

DESIGN, IMPLEMENTATION, AND FLIGHT TESTING OF A STABILITY AUGMENTATION SYSTEM FOR COAXIAL ULTRALIGHT ROTORCRAFT

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ABSTRACT

This paper describes the first ever flight demonstration of a stability augmentation system using electromechanical inline actuators in a rotorcraft with a reversible control train, i.e. without a hydraulic system. A coaxial ultralight helicopter equipped with the stability augmentation system is used as a flight demonstrator. The paper covers the design and architecture of the system, technical challenges, the algorithms including the control laws, safety features, flight testing, and flight test results. The flight test campaign includes scaled down benchmark maneuvers from literature. The flight demonstration proves that the proposed SAS can be integrated safely into small rotorcraft despite the space and weight limits and the actuation force requirements. Measurement data and pilot feedback indicate that the stabilization is effective and workload is reduced. To emphasize the scientific contribution of the flight demonstration, the paper includes a brief discussion of the dynamics of reversible control trains and the associated challenges and open questions.

NOTATION

Acronyms

CAN	Controller area network
DRP	Disturbance rejection peak
EASY	Electromechanically actuated SAS
EMA	Electromechanical actuator
FCC	Flight control computer
FHA	Functional hazard assessment
FTA	Fault tree analysis
GVE	Good visual environment
HQ	Handling Qualities
IIR	Infinite impulse response
IMU	Inertial measurement unit

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INS	Inertial navigation system
LED	Light emitting diode
PLC	Programmable logic controller
PTC	Preliminary type certificate
SAS	Stability augmentation system
SDO	Service data object
SISO	Single input single output
SSA	System safety assessment
ULH	Ultralight helicopter

Latin symbols

g	$[\text{m/s}^2]$	Gravitational constant
k	$[1/\text{mm}]$	Allocation gain
k_I	$[1]$	Integral gain
k_P	$[s]$	Proportional gain
y		Output
f	$[\text{N}]$	Force
p	$[\text{rad/s}]$	Roll rate
q	$[\text{rad/s}]$	Pitch rate
s_{Act}	$[\text{mm}]$	Actuator displacement
t	$[\text{Nm}]$	Torque

t	[s]	Time
u	[m/s]	Longitudinal velocity
v	[m/s]	Lateral velocity
v_{Act}	[mm/s]	Actuator velocity

Greek symbols

ω_{Act}	[rad/s]	Actuator cutoff frequency
ω_e	[rad/s]	Damped natural frequency
φ	[rad]	Phase angle
ζ	[rad/s]	Damping
$d\alpha$	[%]	Lateral cyclic input
$d\beta$	[%]	Longitudinal cyclic input
$d\theta$	[%]	Collective input
ϕ	[rad]	Roll angle
θ	[rad]	Pitch angle

Superscripts and subscripts

5 %	5 % Quantile
95 %	95 % Quantile
Act	Actuator
BS	Ball screw
BW	Bandwidth
c	Commanded value
H	Highpass
L	Lowpass
Lat	Lateral motion
Lim	Limit
Lon	Longitudinal motion
m	Motor
P	Pilot
r	Requirement / required value
S	Spring
TR	Threaded rod

INTRODUCTION

Ultralight helicopters (ULH) are an affordable alternative to CS-27 rotorcraft in Germany. They are lightweight, mechanically simple, and cheaper to maintain than their larger counterparts, making them attractive especially for hobby pilots. However, like most rotorcraft, they exhibit a pitch and roll instability in hover, posing a challenge for pilots with little to no experience. In larger rotorcraft, an automatic flight control system (AFCS), which typically includes a stability augmentation system (SAS), is used to stabilize the phugoid and dutch roll in hover. However, two things used to prevent SAS development for ULH



Figure 1: Flight demonstrator: edm aerotec CoAX 600 equipped with the SAS EASY

until now: (1) Unlike larger helicopters, ULH have no hydraulic force amplification. Therefore, powerful and thus possibly heavy actuators are needed, and (2) the use of actuators in ULH is currently prohibited by German law.

In 2020, the authors successfully demonstrated that a stability augmentation system using electromechanical actuators (EMAs) mounted as inline-actuators in the control train is technically feasible. The demonstration included a coaxial ultralight helicopter mounted in a metal frame (Ref. 1). During these tests, commercial off-the-shelf (COTS) hardware from automation, UAV electronics, and inhouse developed electric interfaces were used. Due to the metal frame tying the helicopter to the ground, the system could be tested without any certification or formal proof of safety. However, the flight characteristics are altered substantially if a rotorcraft is tethered to a metal frame (Ref. 2). Especially the phugoid and dutch roll modes and the couplings between rotational and translational motion are affected by the additional inertia of the tethering frame.

Therefore, a proper flight demonstration was needed to prove the certifiability, feasibility, and effectiveness of the proposed inline architecture in a real rotorcraft. To achieve that, an improved version of the SAS was designed and implemented. The flight demonstrator is based on the edm aerotec CoAX 600 coaxial ultralight helicopter shown in Figure 1. To ensure safety, all SAS components were replaced by airworthy hardware and an independent non-complex locking mechanism was developed and implemented. In cooperation with national aviation authorities, the way was paved for a flight test with the SAS-equipped flight demonstrator. This paper contains the design of the airworthy SAS, the certification, and a description of the flight demonstration itself.

To the best knowledge of the authors, this is the first time electromechanical inline actuators were used in the primary control train of a helicopter with no hydraulic system. The scientific contributions are the answers to the

following questions:

1. Can a stability augmentation system be integrated into an ultralight rotorcraft despite the limited space, mass, and energy budgets?
2. Do inline EMAs stabilize the helicopter effectively despite their large actuation forces and the inherent feedback forces of reversible control trains?
3. Can safety be ensured and proven in terms of certification?

To illustrate why these answers are a scientific contribution and how the aforementioned goals are achieved, the paper is structured as follows: In the next chapter, the dynamics of reversible rotorcraft control trains are discussed briefly and compared to those of control trains with a hydraulic actuator to highlight the scientific contribution of the flight demonstration. In the subsequent chapters, the functional and safety-relevant requirements are listed, the synthesis of actuator requirements from flight test data is shown, and the design and testing of the SAS actuators is illustrated. In the remaining chapters, the system architecture, the SAS algorithms and control laws, and the safety concept for the flight demonstration are discussed, followed by a description of the flight demonstration itself, a presentation of flight test results, and a brief discussion of flight test data and pilot feedback.

DYNAMICS OF REVERSIBLE ROTORCRAFT CONTROL SYSTEMS

Most modern helicopters are equipped with an AFCS in combination with a hydraulic system like depicted in a simplified manner in [Figure 2a](#). In these configurations, the control forces f_{Hyd} required to counteract the blade feathering forces f_{R1} and f_{R2} are provided by hydraulic actuators. The electromechanic AFCS-actuators are typically mounted inline between the cyclic stick and the hydraulic actuators, where the required forces are comparatively small. The resulting AFCS actuator feedback forces f_{Act} are usually negligible because they are masked by centering springs at the cyclic stick (f_s). The pilot therefore only has to apply small control forces f_{Pil} to control the rotorcraft.

In ultralight helicopters, the mechanical control train directly connects the controls in the cockpit to the swashplate through push rods and levers. [Figure 2b](#) depicts the topology of a typical reversible cyclic control train of an ultralight helicopter with an inline actuator to illustrate this. f_{Pil} is the control force applied by the pilot, f_{R1} and f_{R2} are the forces from the rotor blades. Note that there is no hydraulic force amplification. Also note that some

small rotorcraft with reversible control trains have a trim system or cyclic centering spring, but the flight demonstrator does not. Due to the lack of a trim or centering mechanism, the forces from the rotor system propagate through the control system including the cyclic stick unfiltered and have to be overcome by the pilot to make a control input. Therefore, inline actuators mounted in ultralight rotorcraft also have to be powerful, yielding additional feedback forces f_{Act} at the cyclic stick. In order for the actuator to have any effect at the swashplate, the pilot has to resist these forces and hold the stick at its position, all whilst applying trim and control forces.

In short, the stabilization effectiveness varies depending on how well the pilot is able to counteract the actuator forces and if the feedback forces affect pilot workload. Both questions have yet to be answered. This paper is the first step towards an answer by practically demonstrating the feasibility of the concept.

The Control Train in the Flight Demonstrator

The flight demonstrator features a coaxial teetering rotor system used to control all four primary control axes. [Figure 3](#) is a close up image of the coaxial rotor. Collective and cyclic control is achieved through a system with two swashplates. The lower swashplate sits below the lower rotor and connects it to the control system in the fuselage, the upper swashplate is located between the two rotors. Yaw control is achieved by varying the collective angle of the upper rotor, yielding a yaw-collective coupling. To increase yaw authority in forward flight, the vertical stabilizer features a rudder connected to the yaw control train. Apart from the yaw-collective coupling, the couplings relevant in conventional single main rotor helicopters are negligible in the case of the flight demonstrator operating in hover and slow flight

The control system itself uses three control rods attached to the lower swashplate at an angle of 90° . [Figure 5](#) illustrates the attachment points. Control rod 1 is attached at the front and used for cyclic pitch control, control rods 2 and 3 are attached on the left and right side and are used for collective and cyclic roll control. Due to the control rod attachment points at the lower swashplate, only the two roll control rods carry the collective feathering forces from the rotor blades. Control rod 1 moves in conjunction with the other two in the case of collective inputs, but does not carry any collective force.

FUNCTIONAL SAS REQUIREMENTS

As a prerequisite for the system design process, the functional requirements for the SAS are listed in this section. They are

1. Stabilize the unstable pitch and roll motion in hover with the least amount of authority necessary.
2. Provide intuitive pilot interfaces, i.e. display information about the system state in a compact form in the cockpit.
3. Detect and display hard- & software failures in the complex system and stop actuators automatically.
4. Provide means for the pilot to safely deactivate and lock the actuators at any time through a non-complex system.

Most importantly, safety has to be ensured for all possible failures of the system either through system design or through additional mitigating factors. To define specific requirements for the SAS, a Functional Hazard Assessment (FHA) is conducted for the helicopter equipped with the SAS. To ensure safety, the SAS is combined with an additional non-complex mechanism to lock the actuators if a failure occurs. With an independent locking mechanism as a mitigating factor, it follows that flight safety can be ensured whilst the FCC firmware does not need to be certified formally. It is assumed that the SAS will fail with a probability of 100 % and safety is ensured through the noncomplex locking system, which is operated by the pilot if necessary. To expedite the certification process, several non-technical mitigating factors are introduced in addition, such as limiting the flight envelope to hover and restricting any flight testing to good weather conditions.

For the SAS itself, a System Safety (SSA) is performed based on the preliminary system architecture using fault trees. Assuming a good visual environment and calm winds, the functional hazard assessment showed that any failure of the SAS will only lead to a hazardous situation if any of the actuators move in a hazardous manner as a result. If a system failure occurs and the pilot does not recognize it and just continues flying, the failure is not critical by our definition. This is justified for a limited authority SAS and the restrictions mentioned above. This reduces the set of relevant failures to the following actuator failures:

1. **Power loss:**
Power supply of one or more actuators fails and actuator moves freely.
2. **Out of control failure:**
One or more actuators move erratically at a high frequency
(i.e. due to incorrect sensor data).
3. **Undetected erratic motion:**
One or more actuators drift slowly

resulting in a change in cyclic and collective trim (i.e. due to insufficient power or incorrect sensor data).

Figure 6 shows the fault tree for a complete loss of cyclic control caused by an actuator failure from the SSA. With an additional locking mechanism, safety can be ensured if the pilot detects the failure and triggers the locking mechanism in time.

ACTUATOR REQUIREMENTS

The actuators are the most critical part of the SAS. They need to be lightweight, compact, but powerful enough to control the rotor system, and withstand the vibrations in the control system. To ensure safe operation, a certain safety margin is needed. There is – however – little room for oversizing in ULH, so precise knowledge about the forces and movements in the control train is crucial. To define precise requirements, the rotorcraft that later becomes the flight demonstrator is equipped with a measurement rig and a dedicated flight test campaign is conducted. The gathered flight data is used to define the requirements.

The following sections illustrate the actuator placement, the collection of flight data and the synthesis of requirements from flight data.

Actuator Placement in the Control Train

Due to spacial constraints, the SAS actuators are mounted right below the swashplate, replacing the three aforementioned swashplate control rods. Figure 16 depicts the actuators mounted in the flight demonstrator. The placement right at the swashplate has the following advantages:

1. The actuators can be accessed easily for maintenance.
2. The lever arms are longer than in the lower control rods used in (Ref. 1), yielding lower required actuator forces.
3. Collective control is theoretically possible with the actuators at this position, giving the opportunity of adding collective control in the future.
4. Mounting the actuators here puts them far away from the cyclic stick and close to the rotor system, yielding the largest possible amount of inertia between the cyclic stick and the actuators, possibly mitigating the actuator force feedback.

Flight Data Collection

The measurement system uses proven hardware from the tethered tests in (Ref. 1) and consists of

- an commercial-off-the-shelf FCC including an IMU, GPS, and an EKF for state estimation,
- potentiometers in the primary control system measuring pilot inputs and control rod motion,
- strain gauges at the cyclic stick and collective lever to measure the control forces,
- and strain gauges near the planned actuator positions.

Figure 4 illustrates the position of strain gauges and displacement sensors in the control train. Figure 7 shows the measurement electronics in the baggage compartment under the seats. The measurement system is operated using a laptop by a flight test engineer on the co-pilot seat.

Flight Test Campaign

Recall from the previous section that, to simplify the certification process, the flight envelope is restricted to hover for the SAS tests. However, an SAS might also be useful outside hover, so the goal is to build a system that is technically usable within the complete flight envelope of the CoAX 600. Therefore, the flight plan for the measurement campaign includes maneuvers from across the flight envelope. Additionally, step inputs and quick stops are included because the actuators must also withstand the large forces occurring when the pilot makes sudden violent control inputs. The test plan includes maneuvers from the following categories:

1. **Hover maneuvers**
stationary hover, sidesteps, repositioning, combined yaw and repositioning maneuvers
2. **Normal operation outside hover**
traffic patterns, turns in cruise, approach into hover from cruise
3. **Dedicated test maneuvers and inputs**
Step inputs, quick stops

Definition of Actuator Requirements

To define the actuator requirements, the flight data from the three maneuver categories are analyzed separately. The goal is to find

- the required actuator motion range s_r in mm,

- the required actuator speed v_r and bandwidth ω_{Act} ,
- and the required actuation forces f_r .

For all three of these, a compromise has to be found. From a safety standpoint, small values are preferable as they reduce the safety effect in the case of a failure, but adverse effects due to insufficient authority, bandwidth, rate, or actuation force impede stabilization. The exact requirements depend on the position of the actuators in the control train between the cockpit, the mixing unit, and the rotor system. In (Ref. 1), the actuators were mounted in the bottom control rods between the mixing unit and the swashplate because space at the top was limited due to the tethering frame. Like mentioned before, the top control rods are chosen instead in this work due to longer lever arms and thus lower required actuation forces, and for better maintainability.

For the required authority, the measured pilot inputs in hover maneuvering are taken as a reference. The idea is that an SAS will need approximately the same authority as a human pilot to stabilize the rotorcraft. As far as flight regimes are concerned, only hover data is used to define authority requirements. To calculate the authority, each hover maneuver is analyzed separately and the motion range is extracted for each primary control axis. Figure 8 depicts a 63 second long hover dataset to illustrate the approach. The minimal and maximal values are marked with solid gray lines. Note that most of the data points lie in a smaller band, and the minimal and maximal values are mostly outliers. Moreover, in the tethered tests presented in (Ref. 1), stabilization was successful despite occasional actuator saturation. To strike a balance between the required authority and safety in the case of a failure, the 5%- and 95% quantiles are therefore used to calculate the required authority. The quantiles are marked with dashed red lines in Figure 8.

To find the required actuator rate and bandwidth, the displacement measurements at the actuator position are used because they include both cyclic and collective inputs. The resulting requirements based on these measurements should therefore suffice for combined cyclic and collective actuator inputs. Firstly, the required rate is extracted from the flight test data in a two step process. Afterwards, the attainable bandwidth is estimated.

(1) In step one of the rate calculation process, the displacement data is filtered to remove undesired signal parts. Figure 8 shows the raw displacement data for the pitch actuator control rod in hover as a black line. The frequency spectrum of the displacement data is shown in black in Figure 9. It is apparent that the signal consists of low frequency control inputs between 0 and approx. 6 Hz on one hand, and high frequency vibrations at the rotor harmonics n/rev on the other. The actuators must only

be able to follow the low frequency inputs. To remove the high frequency vibrations from the signal, the pilot input data is lowpass filtered with a forward-backward filtering algorithm using a lowpass filter with a cutoff frequency of 6 Hz, yielding the green lines in [Figure 8](#) and [Figure 9](#). (2) In step two, the filtered displacement data is differentiated numerically, yielding the rate of the filtered signal, and the highest absolute rate is calculate. This is performed for each maneuver, yielding [Figure 10](#). The velocity estimates from numeric differentiation of the raw data is shown as a black line, illustrating the impact of n/rev vibrations. The rate estimate from the filtered data is drawn in green. The dashed red lines indicate the minimum and maximum values of the filtered control rod speeds.

To select a motor, the maximum rotational speed is used as a requirement because most electric motor manufacturers give a speed limit instead of a bandwidth in their data sheets. To plausibilize the speed requirement, check for rate limits in the actuator during testing, and synthesize preliminary actuator models for controller design, an estimation of the attainable bandwidth from a given motor speed is performed in the following paragraph.

For harmonic position commands of the form

$$s_c(t) = \hat{s}_c \cdot \sin(\omega t), \quad (1)$$

the position response of the actuator is

$$s(t) = \hat{s} \cdot \sin(\omega t + \varphi). \quad (2)$$

The maximum rate of the input signal at a given frequency is

$$\hat{v}_c = \hat{s}_c \cdot \omega. \quad (3)$$

If the peak rate $\hat{v}_c$ is smaller than the actuator rate limit, the actuator acts as a linear system. With this information, a rate limited bandwidth can be calculated.

$$\omega_{\text{BW,eff}} = \frac{v_{\text{Act,Lim}}}{\hat{s}_c}. \quad (4)$$

To plausibilize the rate requirement, the authority requirement from [Table 2](#) is used. It is assumed that the input signal follows [Equation 1](#) and uses the complete actuator motion range, yielding an amplitude $\hat{s}_c$ of half the motion range s_r . A peak rate of 35 mm/s at an amplitude of $\hat{s}_c = 0.5 \cdot s_r$ yields:

$$\omega_{\text{BW,eff}} = \frac{35 \text{ mm/s}}{0.5 \cdot 6.9 \text{ mm}} \quad (5)$$

$$= 21.7 \text{ rad/s} = 3.45 \text{ Hz}. \quad (6)$$

This is slightly above 2.5 Hz, which is considered the bandwidth of a human pilot (Refs. [3,4](#)). An actuator with a similar bandwidth should be able to stabilize the flight demonstrator. Note that this is assuming the maximum

Table 1: Actuator requirements from measurement data

Actuator	$\mathbf{f_r}$ = $ \mathbf{f_{max}} $	$\mathbf{s_r}$ = $s_{5\%} - s_{95\%}$	$\mathbf{v_r}$ = $ \mathbf{v_{max}} $
1 (Pitch)	192.2 N	7.08 mm (13.3%)	28.7 mm/s
2 (Roll)	539.7 N	4.01 mm (8.6%)	32.6 mm/s
3 (Roll)	487.6 N	4.01 mm (8.6%)	30.2 mm/s

input amplitude. The attainable bandwidth is inversely proportional to the input amplitude, so smaller input amplitudes yield a larger effective bandwidth. Moreover, smaller amplitudes should theoretically be sufficient for stabilization due to the faster reaction time of a digital system in comparison to a human pilot.

For the required actuation force, the maximum forces occurring at the actuator control rods throughout the flight campaign are used, i.e. including the maneuvers outside hover. Unlike occasional saturation, insufficient actuator forces render the SAS useless altogether. In one of the roll actuators, any unintentional compression due to insufficient actuator force affects roll, pitch and collective control at the same time. The resulting descent might end with the rotorcraft touching the ground at a nonzero attitude and possibly tipping over. To illustrate these couplings, [Equation 7](#) shows the linearized control system dynamics and the effects of each actuator on cyclic and collective inputs. s_i are the actuator displacements, $d\beta$ and $d\alpha$ are cyclic pitch and roll inputs, and $d\theta$ is a collective input. k_j are the effectiveness values of the actuators on the respective control axis. Due to the swashplate attachment points, actuator 1 only affects cyclic pitch inputs, but actuators 2 and 3 affect all three axes. If the actuators move in conjunction, their effects on cyclic inputs cancel out, leaving only a collective input. If actuators 2 and 3 move in opposite directions, they make a roll input.

$$\begin{bmatrix} d\beta \\ d\alpha \\ d\theta \end{bmatrix}_{\text{SAS}} = \frac{1}{2} \begin{bmatrix} 2k_{d\beta} & -k_{d\beta} & -k_{d\beta} \\ 0 & k_{d\alpha} & -k_{d\alpha} \\ 0 & k_{d\theta} & k_{d\theta} \end{bmatrix} \begin{bmatrix} s_1 \\ s_2 \\ s_3 \end{bmatrix} \quad (7)$$

Requirement Harmonization

To harmonize the authority requirements for the roll and pitch actuators, the following steps are taken:

1. Calculate the required authority from cyclic pilot inputs like shown above.
2. Assume that the required authority in % is similar for the roll and pitch axis. Round the authority to an integer percentage between the two.

Table 2: Actuator requirements – simplified

Actuator	f_r	s_r	v_r
1 (Pitch)	192.2 N	6.9 mm	30 mm/s
2 (Roll)	539.7 N	6.2 mm	35 mm/s
3 (Roll)	487.6 N	6.2 mm	30 mm/s
Max.Val.	539.7 N	6.9 mm	35 mm/s

3. Add $\pm 3\%$ for collective inputs to the calculated authority to allow for combined pitch/roll and collective control in the future. This is what was required in hover flight tests to hold the height above ground whilst maneuvering.
4. Convert the authority requirement in % back to the required displacement in mm at the actuator position.

The authority requirements in terms of pilot inputs for each actuator are given in brackets in Table 1. Rounding to the nearest multiple of 5 % yields 15 % for the pitch and 10 % for the roll axis. To simplify the requirements and limit the safety effect in the case of an actuator failure, 10 % are used as an authority requirement for both axes from this step onwards. These 10 % are consistent with literature (Ref. 5). To enable combined cyclic and collective inputs in the future, $\pm 3\%$ are added to the requirement. This is the collective motion observed in hover maneuvers. The 13 % are then converted back to actuator displacements in mm, yielding the values in Table 2. Note that these requirements are based on measurement data, but several assumptions were made to simplify and harmonize the requirements. Therefore, an adjustable hardware authority limit is added to the requirements to have the option to increase authority if necessary.

The required actuator speed is rounded to the nearest multiple of 5 mm/s. Table 2 summarizes the requirements for the three actuators. Note that the force requirements differ between the three actuators. The roll actuator force requirements are a lot higher than those for the pitch actuator, because the roll control rods carry the collective feathering forces. The force requirements also differ notably between the roll actuators. This can be attributed to manufacturing tolerances yielding different amounts of friction in the two roll control paths, asymmetric wind conditions, and measurement inaccuracy. To design one actuator that can be used for all three positions, the maximum values (Max. Val.) are taken as the final requirements for the actuator design. They are given in the bottom row of Table 2.

In addition to these requirements, a safety mechanism is necessary to deactivate and lock the actuators in the case of a failure. Due to the vibrations at the swashplate, a

brake using friction is deemed insufficient, because extensive testing would be necessary to positively prevent actuator creeping in all cases. An interlocking mechanism is therefore added as a requirement.

Unfortunately, no available COTS actuator could provide the large required forces in combination with the small range of motion and with the required interlocking mechanism in a sufficiently small and lightweight housing. Therefore, new actuators are designed specifically for the flight demonstration. The following section provides an overview of the design decisions and testing process of the actuators.

ACTUATOR DESIGN AND TESTING

The safety analyses in the previous section illustrate that the actuators are the most critical part of the SAS. Therefore, special emphasis is put on them in the design process. The following sections illustrate the actuator requirements, design, and testing.

Actuator Design

The actuator design is based on the requirements in Table 2. In (Ref. 1), off the shelf actuators with a ball-type linear drive and a belt drive connecting the motor were used. In the case of a belt failure, the low friction ball-type linear drive would have been able to move freely, possibly resulting in hazardous actuator flutter. To prevent this in the flight demonstrator, a rigid connection between the motor and the ball screw is used, with no belts that could suddenly fail.

Figure 11 shows the inner workings of the actuator. The parts involved in actuation are a threaded rod (A), a ball screw (B), a planetary gearbox (C), and a brushless motor (D). Each actuator is equipped with two displacement sensors. An encoder (E) measuring the angle of the motor shaft is attached to the motor and used for the positioning control loop in the actuator. To ensure safety, an additional linear variable differential transformer (LVDT,F) is attached to the threaded rod. It precisely measures the position of the threaded rod and is used to detect failures in the ball screw and the gearbox. This architecture yields a rigid connection between the motor and the threaded rod. To detect a mechanical failure in the transmission, the measurements from the encoder and the LVDT are used. They are related through Equation 8, where n_{GB} and n_{BD} are the transmission ratios of the gearbox and the ball-type linear drive, respectively.

$$s_{TR} = n_{GB} \cdot n_{BD} \cdot \varphi_m \quad (8)$$

If the two values are no longer consistent within sensor accuracy, a mechanical or sensor failure has occurred.

This method to detect transmission failures is included in the flight control algorithms.

For the necessary **range of motion** s_r , the requirement is 6.9 mm. This value is rounded up to 10 mm. The authority limit is realized through a metal pin attached to the threaded rod. The pin moves in an elongated hole with a length of 10 mm. Through metal inlays, the length can be reduced if a smaller authority is needed.

To meet the high **actuation force** requirement without excessive power consumption, a ball-type linear drive with a small thread pitch is combined with a planetary gearbox and a high speed brushless motor. The ball-type linear drive enables precise positioning with little friction and enables high actuation forces and actuator speeds at moderate supply voltages and without requiring a large torque motor. High speed brushless motors have a smaller diameter than their high-torque counterparts and are therefore better suited for an inline actuator.

Motor Selection

To select a motor, the available thread pitches and gearbox transmission ratios have to be taken into account. The required motor speed ω_m is given by

$$\omega_m = \frac{v_r}{n_{GB} \cdot n_{BD}} \quad (9)$$

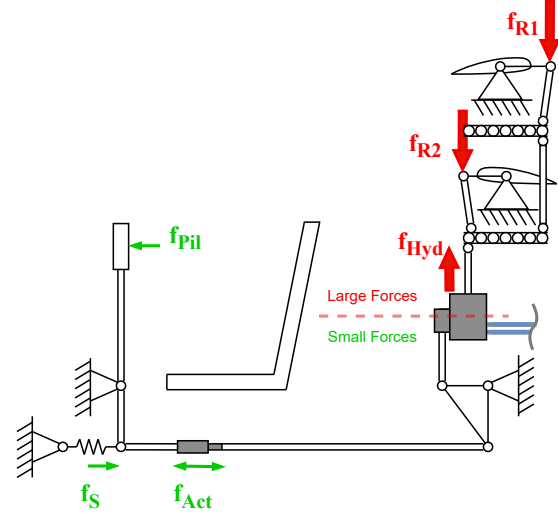
and the required motor torque t_m is

$$t_m = n_{GB} \cdot n_{BD} \cdot f_r. \quad (10)$$

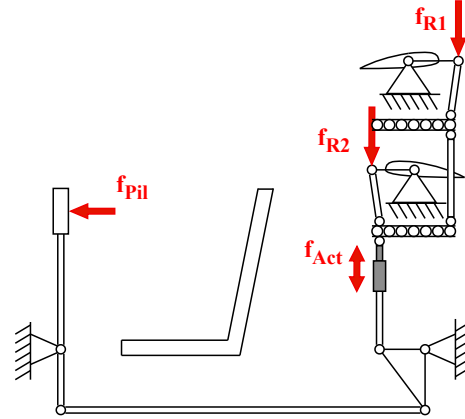
In our design, the total transmission ratio $n_{GB} \cdot n_{BD}$ of both parts combined is $8.06 \cdot 10^{-2} \text{ mm/rad}$. For the motor requirements, losses due to friction are not taken into account explicitly. Instead, a motor that oversatisfies the requirements is chosen to prevent thermal issues and ensure acceptable performance throughout the actuator lifetime.

The resulting motor requirements ω_m and t_m are shown in column two of **Table 3**. Column three shows the parameters of the chosen brushless motor. Note that the peak loads are used for the required forces f_r , resulting in a peak motor torque requirement t_m . The chosen motor oversatisfies this peak torque requirement by 422 % (peak torque) or 74 % (continuous torque), leaving plenty of room to overcome friction and to prevent overheating. The speed requirement is oversatisfied by 93 %.

A compatible motor driver and control board from the motor manufacturer is used to enable positioning an communication via CANopen.



(a) Control train with hydraulic system and centering spring



(b) Control train without hydraulic system or centering spring

Figure 2: Control train architectures with inline actuators

Table 3: Actuator motor requirements

	requirement	selected motor	margins
ω_m	434 rad/s (4148 rpm)	838 rad/s (8000 rpm)	93 %
t_m	43.52 mNm (Peak: 227.3 mNm)	75.8 mNm 422 %	74 %



Figure 3: The rotor system of the flight demonstrator

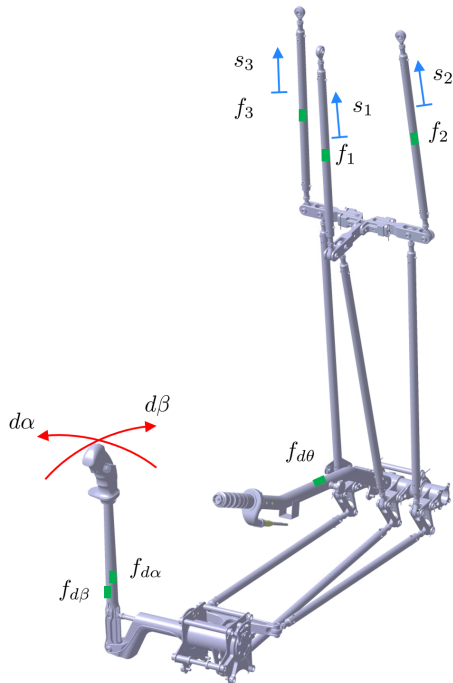


Figure 4: CAD model of the control train

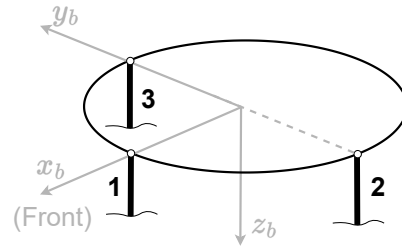


Figure 5: Control rod attachment points at the lower swashplate

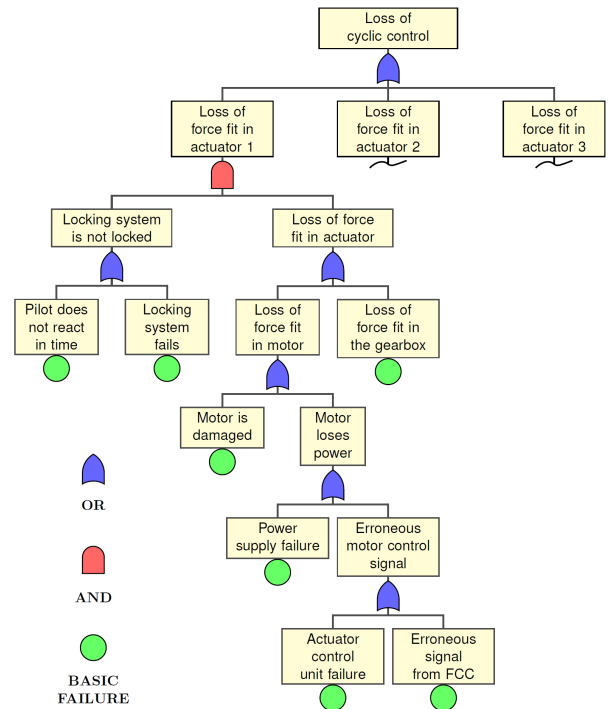
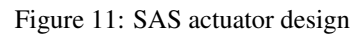
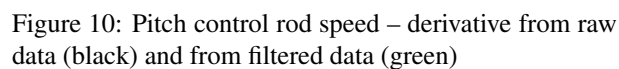
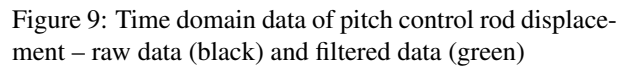
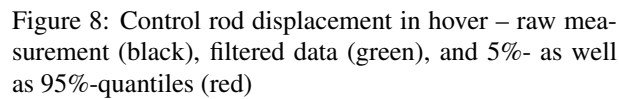


Figure 6: Fault tree analysis – Loss of cyclic control caused by actuator failure



Figure 7: Measurement rig in the helicopter



For the locking mechanism, different existing concepts of interlocking mechanisms were analyzed and reviewed based on the following requirements:

- 10

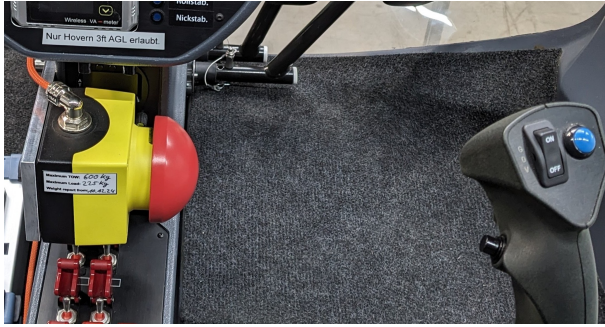


Figure 12: Actuator Killswitch (left)

check the system after each activation of the locking mechanism.

The layout and functionality of the final mechanism is described in the following paragraph. Each actuator has two locking mechanisms, as depicted in Figure 11. On the left side of Figure 11, the mechanism is open and enables actuator motion. On the right side, the mechanism is locked. The locking mechanism is triggered through a killswitch operated manually by the pilot. If the killswitch is pushed, the actuators are locked, yielding rigid actuator control rods. Figure 12 depicts the killswitch in the cockpit. It can be pushed by the pilot with his left leg. To satisfy requirement 1, the locking system uses electromagnets (G) in a direct current circuit. In nominal flight, the magnets are engaged and hold a locking bolt (H) in place against the force of a spring (J). If the pilot pushes the killswitch (requirement 2), power to the magnets is cut and the locking bolt is pulled into the locked position. To prevent the locking bolt from becoming wedged (requirement 3), it is not directly locking the actuator. Instead, a steel ball (I) is used to lock the actuator. The ball screw (B) is equipped with conical notches around its circumference. When the locking system is triggered, the steel ball is pushed against the ball screw and eventually lands in one of the notches, locking the ball screw in place and preventing it from rotating. This provides a smooth-running locking mechanism, which reliably locks the actuator even while the spindle rotates. To satisfy requirement 4, releasing the kill switch reactivates the electromagnets, but won't unlock the actuator. The locking mechanism has to be reset manually from outside the cockpit. This forces the crew to land and check the system after the locking mechanism was triggered. Figure 13 shows the actuators mounted in the rotorcraft. The blue tubes are part of the system used to manually unlock the actuators.

Actuator Testing

Before the three actuators for the flight demonstrator are built, a prototype of the SAS actuator is built and tested

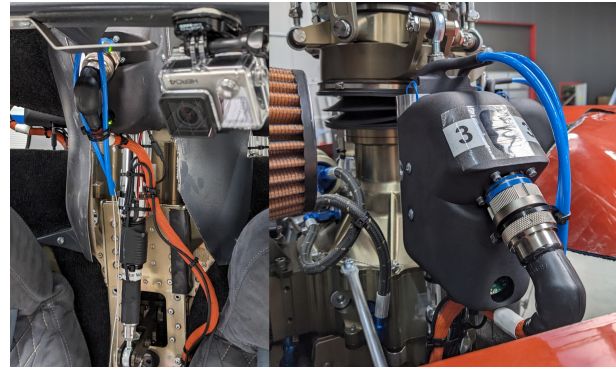


Figure 13: Actuators mounted in the helicopter

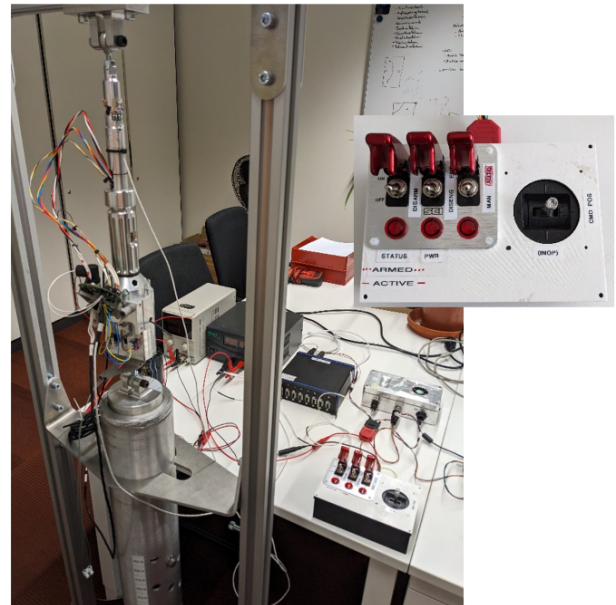


Figure 14: Actuator test rig

in a test rig. The test rig is shown in Figure 14. It includes a metal frame where the actuator is mounted, a control unit running a microcontroller with a proprietary firmware, strain gauges to measure the actuator forces, and a control panel including switches to choose a testing profile, status LEDs, and a joystick to control the actuator manually. The control unit runs a proprietary firmware that includes automatic testing profiles and logging algorithms. The communication protocols emulate those in the flight demonstrator. The test rig uses a steel spring to generate a compression force similar in magnitude to the peak force measured in the roll control train. Figure 14 shows the testing process of the actuator prototype. Apart from the **communication algorithms** and actuator error detection, the bench testing program includes

1. tests where the actuator is controlled manually using a joystick to check for **basic functionality**,
2. step responses and sine sweep tests at different am-

Table 4: Actuator bandwidths for different sweep amplitudes.

Amplitude [mm]	ω_{-3dB} [rad/s]	f_{-3dB} [Hz]
3.45	21.7	3.45
1.25	55.4	8.82
2.5	31.9	5.08
3.75	23.1	3.68
5	18.2	2.90

plitudes to check the **dynamic behavior** and **provide a dynamic model** of the actuators,

3. constant frequency sines, ramps, and sawtooth profiles to check for **thermal issues** and **endurance** under load,
4. static and dynamic tests of the locking mechanism.

To synthesize an actuator model for controller design, sine sweep data is used. Sine sweep data from tests with different input amplitudes is compared to check (1) if the actuator bandwidth is limited by linear effects or by the rate limit and (2) if the bandwidth is sufficient. **Table 4** lists the -3 dB bandwidths identified from the data and answers both questions: (1) The bandwidth is inversely proportional to the input amplitude, which means that the actuator bandwidth is likely rate limited. (2) At maximum authority, the bandwidth is 2.9 Hz, which is still higher than the upper end of the stabilization frequency band from literature (Ref. 4). At smaller amplitudes, the actuator bandwidth lies beyond 5 Hz, yielding a sufficient bandwidth margin.

For controller design, the actuator dynamics are represented as a second order system with a damping of $\sqrt{2}/2$. Based on the information in **Table 1** and with an authority limit of 10 % of pilot authority, it can be assumed that the stabilization activity will primarily include amplitudes in the range of ± 2.5 mm in hover. At this amplitude, the actuator has a bandwidth of 5.08 Hz according to **Table 4**, corresponding to a slightly smaller natural frequency. Therefore, the undamped natural frequency of the actuator model is set to 5 Hz = 31.4 rad/s:

$$G_{Act}(s) = \frac{\omega_0^2}{s^2 + 2 \cdot \zeta \cdot \omega_0 \cdot s + \omega_0^2},$$

$$\omega_0 = 31.4 \text{ rad/s } (= 5 \text{ Hz}), \zeta = \sqrt{2}/2 \quad (11)$$

SYSTEM ARCHITECTURE

Hardware Architecture & Operation

Figure 15 illustrates where the SAS components are mounted in the flight demonstrator. As mentioned before,

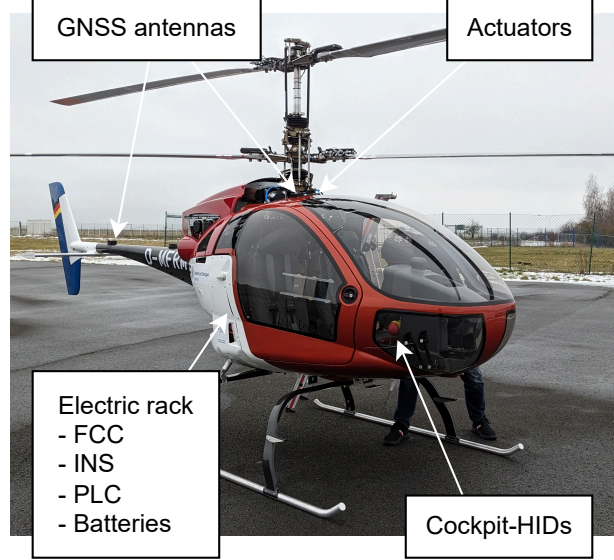


Figure 15: SAS components in the flight demonstrator

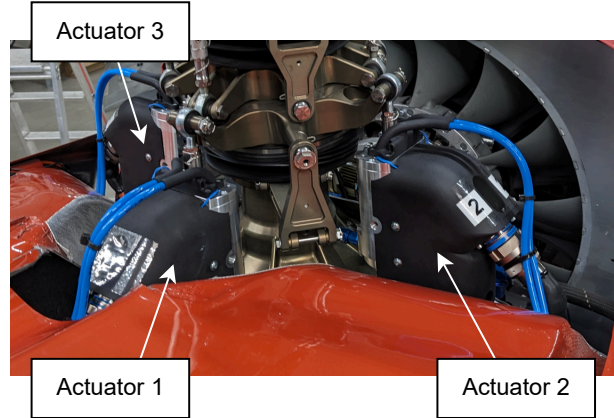


Figure 16: Actuators mounted in the flight demonstrator

the actuators are mounted directly to the lower swashplate, replacing the three upper control rods used for cyclic and collective inputs. **Figure 16** shows the upper parts of the actuators and the lower swashplate.

Apart from the proprietary actuators and an FCC with a custom firmware, the SAS uses COTS components, some of which with a customized software. The other subsystems include

- **An airworthy FCC**
running a firmware from the University of Stuttgart
- **An airworthy sensor package**
(IMU, Dual-GNSS, INS onboard)
- **CAN-compatible programmable logic controllers (PLCs)**
to connect the switches, buttons, and LEDs

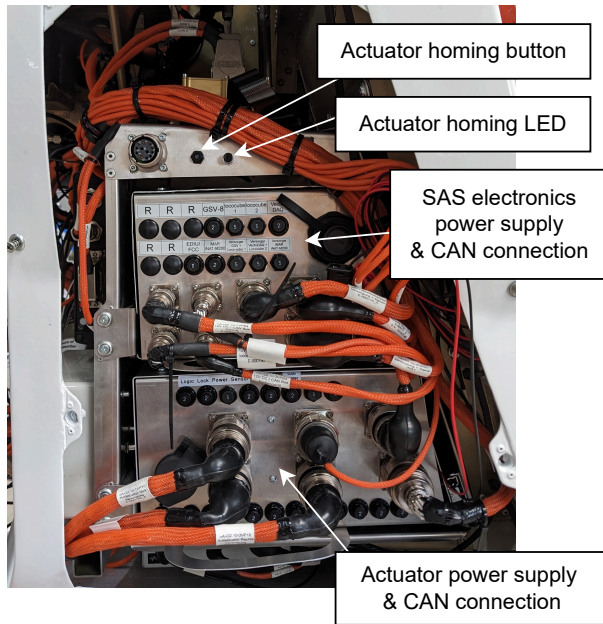


Figure 17: Electric rack

- A **proprietary control & display unit** developed and built specifically for the SAS and used by the flight test engineer
- A **CAN data logger**

Most of the electronics including the IMU and the FCC are located in an electric rack behind the pilot seat, close to the center of mass of the rotorcraft. The electric rack is shown in [Figure 17](#). A GNSS antenna is mounted on top of the cabin, another one is mounted on the tailboom. In the cockpit, there are several interface devices that are explained in the following paragraphs. [Figure 18](#) gives an overview of the SAS interfaces in the cockpit. They include four light emitting diodes (LEDs) and a battery status display ([Figure 19](#)), two buttons on the cyclic stick ([Figure 20](#)), a kill switch, and a control and display unit ([Figure 21](#)).

The SAS uses a single lane architecture. Communication is realized through CAN-buses to keep everything modular and to enable future extensions to the system. In contrast to (Ref. 1), all hardware is replaced by parts suitable to obtain a preliminary type certificate (PTC) for a flight demonstration. The ground station that used to be connected wirelessly to monitor the system is replaced by a wired control & display unit in the cockpit.

The SAS is designed for two people operation. The pilot engages and disengages the SAS and operates the locking system. All other tasks, such as switching between stabilization modes, arming the system and monitoring actuator activity are fulfilled by the flight test engineer. [Figure 19](#) depicts the four status LEDs in the cockpit. The



Figure 18: HID in the cockpit

two red LEDs indicate abnormal conditions. If the actuators are locked, the locking LED lights up. If the FCC detects an error in the complex system, the error LED lights up. The two lower LEDs indicate the status of the roll and pitch axis stabilization. The SAS modes and corresponding LED patterns are listed in the software-section of this paper. To activate and deactivate the stabilization, the pilot uses two buttons on the cyclic stick, as shown in [Figure 20](#). All other switches are operated by the flight test engineer.

The flight test engineer operates and monitors the system using the control and display unit shown in [Figure 21](#). It has a switch to arm/disarm the system, two switches to select a stabilization mode, and an event marker to mark important episodes in the measurement data. In addition, a display shows information about the system health. [Figure 22](#) illustrates what data is displayed. In normal operation, the display shows the position of the roll and pitch actuators, the current state of the SAS, and the selected stabilization mode combined with a heartbeat signal. If an error occurs, the actuator position data is replaced by information about the affected subsystem and error category at the top and an error code at the bottom. A list of the errors that are displayed is shown in the appendix. To configure and parametrize the controller, a laptop can be connected to FCC via CAN. An inhouse developed app lets the flight test engineer change the controller gains, the filter gains, and the SAS authority without the need to flash a new firmware, and save all parameters to an SD-card mounted in the FCC. [Figure 23](#) shows a screenshot of the parametrization app.

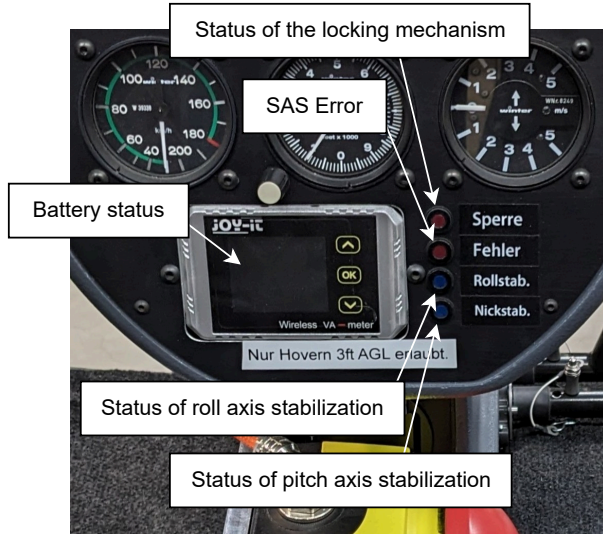


Figure 19: Status LEDs in the cockpit

FCC Algorithms

The FCC runs a proprietary firmware. To ensure that only necessary functionality is included, the algorithms are developed implemented from scratch at the University of Stuttgart. The FCC has three CAN bus interfaces running different protocols according to the requirements of each connected device. One channel is used to connect the INS, one is used for the HID, and one is exclusively used for communication with the actuators.

The controller algorithms are implemented in C and include the control laws, a state machine, allocation algorithms, a parameter management module, and Health and Usage Monitoring Systems (HUMS). The HUMS functionality includes monitoring of the bus traffic as well as the sensor data and the status of all system components. [Figure 24](#) illustrates the modules running on the FCC and the data flows. The main control loop is shown in the second row. Sensor data is received on Lane A and actuator commands are transmitted on Lane C. Lane B is used to communicate with the PLCs connecting the switches, buttons, display, and LEDs.

[Figure 25](#) illustrates the SAS state machine. After booting, the actuators require a homing procedure to initialize the encoders. The initialization is triggered with a button on the electric rack ([Figure 17](#)). Afterwards, the system switches into IDLE. From there, the flight test engineer can choose a stabilization mode and arm the system. Once the SAS state is ARMED, the pilot can engage it with a button on the cyclic stick ([Figure 20](#)). The activation includes a two second fade in phase to prevent abrupt actuator motion. Similarly, disengaging triggers a two second fade out phase. If the FCC detects any error, it switches into the ERROR state, stops all actuators, and activates the error LED ([Figure 18](#)).

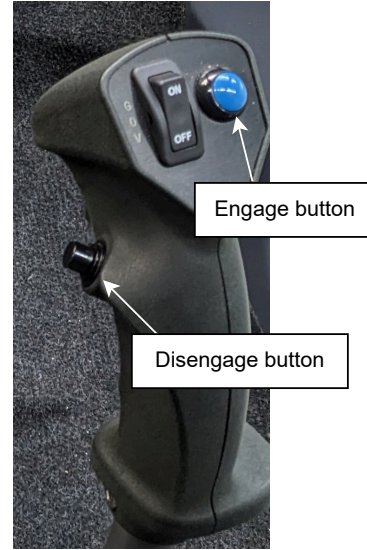


Figure 20: Cyclic stick buttons

Implementation Aspects

The base sample rate of the FCC software is 1 kHz, all controllers and filters run at a sample rate of 100 Hz, resulting in ten base cycles per control cycle. To ensure deterministic execution, different modules (data acquisition, error detection, data filtering, calculation of controller commands, and allocation) are each assigned to one of these ten cycles. This keeps the controller delay at less than 10 ms whilst ensuring consistent cycle times. The filters are implemented as digital IIR filters. The controller module includes anti-windup functionality through integrator clamping. The FCC uses an SD card to store the controller parameters. A Fletcher checksum is used to ensure data integrity ([Ref. 6](#)). The parameters can be changed through a CAN-connected app that uses a proprietary protocol inspired by the CANopen service data object (SDO) protocol.

CONTROL LAWS

This section covers the SAS control loop, shows the plant model used in controller design, and includes the synthesis and tuning of the control laws and the required filters. During the testing campaign, the filters and the controllers were updated several times. This paper contains the final SAS configuration.

Plant Model

Due to the coaxial rotor system, the hover pitch and roll dynamics of the flight demonstrator are largely uncoupled ([Ref. 7](#)). Therefore, single-input-single-output (SISO) control loops are deemed sufficient, which simplifies controller design. To synthesize the control laws,

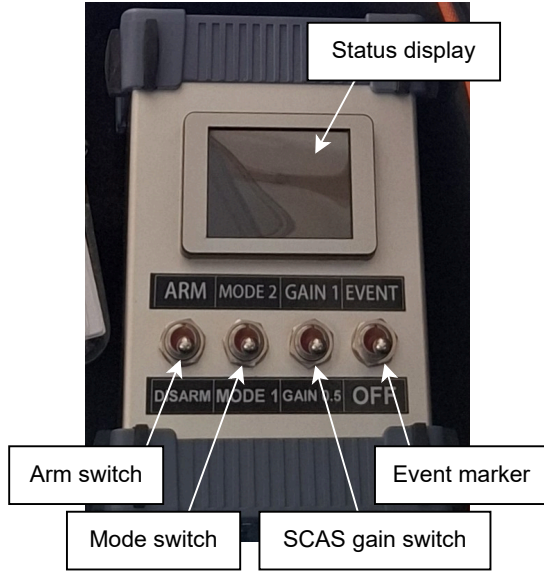


Figure 21: Control and display unit

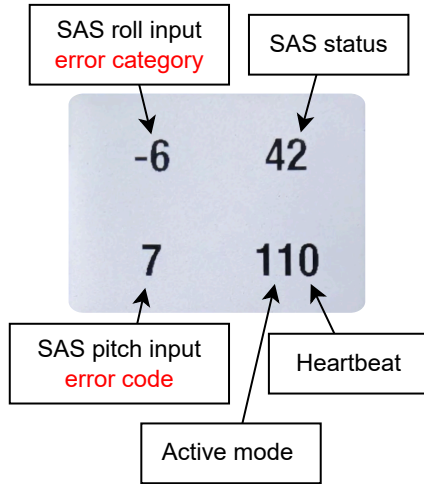


Figure 22: SAS display data example

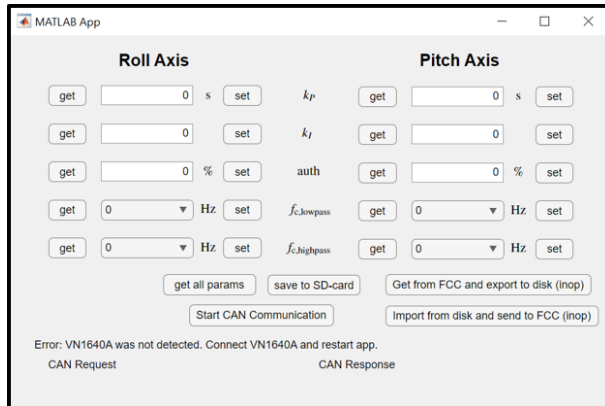


Figure 23: SAS parametrization app

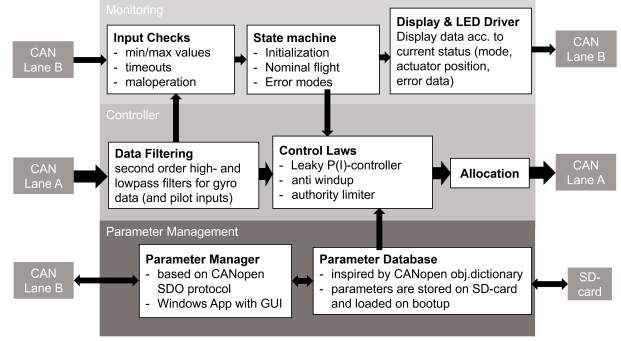


Figure 24: Controller algorithms

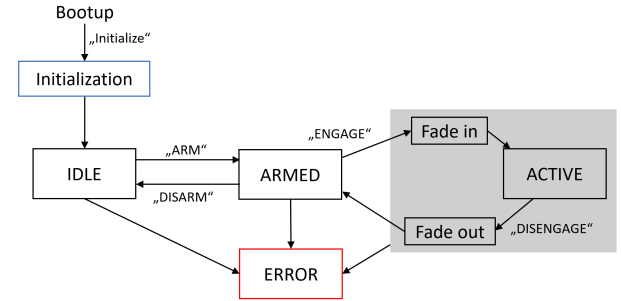


Figure 25: SAS state machine

linearized SISO plant models for the roll and pitch motion based on (Ref. 7) are used. The model structure is given in Equations 13 and 15.

$$\begin{bmatrix} \dot{u} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} X_u & X_q & -g \\ M_u & M_q & 0 \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} u \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} X_{d\beta} \\ M_{d\beta} \\ 0 \end{bmatrix} d\beta \quad (12)$$

$$y_{Lon} = q \quad (13)$$

$$\begin{bmatrix} \dot{v} \\ \dot{p} \\ \dot{\phi} \end{bmatrix} = \begin{bmatrix} Y_v & Y_p & g \\ L_v & L_p & 0 \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} v \\ p \\ \phi \end{bmatrix} + \begin{bmatrix} Y_{d\alpha} \\ L_{d\alpha} \\ 0 \end{bmatrix} d\alpha \quad (14)$$

$$y_{Lat} = p \quad (15)$$

To find the model parameters for the CoAX 600, the system identification from Richter et al. is revisited and the results are updated. Similar to the approach in (Ref. 7), an equation error method with precomputed state derivatives is applied. However, following Tischler (Ref. 8), the angular and translational motion are identified as separate scalar first order SISO systems from different measurement data. The coupling derivatives between angular and translational motion are calculated based on kinematic considerations, yielding the bare airframe poles listed in Table 5. For controller design, the rotorcraft dynamics are expressed as transfer functions $G_q(s)$ and $G_p(s)$, respectively.

$$G_q(s) = s \cdot \frac{M_{d\beta}s + M_u X_{d\beta} - M_{d\beta} X_u}{s^3 - (X_u + M_q)s^2 + (M_q X_u - M_u X_q)s + M_u g} \quad (16)$$

$$G_p(s) = s \cdot \frac{L_{d\alpha}s + L_v Y_{d\alpha} - L_{d\alpha} Y_v}{s^3 - (Y_v + L_p)s^2 + (L_p Y_v - L_v Y_p)s - L_v g} \quad (17)$$

Controller and Filter Design

The SAS uses SISO controllers for the pitch- and roll-rate to set the damping and the natural frequencies of the phugoid and dutch roll. **Figure 26** depicts the control loop used to tune the controllers. Parametrized high- and lowpass-filters are added to attenuate rotor vibrations and remove trim values from the rate measurements, resulting in leaky controllers, which are less prone to saturation in the case of trim value changes and sensor bias. The low-pass (L), highpass (H), and resulting bandpass (BP) filter transfer functions are

$$G_L(s) = \frac{\omega_0^2}{s^2 + 2 \cdot \zeta \cdot \omega_L \cdot s + \omega_L^2}, \quad \omega_L = 2 \cdot \pi \cdot (1 \dots 5) \text{ rad/s}, \quad \zeta = \sqrt{2}/2, \quad (18)$$

$$G_H(s) = \frac{s^2}{s^2 + 2 \cdot \zeta \cdot \omega_H \cdot s + \omega_H^2}, \quad \omega_H = 2 \cdot \pi \cdot (0.01 \dots 1.28) \text{ rad/s}, \quad \zeta = \sqrt{2}/2, \quad (19)$$

and

$$G_{BP}(s) = G_H(s) \cdot G_L(s). \quad (20)$$

The natural frequencies in brackets show the possible values prepared in the FCC firmware.

The control laws use PI controllers:

$$K_q(s) = \frac{k_{qP} \cdot s + k_{qI}}{s} \quad (21)$$

for the longitudinal and

$$K_p(s) = \frac{k_{pP} \cdot s + k_{pI}}{s} \quad (22)$$

for the lateral motion.

The closed loop control transfer functions are

$$T_q(s) = \frac{G_q(s)}{1 + G_q(s)G_{Act}(s)K_q(s)G_{BP}(s)} \quad (23)$$

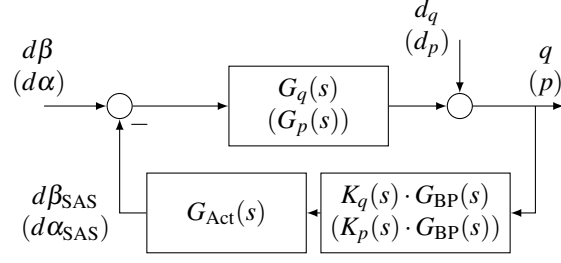


Figure 26: SAS control loops incl. filters and actuator dynamics for controller design.

and

$$T_p(s) = \frac{G_p(s)}{1 + G_p(s)G_{Act}(s)K_p(s)G_{BP}(s)}. \quad (24)$$

Note that the filter cutoff frequencies, helicopter Phugoid and Dutch Roll poles, and controller zeros all lie in a small frequency band between 0.01 Hz and 5 Hz. Therefore, the filter and actuator dynamics have to be included in the controller design, as illustrated in **Figure 26**.

The controllers are parametrized through pole placement, yielding the values in **Table 5**. The table also lists the resulting closed loop Phugoid and Dutch Roll parameters. Three controller configurations are part of the flight campaign. The corresponding closed loop Phugoid and Dutch Roll poles are shown in **Figure 27** along with the limits for predicted Handling Qualities (HQ) Levels 1, 2, and 3 according to ADS33E (Ref. 9).

Configuration 1 is a P-controller used to test the basic functionality and check for adverse effects due to possible actuator force feedback. This controller does not yield stable closed loop Phugoid and Dutch Roll modes in control theory terms, but the closed loop poles satisfy ADS-33E Level 1 predicted HQ.

Configuration 2a is a PI-controller that retains the damped natural frequencies of the Phugoid and Dutch Roll mode, but yields closed loop dampings of 0.5 for both modes.

Configuration 2b is a PI-controller that yields a damped natural frequency that is 60 % of the Phugoid and Dutch Roll mode of the bare airframe, and a damping of 0.5. The idea is to slow down the Phugoid and Dutch Roll whilst retaining the ratio between Phugoid and Dutch Roll frequency that the pilot is used to.

Figure 30 shows the bode diagrams of the sensitivity function $S_q(s)$ for the pitch control loop for a highpass filter frequency ω_H of 0.01 Hz and lowpass filter frequencies ω_L of 1, 3, and 5 Hz. Note a distinct disturbance rejection peak (DRP) at 3 rad/s for filter frequencies of 1 Hz (2π rad/s). At filter frequencies of 3 and 5 Hz, the DRP moves to approx. 5 and 7 Hz, respectively. **Figure 31** shows the same bode diagrams for the roll axis. On the

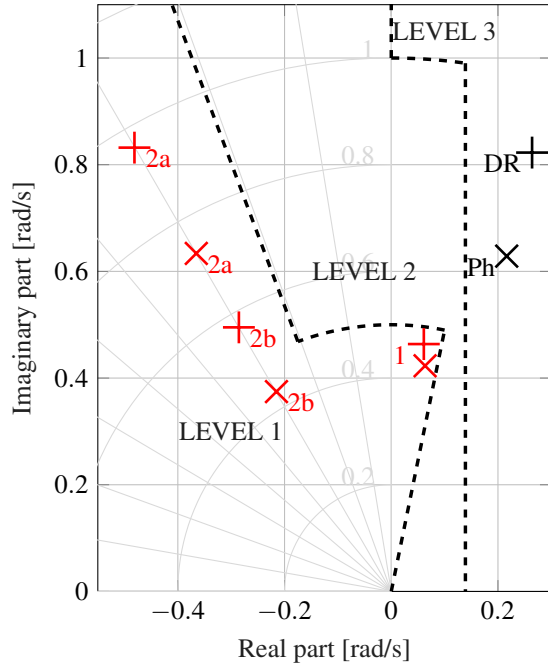


Figure 27: Phugoid ($\times$) and Dutch Roll ($+$) poles of the bare airframe (black) and the closed control loops (red)

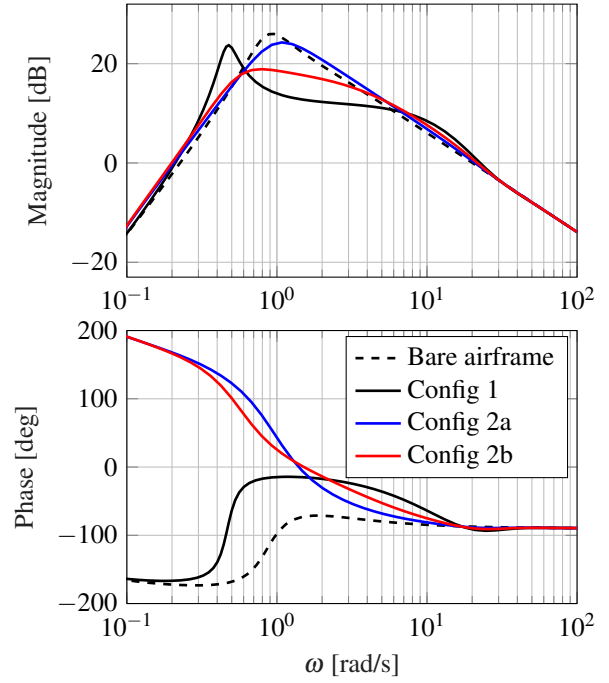


Figure 29: Bode plots of $G_p(s)$ (dashed) and $T_p(s)$ (solid) for the different SAS configurations

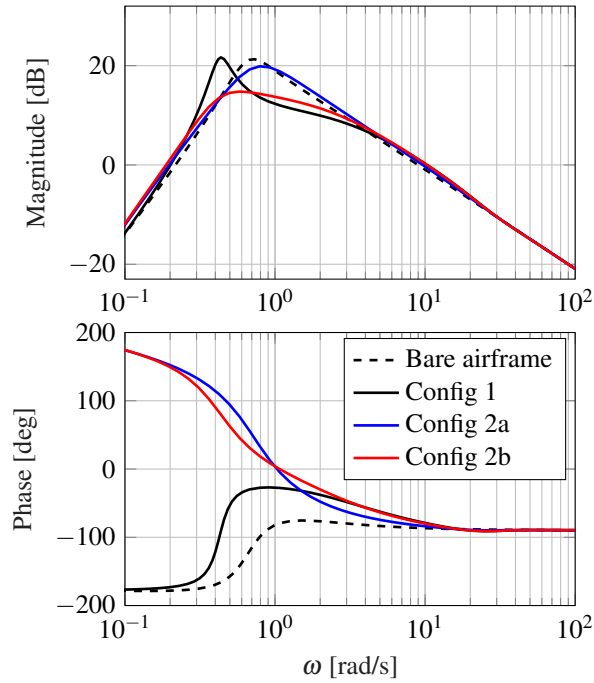


Figure 28: Bode plots of $G_q(s)$ (dashed) and $T_q(s)$ (solid) for the different SAS configurations

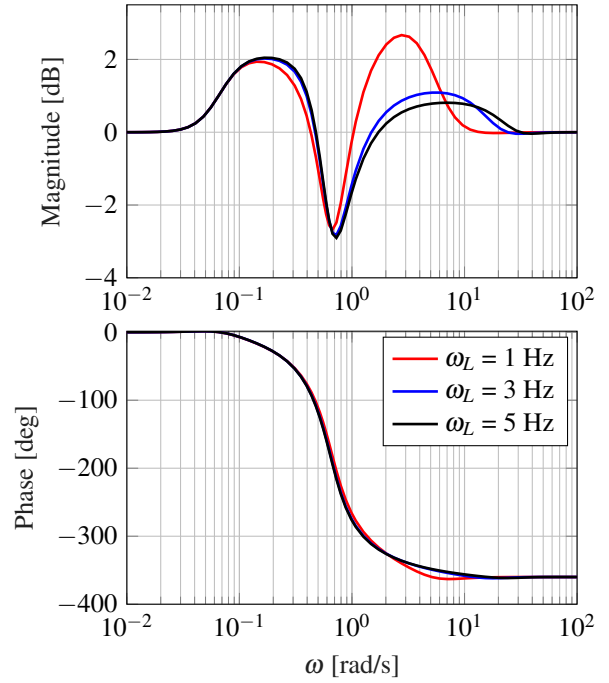


Figure 30: Bode plots of the pitch axis sensitivity function for SAS configuration 2b and lowpass filter frequencies from 1 to 5 Hz

Table 5: SAS configurations and closed loop poles

Roll Axis			
No.	Configuration	ω_e [rad/s]	ζ
0	Bare airframe	0.82	-0.31
1	P-controller	0.46	-0.15
2a	PI-controller	0.82	0.5
2b	PI-controller	0.49	0.5
Pitch Axis			
No.	Configuration	ω_e [rad/s]	ζ
0	Bare airframe	0.63	-0.33
1	P-controller	0.42	-0.13
2a	PI-controller	0.63	0.5
2b	PI-controller	0.38	0.5

roll axis, the DRP is even more pronounced at $\omega_L = 1$ Hz. This is consistent with testing in the helicopter. In conditions where the flight demonstrator faced translational lift, lowpass filter cutoff frequencies of 3 Hz or less lead to oscillations in the control system which were disturbing the pilot. A lowpass filter frequency of 5 Hz struck the best balance between the suppression of rotor vibrations and the prevention of undesired resonances and was kept for all further testing.

Allocation

The control laws output equivalent cyclic inputs. To distribute them among the actuators, an allocation function is needed. Linearizing the control train kinematics were given in Equation 7. Calculating the inverse of the effectiveness matrix and ignoring the influence on collective inputs yields Equation 25, which is implemented on the FCC. Note that – despite the non quadratic allocation matrix – the primary control train as a whole is not an over-actuated system. If the roll actuators move in the same direction, they make a collective input. The column corresponding to collective inputs is not shown here for the sake of simplicity.

$$\begin{bmatrix} s_1 \\ s_2 \\ s_3 \end{bmatrix}_c = \begin{bmatrix} 1/k_{d\beta} & 0 \\ 0 & 1/k_{d\alpha} \\ 0 & -1/k_{d\alpha} \end{bmatrix} \begin{bmatrix} d\beta_{SAS} \\ d\alpha_{SAS} \end{bmatrix}_c \quad (25)$$

SAFETY CONCEPT

The safety concept covers technical measures, operational measures, and flight test preparations, each of which are described briefly in the following paragraphs. Proofs of safety are brought forward through formal methods adapted from ARP2754 (Ref. 10) and practical testing in the tethered rotorcraft. As the current German regulation prohibits any actuator from being mounted in the primary control train of ultralight rotorcraft, extensive coordination and communication with the German

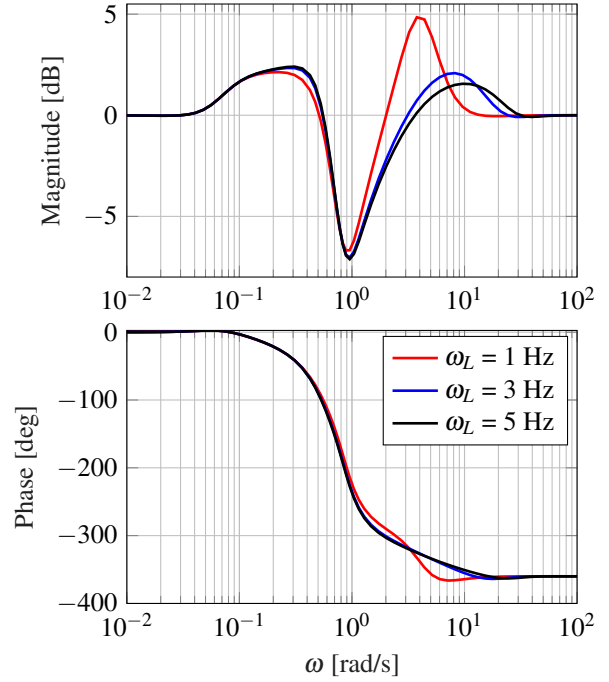


Figure 31: Bode plots of the roll axis sensitivity function for SAS configuration 2b and lowpass filter frequencies from 1 to 5 Hz

Luftfahrtbundesamt (LBA) and the German Association for Ultralight Aircraft (Deutscher Ultraleichtflugverband, DULV) were necessary to obtain a preliminary type certificate (PTC) for the flight demonstrator. The following paragraphs give a brief description of the safety measures and the proofs of safety required for the PTC.

Technical Safety Measures

- The SAS in the flight demonstrator is separated from the existing avionics. It uses its own CAN buses and runs on its own power supply. If an electrical error occurs in any of the SAS components, it cannot affect the stock avionics in the rotorcraft required for safe operation.
- The interfaces for the pilot are intuitive and designed for hands on collective and stick (HOCAS) operation to minimize the reaction time in the case of a system failure. Two buttons on the cyclic stick trigger the activation and deactivation of the SAS. Information about the system status is provided to the pilot in a concise manner via four LEDs on the dashboard.
- The pilot can always deactivate the stabilization at any time by pushing the kill-switch with his left leg. Pushing the stick instantly locks the actuators at their current position.

- All other interfaces and subsystems of the SAS are designed to be operated by the flight test engineer so that the pilot can concentrate on flying, assessing the HQ, and locking the actuators if necessary. He can monitor the SAS using a combined switch-and display-device on the left side of the cockpit. The device includes a dedicated status display that indicates the actuator position, the current mode of operation and any error that might occur in the complex system, thus removing the need for a laptop in the cockpit.

Operational Safety Measures

- The flight demonstrator is designed for two people operation: This way, the increase in pilot workload that arises from operating the SAS is kept at a minimum. This enables safe and efficient testing. All tasks not directly connected to piloting the helicopter are attended to by the flight test engineer. He selects different stabilization modes and controller settings through an additional interface in the cockpit, monitors the system status through a dedicated status display, keeps track of the testing schedule, interviews the pilot after each test item, takes notes, and controls the logger.
- To limit the safety effect of system failures, the flight envelope is limited to hover in ground effect at speeds no greater than what is required for taxiing at airfields.
- Testing is performed at a large airfield with little traffic, leaving a lot of safety clearance in all directions.
- Testing is limited to good visual environments (GVE) and reasonably calm wind conditions.

Flight Test Preparations and Training

To prepare for the flight demonstration, a three step training program is developed to ensure all members of the crew are prepared in the best possible manner. Step 1 includes briefings about the system architecture, nominal operation, possible failures and appropriate action in the event of a failure. Step 2 includes tests in a virtual reality flight simulation to illustrate and review the concept of operation, the effect of the stabilization on HQ, and the effect of system failures. The flight simulator is based on a converted edm CoAX 2D and is described thoroughly in (Ref. 11). For step 3, the flight demonstrator with the SAS is mounted in a metal hover frame equipped with hinges that limit the pitch and roll angles but enable rotation around all three axes of motion and enable hover



Figure 32: The tethered helicopter used in the safety trials

in ground effect whilst adding sufficient inertia to prevent the helicopter from tipping over or climbing higher than 2 m above ground level. The tethered helicopter is shown in Figure 32. This provides a platform to test the SAS safely under real circumstances and test the system functionality, give the pilot an impression of the feedback forces from the actuators in nominal operation, to check for undesired resonances in the augmented control system and to train the crew for system failures. In addition, the tethered helicopter is used to prove that the system is safe under all assumptions taken in the formal safety assessment, i.e. to prove that all mitigating factors and assumptions are justified. The hover frame is described thoroughly in (Ref. 2).

Proof of Safety

To prove that the safety measures are sufficient, the tethered flight demonstrator is used. To simulate the relevant failures identified by the formal safety analyses, the flight control computer is flashed with a custom firmware with most of the failure detection functionality turned off. Instead, the firmware includes the possibility to trigger system failures through a laptop connected via CAN. Moreover, actuator motion range is set to the maximum possible value of ± 5 mm, corresponding to an authority approx. 19 % on the pitch axis and 21 % on the roll axis to ensure the system is safe even at its maximum authority. This way, the authority can safely be increased later beyond 10 % if necessary for the flight test. Figure 33 shows a cockpit view from the safety trials. A Simulink model running in realtime enables communication with the FCC, manually moving the actuators, and simulating failures. The laptop running the model is shown on the left side of Figure 33.

In the formal safety analyses, the actuators were identified as the most critical component of the SAS. There-



Figure 33: Failure simulation through laptop interface in the tethered flight demonstrator

fore, the safety tests focus on actuator failures. Recall the three relevant failure modes for the actuators are

1. **Power loss** yielding a loss of force fit in one or more actuators
2. **Out of control failure** involving one or more actuators moving erratically and possibly destabilizing the helicopter
3. **Undetected failure** involving one or more actuators slowly drifting, changing the required trim input over the course of several seconds

In the test campaign, each of these cases was tested in all possible actuator combinations and with and without an announcement before triggering the failure. The following paragraphs give a brief description of the relevant failure tests.

Power Loss: To simulate a power loss, the power module of the actuators is turned off via CAN. This is shown for a simulated failure of actuator 3 in Figure 34. In this test, actuator 3 is turned off at $t = 5$ s. The collective feathering forces slowly compress the failed actuator, yielding a slow shift in cyclic roll trim. It takes 18.8 s until the actuator reaches the end of its motion range, which corresponds to a roll input shift of 0.53 % of pilot authority per second. The asymmetry in roll input results in a 9 % decrease of collective input, corresponding to -0.48 % of pilot authority per second. These figures give the pilot plenty of time to detect and react to the actuator failure. Also note that, in the flight test campaign, the FCC includes functionality to detect and display actuator power loss or tracking errors so the pilot can react immediately. In the case of the pitch actuator, the control forces are not sufficient to move the actuator ballscrew at all. Apart from the pitch stabilization being inactive, there are no safety effects. The position of actuator 1 during a simulated power loss is shown in Figure 35.

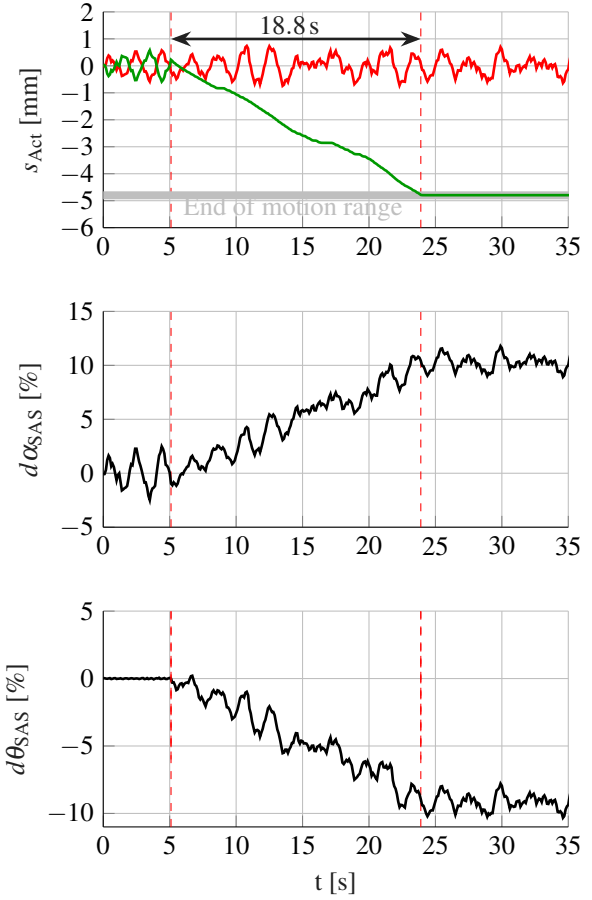


Figure 34: Simulated power loss in actuator 3 while SAS is active: Position of actuator 2 (red) and 3 (green), SAS roll input $d\alpha_{SAS}$, and collective input $d\theta_{SAS}$

Out of control failures: To simulate an out of control (OOC) failure, erratic actuator setpoints are generated on the laptop and sent to one or more actuators. The test signals are implemented as lowpass filtered random numbers. The signal gain is adjusted to cover the whole actuator motion range and occasionally saturating the actuator. In situations where both the roll and the pitch axis are tested, different seeds are used to yield uncorrelated

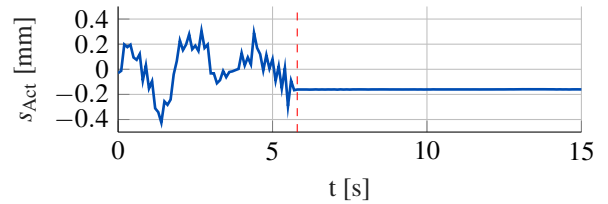


Figure 35: Simulated power loss in actuator 1 while SAS is active: Actuator 1 position

In all cases, the pilot was able to detect the failure before the helicopter motion was affected significantly, either as an absence of the stabilization or as a slow drifting similar to a shift in wind direction. In the case of an undetected failure, it took the pilot longer to detect the error, but with no safety effect. To summarize the tests, even with the failure detection algorithms turned off, none of the simulated failures resulted in excessive pilot workload or hazardous situations.

In the testing campaign, all maneuvers are flown at a height above ground between 0.5 and 1.5 m (1.6 and 4.9 ft) to limit the safety effects of a failure. In this paper, flight data from stationary hover, doublet inputs, and the Hover MTE is shown to give an impression of hover

Table 6: Flight characteristics rating scale

Rating	Flight Characteristics
3	a lot better
2	better
1	a little better
0	unchanged
-1	a little worse
-2	worse
-3	a lot worse

Table 7: Force feedback rating scale

Rating	Force Feedback	
	Severity	Annoyance
3	very noticeable	very annoying
2	noticeable	annoying
1	barely	a little annoying
0	not noticeable	not annoying

flight instructor for the CoAX. However, he is not a certified test pilot and has no experience working with HQ rating scales. Therefore, a proprietary rating system is used, where the stock CoAX 600 is the baseline. The rating system is described in the next paragraph.

Rating Scales

To quantify the feedback from the pilot and compare the different control laws, proprietary rating scales are used. The approach is similar to the one taken in the tethered flight demonstration (Ref. 1) and covers the flight characteristics and the severity and workload impact of the actuator force feedback. Three rating scales are used. Table 6 shows the light characteristics rating scale. Note that this is a relative rating scale where the unstabilized helicopter is the baseline. Table 7 shows the two scales used to assign a rating for the actuator force feedback. One of the scales covers the severity of the force feedback, the other quantifies if the force feedback affects pilot workload.

FLIGHT TEST RESULTS

Flight Test Data

The following paragraphs include flight test data and highlight the differences between flights with the SAS off and on. Each maneuver is discussed separately, focusing on roll and pitch oscillations and pilot inputs.

To compare pilot workload between maneuvers, frequency spectra of the pilot inputs are used. Following Zander et al., pilot activity is analyzed separately for different frequency bands (Ref. 4). Guidance (0 - 0.5 Hz), crossover (0.5 - 1.0 Hz), and stabilization (1.0 - 2.5 Hz) bands are used in this analysis.

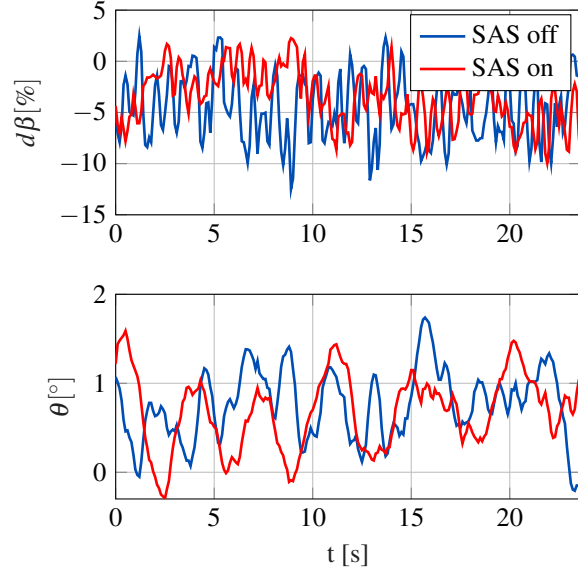


Figure 38: Stationary hover with and without SAS configuration 1 – frequency spectra of pilot and SAS pitch inputs and pitch rates

Stationary Hover: Figure 38 and 40 show the pitch and roll motion in stationary hover with and without the SAS in configuration 1 (P-controllers). Roll inputs range from -100% (full left) to 100% (full right), pitch inputs range from -100% (full fwd) to 100% (full aft). For both axes, the amplitude of pilot inputs and angular oscillations is smaller with the SAS active. The reduction is more pronounced on the roll axis. Figure 39 and 41 show the frequency spectra of pilot inputs, actuator motion, and angular rates for both axes in the same hover maneuver stationary hover.

With the SAS turned on (red lines), cyclic pitch pilot activity between 0.3 and 1.3 Hz is reduced significantly, covering the upper end of the guidance band, the crossover band and the lower end of the stabilization band, as shown in the upper part of Figure 38. Pitch oscillations are reduced between 0.4 and 1.2 Hz with the SAS on (bottom part of Figure 38), but there is a peak in the frequency spectrum at 0.3 Hz, which has to be investigated further. On the roll axis, cyclic pilot inputs are less severe between 0.3 and 1.7 Hz, also covering the upper end of the guidance band, the crossover band and the lower end of the stabilization band. Roll oscillations are reduced between 0 and 1.5 Hz, as shown in the bottom part of Figure 41.

Pitch Doublet: To verify the on-axis stabilization by the SAS and to check for cyclic cross couplings and adverse effects with and without stabilization, doublet inputs are used. As an example, a pitch doublet without stabilization is compared to one with SAS configuration 1 engaged. Figure 42 shows the pilot input and the pitch an-

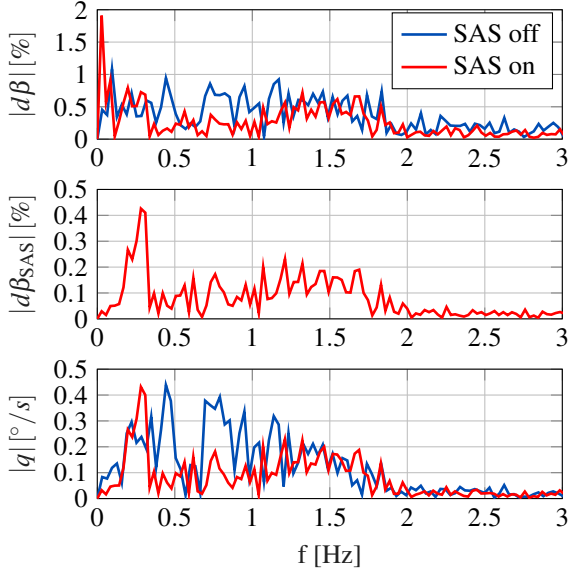


Figure 39: Stationary hover with and without SAS configuration 1 – frequency spectra of pilot and SAS roll inputs and roll rates

gle during the maneuver. From stationary hover, the pilot initiates the maneuver by moving the stick from $\approx 10\%$ to $\approx 40\%$ at $t = 2.3$ s. Due to the feedback forces and vibrations, the plot differs from a textbook doublet, but it serves its job in exciting the pitch motion. Figure 43 shows the cyclic roll input and roll angle during the same maneuver. Without stabilization, severe oscillations are visible in both the roll inputs and the roll angle. With the SAS engaged, these cross couplings are reduced drastically.

Hover MTE (adapted from ADS33E): In this maneuver, the rotorcraft starts in diagonal forward flight at a speed between 3.1 m/s and 5.1 m/s (6 and 10 knots) towards a defined spot on the tarmac marked with five sets of cones (Figure 36). The goal is to decelerate with moderate aggressiveness and hover at this spot for 30 seconds. Figure 44 shows the cyclic pitch inputs from the pilot and the SAS and the pitch angle during the maneuver. Figure 45 shows the same data for the roll motion. For better comparability between the data, the trim pitch angles were subtracted from the data. Both with and without the SAS, it takes between 10 and 12 seconds to stabilize the helicopter after the deceleration is initiated. In both the pitch and roll angle data, an oscillation with a frequency of approx. 0.5 Hz occurs phase between $t = 5$ and 14 s with the SAS turned off. With the SAS turned on, this oscillation still exists on the roll axis, but with a lower magnitude (red line in the bottom plot of Figure 45). On the pitch axis, the oscillation is gone completely with the SAS turned on (red line in the bottom plot of Figure 44).

Figures 46 and 47 depict the frequency spectra of pilot

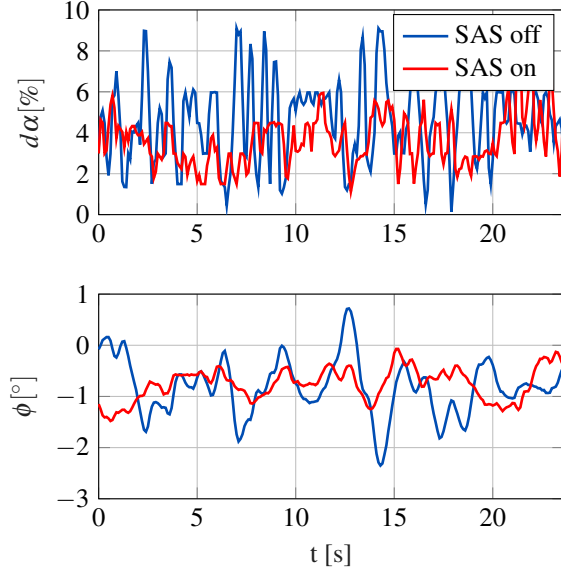


Figure 40: Roll motion during hover with and without SAS configuration 1

and sas inputs as well as pitch and roll angles during the hover phase. Similarly to the hover data of SAS configuration 1 (Figures 38 - 41), both the activity at the cyclic stick and the angular helicopter motion are attenuated if the SAS is active.

Pilot Feedback

Pilot feedback from the test campaign includes ratings using Table 6 and Table 7 on one hand and additional comments after each maneuver on the other.

Table 8 lists the average pilot ratings for each maneuver and SAS configuration tested. Note that configuration 2a was not rated because pilot feedback after the first tests was that the rotorcraft felt twitchy with configuration 2a. It was therefore not included in further tests and – instead – configuration 2b was added.

For the hover MTE hover phase, an additional rating is given in brackets. In this trial, the performance was worse with the SAS, but this test took place in a particularly gusty moment and the rating is therefore considered an outlier. This underlines the necessity to revisit the filter design to mitigate the SAS reaction to gusts.

The following additional feedback was given by the pilot:

1. The helicopter is steadier with the SAS active. This is especially noticeable in hover.
2. The actuators dampen the cyclic stick. Without stabilization, rotor vibrations due to gusty wind are more severe if the filters are tuned properly.

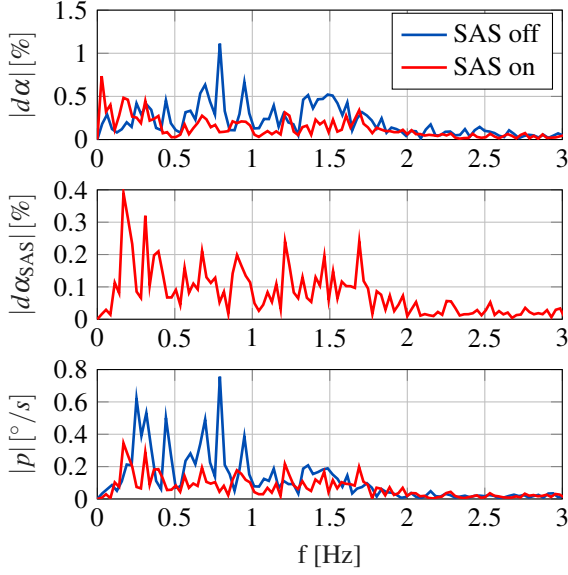


Figure 41: Frequency spectra of roll motion during hover with SAS configuration 1

Table 8: Pilot ratings from the flight campaign with configuration 2b (outliers in brackets)

Maneuver & Phase	Flight char.	Rating Forces	
		Severity	Annoyance
Stationary hover	1	0	0
Hover MTE			
· approach (3x)	0	0.5	0
· deceleration (5x)	0.5	0.5	0
· hover (5x)	1 (-1)	0.5 (2)	0 (1)

3. In slow forward flight with sufficient headwind, translational lift in ground effect yields vibrations at the rotor, in the control system, and subsequently in the SAS control loop, which are disturbing.
4. In slow forward flight with a tailwind, both the helicopter and the control system behave very smoothly.
5. In direct comparison across all maneuvers, SAS configuration 1 was the steadiest. Configuration 2a was too twitchy, configuration 2b is a compromise between the other two.

Discussion

Flight Dynamics: The previous section showed flight data from doublet maneuvers, stationary hover, and hover MTE trials inspired by ADS-33E. For the latter two, pilot ratings were given. In most cases, test data indicates that both the cyclic stick and the angular helicopter motion

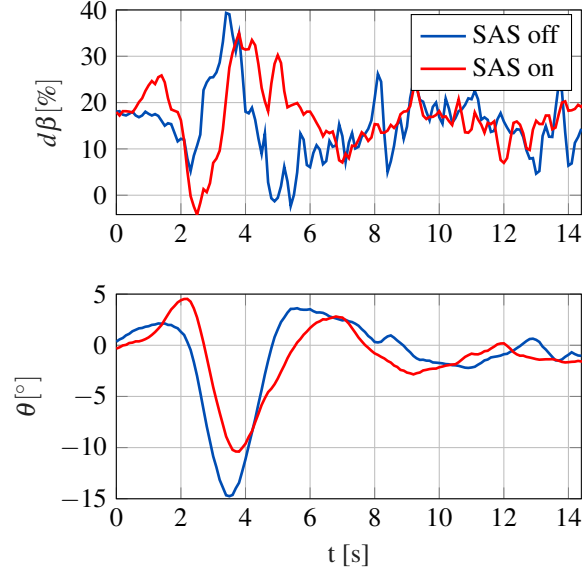


Figure 42: Pitch doublet with and without SAS configuration 1 – pitch inputs and pitch angle

exhibit less oscillations when the SAS is active. This stabilization in most situations is consistent with pilot feedback saying that the rotorcraft felt steadier with SAS configurations 1 and 2b. The numbers given for the rating scales are however very similar between the configurations, leaving room for interpretation. Moreover, configuration 1 felt the steadiest for the pilot, whereas – in theory – configuration 2b should yield a higher closed loop damping than configuration 1. This discrepancy between theory and pilot feedback could be due to inaccuracies in the plant models (Equations 13 and 15), yielding different closed loop poles in reality, or due to pilot bias. Regarding authority, there is still headroom for an increase of the controller gains to improve the performance.

Note that – due to the mechanical control system – the oscillation mitigation achieved by the SAS does not automatically mean that the workload is lower. Unlike in hydraulically driven control systems, the oscillations observed here do not solely come from pilot inputs but also directly from the rotor. The impact of control train damping through the actuators has yet to be investigated. Moreover, any damping achieved this way heavily relies on the right phase shift between existing vibrations and actuator motion, which warrants further investigation.

Force Feedback: According to the pilot, the actuator force feedback was only noticeable occasionally and only disturbing in a few rare instances. In hover and in situations with a tailwind, the SAS actuators even filtered the rotor forces out of the control train, yielding less vibrations at the cyclic stick if the SAS was active. With the force feedback being barely noticeable, there is room for a more aggressive SAS configuration to improve closed

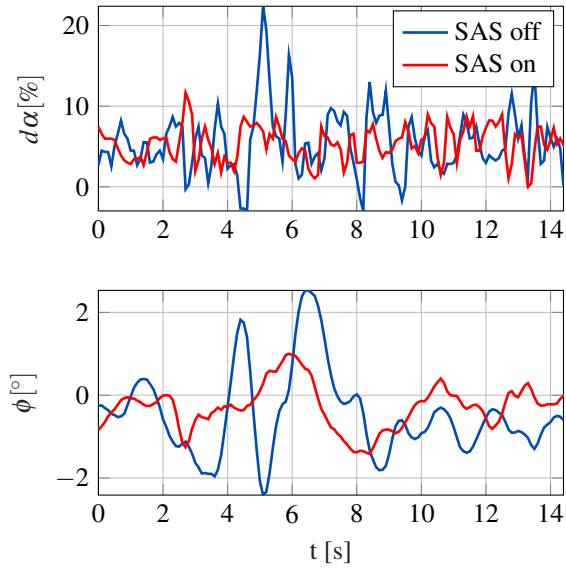


Figure 43: Pitch doublet with and without SAS configuration 1 – roll inputs and roll angle

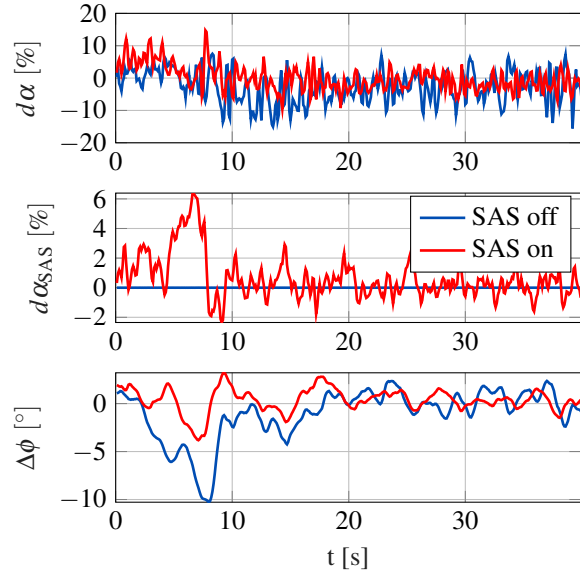


Figure 45: Hover MTE with and without SAS configuration 2b – roll inputs and roll angle

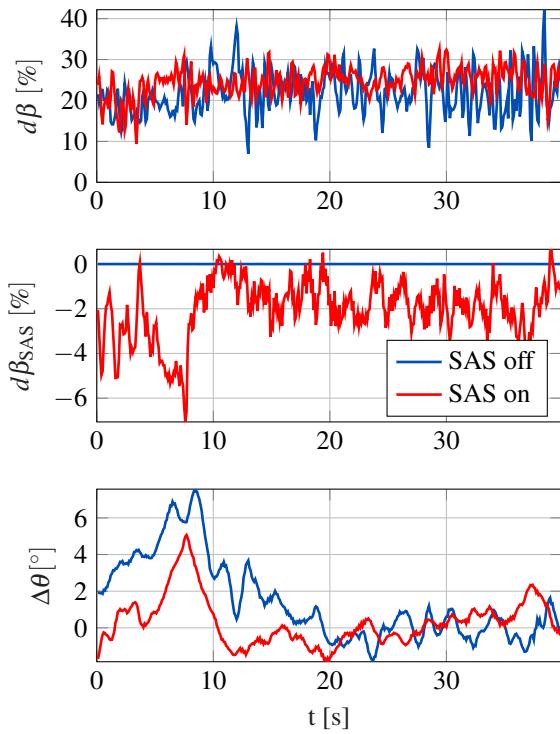


Figure 44: Hover MTE with and without SAS configuration 2b – pitch inputs and pitch angle

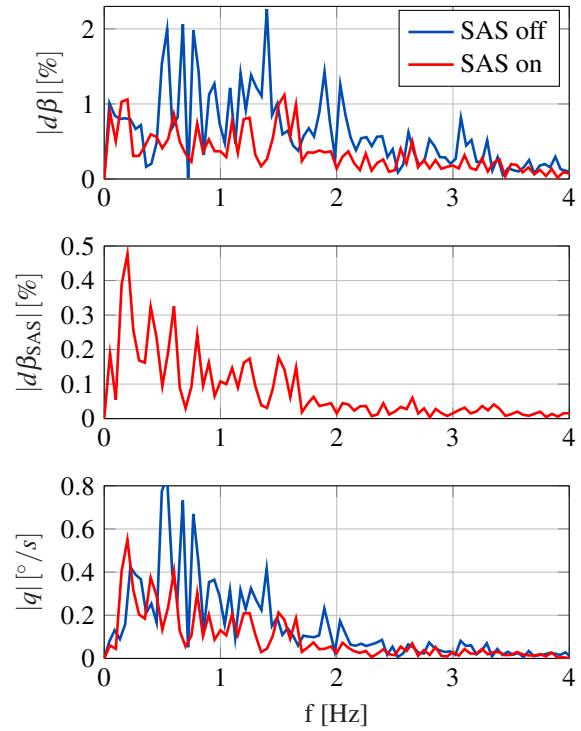


Figure 46: Hover MTE with and without SAS configuration 2b – Hover phase – frequency spectra for the pitch axis

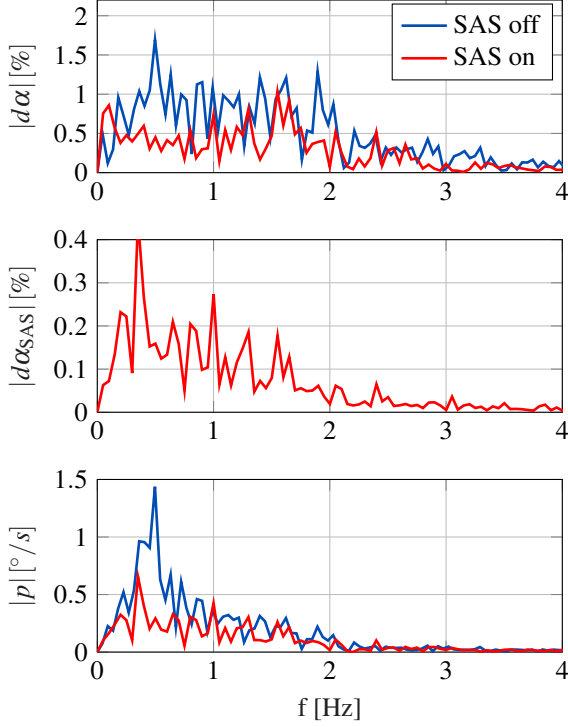


Figure 47: Hover MTE with and without SAS configuration 2b – Hover phase – frequency spectra for the roll axis

loop performance.

However, the filters and control laws have to be parametrized carefully and as a whole to achieve this and to prevent resonance peaks like shown in Figures 30 and 31. If not, vibrations that occur at the rotor can be fed through the control train and into the SAS control loop, yielding an increase in cyclic stick oscillations. Due to the inline architecture that connects the actuators directly to the cyclic stick, this is particularly important to prevent cyclic stick vibrations induced by the actuators. This close interaction between the actuators and the cyclic stick is unique to this architecture and warrants closer inspection.

CONCLUSION

This paper presents the design, implementation, certification, and successful flight testing of the first purely electromechanical SAS for rotorcraft with no hydraulic system. The following conclusions can be drawn from the results in the previous sections, answering the three questions from the introduction:

1. The proposed SAS can be integrated into an ultralight rotorcraft despite the limited mass, space, and power budgets. Further downsizing is possible, e.g. by using the existing power supply in the helicopter

instead of a dedicated battery and by replacing the electric rack by a smaller electronics box.

2. The system stabilizes the helicopter, the feedback forces are only noticeable occasionally, and do not adversely affect pilot workload. Even with only 10 %, actuator saturation is not an issue in hover maneuvers. There is room for improvement regarding the control laws and filters, but the architecture has proven to be effective, even with an authority of only 10 %. Regarding both authority and severity of feedback forces, there is headroom for an increase in aggressiveness.
3. With the addition of a locking mechanism, flight safety is ensured in normal operation and for all relevant system failures, as shown both formally and in practice.

The contributions of the presented work are:

1. For the first time, a rotorcraft with a fully mechanical control system was equipped with an SAS using electromechanical inline actuators and successfully tested in flight. The flight demonstration shows that the concept is effective in flight and mitigates pilot workload.
2. The flight trials identified a subset of relevant system parameters that hinder SAS performance if not parametrized properly. The knowledge about these pitfalls will contribute to future experiments.
3. The flight demonstration results show that electromechanical inline actuators are effective in improving the HQ of rotorcraft with fully mechanical control systems. Pilot feedback and measurement data indicate that the architecture is effective in stabilizing the rotorcraft in hover and the feedback forces do not impede the desired workload mitigation.
4. The certified hardware architecture enables future research including optimization of the SAS control laws, testing other control approaches, increasing automation, and expanding control to all four axes of the primary control train.

APPENDIX

SAS States

Table 9 lists the SAS modes shown in Figure 25 and assigns a number that is displayed on the display in the top right corner in Figure 22.

Table 9: SAS modes

No.	Mode & System Behavior
0	Waiting for Init - All subsystems are on - Actuator motors have no power
1	Init - Actuator homing is in progress - Initialization LED flashes - Stabilization LEDs flash alternately
2	Idle - Actuators are centered - Actuator motors are powered on - All LEDs are off - Engage button is off
3	Armed - Actuators are centered - Actuator motors are powered on - Stabilization LEDs flash - Engage button is active
41	Fading in - Stabilization is active - Stabilization LEDs are on - Actuator commands are faded in
42	Active - Stabilization is active - Stabilization LEDs are on
43	Fading out - Stabilization is active - Stabilization LEDs are on - Actuator commands are faded out
255	Error - Actuators are stopped at current position - Actuator motors are powered on - Error LED is on, other LEDs are off - If Locking is active, locking LED is on

SAS Errors

If the FCC detects an error in one of the subsystems, the system switches into mode 255 (Error). Table 10 lists the error codes displayed in the top left corner of Figure 22 if an error is detected. In the case of an internal actuator error (11, 21, 31), the internal actuator error code is displayed in the bottom left corner of the display unit.

Table 10: Error codes

Code	Subsystem	Meaning
1	Locking system	Killswitch is pressed
2		Mechanism is locked
11	Act. 1	Internal error
12		Encoder and analog sensor mismatch
13		Initialization error
14		Actuator telemetry timeout
15	Act. 2	Actuator is not centered in IDLE mode
16		Actuator tracking error
21		Internal error
22	Act. 3	Encoder and analog sensor mismatch
23		Initialization error
24		Actuator telemetry timeout
25		Actuator is not centered in IDLE mode
26	Gyro roll axis	Actuator tracking error
31		Internal error
32		Encoder and analog sensor mismatch
33	Gyro pitch axis	Initialization error
34		Actuator telemetry timeout
35	Switches	Actuator is not centered in IDLE mode
36		Actuator tracking error
62	Buttons	Value out of range
64		Timeout
72	Buttons	Value out of Range
74		Timeout
84	Buttons	CAN-Timeout
94		CAN-Timeout

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