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Synthesis and Analysis of a Multi-Objective Controller for Tilt-Rotor Structural Load Alleviation

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This paper describes the design and implementation of a Structural Load Alleviation (SLA) system for the Agusta tilt rotor/wing aircraft concept (ERICA), carried out under the European Framework V ACT-TILT project. A multi-objective controller, primarily designed to reduce the in-plane bending at the rotor blade root and the out-of plane flapping, is synthesised using an LQG (Linear Quadratic Gaussian) approach. Evaluations performed using the controller have shown that the use of cyclic control is very effective in suppressing both the build-up of in-plane loads and gimbal flapping; simultaneous suppression results in an inevitable compromise however. The issues associated with this compromise are discussed in the paper. Results from offline non-linear analysis are presented for various flight manoeuvres showing the effect of the SLA system on load suppression and manoeuvre performance.

Nomenclature

$\boldsymbol{d}$	=	Disturbance vector in LQG formulation
D	=	Aerodynamic drag force at a blade element
I_β	=	Flap moment of inertia of a rotor blade about the hub
$\boldsymbol{J}$	=	Cost function in LQG formulation
L	=	Aerodynamic lift force at a blade element
m	=	Blade mass per unit length
$\boldsymbol{m}$	=	Measurements in LQG formulation
$M_{zL,R}$	=	In-plane moment envelope for the left and right rotors
$\boldsymbol{n}$	=	Sensor noise model in LQG formulation
p, q, r	=	Fuselage angular velocity components in body axis
$\boldsymbol{Q}, \boldsymbol{R}$	=	Weighting matrices in LQG formulation
R	=	Rotor radius
u_t, u_p	=	Tangential and normal relative air velocities for a blade element
$\boldsymbol{W}_{act}$	=	Actuator transfer function
$\boldsymbol{x}$	=	State vector in LQG formulation
$\boldsymbol{y}$	=	Output vector for LQG minimisation
α	=	Angle of attack of a blade element
$\beta_{1sL,R}$	=	Lateral disc tilt of the left and right rotors
$\beta_{1cL,R}$	=	Longitudinal disc tilt of the left and right rotors
ζ	=	Rudder deflection
η	=	Elevator deflection
ξ	=	Aileron deflection
θ_{1s}	=	Longitudinal cyclic pitch
θ_{1sd}	=	Differential longitudinal cyclic pitch
θ_{1c}	=	Lateral cyclic pitch
θ_{1cd}	=	Differential lateral cyclic pitch
φ	=	Rotor inflow angle
$\dot{\Omega}$	=	Angular velocity rate of the rotor

I. Introduction

Compared to a conventional helicopter or a turboprop fixed-wing aircraft, the tilt-rotor operates over a broader flight envelope with higher manoeuvre capabilities and a much wider speed range. For existing tilt-rotor aircraft, it has been shown that the loads on some of the structural components during certain manoeuvres result in high fatigue usage and, in some cases, exceedence of the design limit. For example, in the V-22, to minimise the manoeuvre loads in the rotor/drive system and the fuselage, structural load limiting laws were incorporated into the active flight control system. In a similar vein, the smaller civil variant, the BA609, contains load alleviation functions in the active flight control system[1].

The European Commission's Framework V 'critical technology' RHILP project[2] addressed a number of technology issues related to a civil tilt-rotor aircraft and structural load alleviation was one of the four work packages included in the project[3], led by The University of Liverpool. SLA systems for a tiltrotor/wing aircraft have been analysed further under the 5th Framework sister-project ACT-TILT (Active Control Technology for Tilt-Rotor aircraft). This paper will describe the design and implementation of a SLA system for the ACT-TILT configuration, the Agusta tilt-rotor/wing aircraft concept (ERICA - Enhanced Rotorcraft Innovative Concept Achievement[4], Fig 1). A multi-objective controller, primarily designed to reduce the in-plane bending at the rotor blade root and the out-of plane flapping, is synthesised using an LQG approach. Evaluations performed using the controller have shown that the use of cyclic control is very effective in suppressing either the build-up of in-plane loads and gimbal flapping, although simultaneous suppression results in an inevitable compromise that are discussed in the paper. Results from offline non-linear analysis are presented showing the effect of the SLA system on load suppression and manoeuvre performance. This paper is written as ACT-TILT draws to a close and the European Tilt Rotor team anticipates future activities in the form of hardware developments.

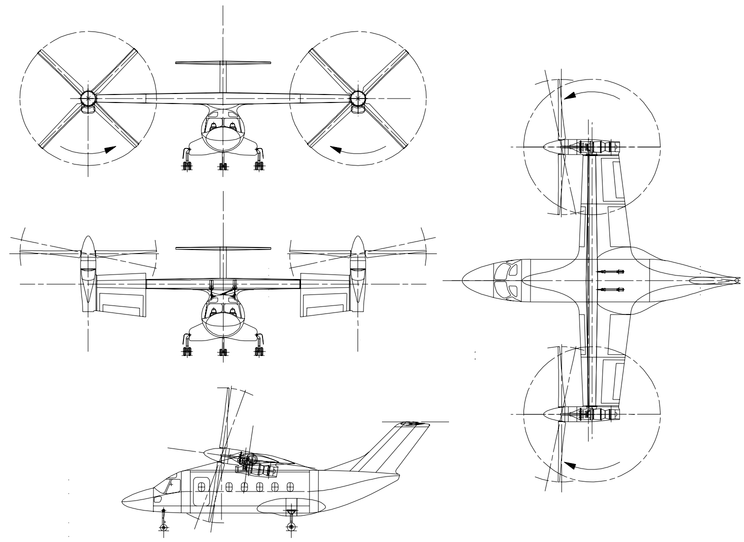

Figure 1: Basic layout of ERICA Configuration

II. The ERICA Configuration and the FLIGHTLAB Model

ERICA features both tilt rotors and tilt wings, the latter obtained by the independent tilt of the outboard portion of the wing. This configuration (Fig. 1) significantly reduces the download due to the downwash of the rotor on the wing in helicopter mode, providing the opportunity to reduce the rotor size to improve the cruise performance. In addition, independent tilt freedom allows the wing to avoid stall and to supply a suitable amount of lift during the conversion phase.

The four-bladed rotor system of ERICA is gimbal-mounted, homo-kinetic and stiff in plane. The prop-rotors are installed in wing-tip nacelles. The nacelles are, in turn, supported by the wing spar, a composite tube designed to give stiffness characteristics to achieve structural stability and to carry the flight loads generated by the prop-rotors in helicopter flight mode and the wing and propeller loads during airplane flight mode.

The drive system has two main gearboxes; each supplying the power from the engines to its respective rotor. They are connected by a shaft to guarantee the speed synchronisation of the rotors, and to prevent the complete loss of power to either rotor due to one engine failure. The engines tilt together with the rotors and the drive system and permit the reduction of rotor rotational speed for airplane mode flight to improve performance.

A. FLIGHTLAB mathematical model

FLIGHTLAB[5] is an advanced modelling and simulation environment for rotorcraft analysis with a modular structure, enabling rotorcraft (and, indeed, fixed wing aircraft) models of varying levels of complexity to be created. The main characteristics of the FLIGHTLAB ERICA model (F-ERICA) are as follows: the aircraft's four-bladed, counter-rotating, gimbal mounted prop-rotors are modelled as if rigid. The homokinetic gimbals are modelled with torsional spring-damper components in pitch and roll. No individual blade flapping is modelled in the F-ERICA. The five different airfoil sections used in the ERICA blade design are represented in look-up tables as functions of angle of attack and Mach number. The air-load calculation is performed on user defined blade segments using quasi-unsteady aerodynamics and a three state dynamic rotor inflow model.

ERICA is equipped with two outboard tiltable wing parts. The fixed wings are attached high to the fuselage with a setting of 3 degrees. The sectional aerofoil coefficients are presented for Mach numbers from 0.1 to 0.6. With these data a lifting line model for the whole wing was created in FLIGHTLAB. Interference effects between wings are calculated with a horseshoe vortex type model.

The fuselage aerodynamics are modelled by the use of aerodynamic coefficients for drag, side force, lift, pitch, roll and yaw moments with respect to angles of attack and sideslip. The horizontal stabilizer and fin aerodynamics are modelled by using look-up tables. A simplified drag model is used for the nacelles. The nacelle inertia and the centre-of-gravity (CG) shift during conversion process are modelled by placing point masses at the nacelles reference points. The aerodynamics interference effects are modelled by using FLIGHTLAB aerodynamic interference components.

A degree of validation of the F-ERICA model was achieved by comparison with the Eurocopter HOST[6] implementation and the Westland/Glasgow Caledonian FMC model [7].

III. SLA for Tiltrotors

The tiltrotor offers the vertical capabilities of a helicopter with the high speed performance of a turbo-prop aircraft. The prop-rotors of a tiltrotor are required to operate both as a lifting rotor and as a propeller and hence the design is usually a compromise between hover efficiency and forward speed performance. The prop-rotors are gimballed in the hub and manoeuvres in airplane or conversion mode can lead to large transient flapping and oscillatory in-plane loads and these aspects are now discussed.

A. Yoke chord bending and flapping

Yoke chord bending is the term used to describe the moment generated at the root of a rotor blade associated with bending of the blades in the plane of the rotor disk. Large oscillatory in-plane rotor loads develop during manoeuvres in airplane and conversion modes, due primarily to the significant lift forces acting in the plane of the rotor disc, on the highly twisted blades.

The in-plane load at the blade root can be expressed in terms of the blade lift L and drag D as follows:

$$M_Z = \dot{\Omega} I_\beta + \int_0^R L \sin\phi r + D \cos\phi dr \tag{1}$$

Where, $I_\beta = \int_0^R m(r) r^2 dr$ is the flap moment of inertial about the rotor hub.

The first term gives the moment generated by the inertial properties of the rotor blade; this contribution is small and transient when compared with the aerodynamic components in the equation.

In airplane mode, due to the high forward velocity, the angle of inflow ϕ is large (Fig 2), leading to an in-plane force component, $L\sin\phi$. Even in steady flight, the large in-plane force component leads to relatively high levels of 1/rev aerodynamic in-plane moment at the blade root in the rotating blade system, whenever the resultant airflow has a component in the plane of the rotor disc. During a pitch (or yaw manoeuvre), the in-plane oscillatory loads are substantially higher than the steady state values. During a pull-up (turn) manoeuvre, for example, changes in the body axis vertical (sideways) velocity alter the incidence (sideslip) of the rotor inflow causing the rotor disc to flap in the same direction as the fuselage pitch (yaw). It has been shown by Miller and Ham[8] that the out of plane moment required to produce a change in flap can be directly related to the in-plane moments. The applied aerodynamic moment is then greater than that required to precess the gimballed rotor,

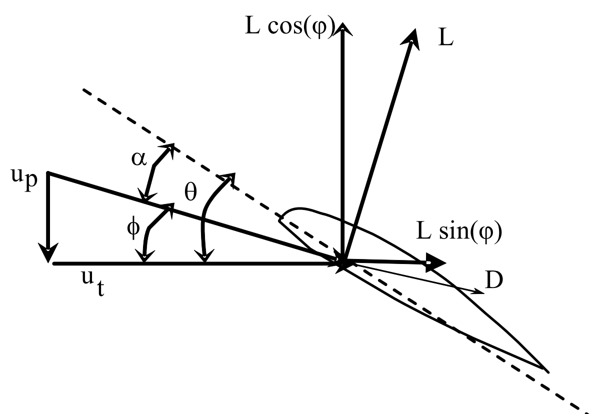

Figure 2. Schematic of a blade element

leading to an increased flapping in the direction of motion. The excessive flapping can cause elastomeric flap bearing loads and rotor mast bending loads. An analysis of the structural loads of a tiltrotor aircraft manoeuvring in airplane mode and a description of a multi-objective LQG controller designed to minimize these loads can be found in Ref 9.

The fundamental cause for the amplification of the in-plane load during a pull-up manoeuvre is that the rotor longitudinal flap rate and the aircraft pitch rate occur in the same direction. Principally the SLA controller needs to reduce the inplane load by forcing the rotor to flap against the pitch rate through the application of longitudinal cyclic. For this kind of SLA system, it was noted that the magnitude of the longitudinal flapping is directly proportional to the applied cyclic angle. In order to address this problem, the controller design described in this work is formulated as a multi-objective minimisation problem with both the in-plane loads and rotor flapping included in the objective function.

For the design of SLA controllers it is necessary to develop linearised models relating the controlled variable (e.g. in-plane bending) to available measurements and controls. An approach, making use of a multi-blade coordinate (Coleman) transformation to convert the in-plane loads from the individual blade coordinates to a hub-fixed coordinate frame was used. This method is described fully in Reference 3.

B. Brief survey of SLA controllers

The problem and the need to minimise these structural loads has been extensively studied and reported for the V-22 tiltrotor aircraft [10, 11, 12, 13]. To minimise the manoeuvre loads in the rotor/drive system and the fuselage, structural load limiting laws were incorporated into the active flight control system. The control laws were developed to limit the loads while maintaining Level 1 Handling Qualities and not unduly penalising the aircraft manoeuvre capability. The oscillatory yoke chord bending load limiter on this aircraft features modifications to the AFCS command model/stability compensation and rate limitation of the longitudinal stick command. The SLA controller described in Refs 8 and 12 used a modified eigen-structure assignment technique. Two different approaches to the development of control laws for rotor yoke chord load alleviation were investigated in Ref 12. The first controller used flapping feedback to regulate longitudinal and lateral cyclic pitch angles, but as this compensator did not meet the stability robustness requirements, a second control system was developed using an eigen-structure assignment methodology. The resulting pitch rate feedback control law, utilising longitudinal cyclic pitch and elevator, provided a favourable match between the desired and achieved short period eigen-structure and was robust to structural mode parameter uncertainties. The controller described in Ref 8 utilised feedback of pitch rate, pitch angle and normal velocity to generate control inputs on rotor cyclic pitch angles. As an observation, the study showed that the use of rotor flapping states in the feedback is not necessary for suppression of the in-plane loads.

Preliminary results from the SLA work package of the RHILP project are presented in Ref 3. The primary focus of attention in this project was the suppression of in-plane rotor yoke loads for longitudinal manoeuvres in airplane mode. Two separate controllers were developed. For the first controller, multi-variable technique using μ-synthesis was used and load suppression of 80-90% was demonstrated using rotor longitudinal cyclic control, albeit at a 20-30% performance penalty. This control law was designed to attenuate in-plane bending moments using longitudinal cyclic. However, use of the cyclic resulted in what were felt to be excessive excursions in gimbal longitudinal flapping. The second controller, using H-infinity techniques, showed how the dual-objectives of suppressing transient load and flapping during manoeuvres were feasible using both rotor cyclic and elevator controls.

C. Design Goals

The general goal of the controller described in the present paper is to obtain the best possible performance whilst using the minimum possible control input; of course these are conceptually conflicting requirements and as such a trade-off must be made between controller performance and control activity. There was an allowance of ±2° cyclic available for use for the SLA, which was not to be exceeded even for the limiting manoeuvre envisaged for a civil tilt-rotor - a 2.5g pull-up manoeuvre during cruise at altitude.

D. Design schematic

The first part for the design of the controller is to provide a linear model for analysis from FLIGHTLAB; this was a 17 state, 9 input model, with 5 outputs trimmed around a straight and level condition at 200 knots equivalent airspeed and 3000m pressure altitude. The linear model contains the 9 rigid body states and the longitudinal and lateral flap states and their rates for the left and right rotors. A schematic of the LQG process is presented in Fig 3.

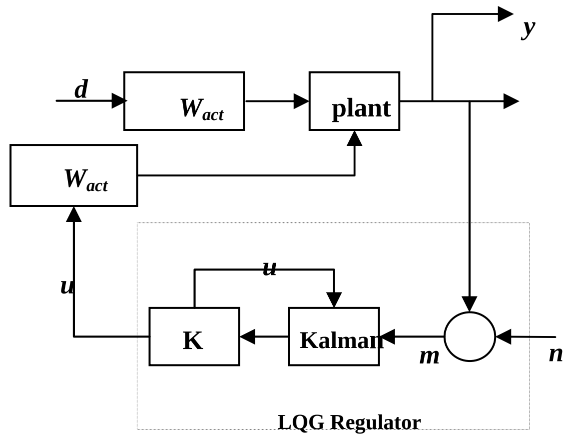

Figure 3: LQG design for the SLA controller

Although all the rotor controls are available, only the rotor cyclic controls were chosen as the output of the controller, u, which is given by:

$$\mathbf{u} = [\theta_{1c}, \theta_{1cd}, \theta_{1s}, \theta_{1sd}]$$

The measurements, **m**, for feedback, were selected as follows:

$$\mathbf{m} = [p, q, r]$$

The output vector used for minimisation is made up of the in-plane bending moment envelopes of the left and right rotors and the longitudinal and lateral flap of both rotors:

$$\mathbf{y} = [M_{zL}, M_{zR}, \beta_{1sL}, \beta_{1sR}, \beta_{1cL}, \beta_{1cR}]$$

Ref. 3 describes the in-plane load envelope. The disturbance vector is made up of the tilt-rotor's aerodynamic controls that would be used in aeroplane mode, namely the ailerons, elevators and rudder:

$$\mathbf{d} = [\xi, \eta, \zeta]$$

A covariance of 10^{-3} models sensor noise and an actuator band-width of 60 rad/s were used (see Fig 2):

$$\mathbf{W}_{\mathbf{act}} = \frac{60}{s+60}$$

E. The LQG design process

In LQG control, the regulation performance is measured by a quadratic performance criterion of the form:

$$\mathbf{J}(\mathbf{u}) = \int_0^{\infty} \{\mathbf{x}^T \mathbf{Q}\mathbf{x} + \mathbf{u}^T \mathbf{R}\mathbf{u}\} dt \tag{2}$$

Where ***Q*** and R are the weighting matrices chosen by the designer so that a reasonable trade-off between performances and control authority is obtained. In addition, the elements of ***Q*** can be chosen such that a trade off between the performance of the different output variables can be obtained. The first design step seeks a state-feedback law $\mathbf{u} = -\mathbf{K}\mathbf{x}$ that minimises the cost function ***J(u)***. The minimising gain matrix, ***K***, is obtained by solving an algebraic Riccati equation. This gain is called the LQ-optimal gain. As in the case of pole placement, the LQ-optimal state-feedback is not implementable without full state measurement. It is possible, however, to derive a state estimate, $\hat{\mathbf{x}}$, such that $\mathbf{u} = -\mathbf{K}\hat{\mathbf{x}}$ remains optimal for the output feedback problem.. This state estimate is obtained by the use of a Kalman filter that estimates the state vector given the measurements ***m***. Finally, the state feedback gain matrix ***K*** and the Kalman state estimator are connected to form the LQG regulator. In the current study, this step was performed by using the MATLAB control system tool box.

F. Controller reduction and descretisation

The LQG controller design process produced a 24 state, 3 input 4 output controller. This was a little large to be implemented in a real-time simulation environment and a reduction was necessary to a more manageable size. The reduction was performed by calculating the balanced realisation and optimal Hankel norm approximation of the controller. The reduction process lead to a 6-state controller which exhibited a slight performance reduction compared to the original 24-state controller. The results presented in this paper are obtained by using this 6-state controller.

The controller was required to be implemented in discrete form within Eurocopter's real-time code generation tool. This meant that the continuous state space controller generated by University of Liverpool needed to be converted into a discrete zero-pole format. This operation was relatively simple to carry out within MATLAB as all the tools for such a transformation are readily available. It can be broken down into two distinct steps, first the transformation of the continuous state space form to discrete state space, and second the transformation from discrete state space to zero-pole format. Both these controllers can then be checked against the original to make sure that numerical errors have not degraded the controller's performance or stability.

IV. Nonlinear Responses

The SLA controller developed was implemented on the F-ERICA nonlinear model and a series of inputs, similar to those used during the evaluation of the linear model, were made. Analyses were carried out at the design point of 200kts indicated airspeed (IAS) at 3000m, and also three off-condition points: 225kts (IAS), 6000m, 165kts at sea-level and 120 kts sea-level at 60° nacelle.

The control input was a 10% pulse input for 0.5 seconds on the pilot controls, elevator, aileron and pedal; this input equates to a 2.5g pull-up at the design point which was the identified evaluation manoeuvre. At the off design points the amplitude of the control inputs are adjusted so that approximately same load factor was maintained.

A. Pull-up manoeuvre

Results for the pull-up manoeuvre (10% control input) are presented in Figs 4-9, showing a very effective reduction in the flapping, in-plane bending and H-forces for only small performance loss.

The Figures show the in-plane loads, the flapping behaviour and the hub moments with and without the SLA controller. Approximately 50% suppression of in-plane loads and longitudinal and lateral flap suppressions of more than 1deg is evident from the results. It can also be seen that the hub moments, which are approximately proportional to the flapping, are also reduced. Note that the hub y-moment with the SLA system turned on is negative (stabilising) for a nose-up attitude compared to the positive (destabilising) value when the SLA system is turned off. The SLA controller not only reduces the loads but it also improves the aircraft stability.

The flight-path angle and pitch rate responses with and without the SLA, shown in Fig 5, give some qualitative indication of the reduction in performance for the same pilot command. The controller actions are shown in Fig 6. A maximum of 1.7 deg combined longitudinal cyclic angle is commanded by the controller. In addition, approximately 1 deg of differential lateral cyclic is applied by the controller. Note that the differential lateral cyclic applies equal amounts of control on the left rotor (clockwise) and the right rotor (counter-clockwise), causing them to tilt in opposite directions.

The response of the controller to a 25% pulse input (corresponding to a similar load factor as the design point manoeuvre) in longitudinal stick at the off-design point of 160kt at sea level with the SAS disengaged, is shown in Fig 7. A similar magnitude of load suppression is evident at this flight condition demonstrating the robustness of the controller. The aircraft responses are shown in Fig 8. The flight path performance is reduced by approximately 10% relative to the SLA-off case. The controller demands are given in Fig 9. The controller commands approximately 2.5 deg of longitudinal cyclic angle and 1.5deg of differential lateral cyclic.

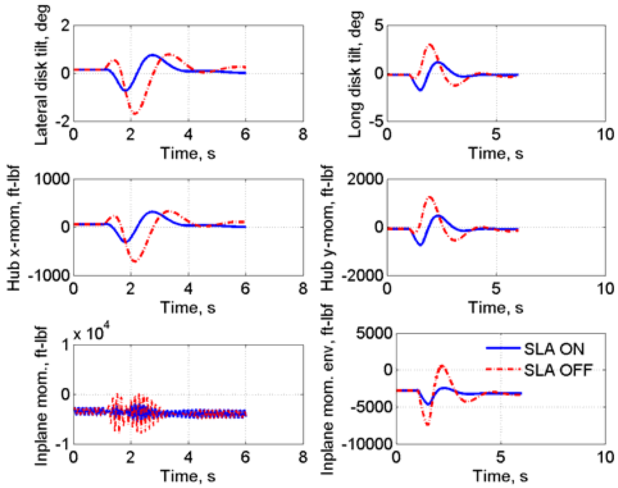

Figure 5: The in-plane moment and flap response for an elevator pulse input at the design point

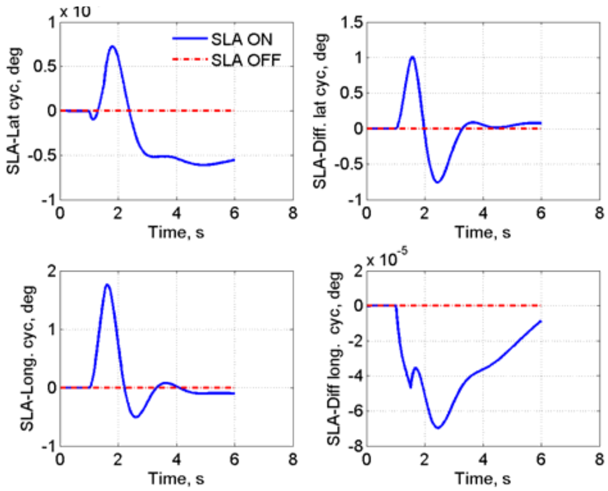

Figure 6: SLA control demands for an elevator pulse input at the design point

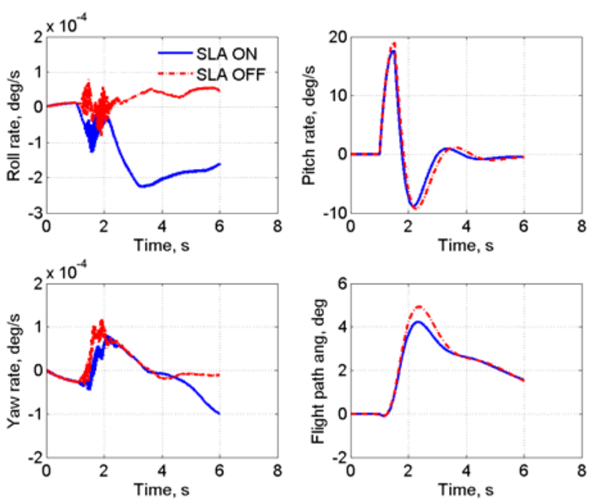

Figure 8: Aircraft response for an elevator pulse input at the off-design point

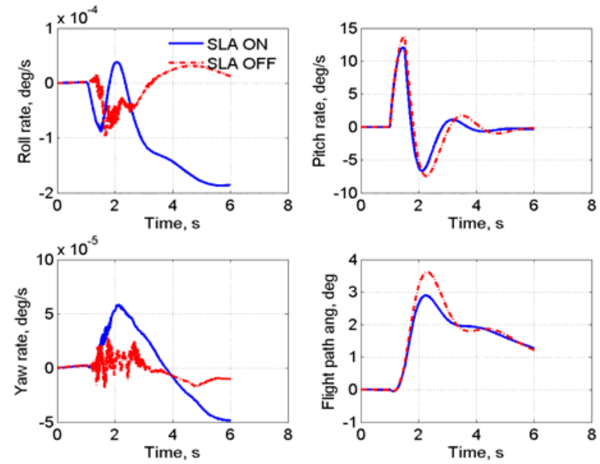

Figure 4: Aircraft response for an elevator pulse input at the design point

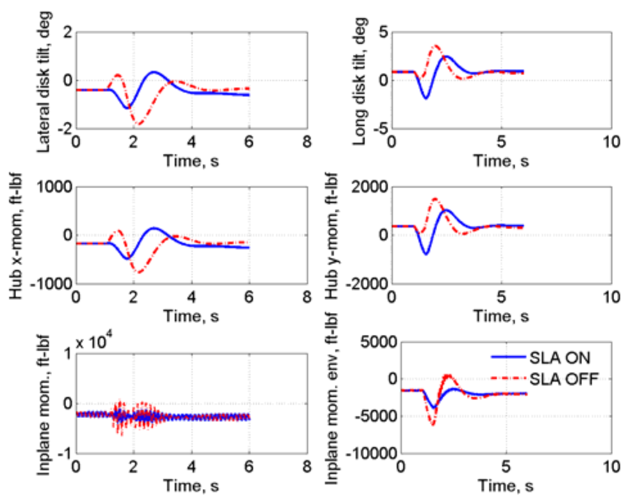

Figure 7: The in-plane moment and flap response for an elevator pulse input at the off-design point

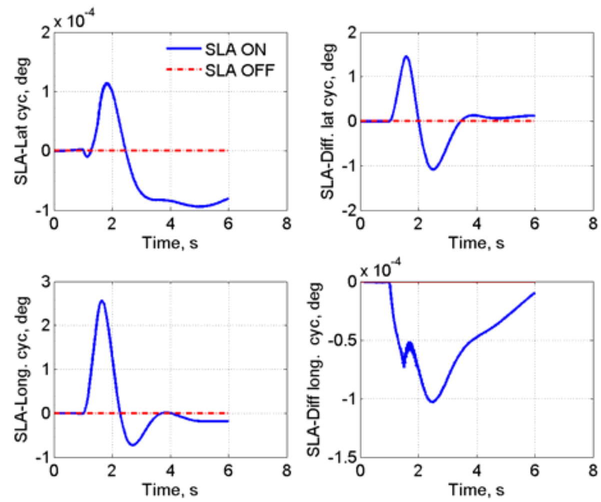

Figure 9: SLA control demands for an elevator pulse input at the off-design point

B. Yaw manoeuvre

Fig 10 shows the responses of the system for a pulse input in rudder with and without the SLA controller at the design point. In-plane load reduction of approximately 50% is achieved with the SLA system. Inspection of the plot showing the gimbal flap angles shows considerable reductions in lateral flap and longitudinal flap angles. It can be seen from Figure 11 that the controller had very limited impact on the primary vehicle responses. Fig 12 shows the controller demands. The controller attempts to cancel out the change in angle of sideslip due to lateral velocity build up by the application of the lateral cyclic angle. The controller also inputs a small amount of differential longitudinal cyclic.

The response of the controller to a 25% pulse input in pedal at the off-design point of 160kt at sea level with the SAS disengaged, is shown in Fig 13. The aircraft responses and the control demands are shown respectively in Figs.

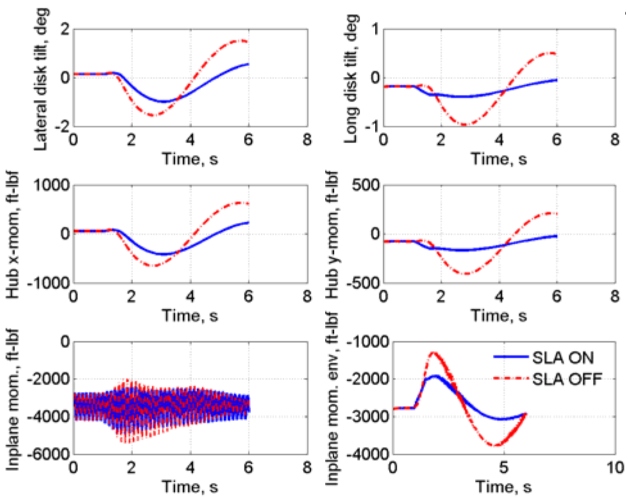

Figure 10: The in-plane moment and flap response for a rudder pulse input at the design point

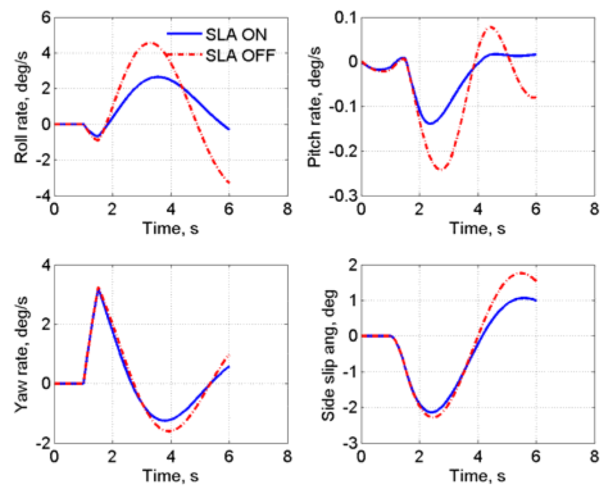

Figure 11: Aircraft response for a rudder pulse pulse input at the design point

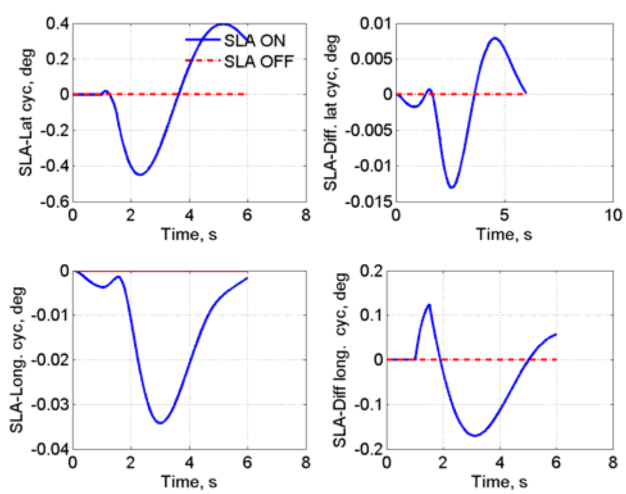

Figure 12: SLA control demands for a rudder pulse input at the design point

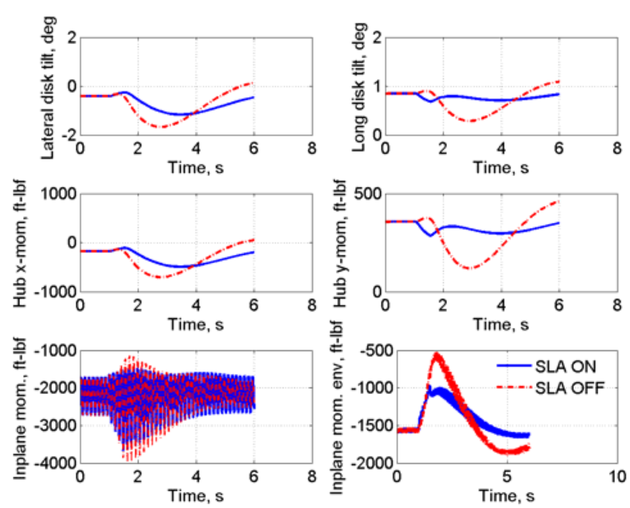

Figure 13: The in-plane moment and flap response for a rudder pulse input at the off-design point

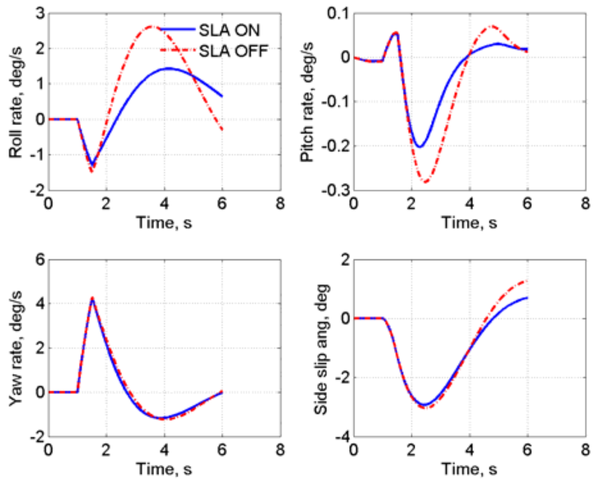

Figure 14: Aircraft response for a rudder pulse input at the off-design point

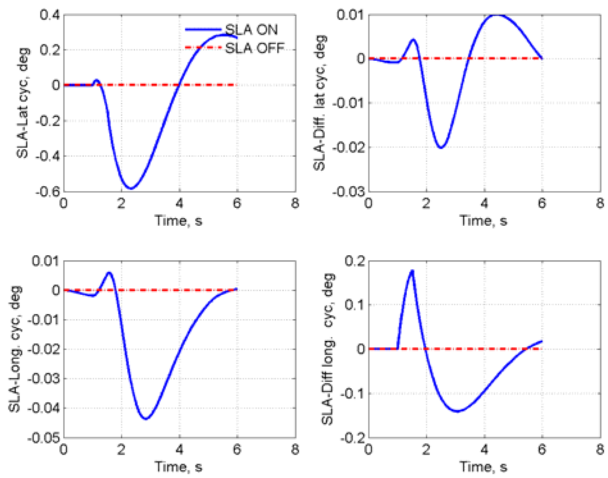

Figure 15: SLA control demands for a rudder pulse input at the off-design point

C. Roll Manoeuvre

The responses of the open and closed loop systems for a pulse input in aileron are illustrated in Figure 16; the fuselage states are shown in Figure 17. The open loop responses suggest that the build up of loads and flap angles is fairly benign for this type of input. Lateral flapping during the roll manoeuvre is driven by a combination of roll rate, and lateral velocity (sideslip). During the initial part of the manoeuvre the roll rate dominates the longitudinal flap response, and once the pulse input is complete the flapping becomes dominated by side slip. This is clearly shown by comparing the plots of side slip and lateral flapping for the roll response. The SLA controller effectively reduces the lateral flapping by an average of 40%.

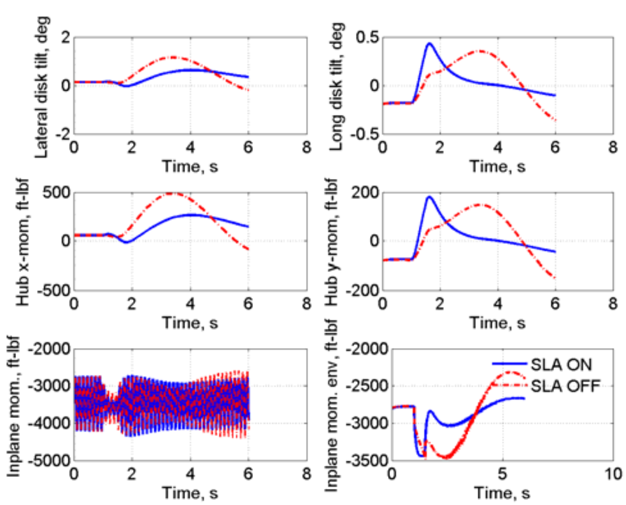

Figure 16: The in-plane moment and flap response for an aileron pulse input at the design point

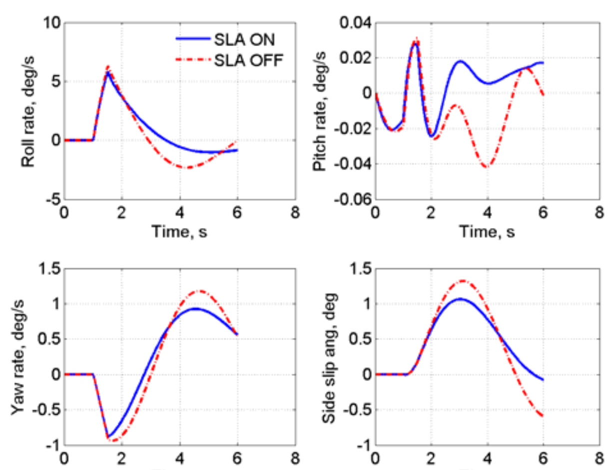

Figure 17: Aircraft response for an aileron pulse input at the design point

V. Concluding Remarks

This paper has presented results from an investigation into the potential of structural load alleviation through active control in tilt rotor aircraft. The work is part of the European Framework V Project, ACT-TILT and the aircraft configuration is the Agusta Tilt Rotor/Wing ERICA. The gimbaled prop-rotors experience large transient aerodynamic loads during manoeuvres, particularly in pitch and yaw, that in turn lead to the build-up of large in-plane loads and gimbal flapping. The load alleviation study has included several different design schemes and in this paper, results have been shown for the application of the LQG design process. To mimimise the complexity of the system, only fuselage rates, p, q and r were used in the measurement system and the controller order was reduced from the design solution of 24 to 6 with little apparent impact on performance and stability. The design case considered was manoeuvring from a 200kts (IAS) condition at 3000m altitude. Simultaneous elimination of transient loads and flapping is physically impossible using only rotor cyclic controls due to the requirement to tilt the disc into the incident airstream to reduce the periodic component of the inplane lift. Approximately 50% reduction in each is achievable however, with the use of only about 2 degrees of cyclic pitch and little loss in performance. Results presented for the off-design conditions at 160 and 225 kts (IAS) provide confidence in the robustness of the design.

Structural load alleviation through active control can be considered as one of the critical technologies in the development of tilt rotor technology. The work conducted within the RHILP and ACT-TILT projects has demonstrated that use of rotor controls in airplane (and conversion) mode offers an effective means of suppressing loads. The continuing work towards realising a European Civil Tilt Rotor aircraft will include refinements to the aerodynamic and structural modelling that will allow further optimization of the control algorithms. Physical implementation aspects will also be considered, along with the important issue of the level of criticality of such functions within an electronic flight control system.

Acknowledgments

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