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SURVIVABILITY OF HELICOPTER WITH INDIVIDUAL BLADE PRIMARY CONTROL FAILURE

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Abstract: The effect of actuator damage on a helicopter rotor with an IBC based primary control system is studied. Such a system eliminates the swashplate and can be accomplished by trailing edge flaps, active twist or full authority IBC. Damage to the collective, longitudinal and lateral cyclic are simulated for one blade, both individually and in combinations ranging from partial damage to complete failure. Numerical results are obtained using a dissimilar blade aeroelastic analysis for hover and at $\mu = 0.1$, 0.2 and 0.3. It is found that the helicopter can be trimmed for all cases with all three controls having failed on the blade with actuator damage thereby showing that the IBC actuated rotor can survive an actuator failure and can be reconfigured by the pilot using the controls on the other blades. However, in case the collective fails and the longitudinal cyclic is present, there are problems in achieving trim at high damage levels at $\mu = 0.3$. Physical explanations of this phenomenon are given. It is also found that the blade tip response of the damaged rotor increases considerably relative to the baseline. In addition, large 1/rev and 2/rev loads result for the damaged rotor. However, power required for flight remains almost unchanged. The high levels of blade response, especially in flap, and large 1/rev and 2/rev hub loads need to be accounted for during design for the helicopter to survive the loss of primary control actuation authority on one blade.

1 INTRODUCTION

Helicopter primary control is typically accomplished using a complex mechanical system consisting of the swashplate, pitch link and push rods and fixed system hydraulic flight control actuators. The swashplate based primary control leads to high weight, aerodynamic drag and reduced mission performance [1]. Eliminating the swashplate reduces the complexity of the rotorcraft which in turn increases reliability and reduces acquisition and maintenance costs. Some researchers have started investigating the swashplateless rotor concept in recent years. The research is motivated by the availability of smart material actuators developed for vibration and noise control. These actuators are typically made of piezoceramic materials and have high bandwidth but low actuation authority. However, with the advancement in materials technology, the actuator authority continues to increase and a swashplateless helicopter rotor is becoming feasible. Three approaches are possible for a swashplateless rotor concept. The two approaches which have been reported in the literature are the trailing edge flap concept [2-3] and the active twist rotor concept [4]. Among these two approaches, the trailing edge flap approach appears to be more feasible using current smart material technology. A third possibility is the use of IBC for primary control. Typically, IBC uses hydraulic actuators which can be heavy and cumbersome. However, recent studies have concluded that combining hydraulic and smart material actuation can result in actuators with high authority [5]. Furthermore, the possibility of using magnetostrictive materials [6] or single crystal piezoceramics [7] also exists and can lead to the development of high authority actuators for use in IBC.

It has recently become clear that whatever mechanism is used for primary control, the control angles needed for achieving trim are quite similar [8]. Therefore, a generic IBC actuator is a good model for the trailing edge flap and active twist concept for research on the effect of actuator failure which is the focus of this paper.

Reliability and fault tolerance are key aspects of a flight control system. A swashplate based primary control system is susceptible to catastrophic failure in the event of swashplate damage. Such damage occurs primarily due to bearing failure between the upper and lower components of the swashplate and is caused by lack of lubrication which can occur due to improper maintenance or a harsh operating environment. Typically, swashplate damage is very rare because the swashplate is investigated routinely during maintenance because of its critical nature. However, the routine maintenance also causes high maintenance costs.

It is possible that an IBC based primary control system will have advantages in terms of survivability and reliability in the event of actuator failure on one blade since each blade has its own actuator. Assuming an IBC based primary controller, it is important to study the effect of damage or failure of the actuator on one blade on the ability of the pilot to reconfigure and trim the rotor. The possibility of trimming a damaged rotor and flying it in level flight is very important from a survivability perspective. The second important issue is the effect of the damaged actuator on the blade response and vibration. These dynamic effects are important for the structural design of the rotor blade and helicopter and must be addressed to take advantage of the reconfigurable rotor concept. Furthermore, by monitoring the changes in blade response and hub loads, it may be possible to detect actuator damage using health and usage monitoring (HUMS) systems which are frequently installed in many helicopters [9].

2 AEROELASTIC ANALYSIS

An elastic rotor blade undergoing flap, lag, torsion and axial motions is considered. The blade equations are derived using Hamilton’s principle and include moderate deflection effects as given by Hodges and Dowell [10]. A linear inflow aerodynamic model is used along with blade element theory in forward flight along with a dynamic stall model. A uniform inflow model is used in hover. The reversed flow effect is included. The elastic rotor blade equations are nonlinear partial differential equations. The blade equations are solved using finite element in space and time by discretizing in the spatial domain (along the blade span) and time domain (along the rotor azimuth). The normal mode approximation is used to reduce the blade degrees of freedom by retaining the first few flap, lag and torsion modes which accurately capture the blade dynamics. Once the blade response is known, blade loads are calculated using the force summation method which involves summing the section aerodynamic and inertial loads and then integrating over the blade span. The hub loads are then calculated by summing the blade loads over all the blades. The steady loads acting on the helicopter are calculated by expanding the hub loads in a Fourier series and then used for the helicopter trim equations.

The trim and blade response equations are solved simultaneously using an iterative coupled trim procedure so that the effect of aeroelastic interactions due to blade deflections on the rotor loads is properly captured. The Newton-Raphson method is used to solve these nonlinear equations. Propulsive trim is considered with the three forces and three moments acting on the helicopter being driven to zero. Six angles are determined following the trim procedure. These are the four rotor control angles: collective pitch, longitudinal cyclic, lateral cyclic, tail rotor collective and two attitude angles: shaft tilt and bank angle. Details of the baseline formulation are given in [11] , [12] and [13].

3 SIMULATION OF ACTUATOR DAMAGE

The simulation of damage to the primary control system is accomplished by using a dissimilar blade analysis. The aeroelastic code is modified such that each blade can have different control angles. Thus, each blade can have different values of collective, longitudinal and lateral cyclic which is fundamental to the IBC based primary control concept. In general, the control angles given to all the blades will be same for an undamaged rotor. In implementation, such control could be given by trailing edge flap actuators, high authority IBC or active twist. Now if the actuator for one blade is damaged, there will be a loss of authority in either the collective, longitudinal cyclic, lateral cyclic or a combination of these three controls on the damaged rotor. For the numerical simulation, one blade is assumed to be damaged and the other blades are undamaged. The blade with actuator damage is assumed to have primary controls given by

$$
\begin{aligned}
\theta_{0d} &= \theta_0(1-D) \\
\theta_{1cd} &= \theta_{1c}(1-D) \\
\theta_{1sd} &= \theta_{1d}(1-D)
\end{aligned}
$$

Here θ_0, θ_{1c} and θ_{1s} are the controls given by the pilot. The blades without actuator damage experience the controls given by the pilot and the blade with actuator damage experiences the controls θ_{0d}, θ_{1cd} and θ_{1sd}. However, it is also possible that not all three controls are damaged, and these cases also need to be considered. When $D = 0$, there is no actuator damage and when $D = 1$, the actuator has failed completely. The use of the damage variable D allows us to study the effects of progressive actuator damage using one scalar parameter. There are several possibilities for damage which are studied in this paper. These involve damage to one control, damage to any two controls and finally damage to all three controls, as shown in Table 1. In the partial damage cases, the other controls are kept at the values experienced by the blades without actuator damage.

Table 1 Damage types and control inputs to blade with actuator damage

Damage	Collective	Longitudinal cyclic	Lateral cyclic
Collective only	θ_{0d}	θ_{1s}	θ_{1c}
Longitudinal cyclic only	θ_{0}	θ_{1sd}	θ_{1c}
Lateral cyclic only	θ_{0}	θ_{1s}	θ_{1cd}
Collective and Longitudinal cyclic	θ_{0d}	θ_{1sd}	θ_{1c}
Collective and lateral cyclic	θ_{0d}	θ_{1s}	θ_{1cd}
Longitudinal and lateral cyclic	θ_{0}	θ_{1sd}	θ_{1cd}
All three controls	θ_{0d}	θ_{1sd}	θ_{1cd}

The dissimilar blade analysis solves for the response of each blade using the different primary controls for each blade. The response of the blade with actuator damage therefore becomes different from the blades without actuator damage. The blade loads for the blades without actuator damage are also different from the blade with actuator damage. The rotor is trimmed considering the dissimilar blades. The blade with actuator damage causes an unbalance which results in the hub loads containing vibration other than the N/rev loads found for an N bladed rotor with identical blades.

There are two major issues caused by the damaged actuator which are addressed in this paper. Firstly, it is important to know if the rotor can be trimmed with actuator damage in one blade. This is important from the survivability and reliability perspective. Note that for the swashplate based control systems, the possibility of compensating for the swashplate failure is not there but is possible for an IBC based primary control. Secondly, the effect of actuator damage on blade tip response and hub vibration is studied to understand the physical effects of actuator failure and the possibility of detection of actuator failure using the measured response and vibration data.

4 BASELINE ROTOR

The rotor properties used in this paper are shown in Table 2. This is a uniform rotor equivalent to the hingeless BO105 rotor and has been used in earlier studies [11, 12]. For the numerical results, the rotor blade is modeled using five spatial finite elements and six time elements. The first nine normal modes are used for the dynamics modeling. These are the four flap modes, four lag modes and one torsion mode. The baseline rotor has identical blades and uses an IBC based primary control system.

5 RESULTS AND DISCUSSION

Numerical results are obtained in hover and forward flight.

5.1 Hover

In hover, the rotor environment is axisymmetric and the results are easier to interpret physically compared to forward flight. A uniform inflow model is used for hover. Hover coupled trim is used which involves only vertical force and yaw moment equilibrium. Thus the cyclics are zero and only the collective pitch and tail rotor control are given by the pilot.

Table 2. Helicopter properties

Number of blades	4
Chord, c	0.2717 m
Solidity, σ	0.07
Lock number	5.20
C_T/σ	0.07
Pretwist	0
Precone	0
Mass per unit length, m_0	6.46 kg/m
Flap stiffness, $EI_y / m_0\Omega^2R^4$	0.0108
Lag stiffness, $EI_z / m_0\Omega^2R^4$	0.0268
Torsion stiffness, $GJ / m_0\Omega^2R^4$	0.00615
Rotor rpm, Ω	383
Blade radius, R	4.94 m

Figure 1 shows the collective pitch (θ_0) of the blades without actuator damages when collective on one blade is damaged. In this study, we will designate the blade with actuator damage as blade 1, and the blades without actuator damage are then blades 2, 3 and 4. The undamaged condition occurs when $D = 0$ in Figure 1, when all the four blades have a collective pitch of 8.39 degrees. As damage increases, the loss of collective authority in the blade with actuator damage is compensated by the increase in the collective pitch of the blades without actuator damage to a final value of 11.18 degrees when the damaged actuator has failed at $D = 1$. The tail rotor collective (θ_{0t}) does not show any change, as vertical equilibrium is maintained and the rotor torque remains the same.

The case of complete actuator failure at $D = 1$ is examined in more detail, and the blade tip response and hub loads for this case are studied. The tip response of the undamaged and blade with actuator damages is shown in Figure 2. For axisymmetric hover condition, only the steady component of the response is present. For comparison, the tip response of the blades of the rotor without any fault is also shown. The no fault or baseline rotor has a flap up (positive), lag back (negative) and nose down elastic twist (negative) condition. These deflections are amplified for the blades without actuator damage and the tip flap, lag and elastic twist increase by about 65, 75 and 21 percent from the baseline case, respectively. However, the blade with actuator damage shows an unusual flap down, lead-lag and nose up elastic twist tip deflection. The blade collective pitch for the blade with actuator damage is now zero and the section angle of attack comes from the inflow angle since $\alpha = \theta - \phi$. Here θ is the pitch angle, ϕ is the inflow angle and α is the section angle of attack. The blade with actuator damage therefore sees a negative angle of attack. Also, for the damage blade the angle of attack is greatly reduced and the effect of centrifugal effects and structural nonlinearities relative to aerodynamic forces will be larger. The net effect of these phenomenon's is that the deflections for the damage blade are in the opposite direction to those of the blades without actuator damage. Figure 3 shows the 1/rev longitudinal (F_x) and lateral (F_y) forces, and rolling (M_x) and pitching (M_y) moments generated due to the damaged rotor. These 1/rev loads are not present in the baseline rotor and occur because of the unbalance caused by the blade with actuator damage. These results show that actuator failure in hover leads to considerable increase in the blade tip deflections for the blades without actuator damage which need to compensate for the loss of aerodynamic forces on the blade with actuator damage. The higher deflections need to be addressed in structural design. In addition, the 1/rev loads will cause an increase in helicopter vibration for the pilot and passengers.

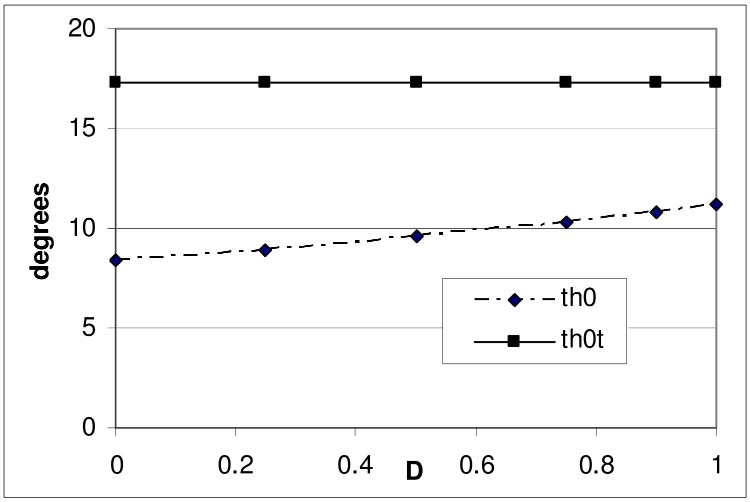

Figure 1 Effect of collective actuator damage on one blade on the collective pitch of blades without actuator damage and tail rotor collective in hover

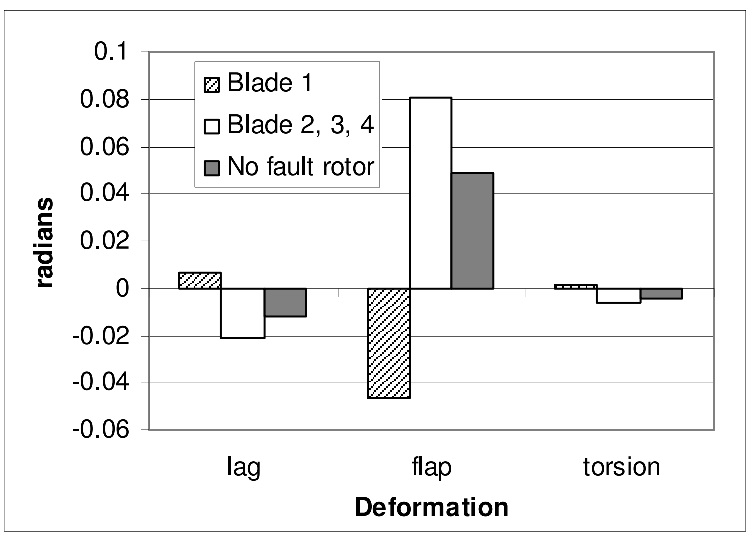

Figure 2 Steady tip deflections of no fault rotor, and damaged (blade 2, 3, and 4) and blades without actuator damage (blade 1) of rotor with fault in hover

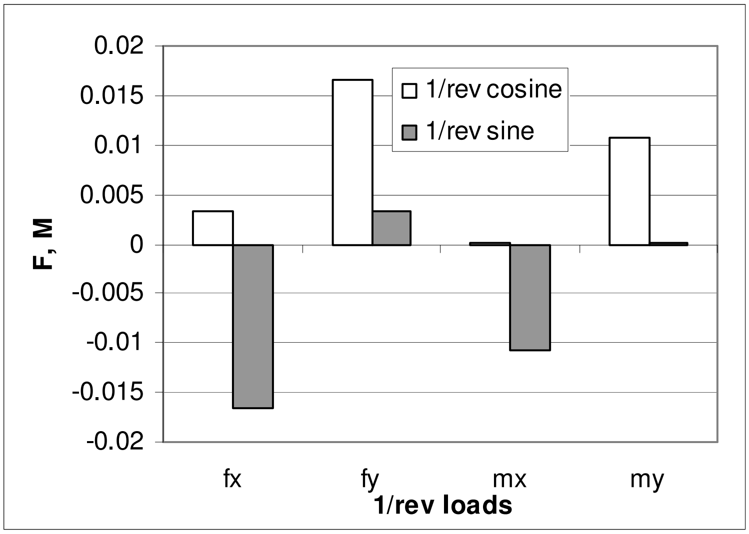

Figure 3 One per rev loads for rotor with actuator fault in hover

5.2 Forward Flight

Three advance ratios are initially considered for forward flight simulation, corresponding to low, moderate and high speed. The possibility of obtaining a trim solution for the damaged actuator is investigated for each flight condition for all the actuator failure possibilities listed in Table 1. The detailed analysis of the blade tip deflections and hub loads is considered for the moderate speed flight condition.

5.2.1 Low speed flight $\mu = 0.1$

All the simulations performed at low speed flight with different damage combinations could be successfully trimmed. This indicates that the pilot can compensate for actuator damage and failure on one blade by adjusting the trim controls for the other blades without actuator damage.

Figure 4 shows the effect of loss of actuator authority in blade 1 on the collective pitch of the blades without actuator damage. The cases involving collective, collective and lateral cyclic, collective and longitudinal cyclic and all three actuator failures show a steady rise of collective pitch for the blade without actuator damage. The other three cases, which do not involve collective damage, show negligible change in the collective of the blades without actuator damage.

The effect of actuator damage on the longitudinal cyclic is shown in Figure 5. The collective and longitudinal cyclic damage case leads to maximum change, closely followed by the “all three” case and the pure longitudinal cyclic case. Lateral cyclic failure has negligible influence on the longitudinal cyclic of the blades without actuator damage. Furthermore, pure collective and “collective and lateral cyclic” failure causes a small change in the required longitudinal cyclic.

The effect of actuator damage on the lateral cyclic of blades without actuator damage is shown in Figure 6. Here four cases consisting of lateral cyclic damage, “collective and lateral cyclic” damage, “all three” and “longitudinal and lateral cyclic” damage show the

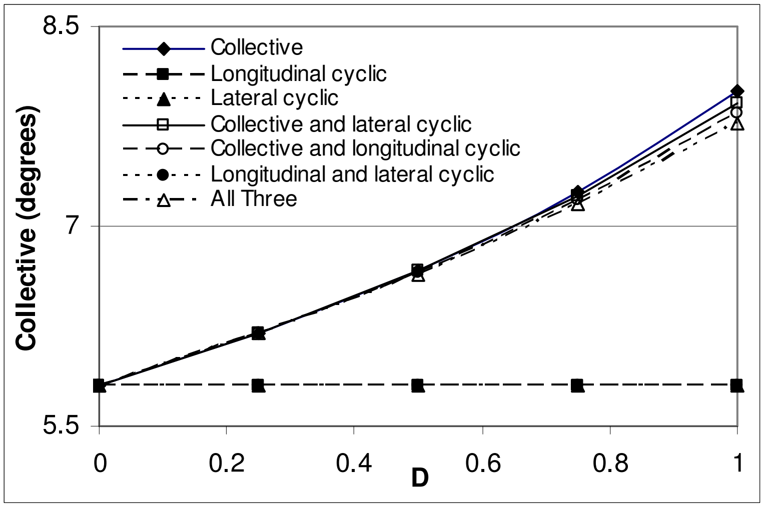

Figure 4 Effect of actuator damage on collective pitch of blades without actuator damage at μ=0.1

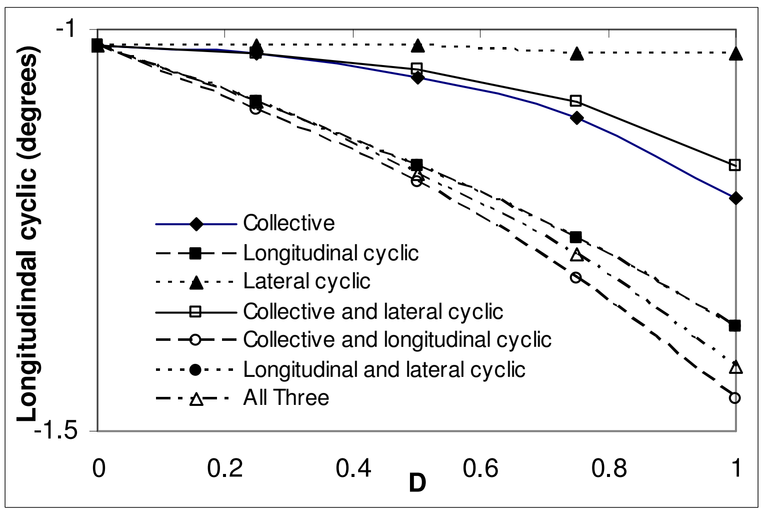

Figure 5 Effect of actuator damage on longitudinal cyclic of blades without actuator damage at μ=0.1

larger change. There is negligible effect of longitudinal cyclic failure on lateral cyclic. There is also a very small effect of collective and "longitudinal and collective" failure on the lateral cyclic.

Figures 7, 8 and 9 show the effect of actuator damage in blade 1 on the tail rotor collective, shaft tilt and bank angle of the helicopter, respectively. The effect on tail rotor collective is negligible. Therefore, adjusting for actuator failure on a main rotor blade primarily involves making changes in the main rotor controls of the other blades. At low speeds the shaft tilt of a typical rotor is negative which means that it is tilted backwards. Figure 8 shows that the four cases where the collective is damaged show a reduction in the shaft backward tilt. Cases where the collective and lateral cyclic are damaged leads the larger change in the bank angles. These changes in shaft tilt and bank angle are in the range of 1.5 degrees which means that there is only a small change in the helicopter

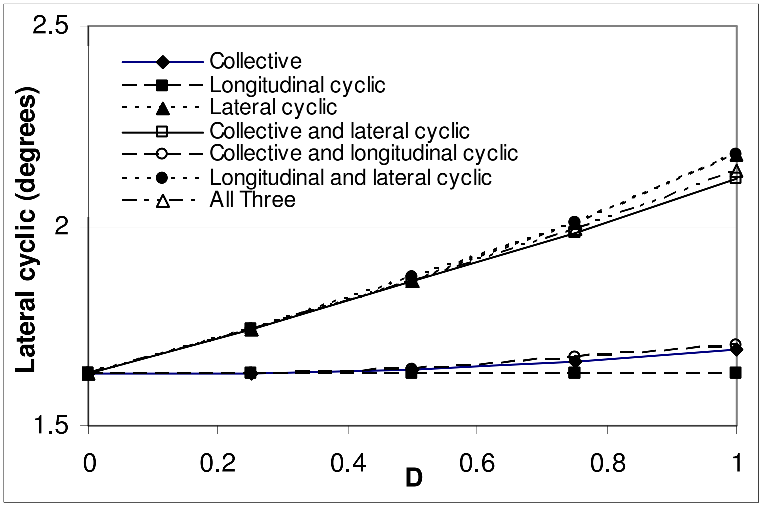

Figure 6 Effect of actuator damage on lateral cyclic of blades without actuator damage at μ=0.1

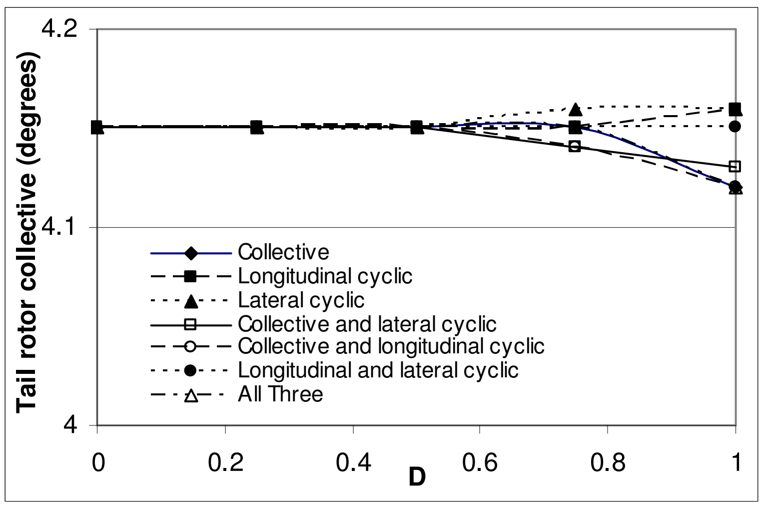

Figure 7 Effect of actuator damage on tail rotor collective at μ=0.1

attitude due to the compensation for actuator failure on one blade by changing the main rotor IBC controls.

Overall, these results show that the primary effect of actuator damage is on the collective pitch of the reconfigured rotor which has to be increased by over 2 degrees to compensate for the actuator failure cases which involve the collective. The increase needed is highest for the pure collective failure and least for the “all three” case, as seen in Figure 4. The pure cyclic faults do not show much change in the trim controls and are easily compensated for by adjusting the cyclics on the blades without actuator damage. These effects get amplified in the moderate speed flight regime discussed next.

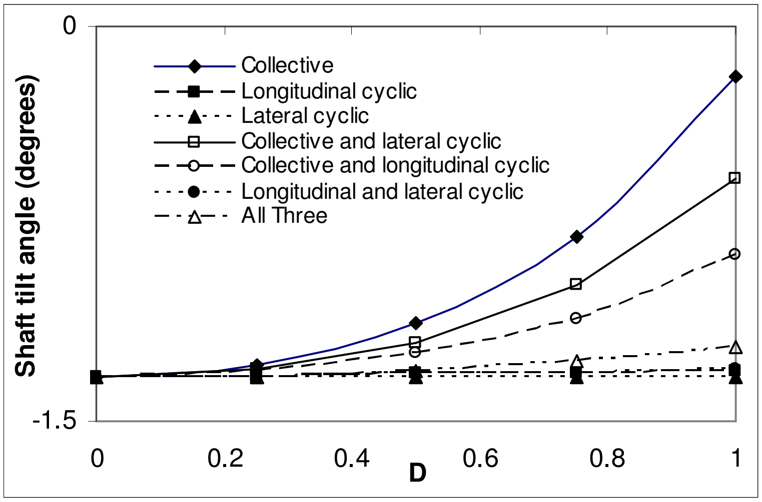

Figure 8 Effect of actuator damage on shaft tilt angle of helicopter at μ=0.1

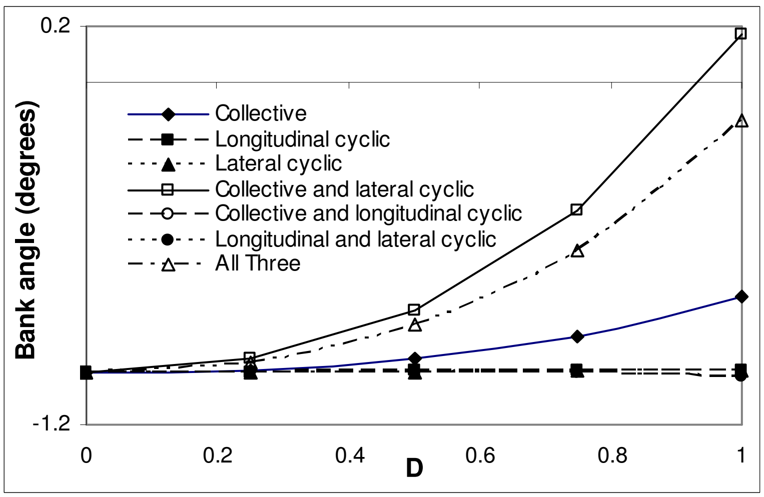

Figure 9 Effect of actuator damage on bank angle of helicopter at μ=0.1

5.2.2 Moderate speed flight, $\mu = 0.2$

For the moderate flight speed, the rotor could be trimmed at all points until actuator failure for all the damage types shown in Table 1. The control angles and the attitude of the helicopter with increasing levels of actuator damage is shown in Figures 10-15. Figure 10 shows the effect of actuator damage on blade 1 on the collective pitch of the blades without actuator damage. The effect is largest for pure collective failure which results in the collective for the reconfigured blades going up by 4.11 degrees and for the collective and lateral cyclic failure which leads to a change of 3.7 degrees. There is negligible impact on the collective for the pure longitudinal and lateral cyclic damage cases and for the "longitudinal and lateral cyclic" damage case. The remaining two cases of "collective and longitudinal cyclic" and "all three" show collective increases of 2.43 and 2.25 degrees. Clearly, the pure collective damage case results in the most significant change for the reconfigured rotor.

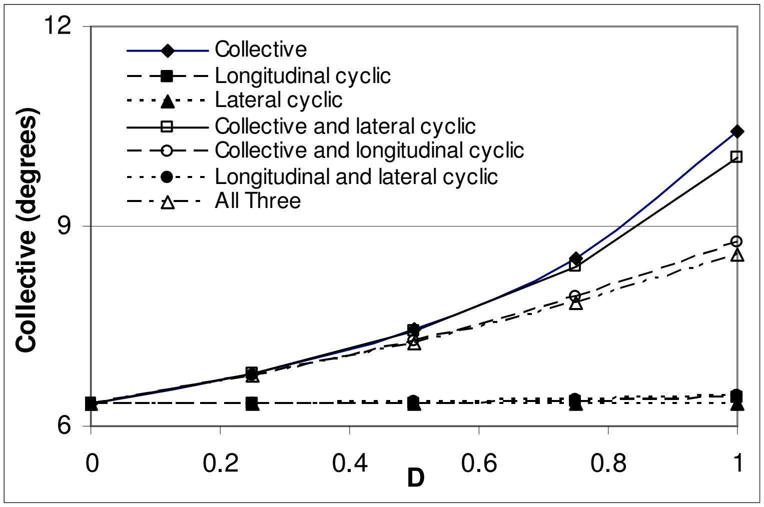

Figure 10 Effect of actuator damage on collective of blades without actuator damage at μ=0.2

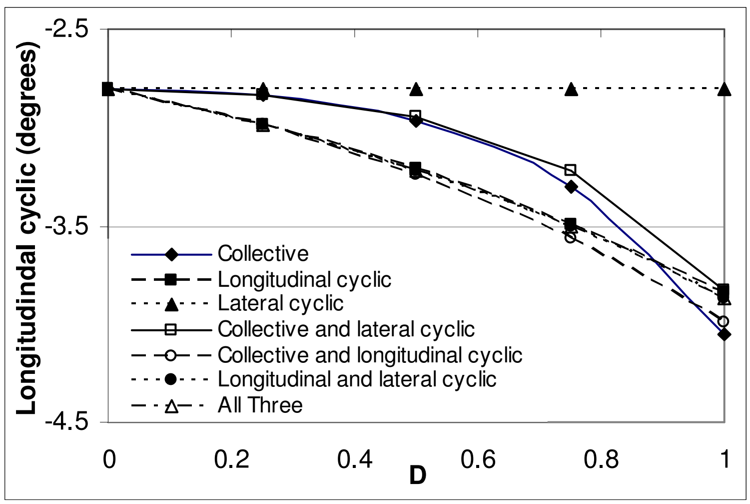

Figure 11 Effect of actuator damage on longitudinal cyclic of blades without actuator damage at μ=0.2

The effect of damage on the longitudinal cyclic of the trimmed rotor is shown in Figure 11. The pure lateral cyclic failure has no influence on the longitudinal cyclic of the blades without actuator damage. All other cases show a change of about 1 to 1.5 degrees. The pure collective and "collective and lateral cyclic" cases show a sharp increase at the higher damage levels from $D = 0.8$ to $D = 1$, which is also seen in Figure 10 for the collective.

Figure 12 shows the effect of damage on the lateral cyclic of the blades without actuator damage. The longitudinal cyclic damage has no influence on the lateral cyclic of the blades without actuator damage. The maximum change of 0.87 degrees occurs for the collective and lateral cyclic failure. The three cases involving pure lateral cyclic failure, "longitudinal and lateral cyclic" failure and "all three" failure are clustered together. The pure collective case shows a sharp increase after $D = 0.8$.

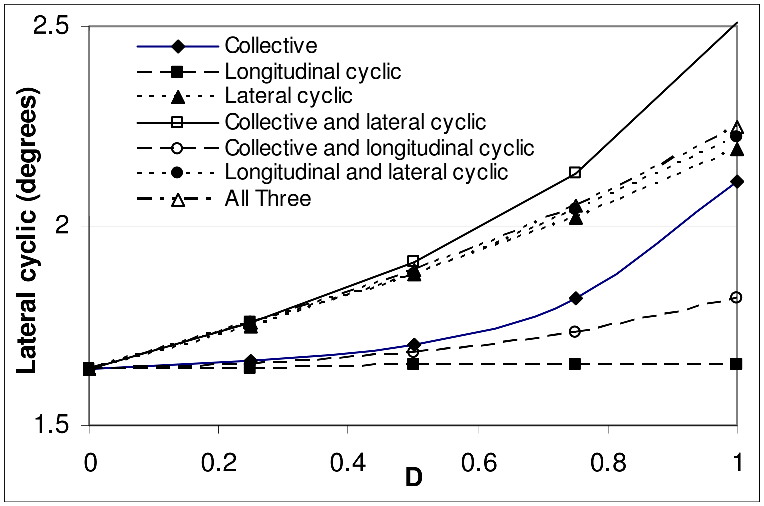

Figure 12 Effect of actuator damage on lateral cyclic of blades without actuator damage at μ=0.2

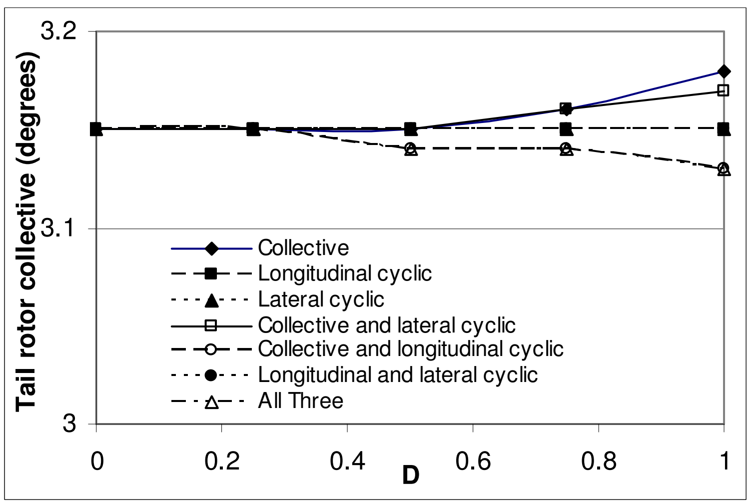

Figure 13 Effect of actuator damage on tail rotor collective at μ=0.2

The tail rotor collective in Figure 13 shows very small changes implying that the pilot can trim the helicopter with the failed actuator using primarily the remaining main rotor blade IBC controls.

The shaft tilt and the bank angles are shown in Figure 14 and 15, respectively. The maximum shaft tilt is now considerably increased compared to the results for $\mu = 0.1$ in Figure 8 but the top two cases resulting in large shaft tilt remain the same as the pure collective failure and "collective and lateral cyclic" failure. The maximum change in bank angle is caused by the "collective and lateral cyclic" failure followed by the "all three" case. As we shall see, these effects will get further amplified at the high speed flight condition of $\mu = 0.3$ discussed next.

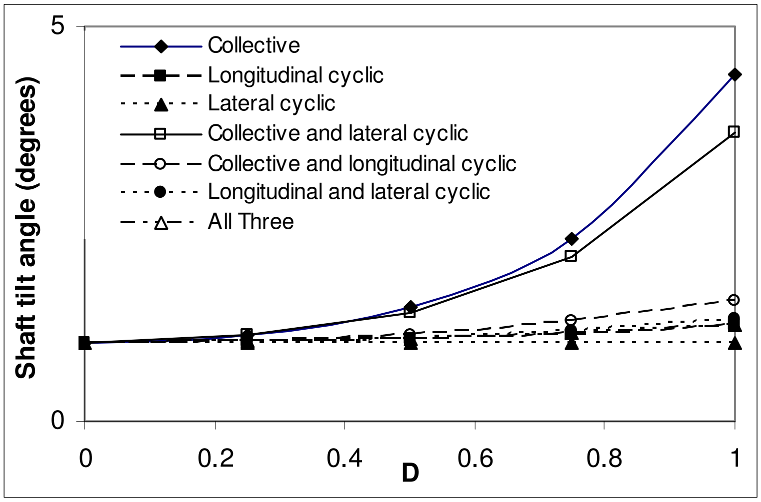

Figure 14 Effect of actuator damage on shaft tilt at μ=0.2

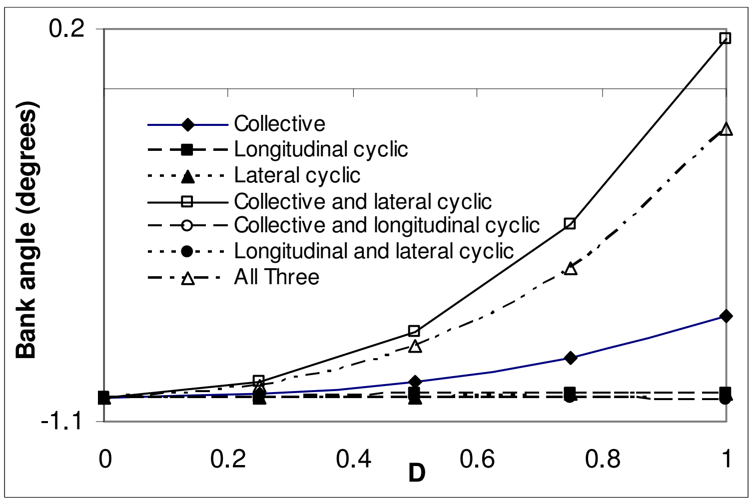

Figure 15 Effect of actuator damage on bank angle at μ=0.2

5.2.3 High speed flight, $\mu = 0.3$

At the high speed flight condition, it is not possible to trim the rotor for the pure collective and "collective and lateral cyclic" damage as the damage size becomes large (greater than $D = 0.8$). However, all other cases including the "all three" case can be trimmed until $D = 1$.

The very high levels of collective needed for the reconfigured rotor for collective and "collective and lateral cyclic" damage is clear from Figure 16. This phenomenon was observed at the low speed in Figure 4 and moderate speed in Figure 10 also but gets greatly amplified in the high speed flight condition. To understand the physics behind this behavior, consider Figures 17, 18 and 19 which show the control angle experienced by the blade with actuator damage for the four cases involving failure of the collective at $\mu = 0.1$, 0.2 and 0.3, respectively. The $\mu = 0.3$ case is for the highest level of damage D at which trim could be achieved. The results obtained at $\mu = 0.1$ and $\mu = 0.2$ correspond to a damage level of $D = 1$. The pure collective and "collective and lateral cyclic" failure cases show the maximum negative pitch near the advancing side ($\psi = 90^0$) and the maximum positive pitch near the retreating side ($\psi = 270^0$) of the rotor disk. Since the dynamic pressure is much higher on the advancing side than the retreating side, there is a downward lift generated in these cases which needs to be compensated by higher collective pitch on the other blades. This effect is primarily due to the presence of the longitudinal cyclic on the blade with actuator damage which increases in magnitude with flight speed. The case involving "collective and longitudinal cyclic" failure shows a pitch governed by the lateral cyclic which is typically much less than the longitudinal cyclic at high speeds. *Therefore, it can be concluded that the presence of a longitudinal cyclic on the blade with actuator damage when the collective actuation has failed makes it very difficult to trim the helicopter in high speed flight.* Also, if the collective were to fail on blade 1, it is best to also set the longitudinal cyclic for this blade to zero. Note that when all three control angles fail on blade 1, the control angle experienced by the blade with actuator damage is zero and the rotor is much easier to reconfigure and trim than for pure collective failure.

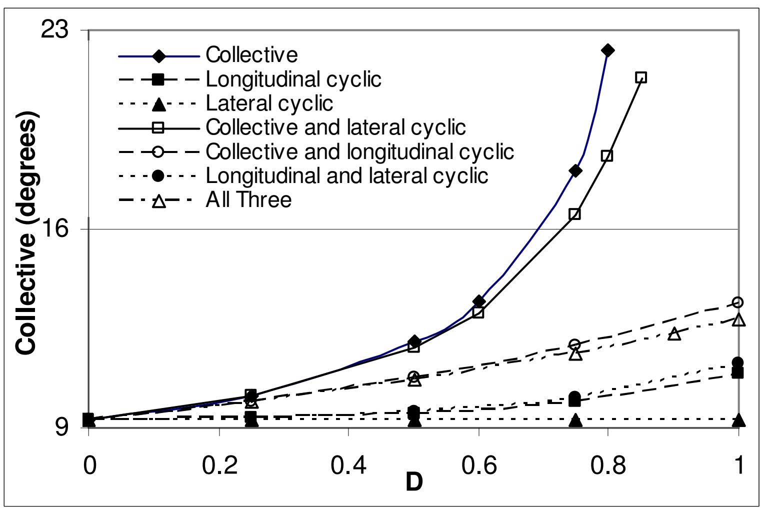

Figure 16 Effect of actuator damage on collective of blades without actuator damage at μ=0.3

Figure 20 shows the effect of actuator damage on the longitudinal cyclic of the blades without actuator damage. As at the lower advance ratios, the lateral cyclic damage has negligible influence on the longitudinal cyclic. All the other damages have considerable influence. The "collective" and "collective and lateral cyclic" damage lead to an increase in the magnitude of the longitudinal cyclic by 7.62 and 7 degrees, respectively, at the last trimmable point. The other four faults also show significant change on 3-4 degrees in the longitudinal cyclic when the actuator fails.

The change in lateral cyclic due to damage for the blades without actuator damage is shown in Figure 21. The lateral cyclic is typically quite small for a baseline undamaged rotor, but shows considerable increase for the "collective" and "collective and lateral cyclic" cases of 4.33 and 4.84 degrees, respectively. As for the lower speeds, there is negligible influence of the longitudinal cyclic damage on the lateral cyclic. The other four faults show a change of about 1-4 degrees in the lateral cyclic at the point of actuator failure. The tail rotor collective of the helicopter is shown in Figure 22. The change in tail rotor collective is small except in the case of the “collective” and “collective and lateral cyclic” damage. However, even for these cases the change is less than 1 degree. Figures 23 and 24 show the shaft tilt and bank angle become very high for the “collective” and “collective and lateral cyclic” damage cases. The other five faults show much less impact on the helicopter attitude.

The main lesson learned from the simulations is that it is possible to trim the rotor after actuator failure in hover and low and moderate speed forward flight for all fault conditions. However, at high speeds the collective and “collective and lateral cyclic” failures should be avoided by setting the longitudinal cyclic of the damaged rotor to zero. The “all three” failure case can be trimmed at all flight conditions considered here and shows the advantage of the IBC based primary control system in terms of reliability, fault tolerance and survivability.

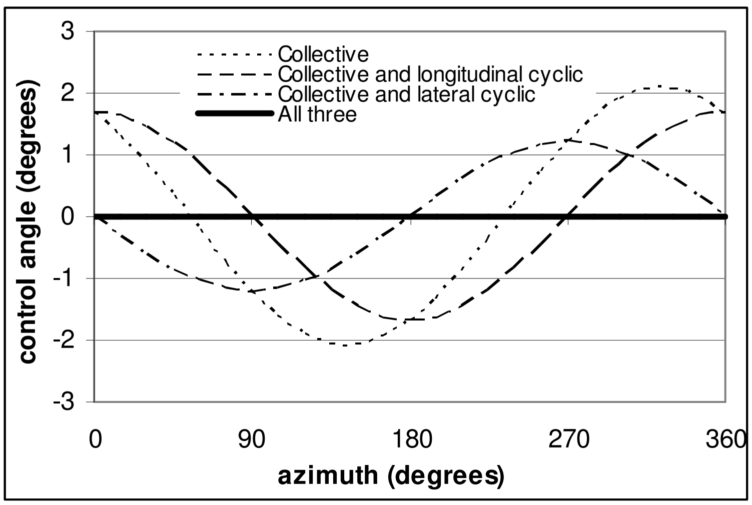

Figure 17 Control angle experienced by blade with actuator damage at μ=0.1

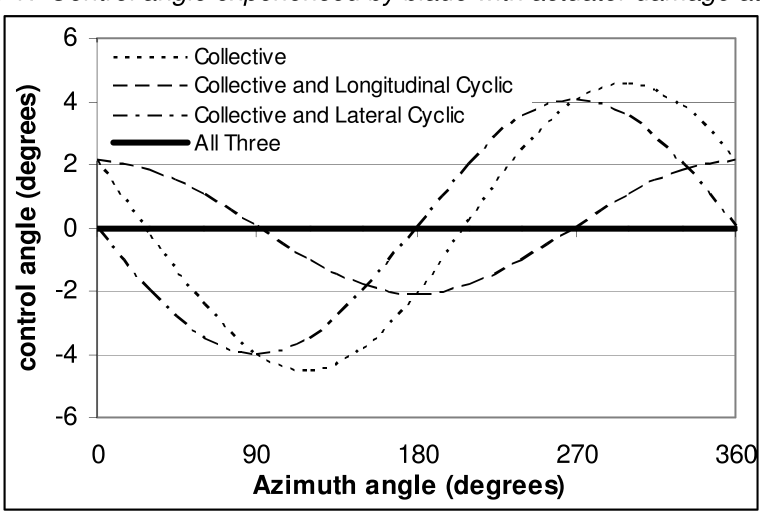

Figure 18 Control angle experienced by blade with actuator damage at μ=0.2

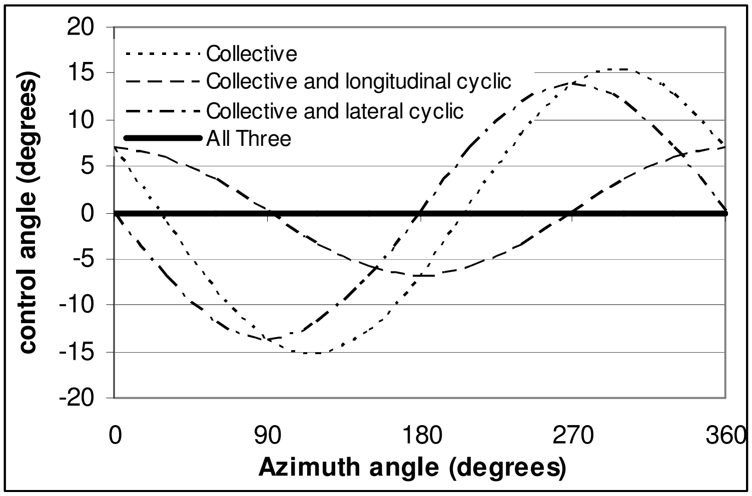

Figure 19 Control angle experienced by the blade with actuator damage at μ=0.3

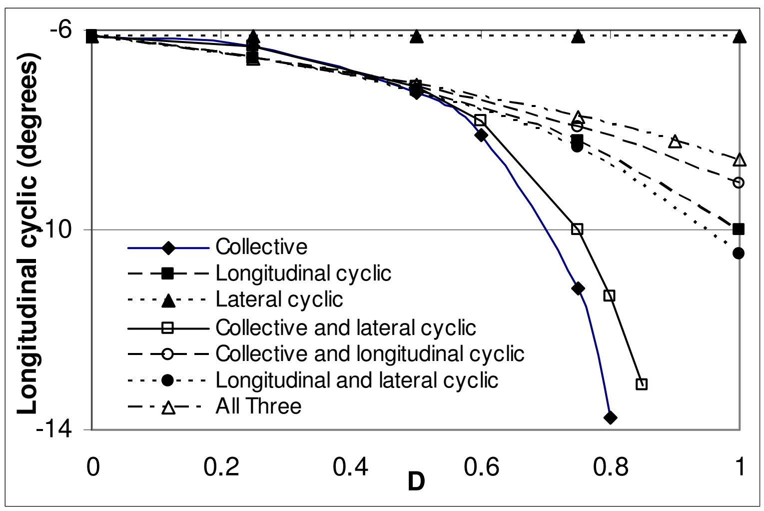

Figure 20 Effect of actuator damage on longitudinal cyclic of blades without actuator damage at μ=0.3

5.3 Response and Hub Loads

To understand the effect of the reconfigured rotor on the blade dynamics and helicopter vibration, we look at the blade tip response and hub loads of the damaged and undamaged rotor at $\mu = 0.2$. The moderate speed condition captures the physics of the problem and is trimmed at all fault conditions. For the numerical results, complete actuator failure or $D = 1$ case is considered for the pure collective damage and the "all three" damage. The collective failure is selected since it is the most dangerous failure as seen from the trim results. The "all three" failure case is most realistic as it simulates a situation where actuator authority is lost. For example, in the situation where a trailing edge flap is used for IBC, it simulates a situation where the flap on the blade with actuator damage does not move from its baseline undeflected position. Consider a flap based primary control which is actuated as

$$\delta(\psi) = \delta_0 + \delta_{1c} \cos\psi + \delta_{1s} \sin\psi$$

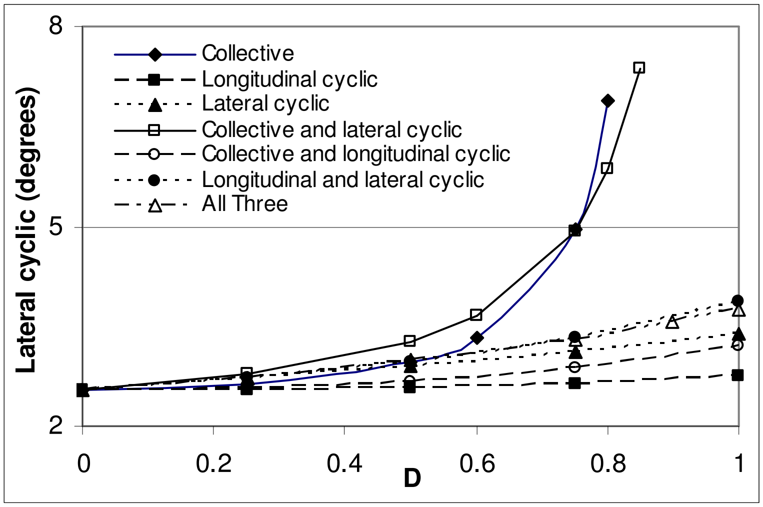

Figure 21 Effect of actuator damage on lateral cyclic of blades without actuator damage at μ=0.3

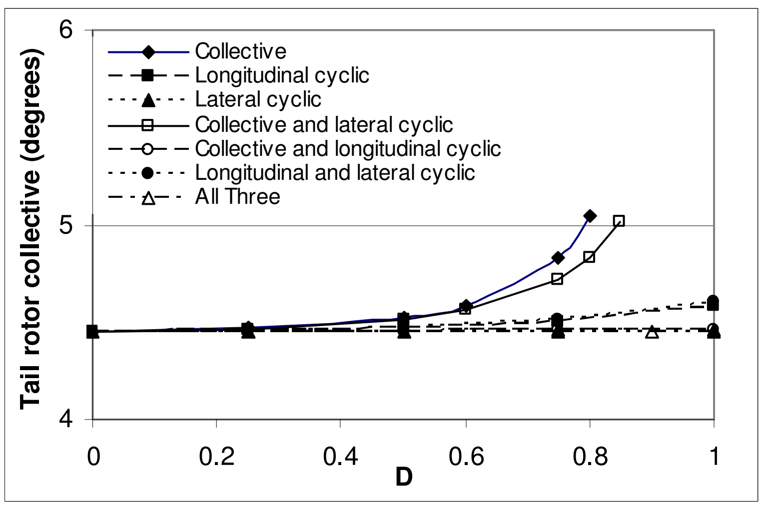

Figure 22 Effect of actuator damage on tail rotor collective at μ=0.3

The "all three" failure then corresponds to the three controls δ_0, δ_{1c} and δ_{1s} being equal to zero for blade 1. The collective failure corresponds to δ_0 being zero. Such a failure is possible if a different mechanism is used to supply the large collective angles which can be done using quasi static actuation using shape memory alloys (SMA). Thus piezoceramic actuators used for higher bandwidth cyclic control may not fail but the SMA based collective may fail.

For the active twist rotor, the three actuator failure simulates the case where the electric power cannot be supplied to the piezoceramic actuators on the blade with actuator damage or the actuators have broken or cracked. Pure collective failure could happen if a quasi-steady SMA based mechanism is used for providing the larger collective deflections.

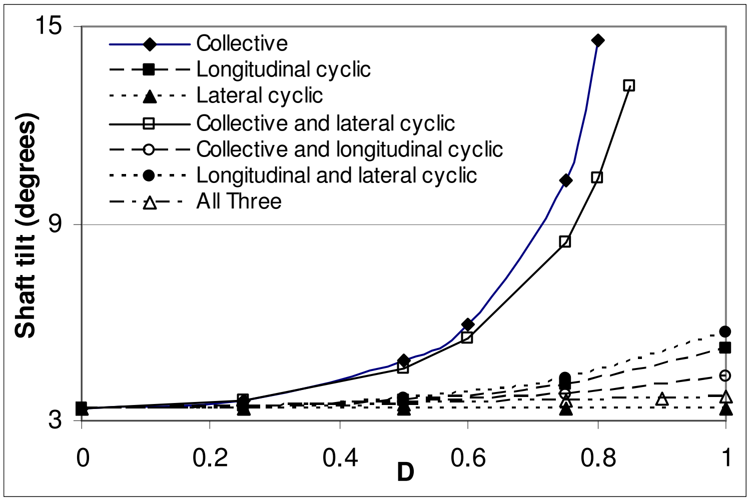

Figure 23 Effect of actuator damage on helicopter shaft tilt at μ=0.3

The tip flap response for the rotor without any faults (baseline case) and the rotor with faults is shown in Figure 25 for collective damage and Figure 26 for the "all three" faults case. For the damaged rotor, the response of the blade with actuator damage (blade 1) is different from the blades without actuator damage (blades 2, 3 and 4). For the collective damage, the blade with actuator damage shows a large flap down response which is primarily 1/rev and reaches a minimum around $\psi = 180^0$. This response is primarily caused by the presence of the longitudinal cyclic on blade 1 which goes through a maximum negative value at $\psi = 90^0$ on the advancing side. To compensate for the down flapping of blade 1, the other 3 blades flap up by a considerable margin. *The important points from these results are that the flapping deflections of the blade can increase by a large amount in the reconfigured rotor which needs to be addressed during blade structural design.* The large flap response is considerably reduced in the case of "all three" fault shown in Figure 26. In this case also blade 1 flaps down and the other three blades flap up beyond the baseline rotor case to compensate for the blade with actuator damage.

The tip lag response of the blades is shown in Figure 27 and 28 for the "collective" and "all three" fault cases, respectively. Again, the "collective" fault case causes a greater change in response compared to the "all three" fault case. The blade with actuator damage shows lead-lag motion for most of the rotor disk for the "collective" damage case. Some lead-lag along with lag is also visible for the "all three" case. The baseline blade has lagging motion throughout the azimuth. The blades without actuator damage show an increase in the lag back to compensate for the blade with actuator damage. *Again, the lag motions of the reconfigured rotor are much larger than the baseline and need to be considered in structural design for the rotor.*

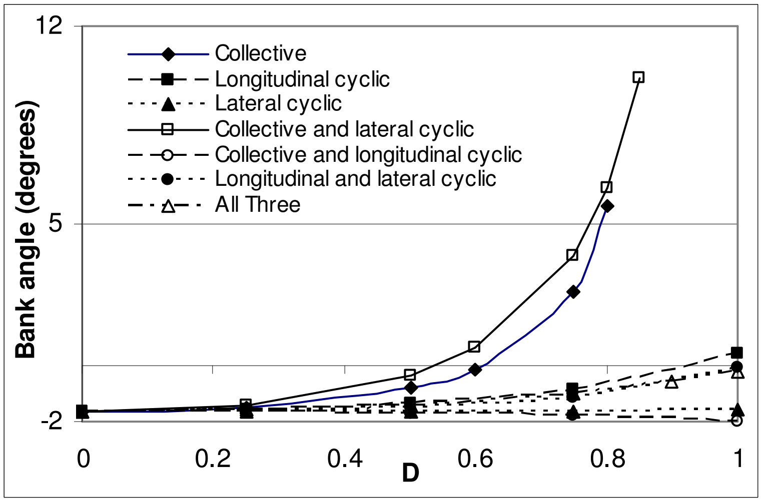

Figure 24 Effect of actuator damage on helicopter bank angle at μ=0.3

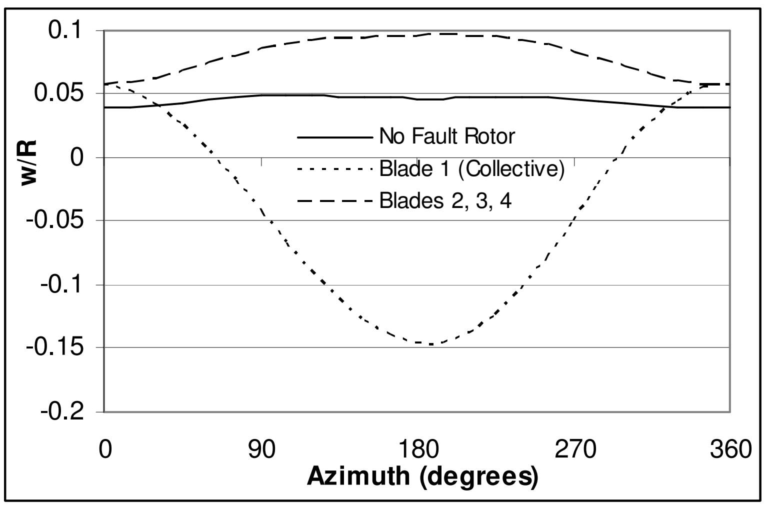

Figure 25 Tip flap response of rotor blades with and without collective fault

The blade tip elastic torsion response is shown in Figures 29 and 30 for the "collective" and "all three" fault cases, respectively. There is a nose up twist for the blade with actuator damage in contrast to the nose down twist observed for the blades without actuator damage and the baseline rotor. Again, the blades without actuator damage show an increase in the magnitude of the response which is more for the "collective" fault and less for the "all three" fault. However, the changes in torsion and lag response are not as large as the change in the blade flap response, which is especially sensitive to the change in blade lift due to change in collective. The results also show that the damaged rotor can be reconfigured to trim with the failed actuators and the price to pay is higher levels of blade deflections. Furthermore, the presence of the negative flap, lead-lag and nose up elastic twist on the blade with actuator damage can be used as an indicator to identify actuator failure.

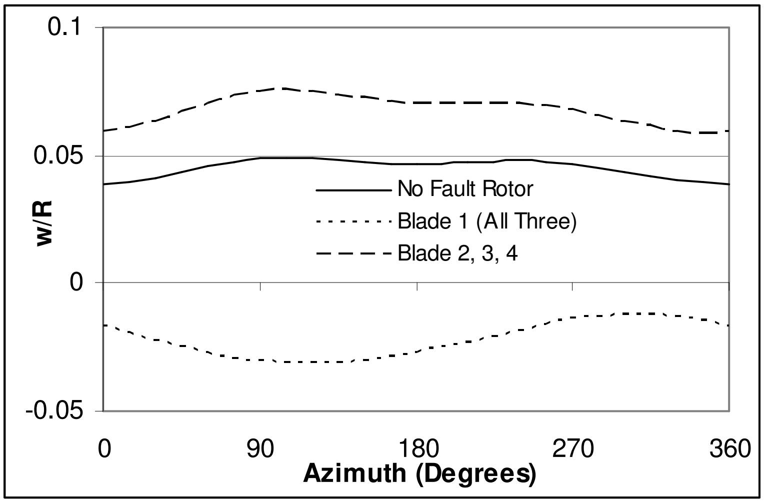

Figure 26 Tip flap response of rotor blades with and without "all three" fault

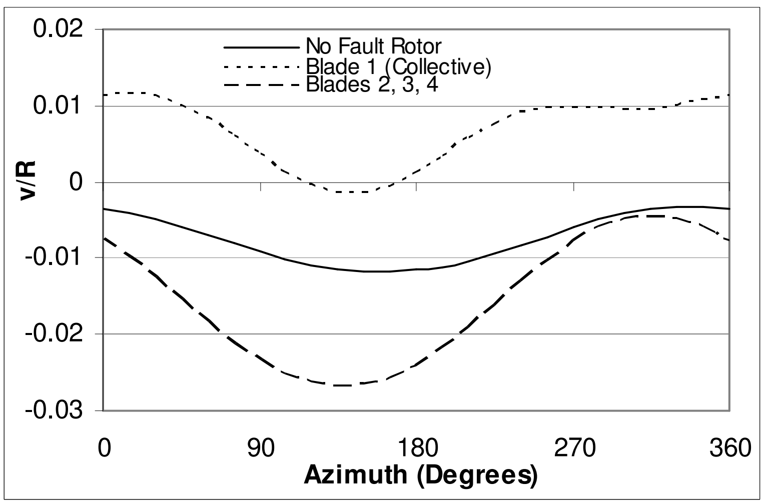

Figure 27 Tip lag response of rotor blades with and without collective fault

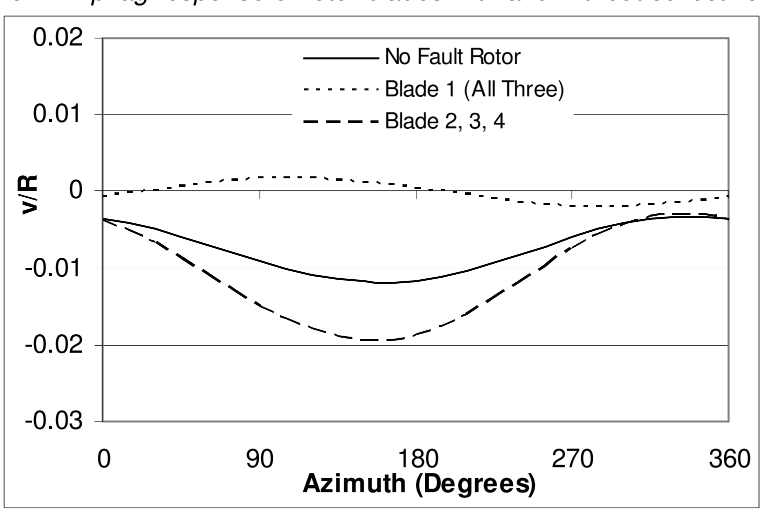

Figure 28 Tip lag response of rotor blades with and without "all three" fault

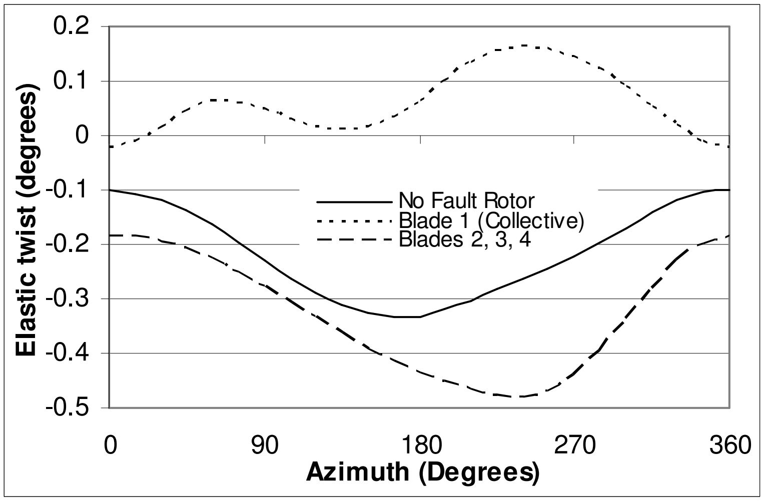

Figure 29 Tip elastic twist response of rotor blades with and without collective fault

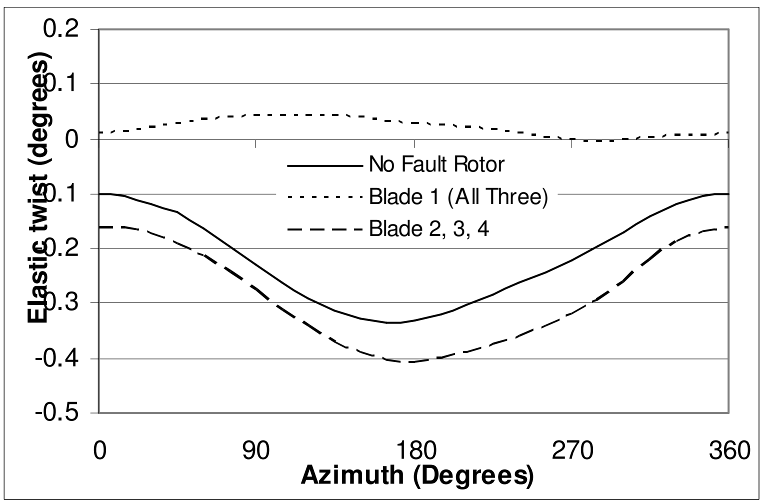

Figure 30 Tip elastic twist response of rotor blades with and without "all three" fault

The steady and first four harmonics of the vibratory hub loads are shown in Figures 31 to 36. The forces and moments are non-dimensionalized by dividing by $m_0\Omega^2R^2$ and $m_0\Omega^2R^3$, respectively. For the no fault or baseline rotor, only the steady and 4/rev hub loads are present for the 4-bladed rotor. The 8/rev loads are much smaller and are not shown. For a typical 4-bladed rotor, the 4/rev loads are the main source of helicopter vibration. The steady loads are used for trim analysis and affect the trim control angles.

The most important effect of the damaged actuator is to create 1, 2 and 3 per rev loads for the damaged rotor. As can be seen from Figures 31-36, the baseline or no fault rotor has steady and 4/rev loads only. Large 1/rev longitudinal, lateral and vertical forces can be observed from Figures 31, 32 and 33, respectively. There is a substantial change in the steady longitudinal force for the collective damage. However, the steady vertical force remains almost same for the rotor as the thrust needs to be maintained at the same level. The rolling and pitching moments also show large 1/rev components. The steady yawing moment is related to rotor torque or power and does not show much change which means that the power required by the damaged rotor remains almost unchanged. The "collective" damage is more severe and leads to changes in the loads of greater magnitude for most cases. However, for the 2/rev longitudinal and lateral forces, the changes due to "collective" and "all three" faults are almost same. For the pure collective fault, the presence of the longitudinal cyclic in the blade with actuator damage is the main cause of the high levels of unbalance. Interestingly, for almost all cases the 4/rev loads do not show much change and are also much smaller than the 1/rev loads caused by unbalance. This is because the actuator failure involves unbalance and changes in steady and 1/rev forcing which do not affect the vibratory loads much.

The results show that the reconfigured damaged rotor has a much higher level of vibration compared to the no-fault rotor. However, the high vibration and blade response are the price to pay for the possibility of surviving damage to any one actuator in an IBC based primary control system. Such survivability following actuator damage for swashplate based helicopter rotors is not possible as all the trim controls are controlled through the swashplate. The problems due to pure collective failure show that it is best to somehow stop the actuator movements completely once the actuator fails so as not to have a blade without collective but with longitudinal cyclic. The "all three" fault case is better from the trim, rotor dynamics and survivability perspective than the "pure collective" or "collective and lateral cyclic" cases.

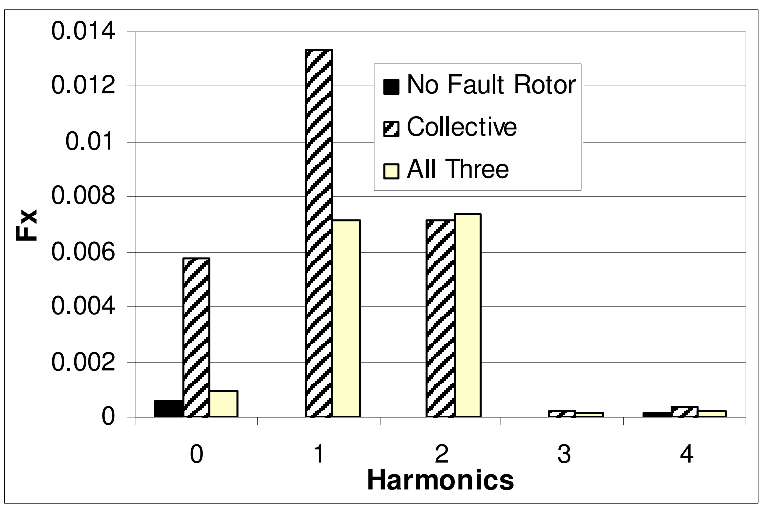

Figure 31 Longitudinal hub force of rotor with and without fault

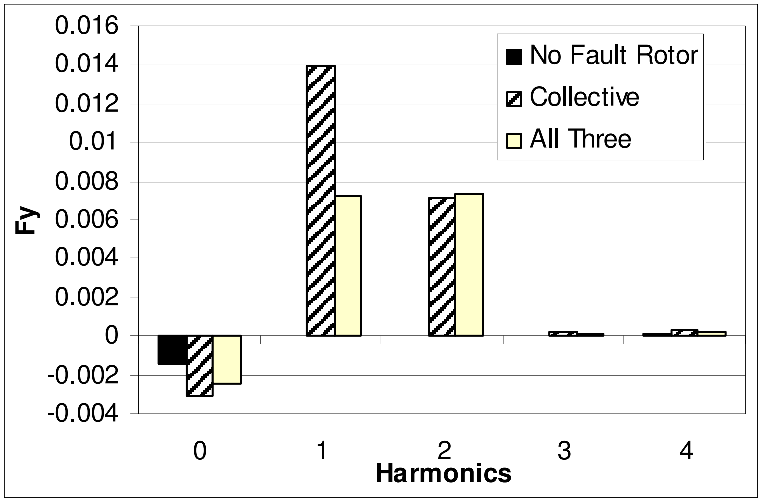

Figure 32 Lateral hub force of rotor with and without fault

5.4 Very high speed flight, $\mu = 0.35$

The most practical failure case is the full primary control failure on blade 1, which is the "all three" case discussed. Figure 37 shows the trim control and vehicle attitude angles for the "all three" fault case at an advance ratio of $\mu = 0.35$. The collective and longitudinal cyclic on the undamaged blades need to be increased to compensate for the actuator damage. It is clear that it is possible to trim and therefore fly the reconfigured helicopter with complete loss of actuator in one blade at even very high speed flight.

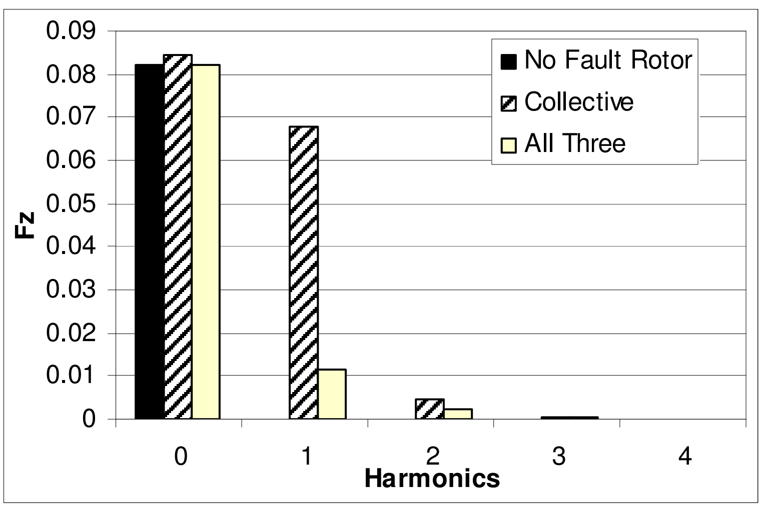

Figure 33 Vertical hub force of rotor with and without fault

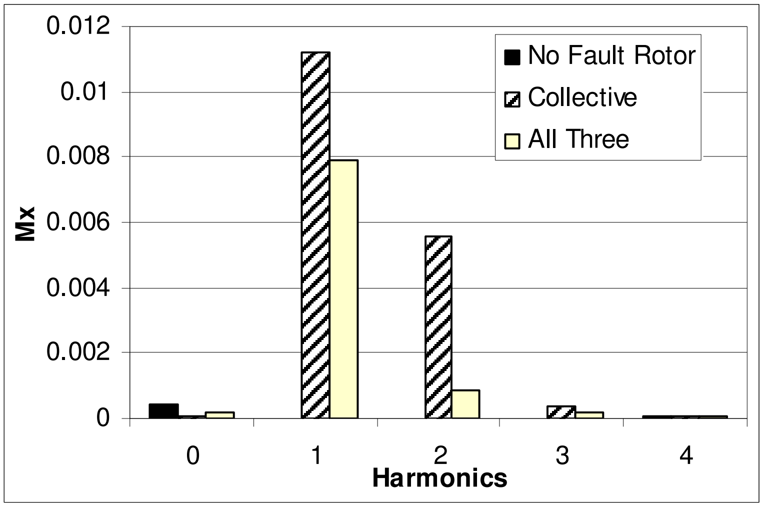

Figure 34 Hub rolling moment of rotor with and without fault

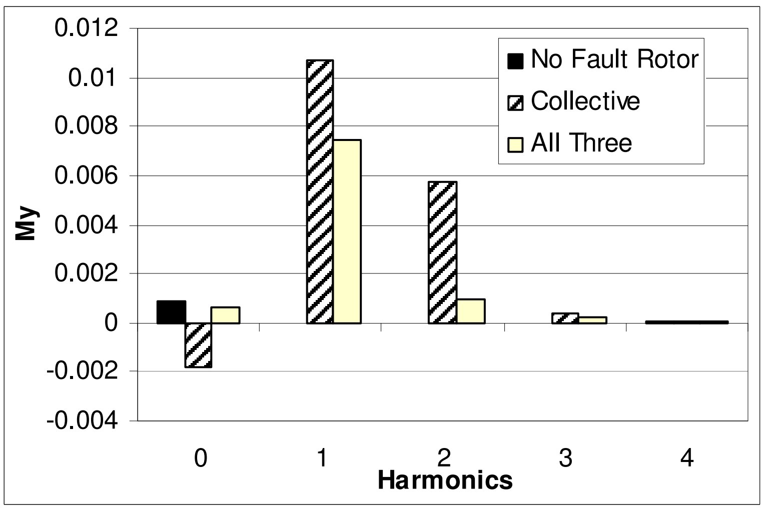

Figure 35 Hub pitching moment with and without faults

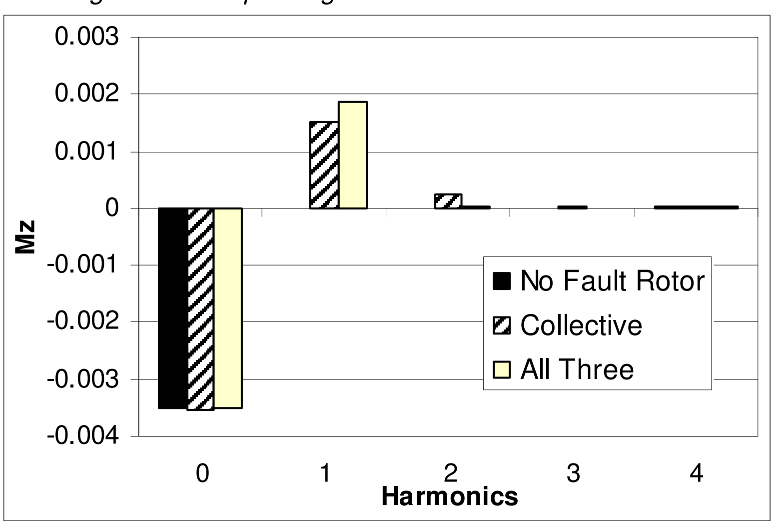

Figure 36 Hub yawing moment with and without faults

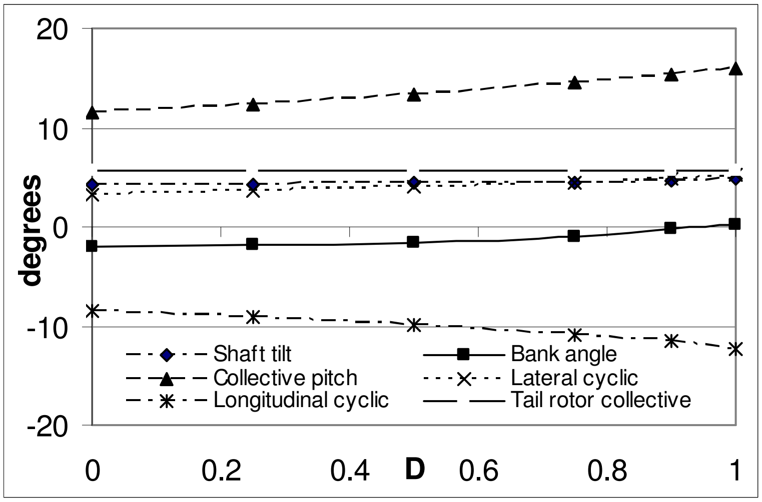

Figure 37 Trim control and vehicle attitude angles for reconfigured rotor with the "all three" fault at μ=0.35

6 CONCLUSIONS

A systematic study of the effect of actuator damage on a 4-bladed hingeless rotor with IBC based primary control is conducted in hover and forward flight conditions. The damage is simulated using different combinations of collective, longitudinal and lateral cyclic. The damage is considered on one blade (blade 1) and the other three blades are assumed to have no actuator failure. Blade response and hub loads are considered for the two important cases of collective failure and "all three" failure on the blade with actuator damage. The following conclusions are drawn from this study.

1. Pure collective damage in hover for blade 1 requires an increase in the collective of the other three blades to achieve trim. The blade with actuator damage is characterized by flap down, lead-lag and nose up torsion deflections. Longitudinal and lateral 1/rev forces and rolling and pitching 1/rev moments result because of the unbalance caused by the actuator damage.
2. For all the damage levels and types considered, trim is achieved for advance ratios of 0.1 and 0.2 which represent low and moderate speed flight. For $\mu = 0.3$, there is difficulty in trimming the damaged rotor for "pure collective" and "collective and lateral cyclic" faults at high damage levels. These two faults are most problematic from trim, response and survivability perspective. The problems for these faults occur because of the presence of longitudinal cyclic on the blade with actuator damage which causes high negative pitch angles on the advancing side where the dynamic pressure is at the highest value.
3. For the case of all primary controls being lost on the blade with actuator damage, the rotor is successfully trimmed at all advance ratios until $\mu = 0.35$. This shows that the IBC based primary control is advantageous from the reliability, survivability and fault tolerant perspective and the pilot can trim the helicopter using the controls on the remaining blades.
4. The presence of actuator faults causes an increase in the magnitudes of the blade response for the blades with and without actuator damages. The blade with

actuator damage tend to have a flap down, lead-lag and nose up torsion response for the critical faults and this requires the other undamaged blades to compensate by having larger flap, lag and nose down torsion deflections than the baseline or no fault rotor. The blade with actuator damage is exposed to negative angles of attack in case of collective failures and the resultant downward lift needs to be compensated by the other blades which experience large flap deflections.

5. The actuator fault introduces blade unbalance which leads to large 1/rev loads. Large 2/rev loads are also present for the longitudinal and lateral hub forces. The change in steady hub moment and therefore rotor power is negligible.
6. The large response of the blades (especially in flap) needs to be considered during structural design to avail of the advantage of the fault tolerance of the IBC actuated rotor to an actuator failure. The changes in blade response between the undamaged and blade with actuator damages and the presence of vibratory loads other than *N*/rev for an *N* bladed rotor can be used by a health monitoring system for detecting actuator damage and failure.

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8 REFERENCES

1. Omistron, R.A., “Aeroelastic Considerations for Rotorcraft Primary Control with On-Blade Elevons”, Proceedings of the 57th Annual Forum of the American Helicopter Society, Washington DC, USA, May 9-11, 2001.
2. Shen, J., and Chopra, I., “A Parametric Design Study for a Swashplateless Helicopter Rotor with Trailing Edge Flaps”, Journal of the American Helicopter Society, Vol. 49, No. 1, January 2004.
3. Shen, J., and Chopra, I., “Swashplateless Helicopter Rotor with Trailing Edge Flaps for Flight and Vibration Control”, Journal of Aircraft, Vol. 43, No. 2, 2006 pp. 346-352.
4. Kim, J., and Koratkar, N.A., “Feasibility Study to Develop a Mach-scaled Swashplateless Rotor Model”, Smart Materials and Structures, Vol. 14, No. 1, 2005, pp. 79-86.
5. Sirohi, J., and Chopra, I., “Design and Development of a High Pumping Frequency Piezoelectric-Hydraulic Hybrid Actuator”, Journal of Intelligent Material Systems and Structures, Vol. 14, No. 3, 2003, pp. 135-147.
6. Friedmann, P.P., Carman, G.P., and Millott, T.A., “Magnetostrictively Actuated Control Flaps for Vibration Reduction in Helicopter Rotors-Design Considerations for Implementation”, Mathematical and Computer Modelling, Vol. 33, No. 10-11, 2001, pp. 1203-1217.
7. Thakkar, D., and Ganguli, R., “Use of Single Crystal and Soft Piezoceramics for Alleviation of Flow Separation Induced Vibration in Smart Helicopter Rotor”, Smart Materials and Structures, Vol. 15, Vol. 2, 2006, pp. 331-341.

8. Gandhi, F., and Sekula, M.K., "Helicopter Horizontal Tail Incidence Control to Reduce Rotor Cyclic Pitch and Blade Flapping", Proceeding of the 60th Annual Forum of the American Helicopter Society, Baltimore, Maryland, USA, June 7-10, 2004.
9. Ganguli, R., "Health Monitoring of a Helicopter Rotor in Forward Flight using Fuzzy Logic", AIAA Journal, Vol. 40, No. 1, 2002, pp. 2373-2382.
10. Hodges, D.H., and Dowell, E.H., "Nonlinear Equation of Motion for Elastic Bending and Torsion of Twisted Nonuniform Blades", NASA TND-7818.
11. Viswamurthy, S.R., and Ganguli, R., "An Optimization Approach to Vibration Reduction in Helicopter Rotors with Multiple Trailing Edge Flaps", Aerospace Science and Technology, Vol. 8, No. 3, 2004, pp. 599-608.
12. Ganguli, R., "Optimum Design of a Helicopter Rotor for Low Vibration using Aeroelastic Analysis and Response Surface Methods", Journal of Sound and Vibration, Vol. 258, No. 2, 2002, pp. 440-447.
13. Bir, G., et al, "University of Maryland Advanced Rotorcraft Code (UMARC) Theory Manual," UM-AERO Report 92-02, Aug 1992.