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ELECTRO-MECHANICAL ACTUATORS FOR GENERAL AVIATION FLY-BY-WIRE AIRCRAFT

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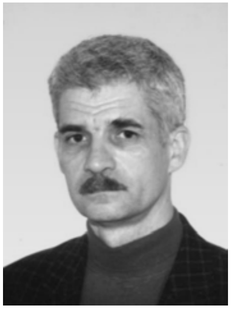

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Abstract. A long-term goal for general aviation aircraft is to reduce or eliminate the use of fluid power systems such as hydraulics from the aircraft. Power-By-Wire (PBW) technology seems to be the next major step in the development of aircraft control. In this solution, control power comes directly from the aircraft electrical system to the Electro-Mechanical Actuator (EMA), which includes the electric motor, controller and gearbox. EMAs have the potential to be more efficient, less complicated, less expensive, and more faults tolerant than actuators based on hydraulic systems.

The construction of a single EMA and set of cooperating electro-mechanical servos actuating airplane control surfaces is presented in this work. The mechanical unit, microcontroller of the EMA, and general information on control algorithms are discussed in this work. Testing methods and laboratory results are also described. Then the in-flight testing results of the complete set of servos installed as a part of a SPS-1 system on board an airplane are presented. SPS-1 is an experimental Fly-By-Wire (FBW) control system for general aviation aircraft designed as a project financed by the Polish State Committee for Scientific Research (2004-2005).

Keywords: electro-mechanical actuator, EMA, controller, fly-by-wire, power-by-wire, flight tests.

Introduction

Fly-By-Wire (FBW) control systems have been used in military and industrial solutions over the last 15 years. The next, natural step seems to be using these systems in general aviation aircraft [4, 8, 9]. This technique extends the possibilities of aircraft control and the stabilization of flight parameters and allows for better, more flexible automatic control. Pilots can do their duty either in a computer operator fashion, or they can control the plane classically by stick. The application of the FBW technique would both make flying safer and minimize pilot workload. The principle of FBW operation is to convert pilot activity (stick movements) into electrical signals. These signals are then sent to flight computers. They will in turn process them and generate signals input into an adequate actuator controller where discrete information is changed to a control surface deflection. This process is performed without any direct mechanical connections between the control lever and control surfaces.

The intricate systems found in large and medium-sized airplanes mostly have hydraulic actuators. Electro-Mechanical Actuators (EMAs) are only applied for quick and accurate control of the boosters. It would however be difficult to install heavy, expensive hydraulic systems on small planes. In most cases it is just impossible. Thus it seems to be rational to use EMA circuits to control the positioning of aerodynamic controls and other control system elements, e.g. power control.

Power-By-Wire (PBW) technology seems to be the next major step in development of aircraft control. Fly-By-Wire flight control systems eliminated the need for mechanical interfaces and Power-By-Wire actuators eliminate the need for central hydraulic systems [6].

Solution of SPS-1 EMA

The EMA presented consists of two general parts (Fig 2) [2, 9]. The first is an electro-mechanical unit (Fig 1) which includes electric engine, gearbox, electromagnetic clutch, frictional clutch, driving roller, and rotary potentiometer. The electromagnetic clutch is used to disconnect the worm gear and roller. The additional frictional clutch is assembled with the gearbox and it disengages the driving roller when the admissible load is exceeded. The rotary potentiometer (connected to analog/digital converter of microcontroller) measures driving roller angle of rotation and realizes electro-mechanical feedback.

The second part of the EMA is the controller (Fig 3). This part is connected directly to the electrical system of the aircraft and the electrical engine is powered through a built-in power output. The microprocessor sets only *power* and *direction* lines into required status and then the power output supplies power to the motor. The controller gets the information on actuator load through a load line. The information on actual deflection is provided to the controller directly from the potentiometer. Control signals (δ_{des}) are delivered to the microcontroller from three independent flilght computers, and one extra line is connected directly to the side stick (emergency control).

The controller is based on microprocessor equipped with a watch-dog timer, an A/D converter, three I/O ports, and a capture compare module used for catching PWM's. The microprocessor is galvanically isolated from input and output circuits by transoptors. Power supply is from DC/DC converter. It is assured by a security chip powered from the power lines of the aircraft. Feedback from deflection is realized by a potentiometer installed in the electromechanical unit and powered by the controller circuit.

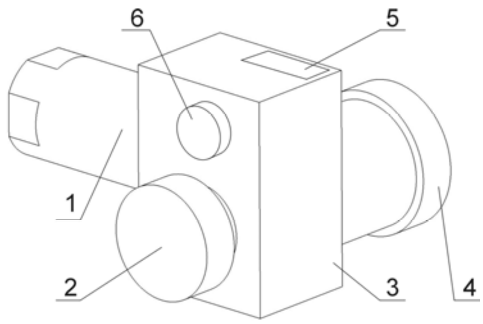

Fig 1. General view of electro-mechanical unit

1 – electric engine,
2 – driving roller,
3 – gearbox,
4 – electromagnetic clutch,
5 – electrical connectors (potentiometer to A/D converter, motor powering, powering of electromagnetic clutch),
6 – potentiometer.

EMA design specification:

- maximum working load 250 [N],
- linear displacement of strand ± 30 [mm],
- peripheral speed of roller 0.03 [m/s],
- precision of position determination 0.2 [mm],
- supply voltage 12 [DVC],
- drive: direct-current motor (brushless),
- restricted temperature range (50°C, + 80°C),
- allowable work in dustiness and high humidity conditions.

δ_{des} – desired deflection of control surface [rad],
U_{eng} – engine powering [V],
δ – actual deflection of control surface [rad],
d – displacement of strand [mm].

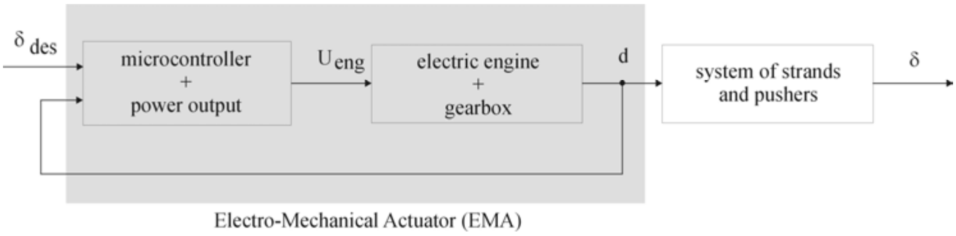

Fig 2. Simple block scheme of EMA

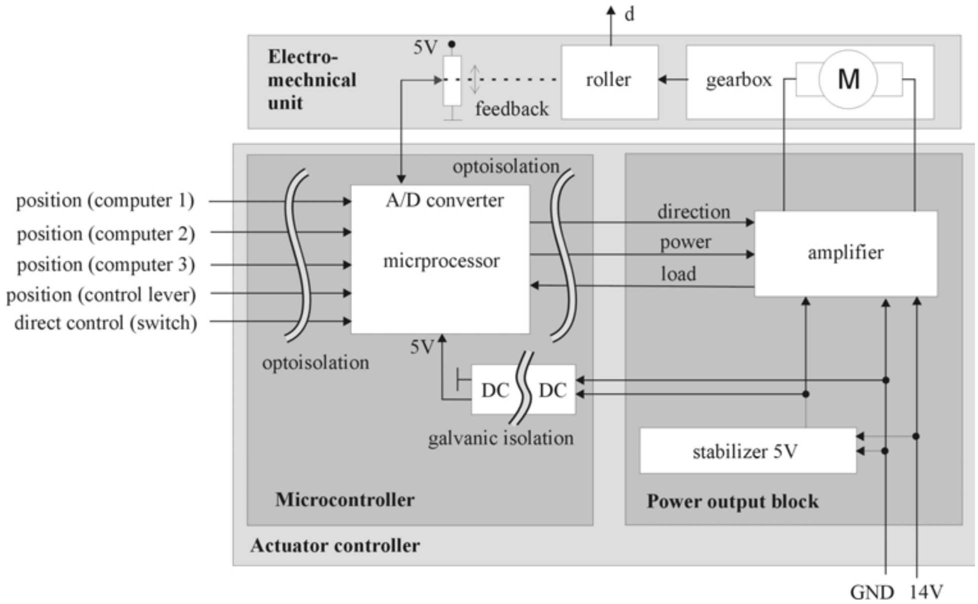

Fig 3. Controller hardware structure

Control algorithm

A heuristic regulator has been applied in the controller solution [3]. It uses expert established dependence between motor powering and actuating error. Heuristic regulator has good static properties and good dynamics under rate saturation. For over rate saturation, an additional phase compensator is needed. Realization of this algorithm is very simple. The dependence between motor powering and actuating error can be formulated as (1)-(3):

$$y_h(k+1) = y_{h\_p}(k+1) + y_{h\_i}(k+1) \tag{1}$$

$$y_{h\_p}(k+1) = kp(e(k)) \tag{2}$$

$$y_{h\_i}(k+1) = \begin{cases} y_{h\_i\_\max}(e(k)) & \text{if} \quad y_{h\_i}(k) + ki(e(k)) \cdot e(k) \cdot T_d > y_{h\_i\_\max}(e(k)) \\ y_{h\_i\_\min}(e(k)) & \text{if} \quad y_{h\_i}(k) + ki(e(k)) \cdot e(k) \cdot T_d < y_{h\_i\_\min}(e(k)) \\ & \text{otherwise} \quad y_{h\_i}(k) + ki(e(k)) \cdot e(k) \cdot T_d \end{cases} \tag{3}$$

T_d - sample time,
k - period of time,
y_h (k) - output signal of heuristic regulator,
y_{h_p} (k) - proportional component of output signal,
y_{h_I} (k) - integral component of output signal,
e (k) - actuating error,
kp (e(k)) - thresholds for gain in proportional path (chosen by expert),
ki (e (k)) - thresholds for gain in integral path (chosen by expert),
$y_{h\_i\_max}$(e(k)) - maximum value of integral component of output signal (dependent on actuation error).

Safety and reliability

As actuators are critical elements of a FBW system, they should be subject to rigorous qualification criteria [1, 7]. They have to be reliable and stable in any situation. This is a very important requirement, because any actuator failure is likely to lead to a serious accident [3]. Any actuator controller should be made intrinsically reliable and insensitive to other equipment failure. The second feature can be achieved by the multiplication of superior control circuits (flight computers). The pilot still should have to be able to deflect control surfaces directly by control lever in an emergency (fig 3, 4).

It has been assumed as a standard that input signals should be able to detect a short circuit or broken external circuit. To achieve this, a pulse width modulation (PWM) signal with two constant zones, 0 and 1 has been adopted. Filling PWM input full is considered as short circuit and a steady zero, as a break (fig 5). Control signals from flight computers are compared and the most believable is selected. This task is performed by software. If all signals from flight computers are wrong, control surfaces are frozen in the last position, and then a pilot via his stick can deflect the controls. The pilot can switch control into stick input at any moment flight (DIRECT=ON).

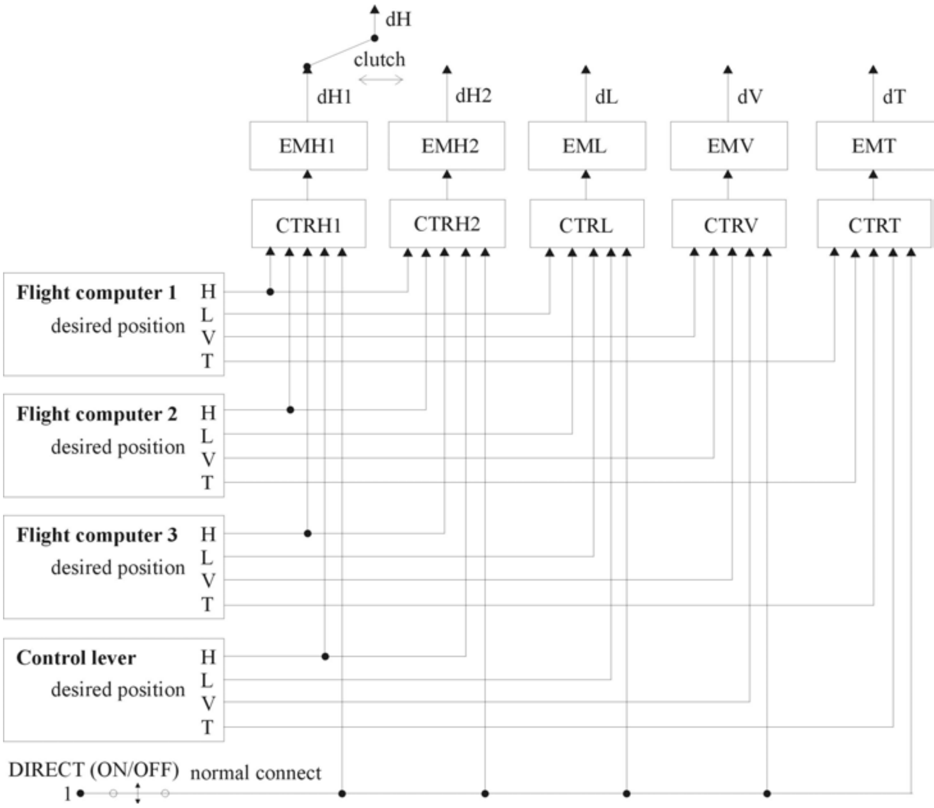

Fig 4. SPS-1 control system equipped with set of EMA: EMH1/H2/L/V/T – electro-mechanical units, CTRH1/H2/L/V/T – actuator controllers, H – elevator, L – aileron, V – rudder, T – throttle

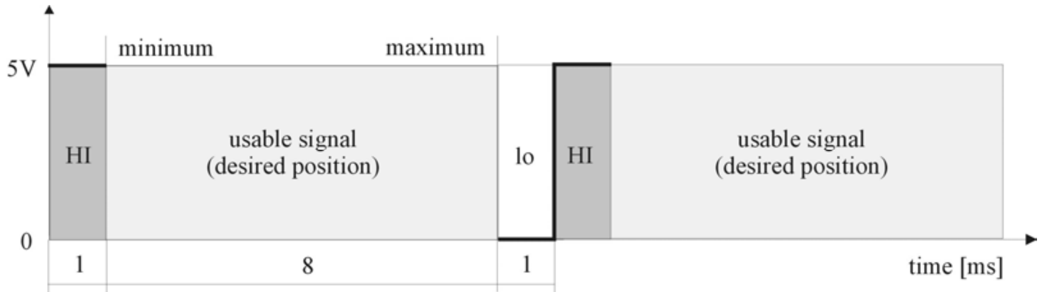

Fig 5. PWM standard established for EMA inputs

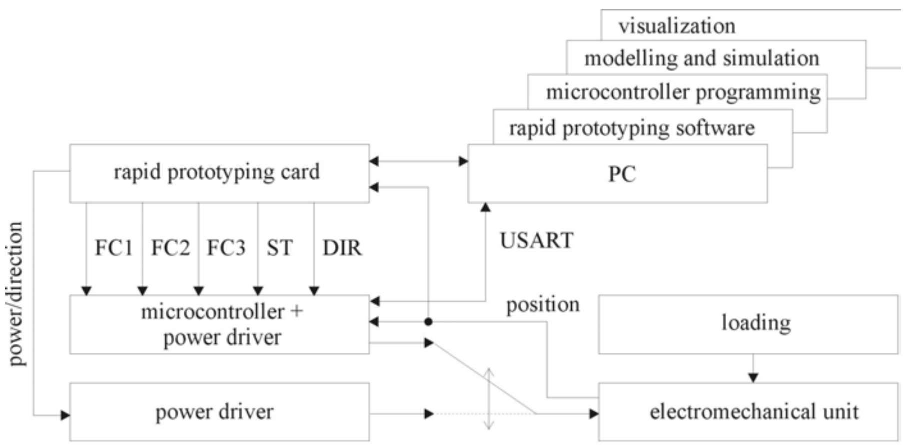

Fig 6. Block diagram of the laboratory stand, FC1/2/3 – simulated control signals from flight computers, ST – stick position signal, DIR – direct control switch, USART – Universal Synchronous Asynchronous Receiver Transmitter

Laboratory stand tests

In order to perform laboratory tests on the actuators, a special test stand was constructed. This stand allows a single actuator at various load conditions and at various driving inputs to be tested. A rapid prototyping PC card is used as the test signal source and data acquisition bus.

A program of static tests included positioning accuracy and allowable load values for various positions of the servo element. A program of dynamic tests included the actuator responses to step, harmonic, and special inputs. The test data were used to adjust control algorithms so that the most favorable static/dynamic properties were obtained. Frequency characteristics of EMA are shown in fig 9.

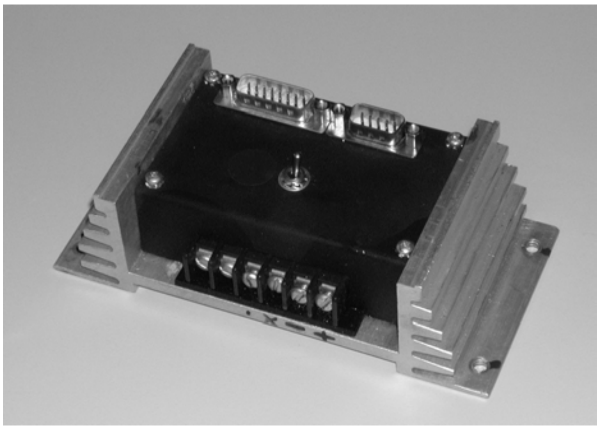

Fig 7. Actuator controller

The tests performed on a laboratory stand showed that the actuator controller is able to position the control surfaces with established static precision of 0.1 degree. EMA was also tested in admitted full load conditions both positive and negative. Control algorithm permits to deflect surfaces with maximal power in wide operating range. Speeds are unfortunately limited by the electro-mechanical design, and due to over-saturation we can only compensate the phase lag. Signal amplitude is suppressed in this case.

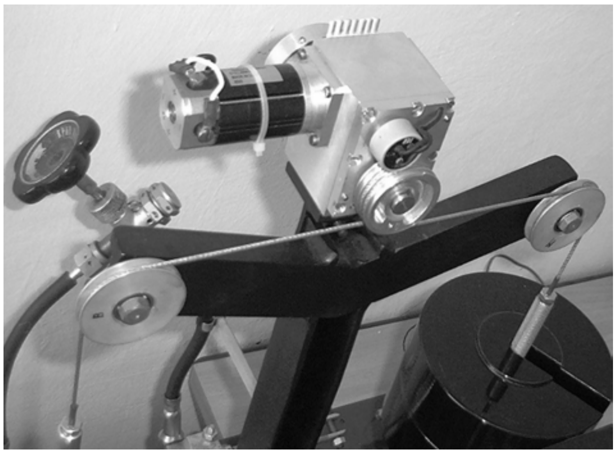

Fig 8. Actuator under loading

In flight tests

The actuator operation has been checked in many possible situations. Positive results of laboratory test [5] made it possible to install it on board a PZL-110 “Koliber” aircraft, as a component of the experimental SPS-1 Fly-By-Wire system.

The servo systems complete with microcontroller were installed on board the PZL-110 “Koliber” aircraft (Fig 12). To achieve a higher safety level in SPS-1 system, a redundant elevator servo and controller system was adopted. A total of 12 flight hours with inputs and responses at various flight conditions were recorded. Tests performed in flight (July-November 2003) supported the laboratory results. The results of aileron and elevator servo testing are shown in figures 10-11.

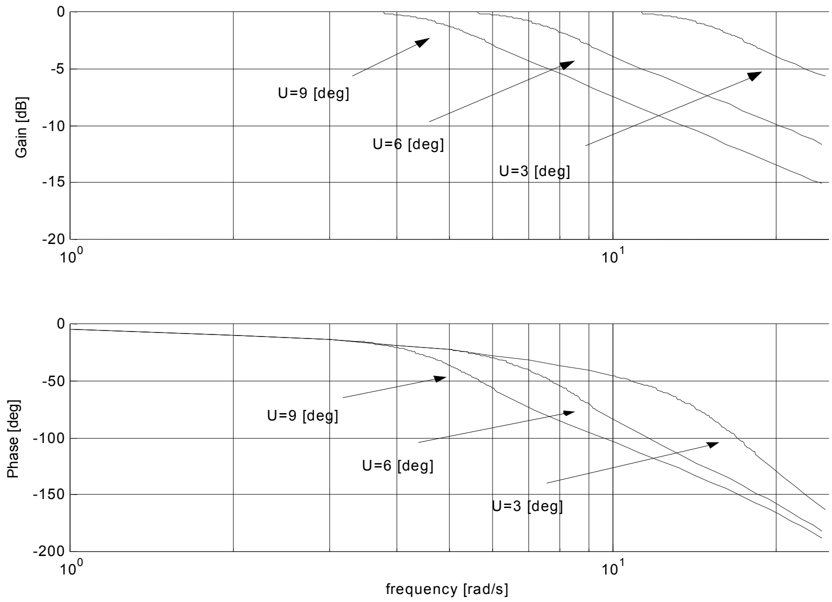

Fig 9. Freqency characteristics of EMA. Characteristics obtained for three different amplitudes of harmonic input function (U)

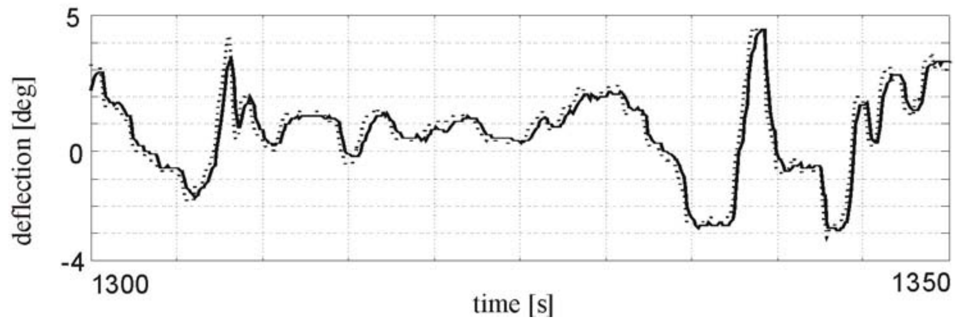

Fig 10. Aileron positioning during flight tests (03-10-24, 08:47:59 GMT); solid –deflection, dotted – desired value

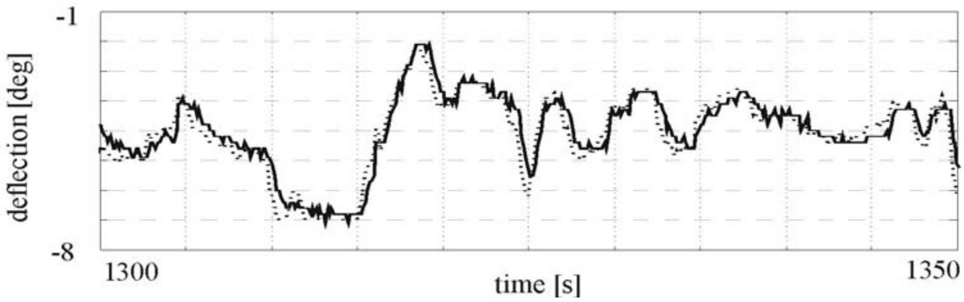

Fig 11. Elevator positioning during flight tests (03-10-24, start: 08:47:59); solid – deflection, dotted – desired value

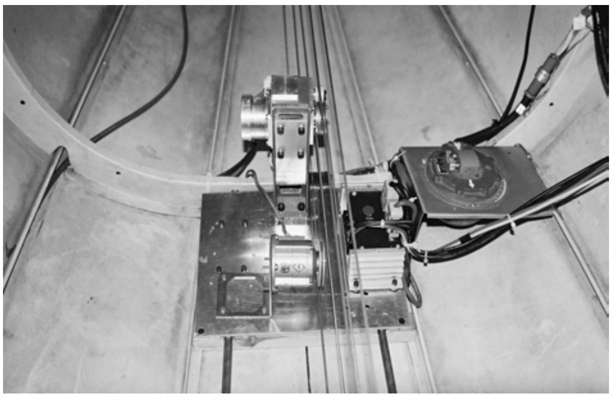

Fig 12. Actuator and controller installed on the board of the PZL-110 "Koliber" aircraft

Fig 13. PZL-110 "Koliber" aircraft: preparation for experimental flight

Future Research

In order to achieve more reliable results and perform more advanced real-time tests, two additional CAN type buses will be applied. They will replace the PWM control signals coming from the on-board computers. As a data transmission protocol, CANaerospace will be used. The next task will be designing a special software for the calibration of operational circuits and testing of the properties of the system operating on this concept. Partial failure of actuators and auto/manual flight commutating will be researched in a further step.

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