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DEVELOPMENT OF AN INTEGRATED ELECTRICAL SWASHPLATELESS PRIMARY AND INDIVIDUAL BLADE CONTROL SYSTEM

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Abstract

It has been shown that a helicopter rotor in forward flight can greatly benefit from more complex pitch control schemes than used today. Thus, control system add-ons have been realized which provide those missing degrees of freedom, usually referred to as Individual Blade Control (IBC). If such systems are added to a conventional control system, however, additional means of actuation have to be placed in the rotating frame, which require their own power and signal transfer links. Instead of implementing such additional separate hardware it might be worthwhile to consider an integrated actuation system designed to fulfill both the primary control as well as the IBC requirements.

This paper discusses the following three aspects: (a) what are the performance requirements of such an integrated control system which is able to fulfill both functions, (b) what actuation technology is best suited for this application and (c) how can the reliability required for a primary flight control system be realized with a minimum degree of redundancy and thus at a competitive system weight?

It will be shown that by using modern customized electro-mechanic actuators and by choosing an appropriate system architecture this type of integrated control system becomes feasible at a very competitive system weight. This paper presents the basic principles, discusses the multi-disciplinary investigations carried out, and describes the approach to demonstrate its feasibility through dedicated bench tests.

Notation

ACU		actuator control unit
AFCS		automatic flight control system
CG		center of gravity
DOC		direct operating cost
EMA		electro-mechanical actuator
FCC		flight control computer
IBC		individual blade control
IGBT		insolate gate bipolar transistor
InHuS		Innovative Helicopter Control
LCC		life cycle cost
M_L	Nm	blade pitch moment
MTOW		maximum take-off weight
N		number of blades
n		blade index (1…4 or 6)
P	kW	power
SGRC		signal generation and reconfiguration computer
VDHP		variable displacement hydraulic pump
β	deg	flap angle
ϑ	deg	blade pitch angle
$\Delta\vartheta_{IBC}=\Sigma A_n cos(n\psi-\varphi_n)$		pitch angle due to IBC
ζ	deg	lead-lag angle
φ_n	deg	n/rev IBC control phase angle
ψ	deg	rotor azimuth angle

1 Introduction and Motivation

The common primary control systems of helicopters comprise a chain of elements that extend from the pilot stick in the non-rotating frame to the blade control horn in the rotating frame. This chain consists of mechanical, hydraulic and increasingly electrical/electronic elements. The pilot control command is the mechanical input at one end and the required blade pitch motion about its feathering axis is the mechanical output at the other end of this chain. In between the control information is somehow conditioned (mixed, decoupled, etc.), commonly superimposed by signals from an AFCS or alike and then used to control the power flow that is required to overcome the aerodynamic and inertial blade pitch loads. Nowadays, the merged stream of information and power is transferred to the rotating frame by a

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mechanical assembly like the swashplate or control spider and then split and distributed to the separate blades. The kinematic properties of the latter devices prevent the individual control of each single blade for rotors of more than three blades. Instead they restrict the blade pitch motions to a certain subset of periodic time histories, which include the conventional (primary) collective and the 1/rev cyclic control.

Since it has been shown that a helicopter rotor in forward flight can greatly benefit from more complex pitch control schemes (which allow for higher harmonic motions for instance), control system add-ons have been realized which provide those missing degrees of freedom, usually referred to as Individual Blade Control (IBC). If such systems are added to the existing structure of a conventional control system, however, additional means of actuation have to be placed in the rotating frame (at the blade root or somewhere else at the blade), which require their own power and signal transfer links.

It is obvious that other morphologies are conceivable which combine the transfer of information and power for both primary and individual blade control. Hence, the basic concept discussed in this paper is to merge the information from the primary and the individual control tasks and likewise merge the required power but keep those two streams separate until they are locally combined at the respective blade. The comparison of the requirements for primary control and IBC shows that they yield very similar actuation speeds. Therefore, it seems to be feasible to use the same type of actuator for both functions. This leads to the idea of using one individual, however inherently redundant actuator per blade. Insofar as modern rotors tend to have more than three blades, the number of actuators will then exceed that of a conventional control system (having conceptually one collective and two cyclic actuators). On the other hand, no additional IBC actuators are required in this case and the ability to control the blades independently even with higher amplitudes introduces some new benefits:

- By applying different constant offsets to the individual blades true in-flight tracking can be realized.
- For rotors with an even number of blades the tip path plane can deliberately be split to reduce BVI noise or to suppress the subharmonic oscillating track (SHOT) phenomenon at high rotor loading conditions.
- Active control can be used to suppress blade instabilities like ground and air resonance. By applying artificial lag damping through closed loop control the installation of mechanical lag dampers is not required anymore, see [1].
- Without the kinematic restrictions of the swashplate it is possible to decouple the cyclic control inputs between the different blades. This allows to reconfigure the primary control function after local failures as partial blade damage or single blade actuator degradation.
- All other known and proven IBC benefits are inherently available also, including the reduction of vibration, noise, power required and/or control loads, comp. [6], [11], [13], [17].

To investigate this concept and to prove its feasibility the program "InHuS" (Innovative Hubschrauber Steuerung = Innovative Helicopter Control) has been established in which four partners from industry, research institutions, and academia contribute their complimentary expertises. The aim is to derive a competitive control and actuation system architecture and demonstrate its critical core elements on a test rig under simulated blade loads.

2 Control System Requirements

Within the above mentioned program a target helicopter has been chosen that provides technical data and geometrical dimensions used for all subsequent investigations and designs. This example helicopter largely corresponds to the CH-53G, because this particular aircraft has served as the testbed for previous IBC flight tests and therefore offers a rich data basis. For more details see Refs. [8], [10], [17]. In few cases the applicability of the developed concepts to a considerably smaller target helicopter similar to the Westland Lynx was checked.

2.1 Primary Control

The Key requirement for any blade pitch control system is the capability to rotate the blade about its feathering axis such that the control amplitudes required to trim and maneuver the aircraft are introduced under all possible aerodynamic loads. Moreover, the motions and corresponding loads from the superimposed IBC input have to be added in our case. The resulting requirement can be represented by a torque vs. rate envelope as shown in *Figure 1*.

The diagram shows (a) the area covered during all of the IBC flight tests (blue), (b) two sample trajectories for representative steady flight conditions (without IBC in red, with 6/rev IBC in green), and (c) the boundaries finally chosen for the actuator and system specification. The considerable margin is required to provide control authority for maximum forward speed, extreme CG conditions and maneuvers. The hyperbolic lines indicate the instantaneous power and give a first impression of the peak power demand. It can clearly be seen that the trajectories cross all four quadrants which means that during part of a main rotor revolution (i.e. 1/rev control cycle) power has to be invested to rotate the blade (Q1 and Q3) while during the rest of the time power can be drained from the blade then driven by the aerodynamic forces. Since the trajectories of the different blades of a rotor are cycled through in a phase shifted way and because it should be technically feasible to store energy at least over one rotor revolution an efficient control system should enable bidirectional power flow and cross supply. This would allow to interchange power between the different actuators and/or a storage element respectively, thereby reducing the over-all requirement for the power supply.

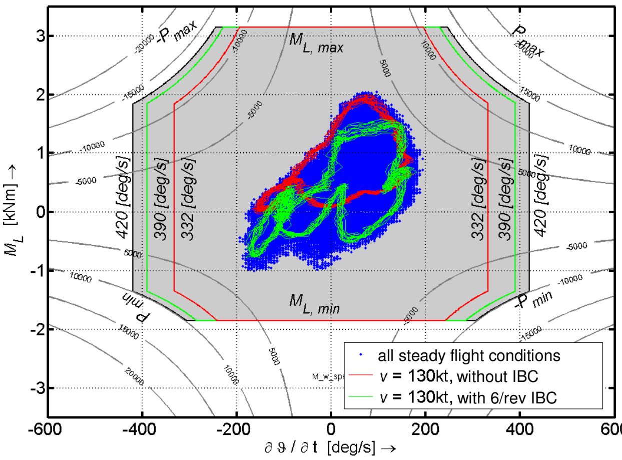

Figure 1: Blade pitch torque – rate envelope

It is worthwhile mentioning that the swashplate uses the same principle by collecting the pitch link loads and balancing as far as possible pulling and pushing forces. The residual portion is converted into a swashplate torque that has to be overcome by the powered rotor shaft. Since this is only a small fraction of the over-all (aerodynamic) rotor torque it is seldom mentioned at all and rarely considered when calculating the power required by the control system.

With the option of an electrical system in mind, which would have to be sized mainly for its adequate thermal behavior, it seemed adequate to define a representative duty cycle which combines power requirements from high amplitude maneuvers along with those from more moderate steady flight conditions. The chosen cycle is shown in *Figure 2*. The different levels correspond to the respective envelopes in *Figure 1*.

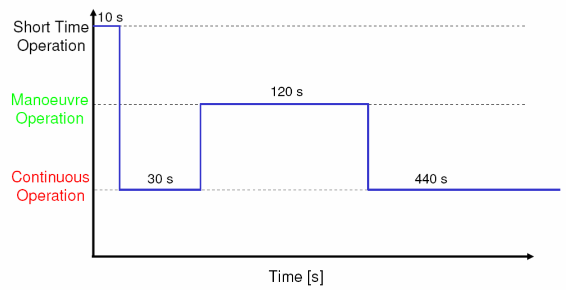

Figure 2: Duty cycle as used for the sizing of the actuation system

One key issue when choosing the appropriate actuator size is related to the wide range between the moderate power requirements (i.e. small cyclic amplitudes) in most areas of the flight envelope and the high power demand in few and short maneuvers at extreme loading conditions.

2.2 IBC

The pitch rates required for the IBC inputs are very similar to those for the primary control, because the higher frequencies used for IBC (usually in the range of 2/rev to N/rev) combine with much smaller amplitudes. However, it can be assumed that any flight critical IBC contributions are much smaller than those desired for performance improvement, vibration and/or noise reduction. Therefore, the actuator power requirements have been defined such that full IBC is available during all "normal" flight conditions but can partly be phased out in the edges of the flight envelope where the primary control requirements become reach extreme values.

2.3 Safety and Reliability Requirements

It is obvious that a complete new control system architecture also rises many new questions with respect to the formal flightworthiness. Whereas the military requirements have traditionally been less stringent the civil limit for a primary control system has been established at a rate of less than 10^{-9} catastrophic failures per flight hour (e.g. JAR part 29). It seems to be questionable whether this low rate could ever be proven by sound analysis for today's conventional control systems. Based on positive past experience, however, the compliance is usually assumed without any further proof if the principle lay-out follows the classical swashplate-type architecture with duplex hydraulic boosters. Obviously, for an InHuS-type control system the certification authorities will demand a stringent proof of compliance with respect to that failure rate. Therefore, all options investigated within this project were continuously checked for their eventual compatibility with this requirement. More details will be presented in chapter 4.

2.4 Operational Aspects

Beside the obvious "hard" requirements mandatory for all control systems as discussed before, each operator imposes several other requirements, which are more or less directly linked to the DOC or LCC. For instance, power consumption and weight of the complete control system have some impact on these costs. As benchmark for this project the weight of an existing conventional hydraulic control system plus the weight for an optimized IBC retrofit system has been defined. Beside those key parameters, especially the contribution from the required maintenance effort is of predominant influence.

Further aspects with rather indirect influence are handling qualities and mission readiness/reliability. Integrated control systems as presented here are based on a fly-by-wire architecture. This provides the necessary freedom to implement state-of-the-art AFCS hard- and software to optimize the handling qualities through active control. On the other hand, the demand for maximum mission readiness may suggest higher degrees of redundancy than required for certification purposes alone, because the ability to limit the degradation even after a second failure helps to improve mission reliability as long as the handling qualities can be kept on an acceptable level.

3 Control System Layout

Two interrelated aspects dominate the over-all system architecture: (1) the availability of suitable actuators, and (2) their geometric/mechanical integration into the rotor(hub). Both aspects have been addressed simultaneously during the early stage of the study and are described in the following sections.

3.1 Available Actuator Technologies

A wide rage of actuation technologies had been thoroughly reviewed in the search for suitable concepts. This comprised traditional as well as modern so-called "smart" concepts. Many modern actuation principles (e.g. piezo-electric, magneto-restrictive), however, were found to be not scalable to the required performance range.

Traditionally, flight control systems have been based on servo-hydraulic actuators due to their high power-to-weight ratio. Therefore, the first of the competing pre-designs was based on conventional valve-controlled hydraulic actuators fed by a constant pressure pump, see *Figure 3*. A refined variation of this lay-out included load sensing and variable system pressure in order to reduce the power consumption during low load flight conditions.

If there is no mechanical element like the swashplate which directly couples the blade control energy flow as mentioned above, other means of energy recovery are vital for a power efficient control system. Thus, a third variant had been derived which combines a central displacement control pump, that is directly connected to the rotor shaft and provides the major part of the 1/rev motion (see *Figure 4*) with individual valve control for the higher harmonic portions. This principle allows partly regenerative operation, because the oil mass flows from/to the different blade actuators are reacted by a single mechanical element and are therefore interconnected similar to the pitch rods via the swashplate.

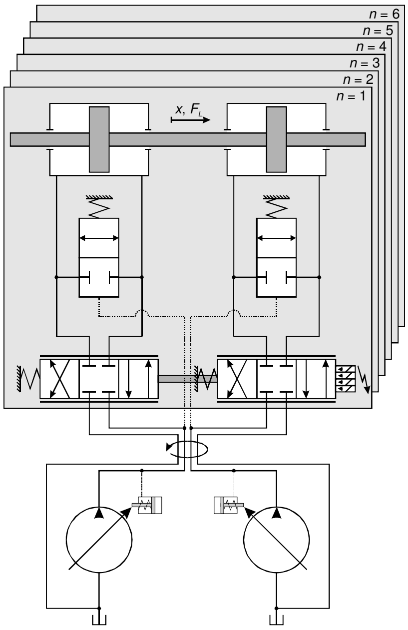

Figure 3: Schematic lay-out of valve controlled hydraulic system

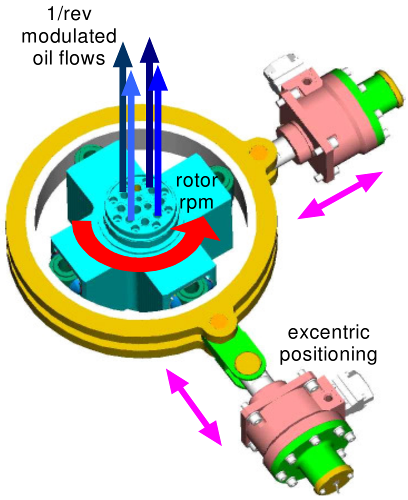

Figure 4: 1/rev displacement pump ("hydraulic swashplate")

Then, a fourth variant has been considered which uses individual variable displacement hydraulic pumps (VDHP), which directly control the oil mass flow into each separate actuator cylinder, see *Figure 5*. This principle allows full regenerative operation because the different pumps, which are mechanically interconnected on their input shaft side, can temporarily serve as hydraulic motors and recover the power generated by the aerodynamically loaded blades. The VDHPs are controlled by a secondary much smaller valve controlled hydraulic system that moves the swash plates inside that pumps according to the required blade pitch control function. Although such VDHPs, as sketched in the principle drawing of *Figure 6*, are in common use, this application would require a continuous operation at a comparatively high frequency.

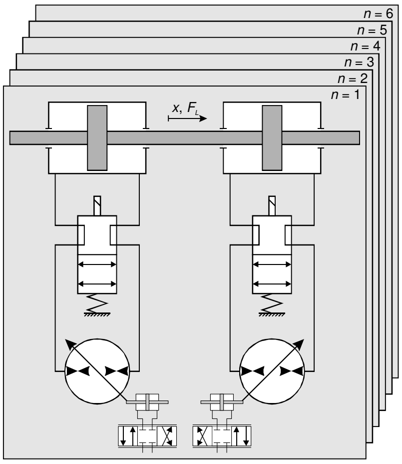

Figure 5: Schematic lay-out of hydraulic displacement control system

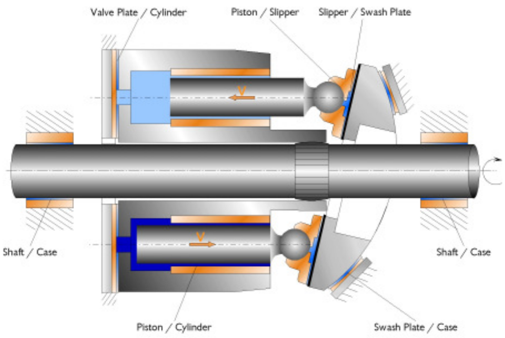

Figure 6: Operating principle of variable displacement hydraulic pump (VDHP)

With regard to the concurrent integration studies it was felt that elegant solutions might potentially result from the application of rotatory actuators. Therefore, customized hydraulic vane motors have been considered in addition to the conventional cylindrical actuators whose translatory motions need to be mechanically converted into the blade rotation. Figure 7 shows the preliminary design of a suitable redundant vane motor, which was used as an option in some of the sample system lay-outs.

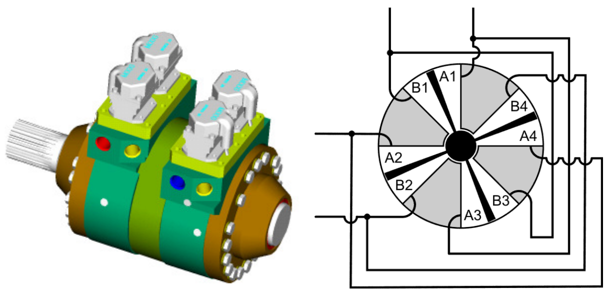

Figure 7: Hydraulic vane motor for efficient blade root integration

From the early stage of the project electrical actuators were considered an attractive alternative to hydraulic solutions. From the variety of possible lay-outs the synchronous drive (brushless DC motor) was found to be the only feasible candidate. The first attempt was to design a direct drive (torque) motor whose output speed is directly compatible to the required blade pitch rates. This variant had the potential to keep the number of mechanical elements to a minimum. However, calculations based on integration studies as described in the next section have clearly shown that a feasible electromagnetic lay-out would lead to an unacceptable actuator weight. It was concluded that an efficient system would require a gearbox with a reduction gear ratio in the order of i = 30 … 50. The basic elements of this concept are shown in *Figure 8*, for more details see [14].

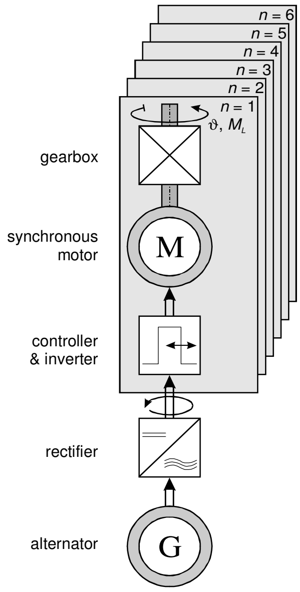

Figure 8: Schematic lay-out of electrical control system

A design study was then launched to search for an integrated electro-mechanical actuator (EMA), which comprises the electrical drive as well as the mechanical reduction stages. With regard to an efficient actuator integration it was expected that the installation of one single unit per blade would emerge as the preferable solution This concept implied that all redundancy provisions had to be integrated within one EMA. This would be somehow similar to the integration of one internally redundant primary booster unit per control degree of freedom within a conventional control system. Thus, in the pursued embodiment the electrical as well as the mechanical part each hat to consist of multiple components arranged to provide the required independent and jam-free operation (decoupled sets of coils, multiple load paths, etc.).

The general decision between a hydraulic vs. an electric solution was based on a rating matrix which compared the above mentioned five hydraulic and one electrical designs. Although the calculated metric showed clear advantages over the conventional control system used as benchmark, the distinction between the different InHuS solutions was only marginal. Thus, it was decided to pursue the electrical version based on the expectation that this concept provides the largest potential with respect to improvements in the electro-mechanical actuator (EMA) technology. Moreover it was felt that the benefits from completely removing the hydraulics from the rotorcraft had been underestimated in the formal assessment process.

Currently, Westland is developing an electrical primary control system for the EH-101, see [15]. In that case the conventional boosters below the swashplate will be replaced by suitable EMAs which basically emulate the performance of the hydraulic ones. Likewise, in the fixed wing community the more-electric aircraft approach is increasingly pursued. One of the main arguments for the reduced maintenance cost is the greatly simplified infrastructure required on the ground. Some of the typical claims are listed hereafter, compare also [20].

- Reduction in flight control system weight of approx. 25%.
- Reduction in maintenance cost of approx. 42%.
- Reduction in Mean Time To Repair (MTTR) of approx. 50%.
- Increase in aircraft availability by 15%.
- Enhanced ballistic tolerance.

3.2 Over-All Architecture

At the beginning different principle system lay-outs had been studied. However, all considered designs showed that any interconnecting elements which would offer the possibility to separate certain control tasks (e.g. collective placed at the non-rotating frame, cyclic at the blade root and IBC via a trailing edge flaps) would unnecessarily increase complexity. Therefore it was decided to assign one customized actuator individually to each blade which then had to cover all control tasks.

The following figures show some examples of an integration study. Different actuator locations, orientations and kinematic linkages to the blades were considered and checked for their general feasibility. Initially, as mentioned above, it was also attempted to use direct drive actuators. To reach an acceptable torque capacity, however, their active elements then had to be placed on a large diameter. *Figure 9* shows a corresponding embodiment with ring-shaped motors mounted around the rotor shaft. But with increasing number of blades the kinematics to connect these actuators to the blade pitch horns become more and more cumbersome. As soon as one accepts the existence of a reduction stage within the actuator, though, the integration options are manifold and the adaptation to most of the commonly used rotor hub types becomes possible.

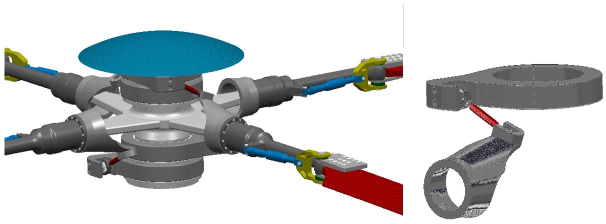

Figure 9: Electrical direct drive ring motors integrated into Lynx-type hingeless rotor

Figure 10 and *Figure 11* show two examples of possible variants. In the first case the EMAs are vertically oriented parallel to the rotor shaft and their rotatory motion is converted into the blade pitch motion by a scissor-like linkage. In the second variant, the actuators are integrated into the blade arm moving with the blade and transfering their torque to the rotor hub structure via a moment strut.

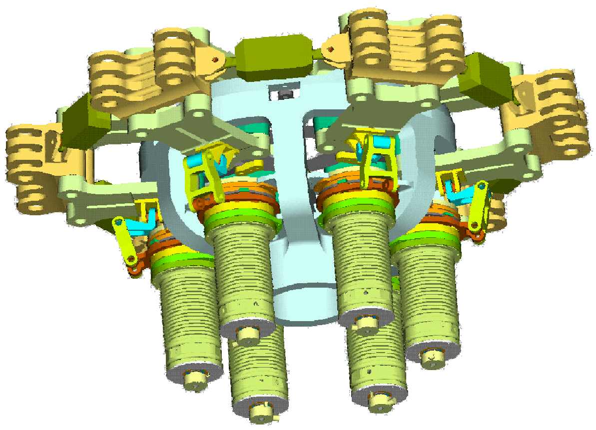

Figure 10: Vertical integration of EMAs into state-of-the-art articulated rotor

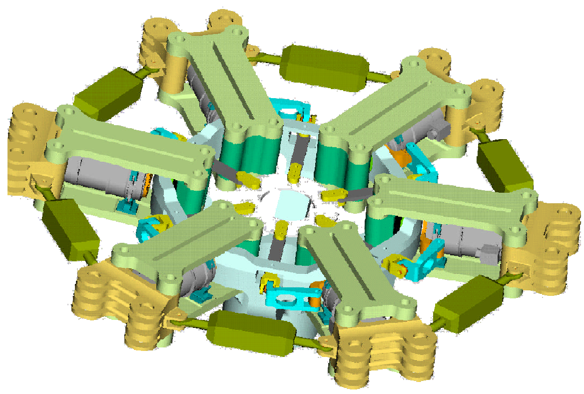

Figure 11: Co-axial integration of EMAs into state-of-the-art articulated rotor

As result of these integration studies and the concurrent system weight estimations the latter variant with the co-axial integration of blade-individual EMAs was chosen for further refinement.

3.3 Chosen Variant

After it had been verified that the EMA concept was feasible, the specifications could likely be fulfilled, and a redundancy architecture was identified which had the potential to achieve the reliability required for civil certification, the scope was widened to the over-all system lay-out. *Figure 12* shows the current design which will subsequently be described in more detail. The rotor features a modern titanium hub with elastomeric flap-lag-pitch bearings. In this embodiment the EMAs are directly placed inside the force-carrying tubes of the blade arms largely protected from environmental influences. The thermal conductivity of the blade arms allow to use forced convection (due to the rotor rotation) for the necessary heat transfer. A scissor unit counteracts the blade pitch moments. Also located within the hub assembly are the electronic power inverters and the actuator control electronics.

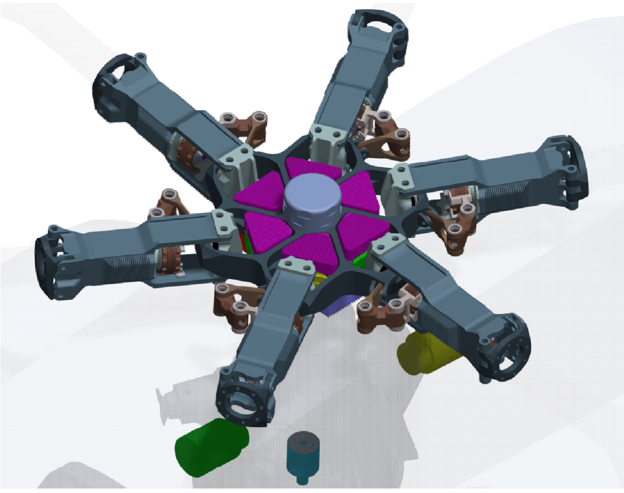

Figure 12: System integration in the rotating frame

The actuator units are the result of several iteration steps that aimed at an optimum choice of the speed/torque capacity of the motor to be mated with a suitable transmission stage, see *Figure 13*.

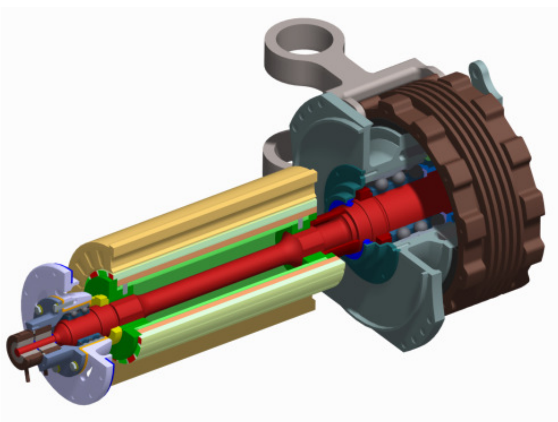

Figure 13: Electro-mechanical actuator layout

The synchronous motor features a passive rotor with high performance permanent magnets and has been designed for a minimum moment of inertia. The special arrangement and connection of the stator coils realize the required redundancy and provide fault-tolerant operation which covers up to six local electrical failures without catastrophic consequence. The attached excentric gearbox (reduction ratio i=29) has a high torsional stiffness, no mechanical backlash, low hysteresis losses and a high shock load resistance. It uses multiple load paths and was likewise designed for a minimum moment of inertia. It also allows four-quadrant operation which is used for power recovery as mentioned above.

The system is completed by the power supply elements as sketched in *Figure 14*. The energy for the system is generated by 3+1 three-phase 115V 400Hz synchronous alternators, two of them mounted at the main and two at the auxiliary gearbox (one of them as switchable spare according to JAR part 29). After rectification to 270V DC the current is transferred to the rotating frame via a set of redundant rollring transducers. The two dual-duplex flight control computers are widely comparable to current fly-by-wire systems, although they have to cover some additional tasks, compare section 4.1.

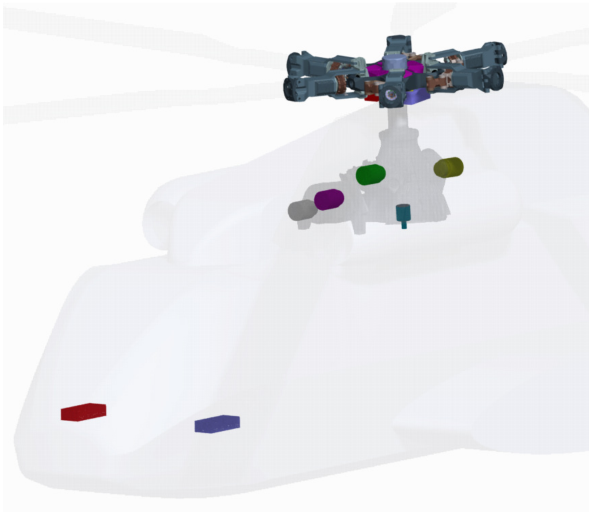

Figure 14: Over-all system integration with supply elements in the non-rotating frame

3.4 Estimated Weight and Power Figures

Based on the presented architecture an over-all system weight of approx. 790kg has been estimated. This falls within 3% of the benchmark weight for a conventional control system including a retrofit IBC installation. The weight break-down to subsystem level, *Figure 15*, indicates the clear dominance of the actuation components. This is where weight saving measures would be most useful.

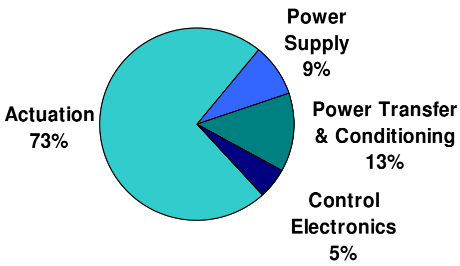

Figure 15: Weight break-down for the proposed system architecture

The power consumption for the complete system is expected to reach values of 16…21kW if the operating condition drives the blade rates and loads to the respective limits in *Figure 1*. This is the same order of magnitude as for a conventional hydraulic IBC system alone. Thus, even at this early stage of that novel technology no weight or power penalty is compromising the expected operational benefits.

4 Reliability Aspects

This chapter describes how the safety and reliability requirements have influenced the system architecture of the entire electro-mechanical control system. The key advantage of the proposed solution results from the internal design of the InHuS EMAs which allows their connection to more than two power supply systems in a straightforward manner. Nonetheless, the number of components is kept smaller than with usual duplex hydraulic systems using tandem cylinders. This leads to a system which can tolerate more simultaneous failures without catastrophic consequences than current designs.

The following principles characterize the safety properties achievable by an InHuS-type control system:

- The first electrical failure (loss of a complete power lane, short circuit of a motor phase, etc.) does not cause any degradation of the control performance.
- The worst second failure (e.g. complete loss of a second power lane) leads to a "mild" degradation, which means that the flight can safely be continued within a restricted envelope (130kts instead 160kts, reduced maneuverability, roughly corresponding to the shaded area in *Figure 1*).
- Most power lane failures can be compensated through activation of the fourth alternator.
- At each EMA a minimum of three and up to five local failures can be tolerated before any degradation is caused.
- Even a considerable degradation at one EMA can be tolerated when counteracted by appropriate reconfiguration measures as will be shown in section 4.2.

4.1 Redundancy Architecture

In the search for a suitable system architecture, different alternative lay-outs have been assessed with respect to their conformity to the mandatory and the desirable requirements on one hand and their complexity on the other hand. *Figure 16* gives an overview of the chosen general system architecture, which covers both the components located in the non-rotating frame (left), and those in the rotating frame (right).

Three power lanes which separately supply one third of each actuator can be recognized. The mandatory fourth alternator can be switched to any one of these lanes. Each of three redundant sections per actuator is driven by its own Actuator Control Unit (ACU) which provides current, rate, and position control. The pitch angle sensors are quadruplex and the two Flight Control Computers (FCC) dual-duplex redundant.

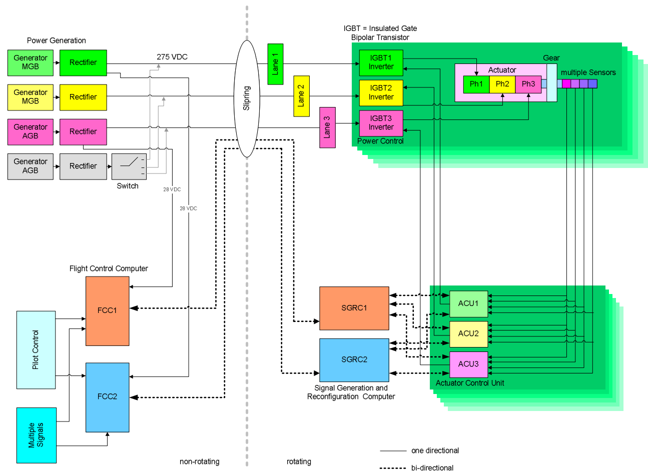

Figure 16: Overall system architecture showing the respective redundancy levels (only one example blade depicted in the rotating frame)

The internal structure of the ACUs is shown in *Figure 17*. One command and one monitor channel (dual-simplex) each use the complete set of position and rate sensor signals to assure failure recognition/tolerance and prevent force fighting. In contrast, only the motor currents of the corresponding power lane are processed.

ACUs and FCCs are linked via two Signal Generation and Reconfiguration Computers (SGRC), see *Figure 18*. These dual-duplex units convert the conventional rotor control inputs into the blade-individual signals required in the rotating frame. Moreover, they add the appropriate IBC signals and allow to decouple the usually identical (only phase shifted) blade pitch time histories if reconfiguration shall be used to counterbalance local actuator degradation as discussed in section 4.2.

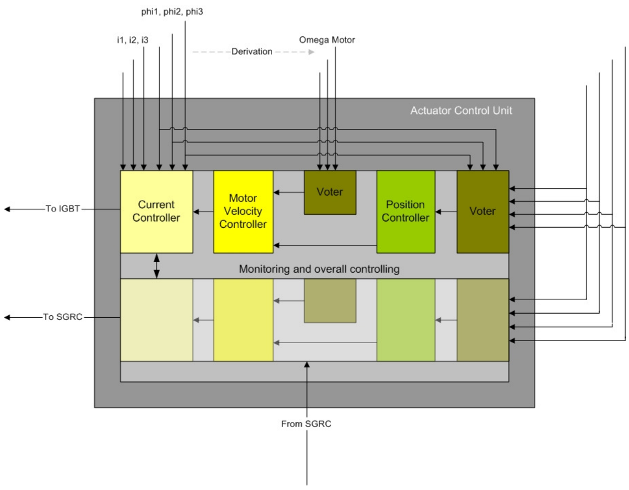

Figure 17: Principle layout of actuator control units (ACUs)

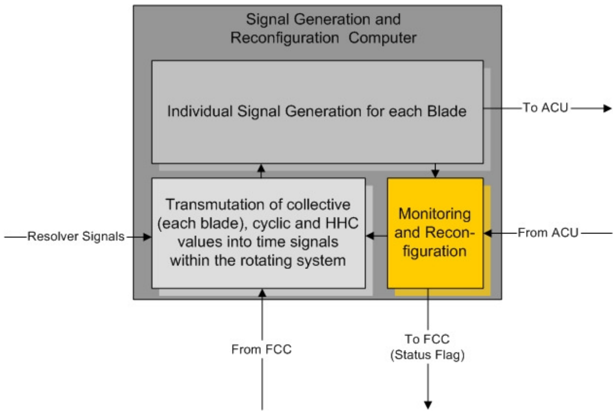

Figure 18: Principle layout of signal generation and reconfiguration computer (SGRC); monitor channels not shown

The FCC is rather conventional and similar to a typical fly-by-wire system except that it does not only compute the conventional control inputs but also the amplitude and phase settings of the IBC/HHC inputs.

To check whether the chosen redundancy architecture satisfies the crucial certification requirements, different methods have been used to estimate and benchmark the over-all reliability figures. In parallel existing architectures have been assessed for comparison with the InHuS solution. On a component level (alternator, IGBT power inverter, gearbox) fault-tree analysis and reliability block diagram methods have been used to identify the critical paths. Due to the customized internal architecture of the EMA, special attention was paid to the determination of the catastrophic failure rate in this area, comp. *Figure 19*. The investigations have confirmed that the chosen EMA architecture should be able to provide the required reliability.

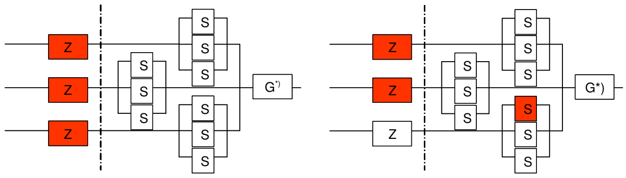

Figure 19: Determination of catastrophic failure probability

For the EMA one failure mode is defined by the exceedance of certain critical temperatures at the permanent magnets and/or the coils. The ability of take over load from a faulty phase by the unaffected phases is primarily restricted by the thermal behavior after such failure. Accordingly, thermal simulations have been used to validate the residual EMA performance after multiple failures. *Figure 20* shows one of the applied models and *Figure 21* gives an example of the transient temperature curves at different locations within the EMA. In this particular case (simultaneous loss of three electrical phases) the maximum temperatures are reached at the coils. But with approx. 100degC (the final EMA version now reaches 120degC), they stay well below the limits of about 200degC for the insulation and magnet materials.

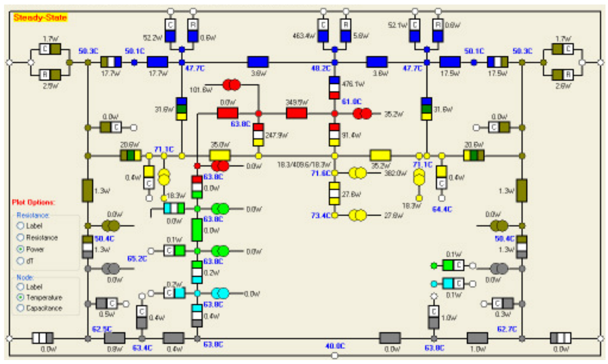

Figure 20: Thermal simulation of synchronous drive with MotorCAD® software

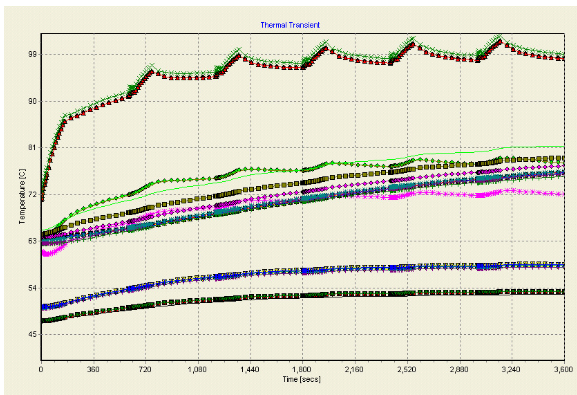

Figure 21: Transient temperature curves at different locations within the EMA after loss of three electrical phases (duty-cycle according to Figure 2)

4.2 <u>Degradation and Reconfiguration</u>

An interesting aspect concerns the question whether the freedom to control every blade independently can be used to counterbalance certain actuator malfunctions or degradations through control reconfiguration. It was felt that this inherent redundancy of a multi-bladed rotor should be utilized to improve safety or even reduce the redundancy requirements on the actuator level. Therefore, various simulations have been carried out to investigate, how the control schemes of the unaffected blades could be altered in order to counteract partly reduced actuator performance at a single blade.

For these generic investigations a basic flight mechanic simulation program has been combined with a Matlab optimization code. The math model describes the 6 degrees of freedom body motion which is dynamically coupled to N rigid blades each having independent flap, lag and pitch degrees of freedom. Pitch input time histories can be imposed separately for each single blade. The aerodynamic model is based on strip theory, look-up tables for the blade loads and 1st order dynamic inflow. Although this model certainly lacks any fidelity for the higher harmonic blade dynamics, it was assumed that the principle countermeasures to suppress the 1/rev unbalance caused by a single blade failure would be represented sufficiently well.

The very first calculations had shown that a rotor with a single blade run in one of the hard stops would not be recoverable in wide areas of the flight envelope. Moreover, failures that would enable completely free feathering motions within the usual blade pitch range were not considered treatable by reconfiguration. *Figure 22* gives an impression of different conceivable degradation scenarios. Therefore, certain failure scenarios have been defined which were mild enough to be counteracted with the remaining blades but on the other hand corresponded to real degradation options for the actuators.

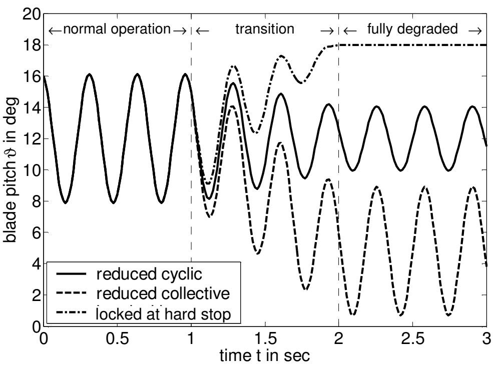

Figure 22: Some fictitious degradation scenarios

For example, if one power lane is lost locally at a single actuator it would still be possible to operate the affected blade at more than half its nominal amplitude. This is based on the fact that the reduction of the control amplitude (and pitch rate!) even though it not always reduces the aerodynamic loads, in any case reduces the inertial loads (especially those generated from the EMA itself), which considerably contribute to the over-all load the EMA has to carry. Thus, in such situations the remaining functionality of the affected actuator still allows to control the blade with respect to both collective and cyclic, however at a reduced cyclic amplitude.

The variety of control input variations did not allow to systematically check all possible options. The definite answer would result from a numerical optimization which allows to change collective, cyclic and arbitrary higher harmonic components independently for all of the non-affected blades. This attempt would have led to extreme computation times. Therefore, the search was restricted to those concepts which were identified as most successful during the initial calculations. The options investigated all followed the idea to restore the symmetry of the rotor by combining certain blades to groups which then where each controlled with the same but modified control scheme. *Figure 23* shows two obvious possibilities for a 6-bladed rotor.

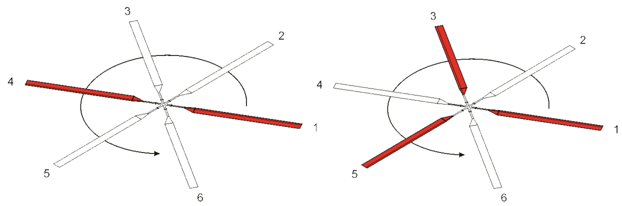

Figure 23: Grouping of blades to restore symmetry for successful reconfiguration

The presented example shows the successful reconfiguration after the loss of 50% cyclic amplitude at blade 1. Two 3-bladed rotors were formed which each got its own set of primary controls. Since blade 1 lost 50% of cyclic per definition, blades 3 and 5 were also operated at this reduced level. Obviously, the lost control moment would then have to be compensated by the blade group 2 – 4 – 6 somehow. The optimization was based on a cost function which contained body translatory and rotatory accelerations. Moreover, the whole aircraft had to maintain the initially trimmed steady flight condition and the blade flap and lead-lag angles had to stay by a sufficient margin within their mechanical limits. The presented example corresponds to level flight at max. forward speed, MTOW, and nominal CG position for the CH-53G-like example helicopter.

The following two figures compare the blade motions (*Figure 24*) and body accelerations (*Figure 25*) for the assumed 50% degradation of actuator 1 (left) with the situation after the reconfiguration (right). Without the appropriate reaction the blade motion of blade 1 rises considerably.

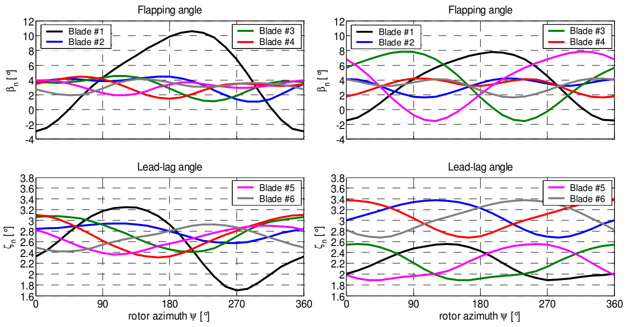

Figure 24: Blade motions (top: flapping, bottom: lead-lag) of all 6 blades without (left) and with reconfiguration (right)

Even more severely, the vertical and roll accelerations of the aircraft reach more than uncomfortable values. The optimized reconfiguration largely suppresses these negative effects. One can clearly recognize the restored symmetry formed by the two distinct blade groups. Although the steady flap motion clearly increases due to the fact that the blade group 1 – 3 – 5 is now restricted in cyclic, no critical blade motions are excited anymore. More important however, the excessive body accelerations are completely attenuated almost down to the non-degraded reference case (not shown here).

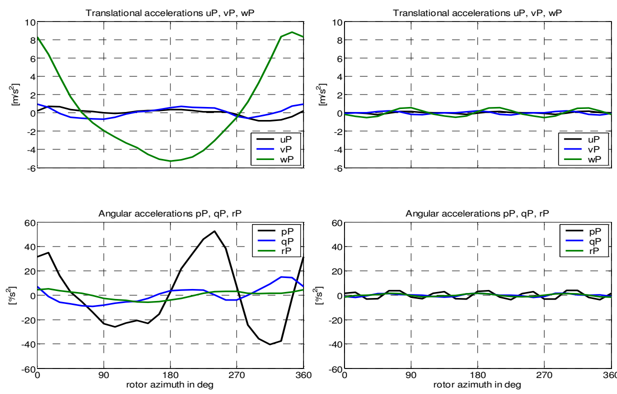

Figure 25: Body accelerations (top: translatory, bottom: rotatory) without (left) and with reconfiguration (right)

The required changes in the control inputs can be visualized as 3D polar plot as shown in *Figure 26*. In the reference case (left) all 6 blades follow the same collective and cyclic control settings. As reaction to the assumed actuator failure, blade 1 (red) would be restricted to only half of the trimmed cyclic input whereas the remaining blades would continue to receive the old trim settings (middle). According to the optimization however, the two blade groups (blue: 1 – 3 – 5 and green: 2 – 4 - 6) should receive different collective <u>and</u> cyclic inputs (right). The required differences amount to approx. 2deg in the collective input, 5deg in the cyclic amplitude, and 22deg in the cyclic phase angle.

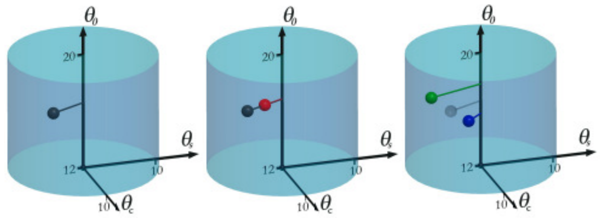

Figure 26: Visualization of control settings for the reference case (left), degradation at one actuator (middle), and the optimized reconfiguration (right)

The results presented above have shown that considerable degradations in the actuator performance can be counteracted by appropriate new trim settings. Obviously, for the real application of this concept one has to assure that the reconfigured control schemes are available and applied fast enough to prevent unacceptable transient accelerations and motions to build up. Therefore, further simulations have been carried out to check how the body accelerations depend upon the reaction time. *Figure 27* compares the time histories for two different reaction times. In the first case, a delay of 1.4s (4 rotor revolutions) was introduced before the reconfiguration inputs were applied (left). In the second simulation only 0.1s was spend before it was switched to the new control settings (right). It is quite obvious that a fast reaction improves the transient behavior. This indicates that in the event of a failure one must limit the resulting degradation to a known scenario and assure that the appropriate pre-calculated control settings are available immediately.

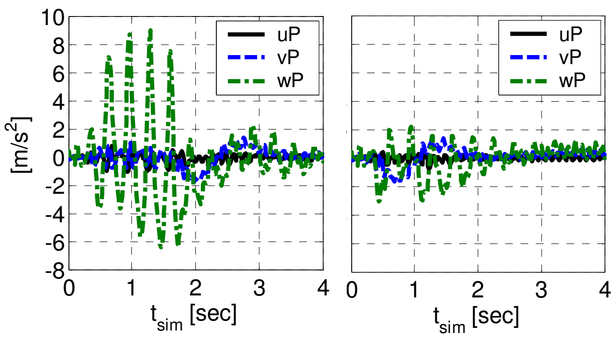

Figure 27: Transient body accelerations for different reconfiguration reaction times (left: 1.4s, right: 0.1s)

5 Conclusions and Outlook

The combined work of the InHuS project indicates that an all-electric swashplate-less control system should be feasible today. The implementation of actuators individually at each blade will provide additional degrees of freedom and new functionalities which today still require separate actuation hardware. The key features of this type of control system comprise:

- Full authority individual blade root actuation separately for each blade
- Primary control system fully integrated into rotor head
- Real Fly-by-Wire/Fly-by-Light capability
- No mechanical mixing units, no swashplate, etc.
- Easy implementation of IBC; actuators can simultaneously be used to introduce primary control and IBC

The system architecture which was found to be most suitable uses rotatory EMAs based on a redundant synchronous electrical motor and a multiple load paths gearbox. These actuators are mounted coaxial to the pitch axis in the blade arm. From the work done so far it is concluded that all major challenges are met by the presented approach. The most crucial questions have been addressed as follows:

- Performance parameters compared to conventional control systems architectures?
  In relation to the total weight of a conventional primary control system plus approx. 1% MTOW for vibration reduction means (and/or other desirable IBC functions), the InHuS architecture with today's electric technology is competitive
  With respect to the power consumption the regenerative operation with power cross supply imitating the mechanical swashplate reaches a similar efficiency as a conventional system; compared to a separate hydraulic IBC system the proposed solution is clearly more power efficient
- Proof of reliability for an electrical actuation system in the rotating frame (10^{-9}/FH)?
  Custom designed electric motors provide multiple redundancy based on smart coil separation; critical mechanical components rely on multiple load path concepts
- Multiple blades provide inherent redundancy; can this be used for reconfiguration?
  Considerable performance degradation (beyond 50% reduction of authority) of a single blade actuator can be tolerated when counteracted by reconfigured control schemes for the unaffected blades

The experimental proof-of-concept will be based on the results of the scheduled bench tests. Three test steps are currently under preparation. The first element is the isolated operation of the electric motor. Simple loads will be simulated and the basic motor position control will be implemented. Then the motor will be mated with the gearbox and this EMA unit will undergo extensive functional and load testing on a customized test rig, shown in *Figure 28*.

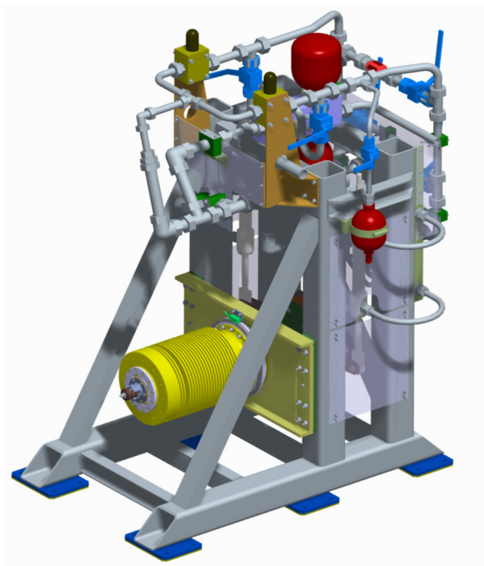

Figure 28: EMA test rig for functional and load testing

The kinematics are closely resembling the intended blade root integration. This rig will allow to precisely simulate the external load time histories over subsequent rotor cycles. It is intended to validate the mechanical as well as the EMA control system hard- and software. A considerable portion of the work will concentrate on failure simulation, recognition, and isolation.

The third contribution will come from a rotating component test stand as partly shown in *Figure 29*, which will be used to validate the operation of the EMA gearbox under centrifugal loads. The gearbox will be loaded mechanically to approximate its working condition in the rotorcraft. The main focus will be placed on lubrication and durability aspects.

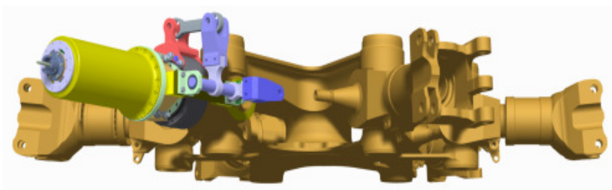

Figure 29: Rotor hub dummy of rotating component test stand for EMA gearbox qualification

As known, other concepts beside blade root control are pursued in the field of innovative rotor control systems. The utilization of trailing edge flaps for certain active rotor control applications has recently reached flight test demonstration stage as presented in [18]. Although these systems do not seem to be feasible for primary control of modern efficient rotors (compare [19]), they follow a similar approach as the InHuS concept. Electrical power and digital signals are transferred to the rotating frame in order to drive blade individual actuators. It might be worthwhile to investigate whether some combination of both systems could use their respective advantages in a complementary way and thereby yield some convincing synergies.

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