Concepts for Morphing Aircraft

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Abstract: This paper gives an overview of a number of concepts for morphing aircraft investigated during the morphing aircraft project led by the author between 2005 and 2008. The concepts cover a range of approaches, from compliant mechanisms, aeroelastic tailoring, multi-stable composites, to active winglets. The objective is to highlight the range of potential concepts, to show that there are still many opportunities for research in this area, and to motivate further investigations.

Key words: Aeroelasticity, Multi-stable Composites, Compliant Mechanisms, Winglets

1. INTRODUCTION

The design of conventional fixed wing aircraft is constrained by the conflicting requirements of multiple objectives. Mechanisms such as deployable flaps provide the current standard of adaptive aerofoil geometry, although this solution places limitations on maneuverability and efficiency, and produces a design that is non-optimal in many flight regimes. The development of new smart materials together with the always present need for better UAV performance is increasingly prompting designers towards the concept of morphing aircraft [1, 5, 11, 15, 16, 17, 18]. These aircraft possess the ability to adapt and optimize their shape to achieve dissimilar, multi-objective mission roles efficiently and effectively. One motivation for such uninhabited aircraft are birds that morph between cruise and attack missions by changing their wing configuration accordingly. Birds also use camber and twist for flight control. The Wright Brothers used wing warping as a seamless flight control in their first flying machine. Morphing wings for flight control bring new challenges to the design of control laws for flight. Because configuration changes move the aerodynamic centre, control of the aircraft during planform morphing requires attention. Hence both morphing mechanisms and control systems must be considered.

The structural technologies available to achieve shape changes in a morphing aircraft fall into two major categories, namely planform changes using rigid mechanisms, and compliance (for example wing twist or compliant mechanisms). Methods using compliance are of interest in this paper, and in particular two approaches will be considered. The first is aeroelastic tailoring, where the stiffness distribution of a conventional wing layout is optimized based on weight or aerodynamic objectives. Composite materials present high specific strength and stiffness ratios, and primary flight structures, such as wings or fuselages, are mainly designed using stiffened panels. Structures made of composite materials can be stiffness tailored, and this is a significant advantage over their metallic counterparts. The second approach is the use of compliant mechanisms to deform the camber of the airfoil section. Such structures may be designed using topology optimization or direct optimization of the geometry of truss structures.

Although it is possible to create morphing structures just by deforming the main structural components, such approaches can require significant energy to overcome the stiffness of the material. Furthermore, significant aerodynamic performance gains are achievable only through large overall changes in the aircraft geometry via wing sweep, area and/or span. At present, in both of these categories, such medium to large scale changes are obtained with complex and sophisticated mechanical devices significantly increasing installation and maintenance costs, as well as the structural weight of the airframe. The possibility to use multi-stable structures to produce large geometry changes will be considered in this paper. The final concept considered is the use of novel control effectors, namely active winglets.

This paper presents an overview of a number of concepts investigated during the morphing aircraft project at Bristol University and funded by an EC Marie-Curie Excellence Grant. The concepts include the use of compliant mechanisms, aeroelastic tailoring, multistable composites and active winglets.
2. COMPLIANT MECHANISMS

The structural systems employed to create the smooth variable camber have ranged from a mechanistic approach, and through a number of patented designs, to the more recent application of compliant and flexible structures. This section investigates statically and kinematically determinate truss structures as candidates for the formation of a conformal trailing edge. The promise of these structures is that by the introduction of a number of active actuated elements in place of passive truss elements then the structure would deform increasing the elastic energy. An approximation of predefined surface form is created, differentiated from the undeflected form by variation of the aerofoil mean line. Two methods of comparing the target and deflected surface forms are presented defined as geometry comparison and shape comparison methods [2]. In addition a number of methods for locating the optimal placement of the active elements are employed that may be categorised as exhaustive, heuristic and iterative improvement methods. The performance of the selection methods with varying actuator strain constraints, actuator quantity and truss topology is discussed.

2.1. Displacement Targets

Two displacement control objectives are considered, denoted as a geometry objective and a shape objective.

Geometry objective. Considering an initial NACA 0012 aerofoil profile, the trailing edge portion of the profile is defined as the region 0.6c to c, where c is the chord length which in this symmetric case is equal to the mean line length. The mean line in this region is defined according a quadratic function. The definition of a conformal trailing edge fixes the mean line and slope at 0.6c, and thus gives a single degree of freedom related to the amplitude of the deformation. In addition we assume that the profile thickness at each point on the length of the mean line remains equivalent in the initial and target profiles. The vector of target displacements may now be defined as those displacements required to transform the surface joints of the aerofoil to the target profile defined by a non zero value deformation. The geometry objective is thus quantified by calculation of the Sum of Squares Error ($SS_{E_y}$) of the displacement from this target.

Shape objective. The shape objective forms the target profile in a similar manner to the geometry objective, however in this case the nodal positions that define the target profile are dependent not just on the mean line function but also the location of the surface nodes of the deflected structure. In this case the target nodal displacement is defined as the displacement required to transform a surface node to the nearest point on the corresponding upper or lower surface of the target profile. The shape objective can now be quantified by calculation of the Sum of Squares Error ($SS_{E_y}$) based on the target deflection vector for the shape objective.

2.2. Actuator Location

The actuation location problem is essentially one of subset selection, where the optimum candidate columns of the candidate input matrix are selected. Neglecting the aerodynamic load for illustration, the objective is to choose the best columns of the candidate response matrix that represents the target deformation [2]. Four methods were applied that either allow searching of the entire model space, searching of a fraction of the total model space or alternatively provide a systematic method to examine a path through the model space.

For problems in which the number of possible solution combinations is suitably small it is possible to implement an Exhaustive Search (ES) procedure for the evaluation of all possible actuator location combinations. Within the confines of each possible subset, a gradient based constrained optimisation routine is employed to evaluate the optimum values of actuator forces. If the search space within all possible subsets is convex then this procedure reaches a global optimum at the expense of high computational cost.

A Genetic Algorithm (GA) is a search method that seeks to improve the selection by applying the principles of natural selection. For similar problems in which the use of an exhaustive search was not feasible then heuristic methods that permit inferior cost moves, such as a GA, were recommended. The limiting features of a GA are the requirement for a large number of objective function evaluations and typically the rapid location of optimal regions within the search space but with slow convergence to an optimum value.

Stepwise Forward Selection (SFS) attempts to provide a path through the search space whereby a single actuator location is added at each step, keeping previously selected locations until a particular termination criterion is met. This may be if insufficient improvement to the model is made by the addition of the variable or the full number of subsets is reached. The column that provides the closest correlation to the current residual vector is selected in turn, based on angles between vectors. The applied actuation force is limited to a range of values dependent upon the real actuator constraints, i.e. the maximum and minimum available strains.

The process of applying the maximum possible regression coefficient or that which results in an orthogonal residual implies a greedy selection process that may overlook possible improved combinations of actuator locations and strains. A number of well established methods are able to provide an improved solution compared to SFS methods. Backward Selection begins with the full model and sequentially removes variables from the starting set until a subset of required size is reached, whilst methods such as WOBI begin with the desired number of variables and sequentially add and subtract variables from the subset. For both methods the application of upper and lower bounds to the regression coefficients proves difficult to implement.

Less aggressive forward selection techniques including the Lasso and Incremental Forward Selection (IFS) algorithms have been developed more recently. IFS is a more cautious version of SFS and at each step the coefficient of the variable most closely correlated with the current residuals is incremented a fixed step. By reducing the step size, improvements in the correlation between the actual and target structure re-
response may be achieved at the cost of an increased computational burden.

2.3. Example: A 14 element truss structure

A simple 14 element truss forming the trailing edge between 0.6c and c (c = 100mm) of a NACA 0012 aerofoil is illustrated in Fig. 1. Each of the 14 elements provides a possible location for actuator substitution with the goal of replicating either the illustrated shape or geometry objective forms. Both of the objectives are defined by the deformation of the mean line resulting in a trailing edge deflection that would require 10° equivalent flap angle in order to achieve a similar aerodynamic force with a discrete hinge device.

Figure 2 shows the effect of varying the maximum and minimum allowable actuation strain on the geometry objective (SSSEg). Three actuators were located in the structure using four methods: Exhaustive Search (ES), Genetic Algorithm (GA), Stepwise Forward Selection (SFS) and Incremental Forward Selection (IFS) and for maximum actuator strains of $\epsilon_{\text{max}}$. The GA provided a match with the optimal ES solution for all values of $\epsilon_{\text{max}}$. For $\epsilon_{\text{max}} = 0.01, 0.025, 0.05$ the search space was constrained by limitations on the maximum absolute actuation strain, however for $\epsilon_{\text{max}} \geq 0.07272$ the search space was constrained only by the number of actuator substitutions. In this instance the minimum obtainable (SSSEg) was 0.57536.

SFS provided the optimum actuator locations and strains when the search space was limited to $\epsilon_{\text{max}} \leq 0.07272$. If $\epsilon_{\text{max}}$ was increased beyond this the method became overly greedy, resulting in a large first step. This aggressive selection method bypassed improved selections, resulting in high applied strains in the first selection step instead of a more even actuation distribution that characterises optimal designs. As the search space became larger the results in this case illustrate the worsening performance of the SFS method when compared directly to results obtained with reduced $\epsilon_{\text{max}}$. The conservative progression of the IFS method did not permit the selection of the optimal locations; for $\epsilon_{\text{max}} = 0.05$ it was the worst performing method. However as the size of the search space was increased IFS did not suffer the performance deterioration that afflicted the SFS procedure.

Figure 3 shows the results for the shape objective ($SSSE_s$) for the 14 element truss with respect to varying $\epsilon_{\text{max}}$. Three actuator substitutions were again selected using the ES, GA, SFS and IFS methods. As with the geometry objective, the GA results matched those obtained with ES and thus they may be considered to be globally optimal. In the cases where $\epsilon_{\text{max}} < 0.07232$ the search space was constrained by $\epsilon_{\text{max}}$. This provided close agreement with the geometry objective solutions. However for $\epsilon_{\text{max}} \geq 0.07232$ the location selections switched to the elements forming the lower facing surface of the aerofoil. Comparison of these shape objective results with those derived using the geometric objective reveals a similar trend, whereby in the region $0 \leq \epsilon_{\text{max}} \leq 0.05$ large reductions in the ($SSSE_g$) and ($SSSE_s$) were achievable with small increases in $\epsilon_{\text{max}}$. The SFS method for $\epsilon_{\text{max}} \leq 0.025$ provided the optimal selection however, as with the geometry objective, as the search space was increased the greedy selection method fell in to a non-optimal path. In each case taking an excessively large first step resulted in highly localised deflections of the structure. The IFS method obtained optimal actuator selections when $\epsilon_{\text{max}} \leq 0.025$, however as $\epsilon_{\text{max}} \geq 0.1$ the method provided optimum locations with non-optimum regression coefficient values.

2.4. Example: Two 1752 element truss structures

Figure 4 illustrates two statically and kinematically determinate truss structures comprising 1752 elements and referred to as A and B. Each structure is geometrically identical and topologically similar; the difference being the connectiv-
ity of elements adjacent to the surface. As with the 14 element truss both form the trailing 0.4c fraction of a NACA 0012 aerofoil.

Figure 5 gives the shape error results for optimising the location of 88 actuators within trusses A and B using the GA, SFS and IFS methods. The results for the geometry error were calculated but are not shown here. Beginning with truss A, the GA provided the best performance, and little improvement in the SSEs value was observed at values of $\epsilon_{\text{max}} \geq 0.1$, when using the GA search method. The GA method outperformed both the SFS and IFS procedures, however here a notable improvement of the IFS method over the SFS method was observed. This is due to the alteration of the target vector as the actuator selection process is progressed. As each actuator is added to the subset the structure is subjected to a displacement that, in the case of the shape objective, alters the target displacement. By taking large steps towards the target vector the SFS method oversteps possible improved selections that may appear as the target alters with the values of the regression coefficients. Truss B exhibited similar characteristics in the selection of optimal locations as for truss A, whereby the GA proved to be the best selection method in terms of reducing the shape error in all cases. Improved results were obtained using the IFS method when compared with the SFS method.

Comparison of the results between the two truss patterns for the GA selections revealed an improved performance of truss A at $\epsilon_{\text{max}} \leq 0.05$. Figure 6 illustrates the deflected structures for both A and B with 88 actuator substitutions and $\epsilon_{\text{max}} = 0.1$. Some common features are evident, including concentrations of actuator substitutions along the upper surfaces and between 0.6c and 0.75c on the lower surfaces of the trusses. The deflected result of truss A experienced large rotations of the elements in the upper surface region between 0.8c and 0.85c leading to a number of overlapping elements. In addition truss B required reduced strain in the upper surface element in the region 0.9c to c due to the increased deflection of the structure forward of this point.
3. AEROELASTIC TAILORING

Composite materials have the potential to be stiffness tailored, which fits in well with the purpose of aeroelastic tailoring. Composites can show anisotropic properties or elastic coupling terms. Two common types of anisotropy are membrane and flexural, and these are mainly related to unbalanced or off axis laminates. Flexural anisotropy is influenced by the laminate stacking sequence, whereas membrane anisotropy is affected by the laminate volume fractions or ply percentages content. For instance, if flexural anisotropy is neglected in buckling analysis, buckling load factors can be conservative. Designing for elastic tailoring is characterized by the use of the composite anisotropy to improve composite structural performance. For example, a composite plate with flexural anisotropy (that is bend-twist coupling terms) under bending loads will twist. Due to practical, manufacturing considerations, composites have been restricted to symmetric or mid-plane symmetric laminates with 0, 90, 45 and -45 degree ply angles.

Over the years, local and global optimization techniques have been developed for composite design. Local optimizations have concentrated on the specific components or part subassemblies of an aircraft structure made mainly of laminated composites. Gradient, Genetic Algorithm (GA) or Gradient-GA based techniques have been developed. Composite materials may be modeled and characterized by their stacking sequences or alternatively by lamination parameters. On the other hand, global optimizations have addressed the wing as a whole accounting for the interaction between components. Several methods have been proposed from single to multilevel approach, accounting for aero-structure interaction having high fidelity models, characterizing the wing as a beam and approximating the aerodynamics, and so on. However, existing studies have limited use of aero-structure coupling, practical design rules, anisotropy in the laminated composites and structural sizing constraints. Herencia et al. [9] reviewed the literature on the optimization of composite lay-ups and the application to morphing aircraft structures.

The aim is to provide a method to design an aircraft wing of laminated composites that possesses morphing capabilities employing aeroelastic tailoring. Morphing is achieved passively so that the aircraft wing will adapt itself to improve its performance during the designed flight conditions. The approach consists of an aeroelastic steady-state scheme with aero-structure coupling embedded within a global optimization. The global optimization is divided into two levels. At the first level, Mathematical Programming (MP) is used to optimize the wing under structural and aerodynamic constraints. The wing-box panels (skins and spars) are modeled using lamination parameters accounting for their anisotropy. The panels are assumed to be symmetric or mid-plane symmetric laminates with 0, 90, 45 or -45 degree ply angles. Each of the wing-box panels is subjected to a combined in-plane loading under strength, buckling and practical design constraints. At the second level, the actual lay-ups of the wing-box panels are obtained using a GA, accounting for manufacture and design practices. Thus the aeroelastic steady-state simulation with aero-structure coupling is embedded within a two level global optimization, which accounts for composite material anisotropy, lift and induced drag variations as well as internal load redistribution under structural and aerodynamic constraints.

The global optimization strategy is divided into two stages similar to that given by Herencia et al. [9]. At the first stage, the wing-box panels are optimized using lamination parameters and gradient based techniques under structural and aerodynamic constraints. The optimum thicknesses and values of the lamination parameters for each of the panels are identified. At the second stage, a GA is used to target the optimum lamination parameters to obtain the actual lay-ups for each of the skin and spar panels.

The objective function at the first stage is the mass of the wing-box panels (skins and spars), and the inequality constraints consist of structural constraints such as strength, buckling or practical design rules, and aerodynamic constraints (lift and induced drag). The design variables are the thicknesses and the membrane and bending lamination parameters of the skin and spar panels. Note that for every optimization cycle at this level, an aeroelastic-steady state scheme is performed.

Classical Laminate Theory (CLT) is applied to the wing-box panels (skins and spars) assuming laminates are symmetric or mid-plane symmetric, and provides the link between the discrete lay-up and the continuous model used for the optimization. However, not all laminates defined by a particular set of lamination parameters can be manufactured, and constraints must be included by defining a feasible region in the space of lamination parameters. Furthermore practical design constraints are applied, for example that at least 10% of each ply angle (0, 90, 45, -45) should be included.

At the second stage, a standard GA is employed at this level to solve the discrete lay-up optimization problem. The lamination parameters from the first optimization level are targeted to obtain the actual lay-ups for the skin and spar panels. Note that at this level the GA is applied separately to each of the wing-box panels.

3.1. Numerical Example

A typical UAV aircraft with a swept back wing configuration was used to test the proposed two level hierarchical approach [10]. For the flight case, which constitutes the up-bending case, it was assumed that the aircraft is flying at 5000m, at a speed of 100m/s, and an angle of attack of 8°. The load factor was assumed to be 2.5 representing a maneuver case. For sizing purposes an up-bending case and a down-bending (70% of up-bending) case were considered. The airfoil used was NACA 4412. The wing had a sweep back angle of 15°. The wing chord and semi-span were 400 and 1200mm, respectively. The front and rear spars were located at 20% and 80% of the wing chord, respectively. The wing-box was made of P100/AS3501 with a thickness of 0.125mm. The leading edge, the trailing edge and the ribs were made of...
an aluminum alloy. The thicknesses of the leading and trailing edges for the 5 rib bays starting at the wing root were: 2, 1.75, 1.5, 1 and 0.875mm, respectively. Each of the ribs had a thickness of 0.25mm.

Firstly, the two level optimization approach was performed with structural constraints and without aerodynamic constraints (i.e. lift and induced drag were not restricted). Then the aerodynamic constraints are included and optimum design showed a reduction of 1.4% in the drag was achieved. Figure 7 compares the weight of optimal solution in both cases, for the continuous (first level) and discrete (second level) optimizations. Clearly there is an increase in weight when the discrete lay-up is realized, and there is a weight penalty for the improvement in aerodynamic performance. Finally Figure 8 shows the deformation in both cases, and highlights that there is a bend-twist coupling that improves the performance when the aerodynamic constraints are included.

4. MULTI-STABLE COMPOSITES

This section summarises a study on alternative structural configurations aimed at realising large changes in the geometry of the aircraft without making use of rigid bodies discretely hinged to the main airframe. The first concept is for the realisation of a variable sweep wing whereas the second and the third aim at changing the profile of the airfoil along the chord and along the span of the wing. Diaconu et al. [6] investigated different concepts to introduce multistable structures into aircraft systems. Mattioni et al. [12, 13, 14] simulated bistable structures using finite element analysis. Anisotropic composites have also been used to tailor the structural properties for rotorcraft blades and morphing wings.

4.1. Variable sweep wing

A wing with a variable sweep angle offers several benefits from the aerodynamic point of view such as the delay in the increased drag at Mach numbers close to unity and buffet onset. From the structural point of view, there is the advantage that by changing the sweep angle structural loads are redistributed along the span reducing the bending moment requirements at the root sections. Variable sweep wings also pose significant disadvantages. The most accepted and widespread design for a variable sweep angle consists of a rigid wing that rotates around a pivot: this constrains all of the aerodynamic loads to act through the pivot producing a massive concentration of stresses. As a result, the large metal pivot needed to move the wings is complicated to build and install. This more often increases maintenance requirements and decreases fuel performance. An aircraft capable of moving its wing forward for fuel-efficient flight could never be as efficient as an airplane equipped with a straight wing. The same is true for aircraft with swept-back wings; they would always be more efficient than aircraft with swing-wings. It is clear that several improvements are needed to reduce these disadvantages.

The concept for a variable sweep wing structure proposed in this paper consists of two spars with an interconnected truss-rib structure. The geometry of the preliminary design in the straight and swept-back configuration is shown in Fig. 9. The main feature of the wing is the represented by the spars. The curved cross-section provides two advantageous properties to the wing, namely the stiffness to resist bending moment and a hinge-like behaviour to the spar. At first, the shell exhibits an elastic deformation. Then, once a critical load is reached, it behaves like an elastic hinge, where for an increase of the rotation angle a constant force is required. If the load is removed, the shell returns to its original straight configuration.

The particular choice of the truss-ribs is due to the kinematics of the structure, where the ribs must not constrain the spars during the sweeping. This translation occurs when the wing is swept-back; the height of the spar changes and the spar cross-section becomes flat, therefore this degree of freedom is essential for the snap-through of the spars. The wing skin could present problems during the transition between the two configurations, as the wing plan-form changes from rectangular to rhomboidal requiring a consistent shear deformation of the wing skin. At present no definitive design for the skin has been investigated, however the latest studies on elastic skins suggest that a reinforced silicone based skin may be a suitable candidate. Such a skin would only carry membrane loads and therefore bending and torsion loads on the structure must be carried by the spars.

The advantage of the proposed spar design is that if the forward speed of the aircraft is below a given value, then the equivalent moment produced by the drag force will not be enough to snap the spars, resulting in the wing behaving like a conventional straight wing without any locking requirement. Once the speed is increased, and the drag force overcomes the critical moment to snap the spars, the wing will sweep back. The configuration thus achieved will result from the equilibrium between the internal elastic reactions and the drag force and the sweep angle will dynamically adapt to the variations of the flight speed. In principle it is possible to achieve both positive and negative sweep angle (i.e. forward swept wing for increased manoeuvrability).

In order to understand which parameters are most impor-
tant for the proposed application, a number of different stacking sequences, aspect ratios and curvature radii for the shell structures were tested numerically. Several simulations of the snap-through of the spar were performed and it emerged that the main parameter of interest for the sweep application is the curvature radius of the shell. This radius determines the bending performance as well as the critical load to be applied to activate the sweep mechanism.

A preliminary wing-box structure, identical to the one described above, was manufactured and tested. The spar was manufactured using unidirectional carbon fibre/epoxy prepreg cured on an aluminium tool with a curvature radius of 53.5mm and a length of 1000mm. The truss-ribs were built using 2mm threaded steel rod with plastic ball-cup joints as hinges. The spar successfully snapped from the straight to the swept back configuration under its own weight plus an additional load of 5N at 220mm from the root section. The swept configuration and the position where the snap occurred matched the one computed with the FEA, as shown by comparing Figs. 10 and 11.

The experimental model was also tested statically to investigate the maximum load that the structure could withstand and also to analyse the critical failure modes when the limit load is reached. The wing-box was clamped to a rigid support and loaded. The force was applied at a distance approximately equal to 0.4 of the total length to simulate the resultant of an elliptical lift distribution. The first buckling instabilities were encountered with a load magnitude of 70.3N. Considering that the spars have a maximum thickness of 0.625mm, the load achieved is considered satisfactory. Moreover the collapse of the structure depends greatly on the effectiveness of the joints in the truss-rib structure and these tended to fail when subjected to a tension load. Thus the load bearing capability of the proposed spars is acceptable and a considerable increase in the maximum load could be achieved by re-engineering the rib joints.

4.2. Bistable Blended Winglet

Bistable structures are obtained by introducing a residual stress field into the structure through pre-stress or thermal stresses. In this paper only thermally induced bistable composites are considered, and these structures may be obtained using unsymmetric laminates made of orthotropic material. Because of the difference in the coefficients of thermal expansion parallel to and transverse to the fibre direction, during the cool-down step of the manufacturing process a residual stress field is locked into the laminate and the unsymmetric stacking sequence produces moments that result in out-of-plane displacements. The main characteristic of unsymmetric laminates is that the internal stresses have more than one position of equilibrium making them bistable or multistable. This creates structures that can be at the same time flexible and stiff and opens up the possibility of combining several bistable components to obtain structures with multiple configurations.

The second application considered realises a high-lift device that can also be used during the cruise part of the flight envelope [8]. The idea is to mount a bistable panel, appropriately tailored, onto the tip of a traditional wing. When the panel is in its flat configuration, the wing span is extended thus increasing the lift, while when the panel is in the deployed configuration it acts as a blended winglet optimising...
the aerodynamic performance for cruise flight. Fig. 12 shows the two configurations that the winglet possesses. The extended configuration has a chord-wise curvature that helps to generate lift (i.e. during take off). When the winglet is deployed the cross section is almost flat and the curvature due to the residual stresses allows the composite plate to behave as a blended winglet. The orientation of the laminate increases the wash-out (i.e. a reduction of the local angle of attack) towards the tip, a feature useful to delay flow separation at the wing-tip. Fig. 13 highlights the effect of rotating the laminate axis and shows how the local angle of attack (i.e. wing sweep) may be reduced quite considerably towards the winglet tip. This will also increase the velocity at which the winglet will snap into the deployed configuration. To validate the concept a carbon fibre winglet has been manufactured and mounted on the tip of a rigid wing. The assembly has been tested in the wind tunnel at different angles of attack and airspeeds and for each configuration the aerodynamic loads and the velocity when the panel snaps have been measured. The snap-velocity is different for different angles of attack: at 0° angle of attack the snap velocity was 21.5 m/s, at 2.5° was 19 m/s and at 12.5° was 15.5 m/s.

After the winglet snaps all of the aerodynamic forces (lift, drag and pitching moment) reduced, confirming the possibility to use such a device to increase the lift in certain flight conditions such as take-off and landing. An actuation system to allow the winglet to go back to the extended configuration is under investigation. This could eventually be used to optimise the wing shape during flight operations.

4.3. Variable Camber Trailing Edge

This application concept is to realise a trailing-edge mounted device to change the camber of an airfoil by morphing its geometry. The bistability of unsymmetric patches of composite material is used to drive the shape change. Fig. 14 shows the geometry and the stacking sequence areas for the device: the dark areas have an unsymmetric stacking sequence while the light areas are symmetric. It is important to note that the laminate is continuous from the upper skin to the lower skin when going around the trailing edge. Fig. 15 shows two of the achievable shape configurations. Two more configurations are obtainable but for aerodynamic applications only the first two shapes are of interest.

A carbon fibre specimen has been built using an aluminium
Figure 12: The bistable winglet in extended and deployed configurations.

Figure 13: The effect of laminate axis rotation for the bistable winglet: straight and rotated axes.

Figure 15: Useful configurations for the variable camber trailing edge: deployed and extended.

The displacements measured on the test structure show a good agreement with those computed using FE analysis: the experimental model achieves an angular deflection of 21° while the numerical estimate is 20°. During actuation, the upper skin is fully clamped and two forces simulate the actuation load by pushing the left edge of the lower skin while the displacement is measured at the trailing edge. During the actuation, the upper skin bifurcates first with a load of approximately 20N, and the lower skin follows when the magnitude of the load reaches 30N. Once the structure is in either configuration and the edges are fixed in translation, the structure has considerable stiffness. When the left edges are constrained, the trailing edge device becomes a triangular structure even when it is in the deployed configuration. Furthermore the continuity of the fibres around the trailing edge ensure that the stresses can be transmitted from the lower skin to the upper skin. A test rig to measure the maximum load the structure can withstand is under construction. The transition between the deployed and extended configurations is highly non-linear and furthermore the force required to extend the trailing edge (30N) is approximately a three times that required for the deployment (10N). This force is a measure of the load bearing capability in the two configurations and highlights the beneficial effect of having the higher stiffness in the deployed configuration. In this case the airfoil will have an increased camber and will therefore generate higher aerodynamic forces and hence higher structural loads.
5. ACTIVE WINGLETS

An alternative for meso-scale morphing for control, is a flying-wing concept using independently-controllable, articulated, split wing-tips to achieve basic manoeuvres. Using such bendable, split wing-tips in this regard disrupts significantly the spanwise and chord-wise loading, resulting in, conceivably, a more efficient method of lateral, directional and longitudinal control than through the articulation of discrete control surfaces into the mainstream. The present concept is in fact an extension of a previously investigated concept [4], shown in Fig. 17, where the tips of a flying wing were folded symmetrically or unsymmetrically to achieve longitudinal and/or lateral/directional control. Though it appeared to be effective, a single pair of folding wing-tips could not substitute for all the conventional control surfaces at the same time if one wanted to obtain a full control envelope: in particular, to perform level turns at arbitrary bank angles, the wing-tip deflection needed to be combined with some elevator to trim the aircraft. The present concept is thus an attempt to control the wing solely with folding wing-tips, removing the elevator from the control line by using a second pair of folding wing-tips instead (see Fig. 18). With four multi-axis effectors, the aircraft is then over-actuated, leading to some redundancy in the flight control system, which could be exploited to optimize secondary objectives (e.g. minimum drag, minimum bending moment) at fixed lift and moments.

The model was installed inside the closed test section (2.1 × 1.5m) of a closed circuit wind tunnel whose maximum operating freestream velocity is 60m/s. The model was investigated at a flow speed of 20m/s, giving a Reynolds number of 3.18×10^5 based on the mean aerodynamic chord of the planar configuration. Analysed configurations included symmetric deflections of the aft wingtips and unsymmetric deflections of the fore wingtips, for angles of attack ranging from −4° to +16°. Achievable dihedral angle magnitudes for the wingtips were −75° to +75°. Figure 19 shows results for the experimental rolling moment for one particular configuration, and compares the results to those predicted by the VLM. Bourdin et al. [3] gives further details and results.

The concept has been modelled using vortex lattice and RANS (see Fig. 20) and the results compared [3].

Figure 16: Experimental model of the variable camber trailing edge.

Figure 17: Experimental model with two active winglets.

Figure 18: Experimental model with split winglets mounted in the wind-tunnel.

Figure 19: Rolling moments at 8° angle of attack as the right front wingtip dihedral angle is changed.
6. CONCLUSIONS

This paper can only give a very superficial overview of the morphing concepts investigated; for further details the reader is encouraged to consult the references. What is clear is that robust approaches to the design of morphing aircraft are not mature and there is still significant research still to be done. The examples shown are still at a very early stage and more development is needed before they can be commercially realised. One issue that has not be consider here is the required for the flexible skin. Various concepts for a morphing skin, for example corrugated and honeycomb skins, have been considered [7], but space precludes their inclusion here.

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