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# DEVELOPMENT OF NUMERICAL ALGORITHMS FOR END-USER APPLICATIONS IN AEROSPACE

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**Abstract.** *In this paper, a cross-section of numerical algorithm development at NLR for end-user applications in aerospace is presented. Aerospace is characterized by high safety requirements and standards, complex development programs and operational processes, and challenging multi-physics phenomena. An example is the certification or qualification of civil and military aircraft, helicopters, Unmanned (Combat) Aerial Vehicles (UCAV), or Expendable Launch Vehicles (ELV). This application requires numerical algorithms that have a high level of physical fidelity, a high level of accuracy, short total turn-around times, and low cost. Three applications driving numerical algorithm development at NLR are presented: the simulation of nozzle buffet loads, for qualified space launchers, the simulation of non-linear flutter properties, for qualified fighter configurations, and the simulation of blade-vortex interaction, for the development of environmentally friendly helicopters.*

## 1 INTRODUCTION

One of the prime motivations for numerical algorithm development is the need for such algorithms dictated by (future) applications. Since the advent of computers in the previous century, numerical algorithm development for aerospace applications in the Netherlands has been an important activity at NLR. The resulting NLR panel methods [1], [12], full potential methods [4], [5], [22], Euler and Navier-Stokes methods [6] have had an important impact on the success of the Dutch Aerospace Industry. Anno 2006, numerical algorithm development is still driven by new and challenging aerospace applications. A selection of such applications driving numerical algorithm development at NLR is shortly discussed below.

- Expendable Launch Vehicles (ELV's) need to be reliable. During ascend to space, space launcher structural components such as the engine nozzle should be capable to withstand the experienced steady and unsteady loads. A newly designed space launcher is usually qualified by showing one or more successful test launches with dummy or non-insured payloads. A non-successful qualification flight is a financial risk, since costly redesigns of the space launcher might be necessary. For this reason, a successful qualification flight is prepared on the ground by large amounts of testing and simulation. The balance between the amount of testing and simulation in a launcher development program is dynamic. The equilibrium is determined by the differences cost structure and fidelity of testing and simulation. Today, unsteady aerodynamic loads are mainly obtained from wind tunnel or in-flight testing. The continuously increasing cost-effectiveness of computers together with the development of hybrid RANS-LES turbulence models will allow future analysis of unsteady aerodynamic loads.

- Aerospace vehicles such as civil aircraft of helicopters need to be safe in the first place. To this end, their design shall be certified according to the applicable means of compliance. A strict certification process is to be followed. This process of delivering the proof to the Aviation Authorities that the design is safe, usually is lengthy and expensive. Up to the present day, testing is the ultimate way of proving compliance to the rules. However, there is a strong pressure from industry to introduce more simulation in the routes to the final test, because cost savings are expected. A building block approach is often pursued. For building aerodynamic databases, this building block approach could look like:
    1. Conduct a sub-scale test of aerodynamic properties.
    2. Conduct a sub-scale analysis of aerodynamic properties, and validate the simulation using the sub-scale test data.
    3. Conduct full-scale analysis of aerodynamic properties (for instance Reynolds extrapolation).
    4. Conduct a full-scale test of aerodynamic properties, to validate the full-scale analysis.
    5. Analyze the impact of shape modifications on the aerodynamic properties using the validated analysis method.

  In the last step, savings can be made if the analysis is more cost-effective than testing, and provided the margins of safety found in the simulation are large enough beyond any doubt. Cost-efficient and physically sound numerical algorithms are crucial, for instance to extrapolate aerodynamic data to flight Reynolds numbers.
- Being safe, civil aircraft of helicopter designs also have to be cost-efficient, comfortable, environmentally friendly (noise, emission, production), etc. in order to survive the fierce competition on the aerospace market. In the end, airlines need aircraft that allow them to make money. Within the entire design cycle, new knowledge on positive (or negative) properties of the design continuously becomes available. On the other hand, because the design is getting more and more detailed, the number of degrees of freedom to modify the design decreases rapidly. To overcome this dilemma, as much as possible should be known of the design as soon as possible in the entire design cycle. Multi-disciplinary design and optimization algorithms incorporating high fidelity simulation algorithms offer a solution, provided they are feasible regarding turn-around times and costs.
- Fighter aircraft need to be qualified for their mission. During a mission, the fighter configuration can change. These changes are due to wing-tip or under-wing stores being released, or due to new weapon integration programs. The distribution of mass over the aircraft as well as the aerodynamic properties change accordingly. For safety, it has first to be proven that, for all down loadings, the fighter will be free of classical flutter in the entire operational envelope. Furthermore, vibrations with excessive amplitude, due to non-linear flutter shall be proven to be absent. The qualification effort to prove all this is

nowadays conducted by flight-testing-instrumented fighter aircraft. In order to avoid endless flight testing, flight test campaigns are supported by simulation through validated algorithms.
- Both Unmanned Aerial Vehicles (UAV) and fighters need maximum survivability. To this end, a low radar cross section is helpful to avoid early detection by a hostile radar. The actual radar cross section of a given aircraft is usually determined through trials. “Hot spots” on the aircraft can be determined using radar-cross-section analysis techniques, such that areas that require special radar absorbing treatments can be identified. For multi-disciplinary design optimization of UAV radar-cross-section analysis techniques are included in the design loop together with the traditional disciplines such as for instance aerodynamics, structural dynamics, flight mechanics, weight and balance, aero-elastics. Certain parts such as untreated engine inlet cavities constitute the usual dominant source of radar wave reflection. Simulation of radar wave multiple reflections in a large and deep cavity at relevant radar frequencies still forms a formidable challenge for both algorithms to solve the Maxwell equations and computer resources such as memory and processor speed.

In this paper, a selection of challenging end-user applications of numerical algorithms is discussed:
- Simulation of nozzle buffet loads, for qualified space launchers.
- Simulation of non-linear flutter properties, for qualified fighter configurations.
- Simulation of blade-vortex interaction, for the development of environmentally friendly helicopters.

All three examples will be described from the view point of the end user, starting from current or future needs, detailing the current modeling and computational barriers, and sketching feasible numerical developments.

The challenge of numerical algorithms for real end-user applications lies in obtaining accurate, physically relevant solutions within acceptable turn-around times, despite the high computational complexity that characterizes such applications. In order to become feasible for the end-user, the authors are convinced that even state-of-the-art numerical algorithms still need to be optimized further. In this session, several examples of optimized algorithms will be presented.

## 3 HIGHLY DYNAMIC AERODYNAMIC LOADS

### 3.1 Application

Aerospace vehicles are subjected to significant vibrational loads during the launch phase of the flight. To establish the effect of vibrations on their design and operation, it is imperative to determine the structural responses, internal loads and stresses resulting from externally applied loads. The loads can be categorized in a transient/shock type due to ignition, tie-down release or stage separation and a periodic/random type due to propulsion fluctuations, rocket noise and aerodynamic buffeting.

The unsteady loads corresponding to buffet are traditionally determined in delicate wind-tunnel experiments during the design phase and are validated by costly flight tests. The wind-tunnel measurements provide the buffeting input-forces on rigid scale models. Scaling parameters that have to be satisfied are the Mach number, Reynolds number and reduced frequency. The Mach number and reduced frequency can be satisfied, but generally the Reynolds number is at a sub-scale value. Uncertainties in the scaling laws that provide the extrapolation to flight conditions have not been firmly established with respect to separation phenomena.

To illustrate the complexity of the buffet phenomenon and the implications for the experimental set-up, the identification of buffet loads as a structural design load for the nozzle of a cryogenic engine is given as an example. The separated wake flow represents a severe buffeting environment for this type of launcher at transonic conditions during atmospheric ascent. As the nozzle is immersed in the massively separated wake of the central tank, the loads on the external wall consist of anti-symmetrical forces imposed by the fully three-dimensional flow that is influenced by the presence of the protuberances and the boosters. The situation can be further complicated by anti-symmetrical forces inside the nozzle due to shock-wave induced boundary layer separation that occur as long as the engine is operating in the over-expanded regime. A coupling between the external induced vibrations and the internal flow may exist.

### 3.2 Modeling barriers

The prediction of buffet loads requires time-accurate turbulent flow simulations. Flow solutions based on the Unsteady Reynolds-Averaged Navier-Stokes (URANS) equations are known to incorporate too much dissipation due to high levels of turbulent viscosity in the separated flow areas. Consequently, the prediction of the length and time-scales of the smallest resolvable turbulent structures is inaccurate. A new hybrid URANS-LES turbulence modelling approach termed eXtra-Large Eddy Simulations (X-LES) holds the promise to capture the flow structures associated with massive separations and enables the prediction of the broad-band spectrum of dynamic loads. Turbulence model validation efforts have indicated an increase in resolved turbulent length scales in building-block applications such as the separated flow over an airfoil and cylinder (Kok et al. [7]). Comparison of URANS and X-LES calculations show a clear increase in frequency content in the Fourier spectrum of the aerodynamic forces.

### 3.3 Computational barriers

The chaotic dynamic behaviour of the flow in the wake requires long time intervals to obtain statistical convergence even in the mean of the integral forces. Moreover, the detailed turbulent patterns either require high resolution meshes or improved accuracy on engineering meshes. The combined increase of temporal and spatial resolution significantly increases the computational complexity of X-LES simulations.

### 3.4 Numerical algorithm developments

Higher-order methods are a well-known way to increase the resolution of flow details on relatively coarse engineering meshes. The development of a higher-order low-dispersion finite-volume discretization is the subject of Kok [8].

The large number of time steps required to reach statistical convergence forces one to develop efficient time integration schemes. In most compressible flow solvers, an implicit time integration scheme is applied, where the system is solved using a multigrid algorithm in combination with a Runge-Kutta smoother. In the LES region of the flow however, the time step restriction is dominated by physical and not numerical considerations. This implies that an explicit time integration scheme suffices in the wake, and in general actually is preferred because of its superior dispersion characteristics compared to an implicit scheme. Since the computational effort to perform an explicit time step is negligible compared to the effort to perform an implicit time step, significant gains in computing time can be expected. In practice, Scheijbeler [14] demonstrated a speedup of more than two on the simulation of the turbulent wake behind a circular cylinder.

Further improvements can be expected by increasing the efficiency of the implicit time integration scheme. In the case of geometries with a single Navier-Stokes direction in the mesh (that is, without corners) a line-implicit scheme can be applied in that direction, relieving the numerical time step restriction near the boundary. In the other two directions an explicit scheme can then be applied.

Even more gains can be expected from time-parallel schemes. The basic idea is that it is not the time-dependent behaviour of the flow we are after, but rather the statistics in the flow. This idea is related to the concept of ensemble averaging. This means that instead of one long simulation, several shorter ones can be simulated in parallel, provided that the starting points of the different simulations are sufficiently uncorrelated. Especially when running on massively parallel systems this may decrease the turn-around time significantly, since a spatial distribution of the computational mesh will result in a severe communication overhead, whereas a time-parallel scheme is embarrassingly parallel.

### 3.5 Examples

The geometry is based on a generic space launcher including two side boosters and a nozzle. Time-dependent flow simulations are performed at transonic conditions: free stream Mach number $M_\infty$=0.73, Reynolds number $Re_\infty$=6 million based on the height of the wind tunnel model. The incidence and side slip angles are set to zero degrees (for more details the reader is referred to Cock et al.[2]).

The flow calculations are carried out for the full configuration utilising a multi-block grid consisting of 138 blocks containing 4.78 million grid cells. It was verified that the largest vortices in the separated flow region around the nozzle are captured by at least 32 cells, following the rule-of-thumb issued by Spalart [16].

The physical time span that must be calculated is determined by the anticipated lowest dominant frequency for the separated flow in the vicinity of the nozzle. The Strouhal number of this frequency was found to be $St \approx 0.23$ (based on the free stream velocity and the base diameter) in the wind-tunnel experiment. Simulations are performed for a physical time span equal to 6 periodic cycles for this Strouhal number. This should be sufficient to get an impression of whether the relevant flow physics are captured, but not enough for statistically convergence.

In order to get an appreciation of the computed dynamics, the Power Spectral Density (PSD) of the buffet pressure is evaluated for a single location on the nozzle. These numerical results are compared to experimental results obtained in the DNW-HST wind tunnel for a much more complex geometrical model. It is emphasised that the experimental data is used only for a qualitative assessment, as there exist essential differences between the computational and experimental configuration. For a meaningful comparison, the PSD's should be computed for the same (non-dimensional) time span. As the X-LES computation spans only a fraction of the time span in the experiment, the presented PSD's are based on the time span of the computation. There are 256 data windows available from the experiment for each of which the PSD can be computed.

Figure 7 compares the PSD of the X-LES computation with the PSD of the 36th window of the experiment, which is found to give a minimum deviation between the two. This shows that the computational and experimental results can have comparable frequency contents in the selected window. There is a first indication that the numerical algorithm is capable of capturing typical dynamic flow physics observed in the experiment.

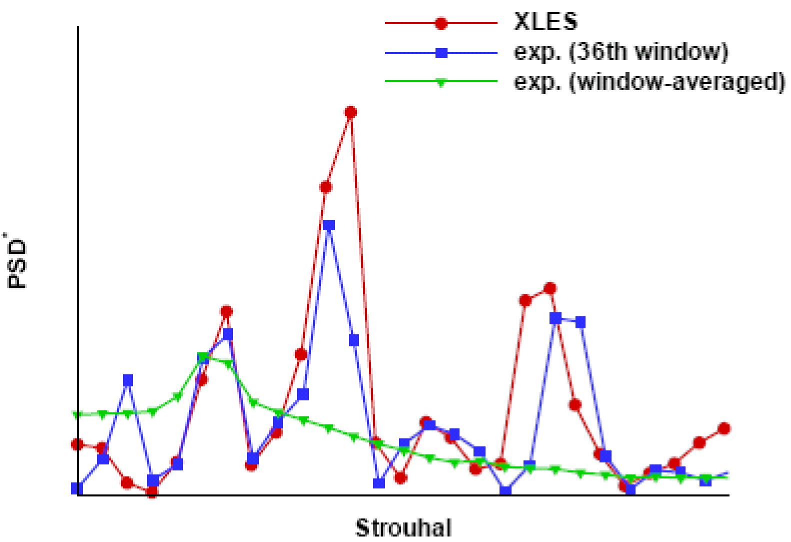


Figure 1 Computational and experimental spectra for the simulation of a generic space launcher with the X-LES model.

# 4 LIMIT CYCLE OSCILLATIONS OF TACTICAL AIRCRAFT

## 4.1 Application

Flutter is a dynamic phenomenon related to the instability of a flexible structure submerged in fluid flow, the so-called aeroelastic problem. Depending on the flight speed and altitude the mutual interaction between the structural vibration and the unsteady aerodynamic forces generated by the motion of the structure can lead to stable or unstable response. Engineers are commonly interested in the specific condition at which the aeroelastic instability begins to occur, i.e. the flutter boundary. The subject of non-linear flutter concerns with two aspects: prediction of flutter boundary and analysis of conditions beyond the flutter boundary, i.e. post-flutter condition. The first issue is related to the response of the aeroelastic system to a (small) perturbation about an equilibrium condition. Non-linearity would influence the flutter boundary through the nonlinearity of the equilibrium condition. The second issue concerns the genuinely non-linear problem in which an interesting post-stall condition is determined by nonlinear phenomena. The second issue will be further discussed in this section.

It is well-known that several types of modern fighter aircraft suffer from sustained vibrations known as the Limit Cycle Oscillation (LCO). This type of vibrations can hamper the effective use of these aircraft, mostly when carrying specific types of stores. For instance, the Royal Netherlands Air Force (RNLAF) operates F-16 fighter aircraft which are susceptible to limit cycle oscillation for certain heavy store configurations. Investigation of limit cycle oscillations is important in association with operational limits, safety aspects and also the influence on maintenance aspects such as fatigue. The repeating loads during LCO may have some influence on the life cycle consumption.

## 4.2 Modeling barriers

Traditionally engineers were interested only in the specific condition at which the aeroelastic instability begins to occur, i.e. the flutter boundary. The flight condition exceeding this flutter boundary has commonly been considered catastrophically dangerous (being a vibration with increasing amplitude causing structural failure) and therefore had to be avoided, i.e. a limit in the flight envelop. However, in the case of fighter aircraft with specific heavy store configuration, practice has shown that the post-flutter condition is in general not disastrous. Instead, the growth of the amplitude of the structural vibration is limited up to a certain, not yet destructive, level.

It is generally accepted that the limiting mechanism is some kind of non-linearity in the aeroelastic system. Computational model to analyse LCO should therefore take into account the relevant nonlinear effects. For a complex aeroelastic system such as a tactical aircraft the nonlinearity may originate from various sources, e.g. structural nonlinearity (stiffness, damping, free-play), aerodynamic nonlinearity (shock-induced flow separation, shock wave motion, vortex dynamics), control-system nonlinearities, etc. A large number of investigations have been, and are, devoted to this topic to investigate the cause of the problem. However, until now a comprehensive insight into the problem has not been reached.

### 4.3 Computational barriers

The aeroelastic simulation of LCO is computationally demanding, because of the following:

1. The multi-physics nature of LCO requires a coupling of structural and aerodynamic modules, which in general requires extra iterations;
2. The dynamic nature of the phenomenon requires a significant number of time steps to be simulated, especially since there is a long transition period until the correct LCO amplitudes are reached;
3. The inherent non-linear nature of LCO may inhibit fast convergence of the numerical algorithms;
4. Since LCO occurs at flight conditions which are unknown a priori, parameter studies have to be carried out.

### 4.4 Numerical algorithm development

The equations governing the mechanics of an aeroelastic system consist of the equations governing the dynamics of the structure of the aircraft and the equations governing the flow field around the aircraft. The global strategy in developing a method suitable for end-user application consists of the following steps:

1. Use a common model and approach as much as possible for each of the subsystems. The nonlinear Euler/Navier-Stokes equations are used to model the flow field around the aircraft. A capability of robust automatic grid deformation is embedded in the aerodynamic method with the input of surface deformation. A finite element method is employed to solve the structural dynamic problem in modal coordinates because LCO phenomenon occurs at a relatively low frequency which can be accommodated properly using a modal approach.
2. Develop a robust conservative synchronous coupling between the subsystems. Two issues involve in the coupling algorithm, i.e. in space and time. The first concerns the transfer of aerodynamic loads from the aerodynamic subsystem to the structural subsystem and the mapping of structural deformation back to aerodynamic subsystem. The latter aspect concerns the synchronisation in time between the subsystems. Different from commonly used staggered schemes, an iterative algorithm is used to let all subsystems converged at the same time. Efficient aerodynamic, structural or prognostic extrapolation methods are employed to ensure fast convergence. In practice 2 to 4 fluid/structural iterations are needed in each time step.
3. Identify the most relevant physics to provide directions for reducing the modelling complexity.

For a more detailed description of the Computational Aeroelastic System (CAS), the reader is referred to Prananta et al. [13].

As an example of reducing the geometrical complexity, consider the F-16 aircraft shown in Figure 2 (the upper left figure). From a geometry point of view, the configuration is extremely complex, i.e. including AIM-9 missiles mounted at the wing tips and at the outboard under-wing attachments, Mk-84 bombs mounted at the mid under-wing attachments, and full 370-gallon fuel tanks mounted at the inboard under-wing attachments. The main assumption underlying the present LCO investigations is the finding in previous work, for example Meijer et al. [9]-[11], that the origin of LCO is the occurrence of shock-induced flow separations at the outer part of the wing. This observation has been used as the basis to consider geometry simplifications which have also been validated using steady and unsteady measurement data. The under-wing stores and the wing tip missiles are not included in the aerodynamic computations. However, the wing tip launcher has been maintained in the model. Also the tail section of the aircraft has been omitted. The reduction of the geometrical complexity is shown in Figure 2. Because the vibration modes which induce the LCO are a global property of the structure, in the structural dynamics module of the CAS system, all components of the configuration are always taken into account in contrast to the aerodynamics module.

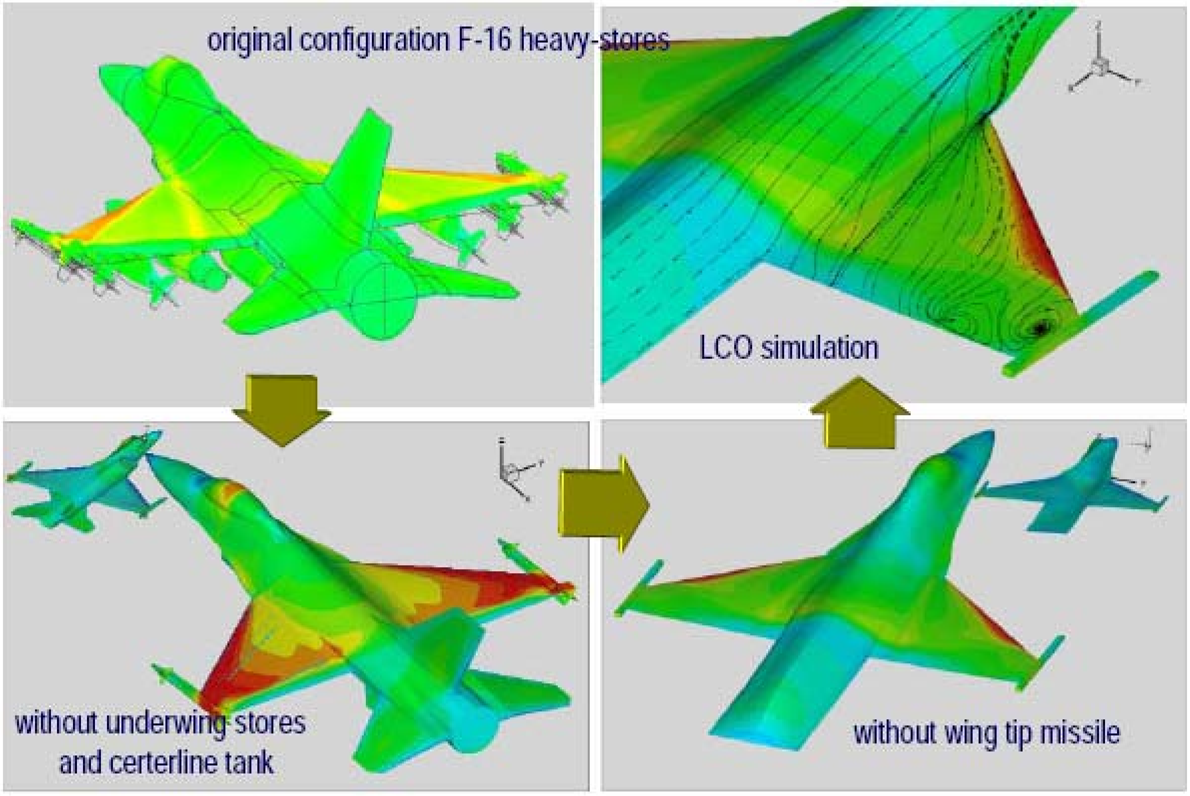


Figure 2 Reduction of the geometrical complexity of the F-16 heavy store configuration for the LCO simulation

### 4.5 Examples

Dynamic aeroelastic simulations have been carried out for a flight speed that corresponds to $M\infty = 0.90$ at an altitude of 10,000 ft and zero angle of side slip. In total 10 symmetrical and 10 anti-symmetrical mode shapes have been used in the simulation. Figure 4 shows the time history of the generalized coordinates for the anti-symmetrical mode shapes during the same simulation. Only the first two anti-symmetrical mode shapes, the first wing bending and first torsion mode shapes, feature oscillations that initially increase up to a certain level and remain at those levels next. The other mode shapes damp out. It can be concluded that the first and second anti-symmetrical modes are coupled by the aerodynamic forces and contribute to the instability.

A series of dynamic aeroelastic simulations have been carried out for varying angle-of-attack with constant small structural damping of 0.002 and zero flaperon deflection. Each simulation required about 48 CPU hours on one processor of NEC SX-5 computer. On a recent top 500 (# 500) this study would require less than one hour. The results of the simulations, i.e. time response and amplitude of LCO, are given in terms of the acceleration at the forward part of the missile launcher. Using the data of the aircraft weight and the dynamic pressure, the normal load factor at each condition can be estimated. Note that the load factor should be considered as indication only because the computations were not carried out in a free-free manner and moreover the horizontal tail plane has not been modeled. The results expressed in term of normal load factor are given in Figure 3.

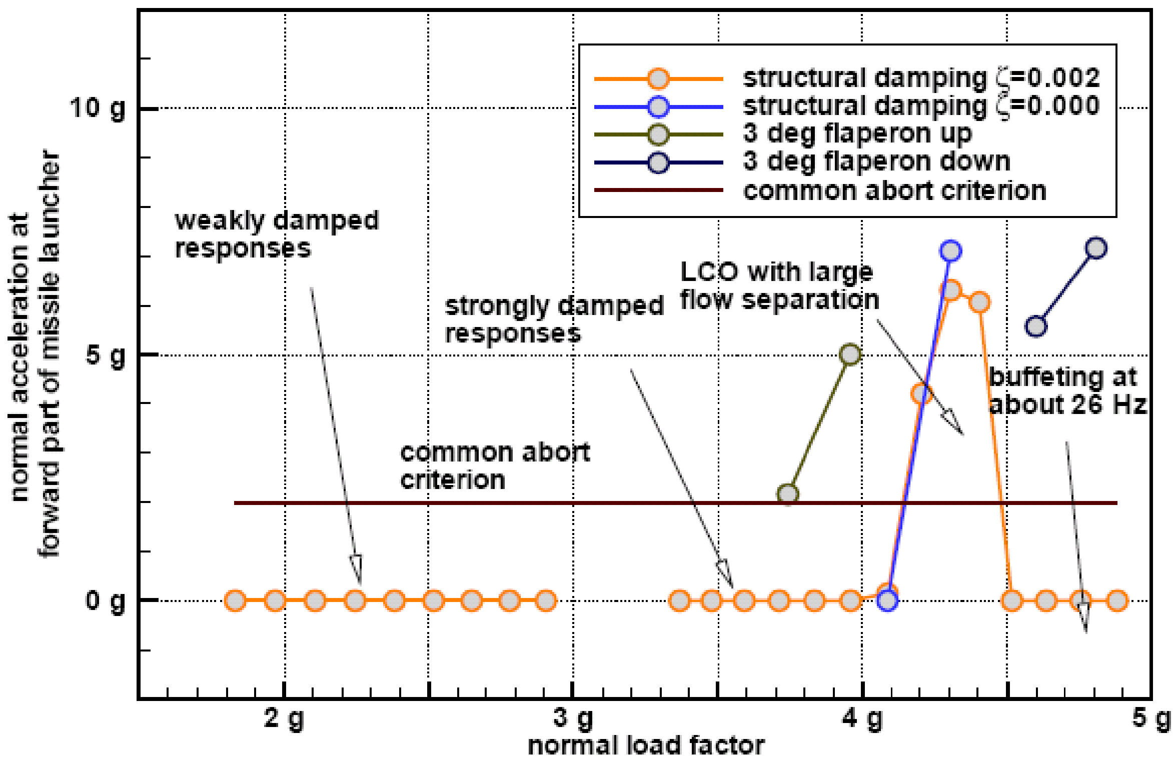


Figure 3 Diagram of LCO amplitude with respect to normal load factor for the F-16 aircraft; $M_\infty = 0.90$, FL100, Reynolds-Averaged Navier-Stokes flow model

It is generally known that certain type of LCO occurs at a narrow band of angle-of-attack. LCO has been found at a range of angles-of-attack between 6.5 degree and 7.5 degree. Strongly damped responses are obtained between angle-of-attack 5 degree and 6.5 degree. At lower angles of attack, where the flow is fully attached, the response are weakly damped. At a higher angles of attack the response are dominated by the buffet flow due tolarge flow separation.

The amplitude at low angle of attack obtained from these simulations are significantly higher than that observed during the flight test of RNLAF. The tendency that numerical simulations produce LCO amplitude which is much higher than that observed in the experiment seems to be quite common, see e.g. Thomas [18] and numerical simulations of the DLR experiment [15]. This suggests the need for improvement in the modelling. Currently, nonlinearity in the structure is investigated.

As is apparent in Figure 4, about half of the time steps are required to attain the steady LCO levels. Since LCO is largely sinusoidal in character, the computational complexity of LCO simulations could be reduced significantly if the simulations were performed in the frequency domain. This is the subject of future research.

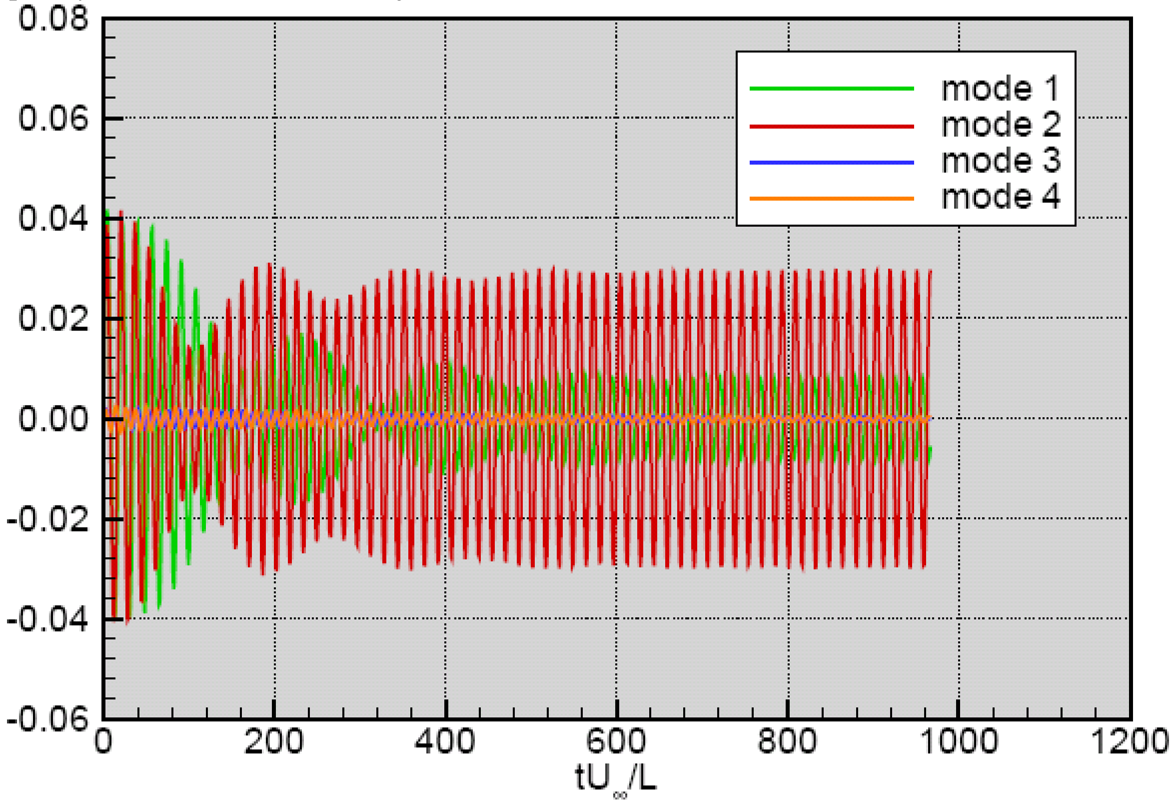


Figure 4 Time history of the generalised coordinates for the anti-symmetrical modes of the F-16 aircraft during the LCO simulation representing the dynamic deformation components; $M_\infty$ = 0.90, $\alpha$ = 7 degree, FL100, Reynolds-Averaged Navier-Stokes flow model

## 5 BLADE-VORTEX INTERACTION FOR HELICOPTERS

### 5.1 Application

The inherently dynamic behaviour of the flow around helicopters poses a great challenge to flow simulation algorithms. Not only the combination of rotating and non-rotating geometries increases the algorithmic complexity, also the presence of a wide range of different scales complicates accurate and efficient simulation of rotor flows. In this section, we will concentrate on the phenomenon of so-called blade-vortex interaction (BVI), where the tip vortex of a preceding blade interacts with a next blade, causing strong pressure fluctuations. These pressure fluctuations are audible as the typical, and annoying, 'wopwop' sound of helicopters. The pressure fluctuations may also cause vibrations running through the elastic rotor blade into the rotor hub, engine, and fuselage, shortening the fatigue life of helicopter components and decreasing the comfort of crew and passengers.

### 5.2 Modeling barriers

Being a vortex phenomenon, BVI is well modeled using Euler equations. The numerical dissipation will limit the vorticity in the vortex core, acting as the physical diffusion which is not modeled in the Euler equations. Essentially, the resolution defines the attainable vorticity levels. In other flow regimes, stability-critical phenomena such as dynamic stall may occur. These phenomena require turbulent models for massive flow separation.

### 5.3 Computational barriers

The different spatial scales for BVI are the vortex core, which is typically a tenth of the rotor blade chord, which itself is typically a tenth of the rotor radius. So the vortex core is one hundredth of the rotor radius. The different temporal scales are the blade passing frequency and the duration of the blade-vortex interaction, which is about as long as it takes the blade to rotate one degree. For a standard four-bladed helicopter, where the blade passing frequency corresponds to a rotation of the blade by ninety degrees (the blades are assumed to be identical), this again leads to a ratio of about one hundred between the largest and smallest scales. In general, numerical methods require ten cells to resolve a wave in one dimension, so a first estimate on the back of an envelope on the required resolution for a BVI simulation would be $(10^3)^3$ spatial elements and a time step of 1/1000 of the blade passing interval. Clearly, the multi-scale behaviour of the flow requires numerical methods which can generate and accommodate highly non-uniform meshes in both space and time.

### 5.4 Numerical algorithm developments

An important step in turning BVI simulations feasible, is the realisation that rotor flows are time-periodic at the blade-passing frequency. As shown in Van der Ven et al.[20], conventional time stepping schemes converge slowly to periodic solutions, even more so when the time steps are small compared to the period. Imposing the periodicity of the solution alleviates this problem. This can be accomplished by solving the flow equations simultaneously for all time steps in a period, on a four-dimensional space-time mesh containing all these time steps, Van der Ven et al.[20]. The fact that the flow equations are hyperbolic in both space and time simplifies the solution process. Standard convergence acceleration techniques such as multi-grid can be applied to the space-time simulation; hence the name multi-time multi-grid algorithm for this four-dimensional solution technique. In essence, the dynamic problem of solving the time-dependent flow equations is reduced to solving a steady-state problem.

The next step is to fully exploit the multi-time multi-grid algorithm. As with fixed-wing aircraft, rotorcraft need to be trimmed in order to obtain the right lift, and for rotorcraft also zero rolling and pitching moment. Since the integral forces can only be obtained at each blade passing, conventional time integration schemes suffer from even longer simulation times than needed to converge to a periodic solution. The four-dimensional algorithm has all trim information directly available. Next to trim, the aerodynamics is tightly coupled with the elastic motion of the blades. Again, conventional time integration schemes require complicated coupling procedures to ensure time accuracy. The steady-state nature of the four-dimensional algorithm simplifies the coupling: if the coupled simulations converge, the temporal accuracy of the coupled solution is equal to the accuracy of the time discretizations of the structural and flow solver.

The conventional way of tackling a multi-scale problem is to apply mesh adaptation in order to obtain the required resolution where it is needed, and only there. Using the four-dimensional algorithm it is also possible to apply mesh adaptation in time, not only refining the mesh where a vortex is present, but also *when* a vortex is present. A discretization method capable of local mesh refinement is the discontinuous Galerkin finite element method, Cockburn [1], which will be applied here in the space-time formulation of Van der Vegt et al. [19].

### 5.5 Examples

The descent flight condition of the HART-I experiments [17] is simulated for a modified BO105 model rotor. The modified BO105 model rotor is selected as being representative for modern helicopter blades. The modified model rotor has a parabolic tip. Flight conditions are a tip Mach number $M_{tip}$=0.64, thrust over solidity CL/$\sigma$=0.056, advance ratio $\mu$=0.152 and a shaft angle of 4.2 degrees, which correspond to HART-I data point 140 flow conditions. Details of the simulation can be found in Van der Ven et al. [21].

A three-dimensional mesh containing about 700,000 grid cells has been generated. Subsequently a four-dimensional mesh is generated by rotating this mesh over the azimuth angle until a complete period (which is one fourth of a revolution) is obtained. The time step on the mesh corresponds with an azimuthal increment of 4.5 degrees, so the final mesh contains 13.5 million cells. Subsequently, the aerodynamic module is coupled with the elastic and trim modules to obtain a trimmed solution with deforming blades.

Because of the limited spatial and temporal resolution this solution does not display the BVI phenomenon, and hence the mesh refinement module is used to locally increase both the temporal and spatial resolution. The effective azimuthal increment in the vortex regions has been decreased to 1.125 degrees. The spatial resolution in the vortex regions has been improved to enhance the vortex persistence. Impressions of the intermediate mesh and the intermediate flow solution are presented Figure 5. The flow clearly exhibits vortices of various ‘ages’.

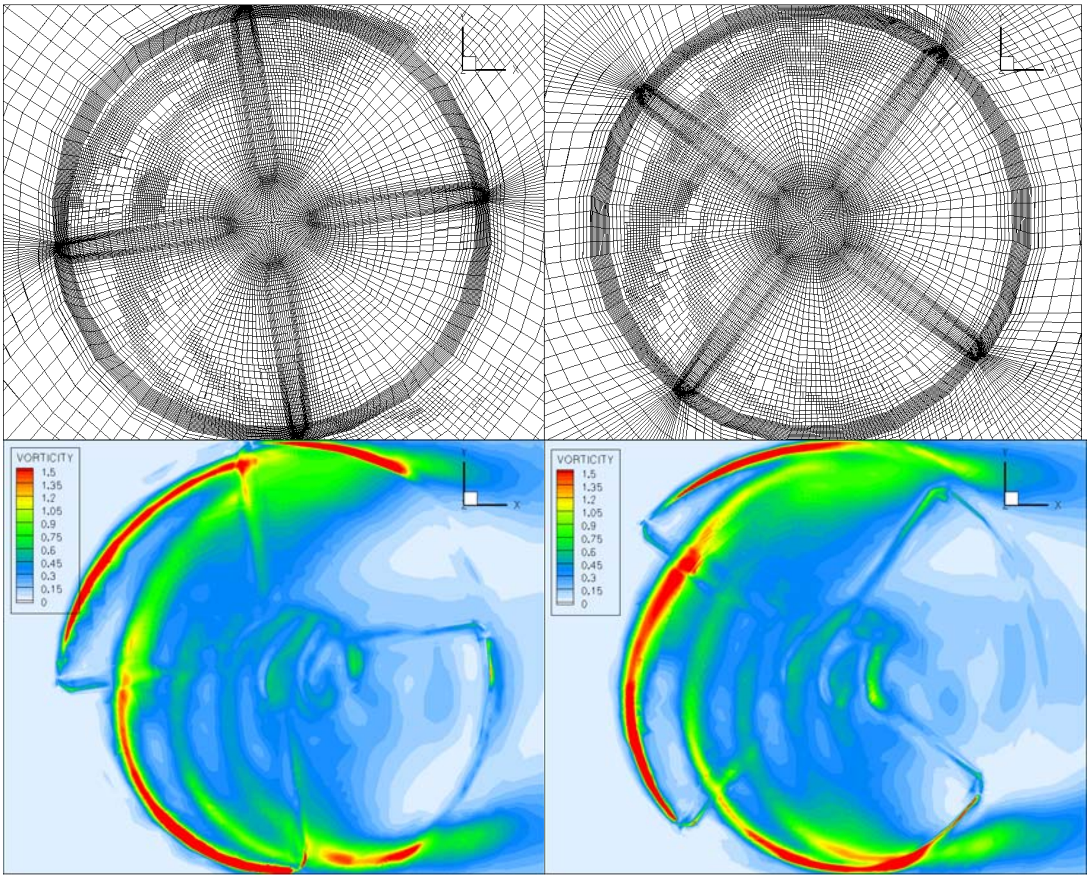


Figure 5 Locally refined meshes and flow results (vorticity magnitude) at azimuthal angle $\Psi=9^o$ (left) and $\Psi=54^o$ (right). Cross-sections are taken at z/R=0.085, where R is the rotor diameter and the hub center is located at z=0. Flow is from left to right.

After refinement the spatial resolution in the vortex regions is 0.005R (where R is the rotor radius) and the temporal resolution is 1.125 degrees. Note that this is, as yet, insufficient to capture BVI. The mesh contains 28 million elements. Please note that the DG method solves five equations per conservative variable, so this mesh represents 700 million unknowns. In order to obtain this mesh, about ten mesh refinements are required and after each refinement the flow solution is converged for another fifty multi-time multi-grid iterations. The current mesh size is at the limit of the current computational capabilities of NLR: roughly 50GB of the NEC SX-5/8B is needed. Even though the NEC SX-5/8B can only be classified as a compute server at best, computing times are relatively modest: 50 iterations on the final mesh take less than one night on four processors.

At the core of the current algorithm is the mesh refinement algorithm. Considering the memory requirements of the MTMG algorithm it is of the greatest importance that cells are only added where and when they are needed. More efficient refinement sensors are required.

## 6 CONCLUSIONS

In this paper three challenging aerospace applications driving numerical algorithm development have been discussed. Two directions can be distinguished: mono-disciplinary and multi-disciplinary applications. For mono-disciplinary applications the focus is on higher fidelity, through higher accuracy in both modeling and numerics. Once the new algorithms become mature, efficiency is the challenge. For multi-disciplinary applications the focus is on efficicieny, both in modeling and numerics. The following developments are foreseen in the near future:

- more efficient time integration schemes and time parallel algorithms,
- robust higher order schemes,
- nonlinear structural modeling,
- frequency domain methods for LCO analysis,
- multi-scale methods (in particular effective adaptation sensors).

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