,123, which is comprised of near-wall 3,832,128 prisms, 1,721,185 tetrahedra in the outer field, 11,810 transitional pyramidal cells, and 2,252,835 nodes. Setup for the initial grid required approximately 45-minutes by an inexperienced user. Subsequent grids could be generated within minutes by changing one or two parameters in the spatial input file and rerunning Heldenmesh[™].

Flow Solver

The NASA computations are performed with a developmental version of USM3D, which is a parallelized mixed-topology cell-centered, finite volume Navier-Stokes flow solver [40]. Inviscid flux quantities are computed across each cell face using various upwind schemes. To improve the convergence and robustness of USM3D solutions, a hierarchical adaptive nonlinear iteration method (HANIM) has been recently implemented in the mixed-element USM3D [41]. Further enhancements to the new USM3D's preconditioner and discretization scheme are reported in Ref. [42]. A prescribed grid motion capability is available that is advanced in time by a 2nd- and 3rd-order time step scheme. Turbulence models currently available are the Spalart-Allmaras (SA) one equation model, SA with Quadratic Constitutive Relation (SAQCR), the two-equation $k-\epsilon$ turbulence model, and the Menter Shear Stress Transport (SST) two-equation model.

C. USAFA

Grid Generation

The computational grid at USAFA corresponds to the half-span MULDICON Design3 configuration with

half span of 769.6 cm and the moment reference point at (600 cm, 0 cm, 0 cm). This grid (shown in Figure 5) was made using Pointwise version 18. Hybrid meshes are generated on surfaces consisting mainly of structured grid meshes connected by triangular meshes in places where structured meshes lead to highly distorted grid cells. The main motivation for using the quadrilateral mesh is to obtain a good grid resolution on the leading and trailing edges of the wing as shown in Figure 5.

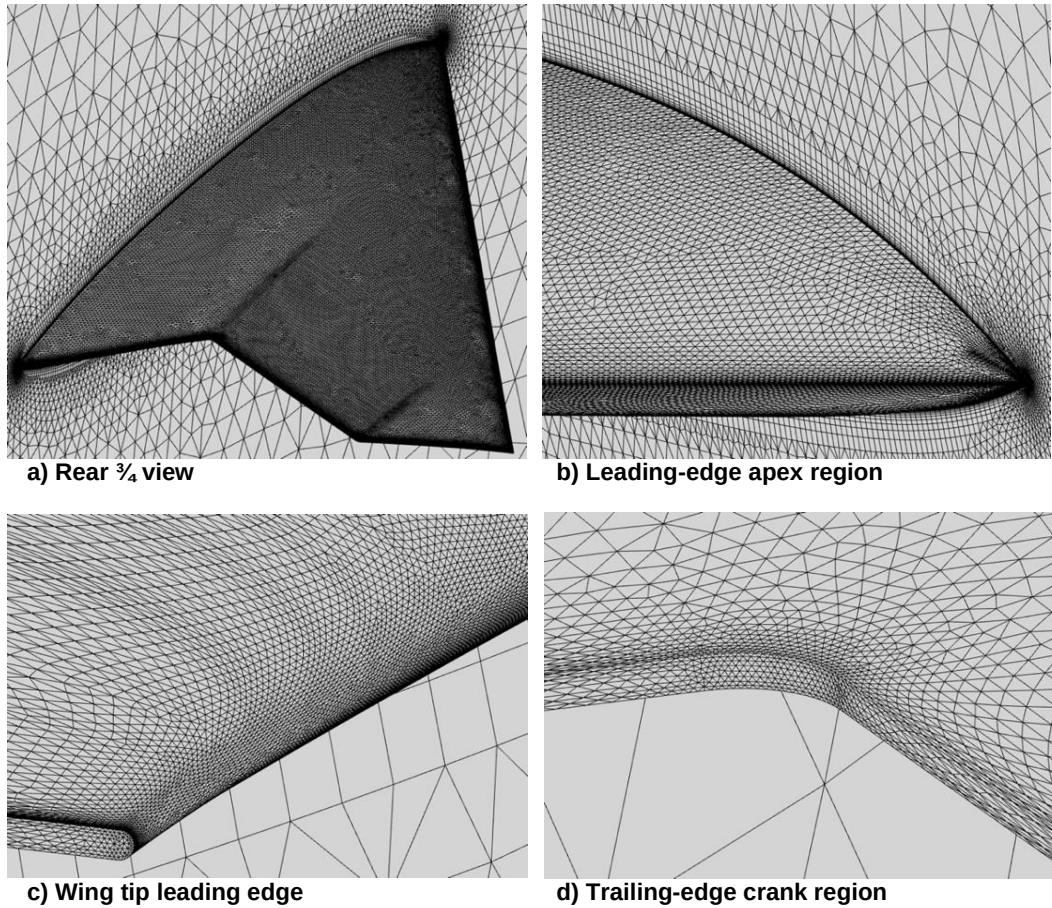


Figure 4: Details of half-span MULDISON Design 3 wing surface and symmetry plane grid used for USM3D computations.

The volume mesh is fully unstructured with 50 prism layers on the UCAV surface. The growing ratio of the prism layers is 1.25 and their growth is terminated when the transition between the prism layer and the tetrahedral mesh is smooth. The grid has around 23.9 million nodes and 35.4 million cells. The averaged first cell grid spacing (y^+) is 0.1 at Mach 0.2.

Flow Solver

The flow solver used at USAFA is the fixed wing computational tool of CREATETM-AV program, i.e. Kestrel [43]. This DoD-developed solver is funded by the DoD High Performance Computing Modernization Program (HPCMP). The CREATETM focuses on addressing the complexity of applying computationally-based engineering to improve DoD acquisition processes. Kestrel version 8.0 is used in this work. The flow solver of the code (kCFD) discretizes Reynolds-averaged Navier Stokes (RANS) equations into a cell-centered finite-volume form. The code then solves unsteady, three-dimensional, compressible RANS equations on hybrid unstructured grids [44]. The code uses the Method of Lines (MOL) to separate

temporal and spatial integration schemes from each other [45]. The spatial residual is computed via a Godunov-type scheme [46]. Second-order spatial accuracy is obtained through a least squares reconstruction. The numerical fluxes at each element face are computed using various exact and approximate Riemann schemes with a default method based on HLLE++ scheme [47]. In addition, the code uses a subiterative, point-implicit scheme method (a typical Gauss-Seidel technique) to improve the temporal accuracy.

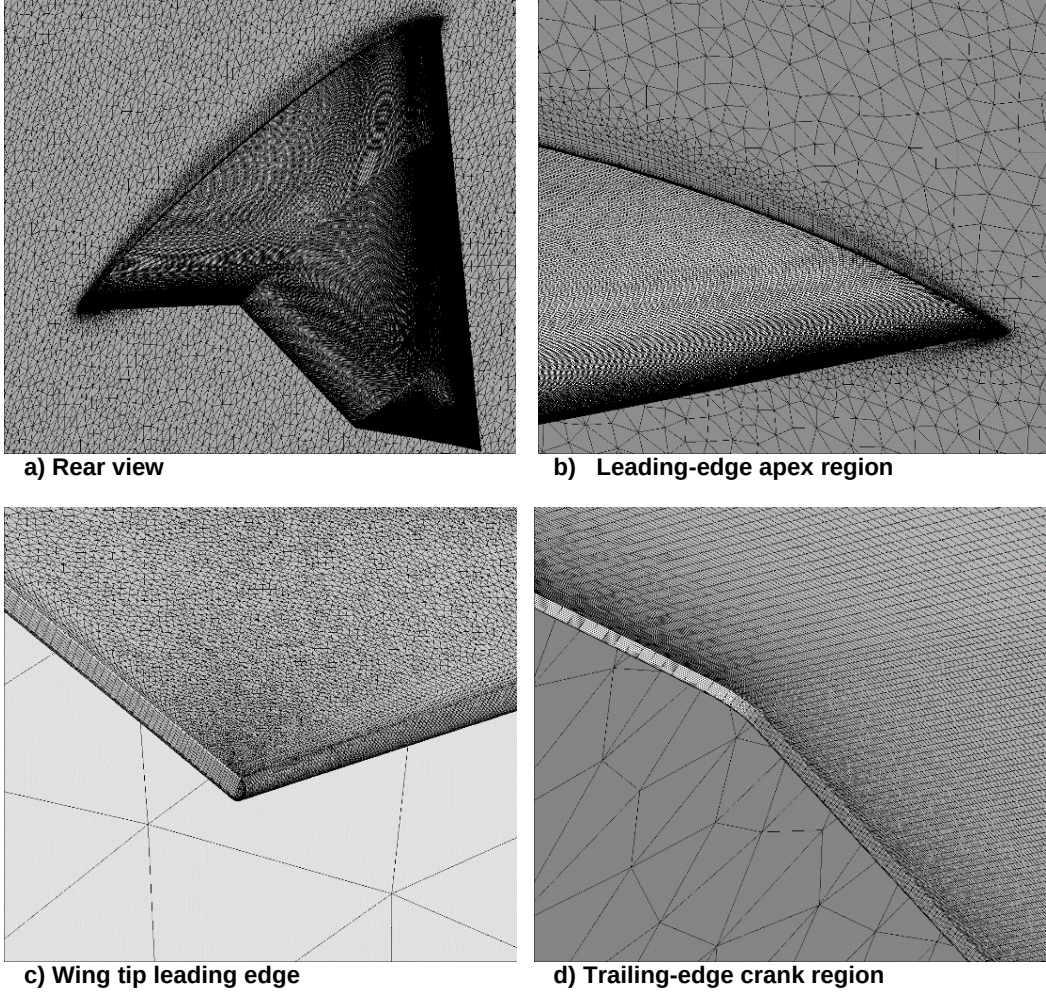


Figure 5: Details of half-span MULTIDICON Design 3 wing surface and symmetry plane grid used for USAFA computations.

VI. RESULTS

The results for the indicial-step and the full-manoeuver solutions from the three flow solvers are described and compared below. All solutions were computed at $Re_{cref} = 27.95$ million and $M_\infty = 0.20$. The indicial responses serve as input to the ROM for computing the normal force coefficient C_N , and pitch moment coefficient $C_{m,p}$. The calculations performed to obtain the response functions are described below.

A. Indicial Solutions

NLR / NLDA

The indicial responses due to a 1° step increase in angle of attack were computed at $\alpha = 14^\circ, 16^\circ, 18^\circ, 20^\circ$,

and 22° . Grid motion was used to achieve a 1° step increase in angle of attack while maintaining the pitch rate $q=0$ rad/s. The response function for a step change in pitch rate was obtained at $\alpha=0^\circ$, for a step size of 20 deg/s (0.349 rad/s), which is the maximum required pitch rate of the MULDISCON. The angle of attack was maintained at $\alpha=0^\circ$ through appropriate grid motion. Motions are defined in a separate input file (motion file), which contains characteristic time, position of the center of rotation, Euler angles, velocity components and rotation rates. The indicial response computations were started from converged time-accurate calculations at the initial angle of attack. A variable time step was employed, ranging from $\Delta t^* = \Delta t(V_\infty/c_{ref}) = 0.005$ up to $t = 0.1$ s, then increasing with a constant factor to $\Delta t^* = 0.05$ in 200 time steps (until $t = 0.446$ s) and remaining constant after that. The total time span covered by the calculation was 2.0 s. This way, the initial transient created by the initialization of the step function is captured accurately while a sufficiently long interval is computed to ensure the step response has reached a steady state, without incurring substantial computational cost. The calculated step responses to change in α do not reach a constant level until $t = 1.5$ s (Figure 6). Persistent unsteadiness is present in the step responses to change in angle of attack for C_N at $\alpha=22^\circ$, and $C_{m,p}$ for all angles of attack.

NASA

The indicial responses to a 1° step increase in angle of attack were computed with USM3D/SA at $\alpha=15^\circ, 16^\circ, 17^\circ, 18^\circ, 19^\circ, 20^\circ, 21^\circ$, and 22° at a characteristic time increment of $\Delta t^* = \Delta t(V_\infty/c_{ref}) = 0.01$ for 500 time-steps, which corresponds to approximately 0.44 seconds. Prior to computing these responses, steady-state time-accurate solutions were generated at the 8 angles of attack as initial starting solutions for the step motions with $\Delta t^* = 0.1$ for 2000 time-steps. At each angle of attack, the vehicle is plunged instantaneously to achieve a 1° step increase in angle of attack while maintaining the pitch rate $q = 0$ rad/s throughout the grid motion. The pitch-rate response is generated in a similar manner with the instantaneous application of a $q = 1$ rad/s pitch-up from the $\alpha=0^\circ$ initial solution as the wing traverses upwards and aft along a trajectory that maintains local $\alpha=0^\circ$.

USAFA

The grid motion approach in Kestrel is used to calculate step responses of the UCAV to a unit change in the angle of attack ($\Delta\alpha = 1^\circ$) and unit change in pitch rate ($\Delta q = 0.349$ rad/s). The step solutions begin at different angles of attack. All simulations are second-order accurate in spatial and temporal directions. The delayed detached eddy simulation (DDES) model with Spalart-Allmaras with Rotation Curvature (SARC) is used. The number of subiterations is five in all simulations. Time step used for indicial responses is 1×10^{-3} seconds corresponding to $\Delta t^* = \Delta t(V_\infty/c_{ref}) = 0.01$. Step responses begin from time zero following 1,500 startup iterations. Note that time remains zero during these startup iterations. Also, Kestrel predicts unsteady flow over the Design 3 Wing for angles of attack above 10° using the SARC-DDES turbulence model even for static simulations. Thus, it is expected that the Kestrel simulations under grid motions and at high angles of attack cause oscillating behavior in forces and moments.

The results are summarized in Figures 6 and 7. Figure 6 shows the indicial responses of C_N and $C_{m,p}$. Responses for C_N predicted by NASA and NLR generally converge to the same average steady state value. As pointed out above, the responses computed by USAFA contain a considerable amount of unsteadiness. The responses for $C_{m,p}$ vary wildly. In the NASA calculations, responses seem to be converging towards a steady state, with pitching moment increasing with α . In the NLR calculations, $C_{m,p}$ response varies per angle of attack. The responses at $\alpha=18^\circ, 20^\circ$ and 22° suggest flow separation and breakdown of over-wing vortices as observed in [3]. USAFA response functions show highly unsteady flow at all angles of attack.

Because experimental results aren't available, the accuracy of the CFD results cannot be assessed here. Figure 7 shows similar responses to change in pitch rate obtained by NLR and USAFA, with the response by USAFA requiring a longer time for oscillations to dampen out. As noted before, the response calculated by NASA is for a larger change in pitch rate. In both the responses to step changes in α

and q , the initial transients can be distinguished clearly. The ability of a ROM to predict these transients is of importance when simulating high speed manoeuvres and control surface deflections.

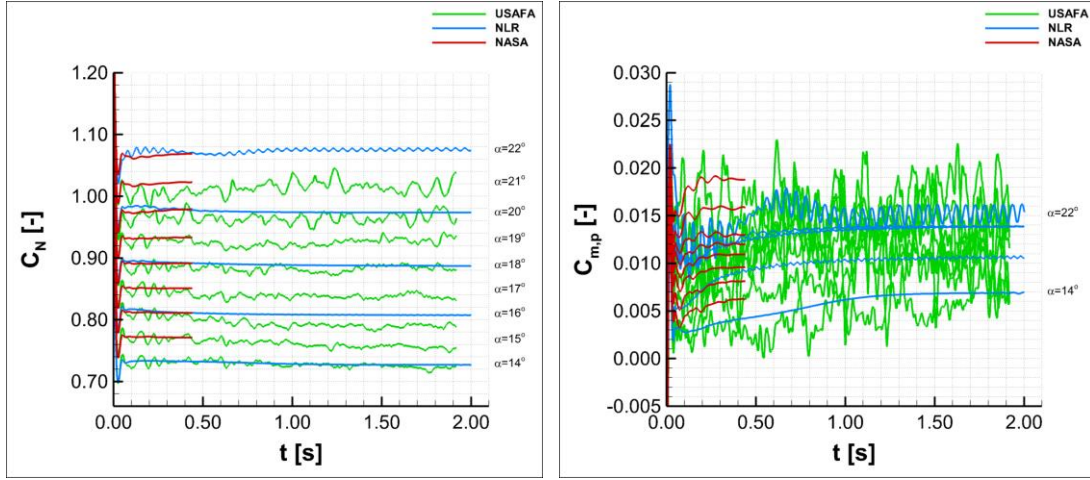


Figure 6: Indicial responses of C_N and $C_{m,p}$ to step changes of 1° angle of attack with $q=0$ rad/s.

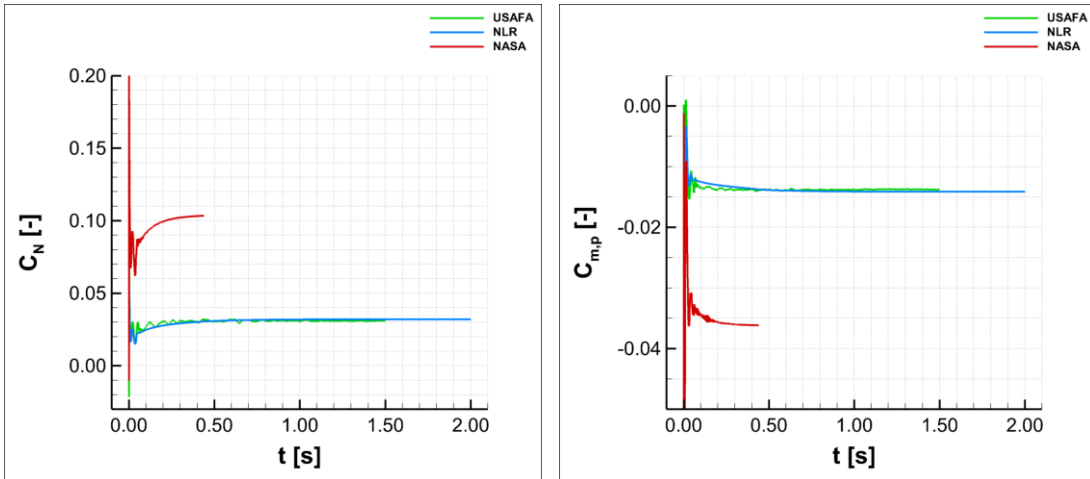


Figure 7: Indicial responses of C_N and $C_{m,p}$ to step change in pitch rate while maintaining $\alpha=0^\circ$ through grid motion. NLR, USAFA: $q=0.349$ rad/s, NASA: $q=1$ rad/s.

B. Low Speed Pull-Up Manoeuvre: CFD and ROM Prediction

NLR / NLDA

The CFD computation of the pull-up manoeuvre was performed with ENSOLV. The motion was prescribed in the form of a motion file containing nondimensionalized time, position of the center of rotation, Euler angles, velocity components and rotation rates for all time steps. The time step was $\Delta t^* = 0.01$, requiring 6560 steps for the complete manoeuvre. Thirty subiterations were used per time step. The computation was started from a converged time-accurate solution obtained at a static angle of attack of $\alpha = 15^\circ$. Both linear and nonlinear ROMs were created. A nonlinear ROM was created using the response functions described above, obtained at $\alpha = 14^\circ, 16^\circ, 18^\circ, 20^\circ$, and 22° for α -step responses, and $\alpha = 0^\circ$ for pitch-rate response. For the linear ROM, the α -step response at $\alpha = 14^\circ$ was used together with the pitch rate response at $\alpha = 0^\circ$.

The CFD results and the ROM prediction of evolution of C_N and $C_{m,p}$ during the manoeuvre are shown in Figure 8. Both the linear and nonlinear ROMs give an accurate prediction of C_N and an

acceptable prediction of $C_{m,p}$. The nonlinear ROM performs slightly better. The existence of rapid fluctuations in the form of the peaks related to step changes in pitch rate are predicted by the ROM, even though their amplitude does not match the CFD results. This could be a result of reduced accuracy of the ROM, or of the full CFD prediction related to the larger time step of $\Delta t^*=0.01$ compared $\Delta t^*=0.005$ at the initial phase of the response function calculation.

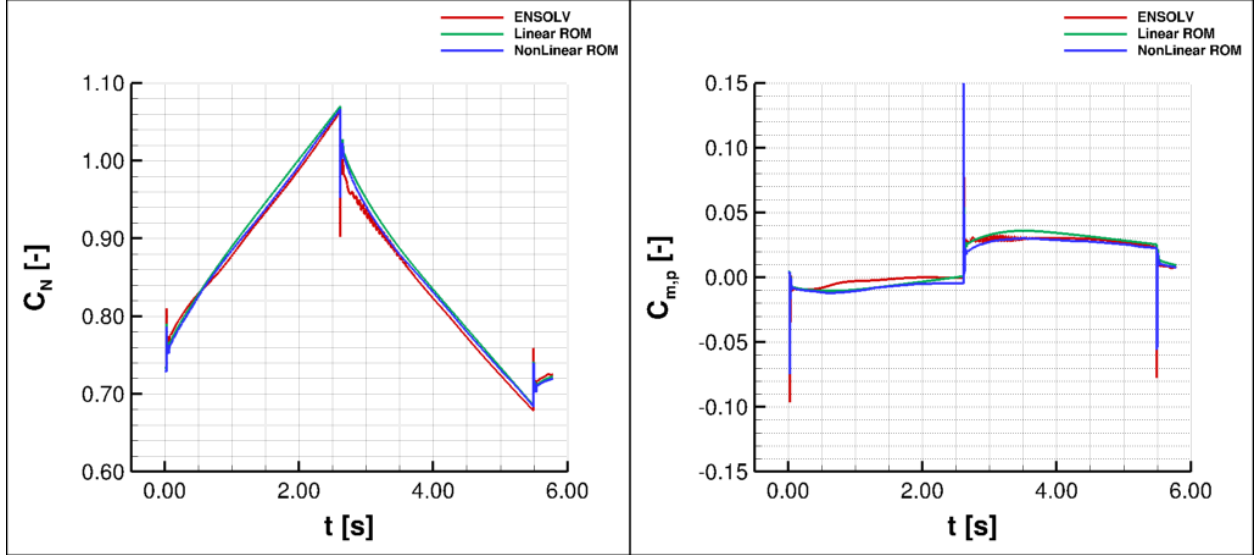


Figure 8: CFD and ROM predictions for the pull-up trajectory (NLR/NLDA).

NASA

The CFD solution for the pull-up manoeuvre was computed with USM3D/SA using $\Delta t^*= 0.01$ for 6560-time steps to complete the 5.8 second manoeuvre. It was restarted from an initial static time-accurate solution at $\alpha = 15^\circ$ that was run for 2000-time steps with $\Delta t^* = 0.1$. For both cases, 20 subiterations were prescribed with a cut-off imposed when the combined residuals reach 10-orders of magnitude reduction, at which point it moves to the next time step. For the initial static solution, this criterion was met after approximately 5 subiterations after the initial startup transients settled out. For the pull-up manoeuvre, this criterion was met on average with about 13 sub-iterations.

The NASA ROM solutions utilized all 8 angles of attack from 15° to 22° of the α -step responses for the nonlinear ROM, and only $\alpha = 15^\circ$ for the linear ROM. The rate derivatives used for the nonlinear and linear ROM are only derived from the $\alpha = 0^\circ$ pitch-rate response. The results are shown in Figure 9. Both the linear and nonlinear ROMs give an accurate prediction of C_N , but for $C_{m,p}$, the linear ROM provides more accurate results, which is unexpected. This could be an effect of insufficient period of time covered by the indicial responses, i.e. a steady state hasn't been reached, that a response of a pitch rate of 1 rad/s was calculated, or both. The ROM predicts the existence of peaks associated with the step changes in pitch rate.

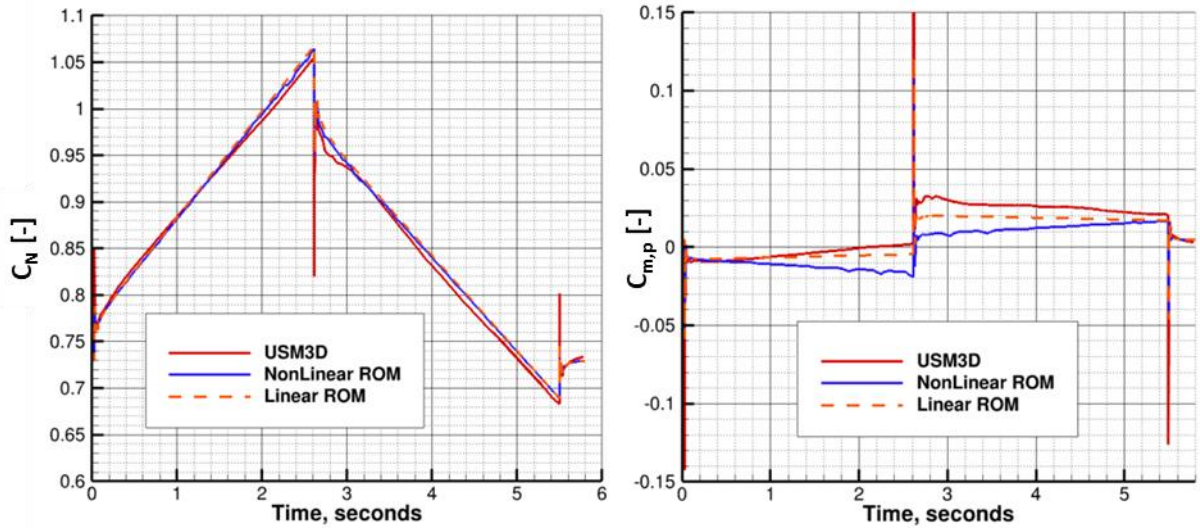


Figure 9: Correlation of Linear and Nonlinear ROM solutions with CFD computed result for pull-up trajectory (NASA).

USAFA

Kestrel static predictions at Mach 0.2 and zero altitude are shown in Figure 10. Time-averaged data are shown by solid lines. The bars in the plots show the maximum deviations from average values. Dashed lines show the linear aerodynamic regimes. The Design 3 wing has a linear and unsteady aerodynamic behaviour up to 8° angle of attack. Above this angle, a small vortex is formed over the upper surface and there is flow separation near the wing tip region. This causes nonlinear and highly unsteady aerodynamic behaviour at higher angles of attack.

The trajectory motion begins from an initial static time-accurate solution at $\alpha=15^\circ$ with 1,000 startup iterations. The time step remains zero during these iterations. The motion file has Euler angles and the position of the center of rotation at each time. The center of rotation at $t=0$ was set to (600, 0, 0) cm. Translation and rotations are defined to have the UCAV set at flow conditions described by the trajectory. The turbulence model is SARC-DDES with time steps of $1 \cdot 10^{-3}$ seconds and using five subiterations. Kestrel prints all motion variables (positions, velocity, Euler angles, and rotation rates) and aerodynamic coefficients at each time step.

The CFD results and the ROM prediction of evolution of C_N and $C_{m,p}$ during the manoeuvre are shown in Figure 11. The nonlinear ROM is built using all calculated α -step responses. The linear ROM is created with using the α -step response at $\alpha = 14^\circ$. The nonlinear ROM gives more accurate predictions for both C_N and $C_{m,p}$, but the difference between CFD and ROM is larger than obtained with ENSOLV and USM3D. This is most likely due the unsteadiness present in the calculated step response functions, which reduce the accuracy of force and moment derivatives with respect to α and q . However, the ROM predicts the presence of high frequency oscillations present in the CFD results.

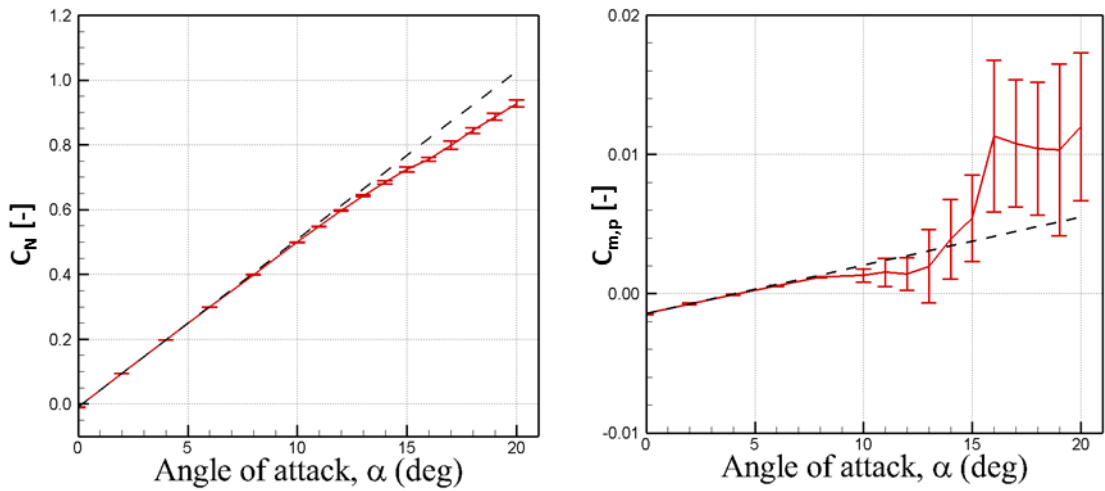


Figure 10: Kestrel static predictions for normal force and pitch moment coefficients of MULDICON Design 3 at Mach 0.2 and zero altitude. Solid lines show time-averaged data; bar denotes maximum deviation from average values; the dashed line shows linear aerodynamic region.

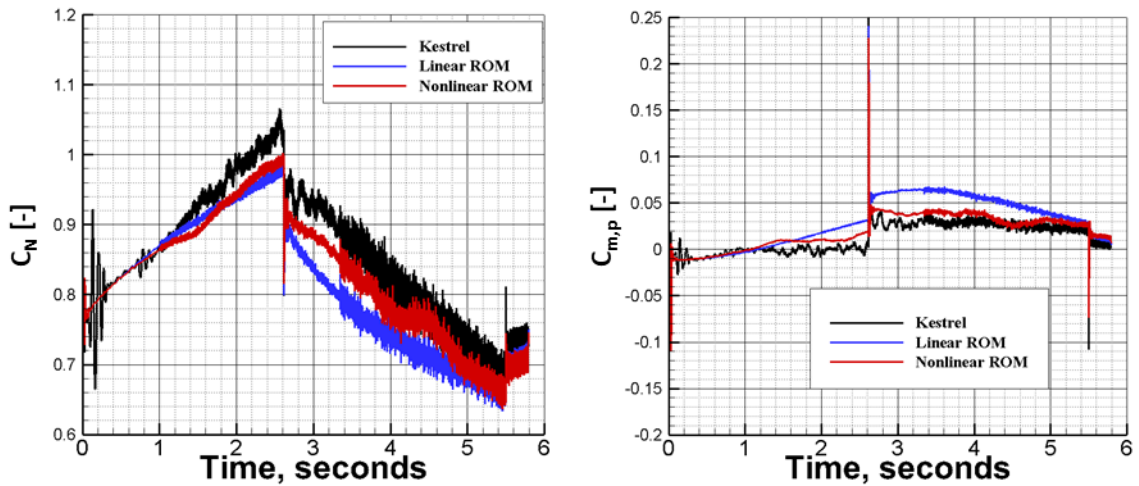


Figure 11: Modeling trajectory with a quasisteady and linear indicial response model.

VII. SUMMARY

Within the framework of NATO STO AVT-251 Task Group on “MultiDisciplinary Design and Performance Assessment of Effective, Agile NATO Air Vehicles,” a collaborative investigation has been conducted on the use of indicial response functions to create ROMs of the dynamic performance of the MULDICON configuration. Four organizations participated in this study, using a variety of CFD codes, computational grids, and turbulence models as well as methods to build the ROM. The tested codes include ENSOLV at NLR, Kestrel at USAFA, and USM3D at NASA LaRC. NLR and NLDA collaborate on ROM method development.

ROMs were created using computed step response functions in angle of attack and pitch rate. Both effects could be separated from each other using a grid motion technique. The ROMs were applied to a pull-up trajectory at constant velocity with varying angle of attack and pitch angle, and with step changes in pitch rate. Implementation of the indicial response method was fairly straightforward for this case. The ROM predictions were compared to CFD results of the same manoeuvre. Some variations were found between the

CFD predictions of both the indicial step responses and the manoeuvre. With the variety of methods used, this was to be expected. Results obtained by USAFA using Kestrel contain a high level of unsteadiness, especially the step responses to change in angle of attack and the pull-up manoeuvre itself. This is a result of the DDES modeling giving inherent unsteadiness, even at lower angles of attack. Without experimental results to compare with, the accuracy of the CFD simulations cannot be assessed.

The goal of the study was to assess whether a ROM could be built that can predict the forces and moments encountered during a pull-up manoeuvre with accuracy comparable to full-order CFD. Generally, the ROM predictions compare reasonably well with the CFD results. It is noted that the pull-up manoeuvre simulated here is a rather benign manoeuvre. On the other hand, the aerodynamic characteristics of the MULDICON show unsteady flows at the high end of the range of angles of attack encountered during the manoeuvre. The indicial response method provides predictions that include transient effects. The occurrence of peak loads due to step changes are consistently predicted by the ROMs. The CFD result of the manoeuvre calculated with Kestrel contained high-frequency unsteadiness. To an extent, this was also present in the ROM prediction. This is encouraging, since a desired attribute of a ROM is to be able to predict unsteadiness.

The accuracy of the ROM predictions rely heavily on the quality of the steady state solutions from which step response calculations were started, as well as the convergence of the step response calculations themselves. The step response calculations need to cover a time span, which is sufficiently long for all transients to disappear. The results presented here show that these points need to be considered carefully, and that each specific case will require a tailored approach. This might become cumbersome, especially when creating six degree-of-freedom models. Even though the indicial response method itself is fairly straightforward to implement, this will likely prevent its use as a general approach to the creation of ROMs.

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