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Electric Actuation For Flight & Engine Control System: Evolution, Current Trends & Future Challenges

Jean-Jacques CHARRIER*
Hispano-Suiza, 92707 Colombes Cedex, France

Amit Kulshreshtha,
Moog Inc. Chatsworth, CA 91311†

The development of reliable electrical actuation systems for aircraft flight & engine control application has been evolving recently to eliminate hydraulic systems from engine and thus improve safety, efficiency, reliability, and maintainability and the common objective to conserve energy, reduce environmental pollution. This paper will review the architecture in electric actuators technology including electro-hydrostatic actuators (EHA) and electro-mechanical actuators (EMA) focusing on electric motor drives-including electronic power converter and electric motors technology. The aim of this work is to analyze and evaluate the possibility of introducing electro-mechanical actuators (EMAs) in more electric aircraft (MEA) and more electric engine (MEE) applications. The overall requirements for performance, availability including fault and jam tolerance, interfaces and physical attributes will be discussed to optimize the weight of such actuator systems with minimum added redundancy in the electrical machine including direct drive and gear mechanism, power converter architecture. The design of the power converter including centralized and distributed design for control and power drive electronics will be described. The EMA design for MEE has to consider the severe environmental constraints regarding to the wide operational temperatures. This constraint concerns the electric motor, the power electronics including possible use of wide band gap electronics such as SiC, the position sensors and the mechanical parts. It will also review electrical power interface issues including electric power and flight control computer electronics interface and the design considerations to minimize weight cost and improve reliability as well as civil certification objectives.

I. Introduction & Background

Aircraft primary flight control and engine control are essential to the safety of the aircraft and require a dependable system where its integrity and continuity of service for the duration of the flight could be assured. It has evolved from a traditional mechanical system with linkage, cables and pulleys connecting pilot controllers with flight control surfaces and engine/throttle control levers, to a powered control system where hydraulic/fuel based actuators or effectors were employed to move the flight control surfaces and engine control based upon the pilot's mechanical inputs. It finally evolved into full authority electrical control system with pilot's inputs going to flight control/ fly-by-wire (FBW) computer, or full authority digital engine control (FADEC) computer which commands and controls flight control and engine control actuators based upon aircraft/engine states/parameters providing safe/reliable mechanism for the control of the aircraft and meeting pilot's handling qualities, passengers comfort, optimizing engine performance & thrust and support/logistics needs for operators. The computerized control of flight and engine controls uses redundancy for fault tolerance in sensors, computers and its actuation system to assure the continuity. This helps in alleviation of pilot's workload under extreme flight conditions with full flight envelop protection of a relaxed stability aircraft design improving performance. It also provides weight saving by elimination of redundant mechanical linkages, reduced maintenance cost due to low/no adjustment of electrical/FBW system and reduced manufacturing cost due to the elimination of installation and assembly associated with low friction mechanical linkages from cockpit to the actuators/power control units (PCUs) installed at aircraft tail/wings etc. An example of B777 Flight Control System is shown in Figure 1 where Primary Flight Control Computers (PFC) command Actuator Control Electronics (ACE) for

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* Head, Research & Technology, 18, bldg Louis-Seguin
† Senior Engineer, Engineering, 21339 Nordhoff St.

controlling Actuators/Power Control Units (PCU) to move flight control surfaces to a desired position at required rate.

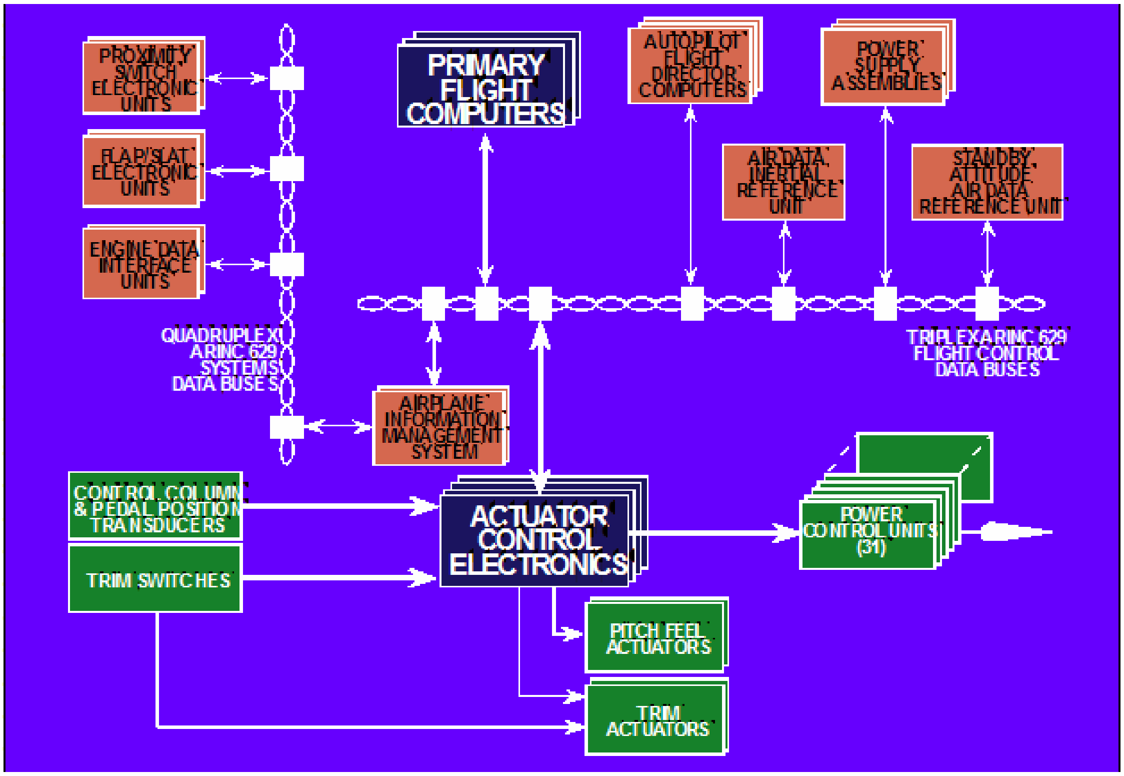


Figure 1. B777 Primary Flight Control System: An Evolution from Mechanical to FBW System

The FBW concept demonstration was done by NASA on F-8 program followed by European evaluation by German DLR during late 60s followed by its demonstration on YF-16 in 1970s and then its first introduction on F-16 production aircraft by General Dynamics during early 70s. Successive military programs adopted full authority FBW system with, or without any mechanical back-up including F/A-18, Swedish JAS-39/Grippen, Rafael, EFA-Typhoon, C-17, B-2, F-22 etc. during 1980~1990 and offered improved aircraft performance with reduced weight and cost. A block diagram of military flight control system is shown in Figure. 2. The wide spread use of FBW drew attention of civil transport aircraft developers and Airbus introduced FBW based Primary Flight Control on A320 and following it on A330/A340 as shown in Figure. 3.

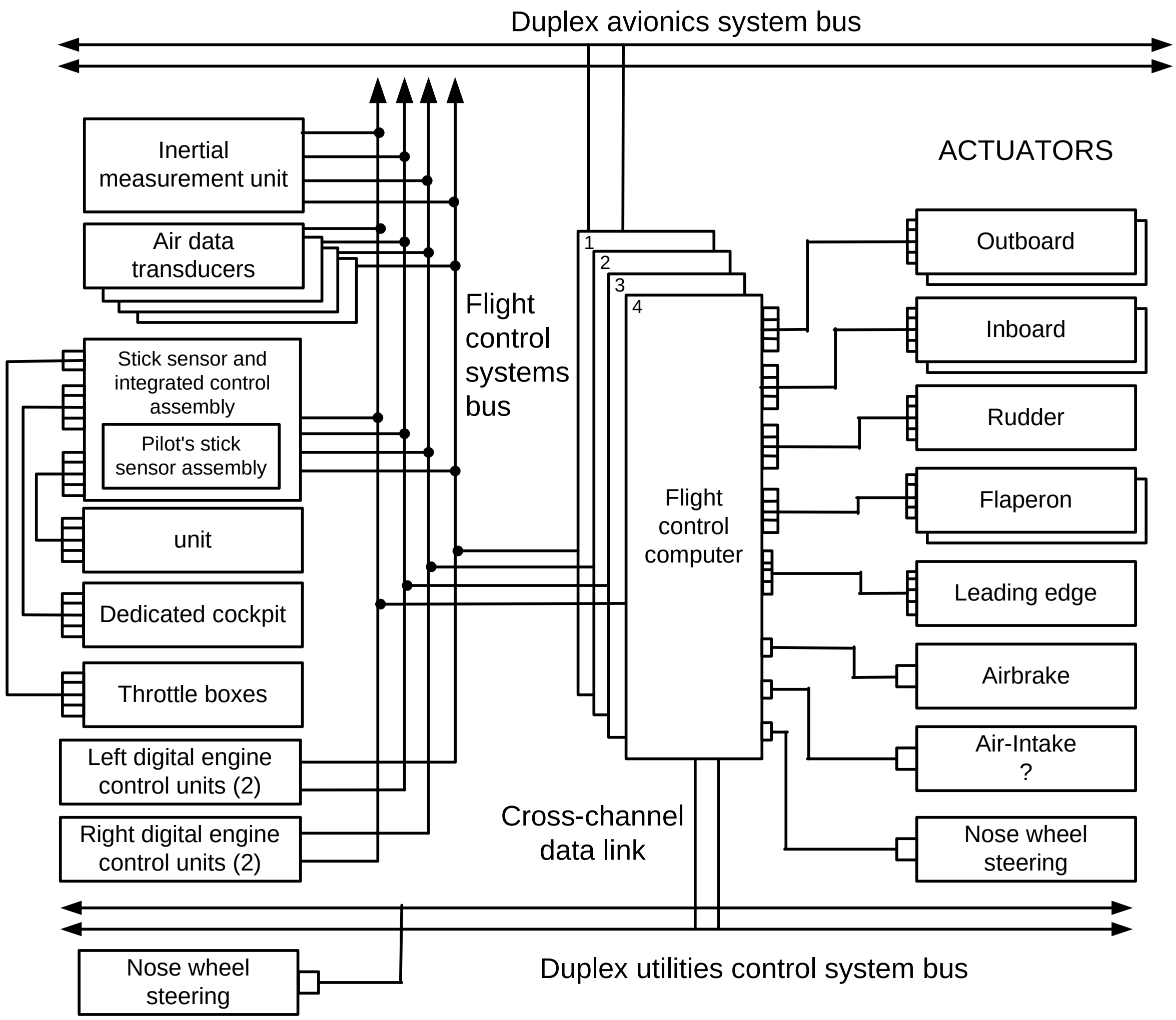


Figure 2. Vehicle Management System including Flight & Engine Control System Architecture with Centralized Actuation Control from FCC & FADEC

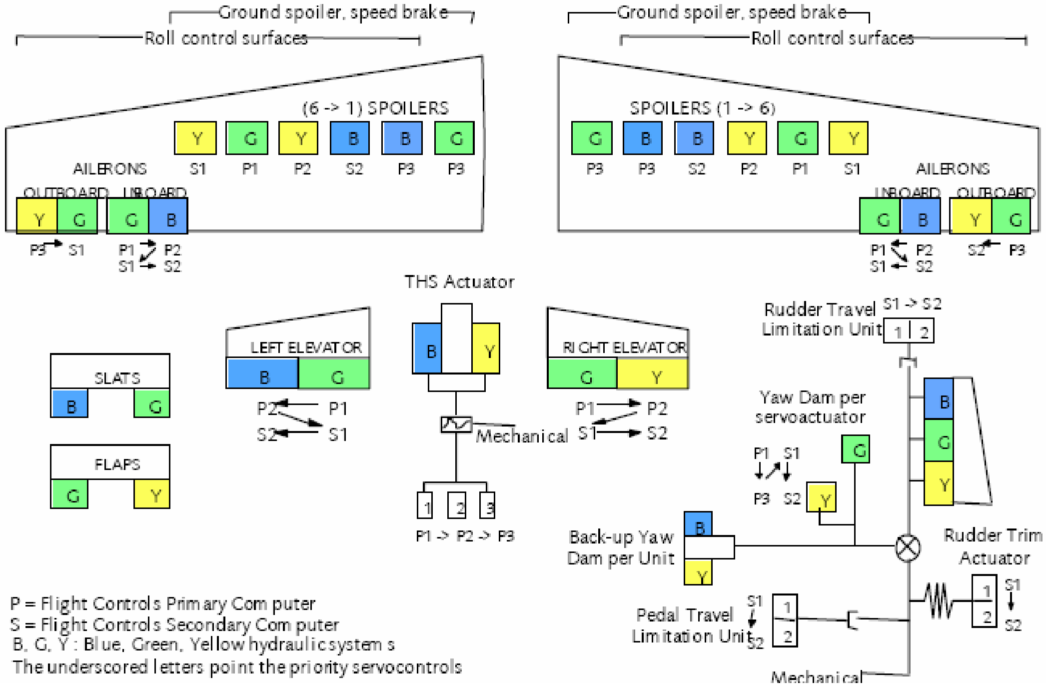


Figure 3. Airbus 330/A340 Actuation Control System using 3 Hydraulic Systems (Y, G & B)

Engine control based upon Electronics' superior accuracy led to early generation analogue electronic control first used by Concorde on Rolls Royce Olympus 593 in 1960. NASA and Pratt & Whitney experimented with FADEC flown on F111 in 1970 and this led to many military and civil engines respectively fitted with FADEC with many systems still employing hydro-mechanical back-up for engine control. The aircraft's thrust lever sends electrical signals to the auto throttle or, FADEC. FADEC calculates and controls the fuel flow rate to the engines giving required thrust. In addition to the fuel metering function, the FADEC performs numerous other control and monitoring functions such as Variable Stator Vanes (VSVs) and Variable Bleed Valves (VBVs) control, cabin bleeds and power off-takes control, control of starting and re-starting, turbine blade and vane cooling and blade tip clearance control, thrust reversers control, engine health monitoring, oil debris monitoring and vibration monitoring. The key elements of Engine Control are shown in Figure. 4 & Figure.5.

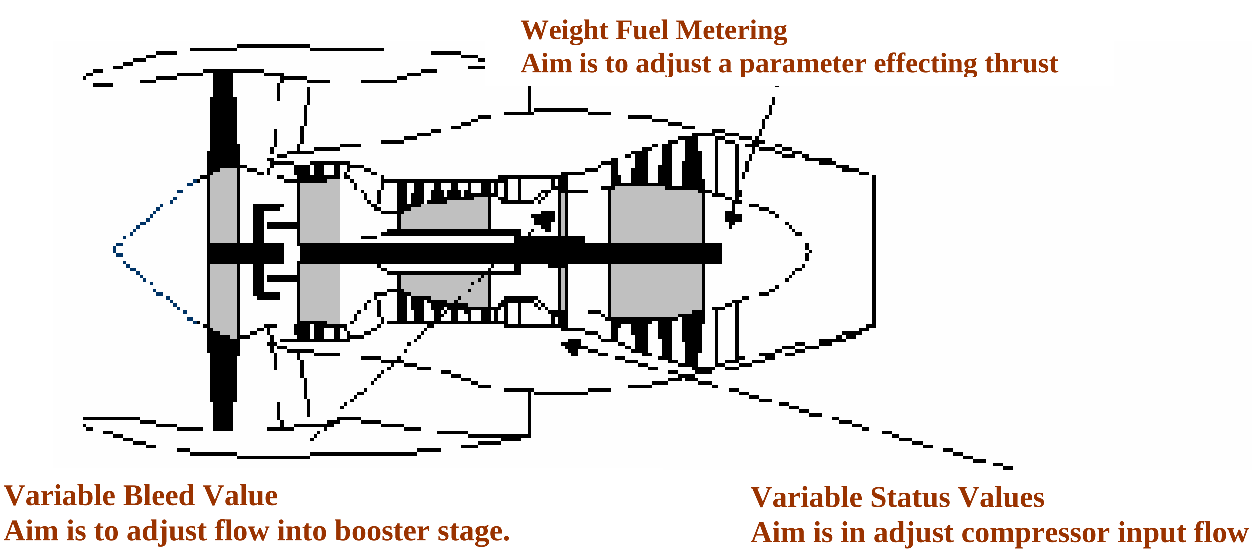


Figure 4. Engine Control System:Variable Stator Valve, Fuel Metering & Variable Bleed Valve Control

The safety requirements for flight and engine control require high availability of the control function and the probability for loss of each of control function i.e., loss for one aileron control ranges from 1E-08~1E-07 ppm for flight control to ~0.5E-06 ppm for engine control function. This requires fault tolerant approach based upon redundancy in control and actuation/effectors. Many of today's FADEC incorporate two separate control channels.

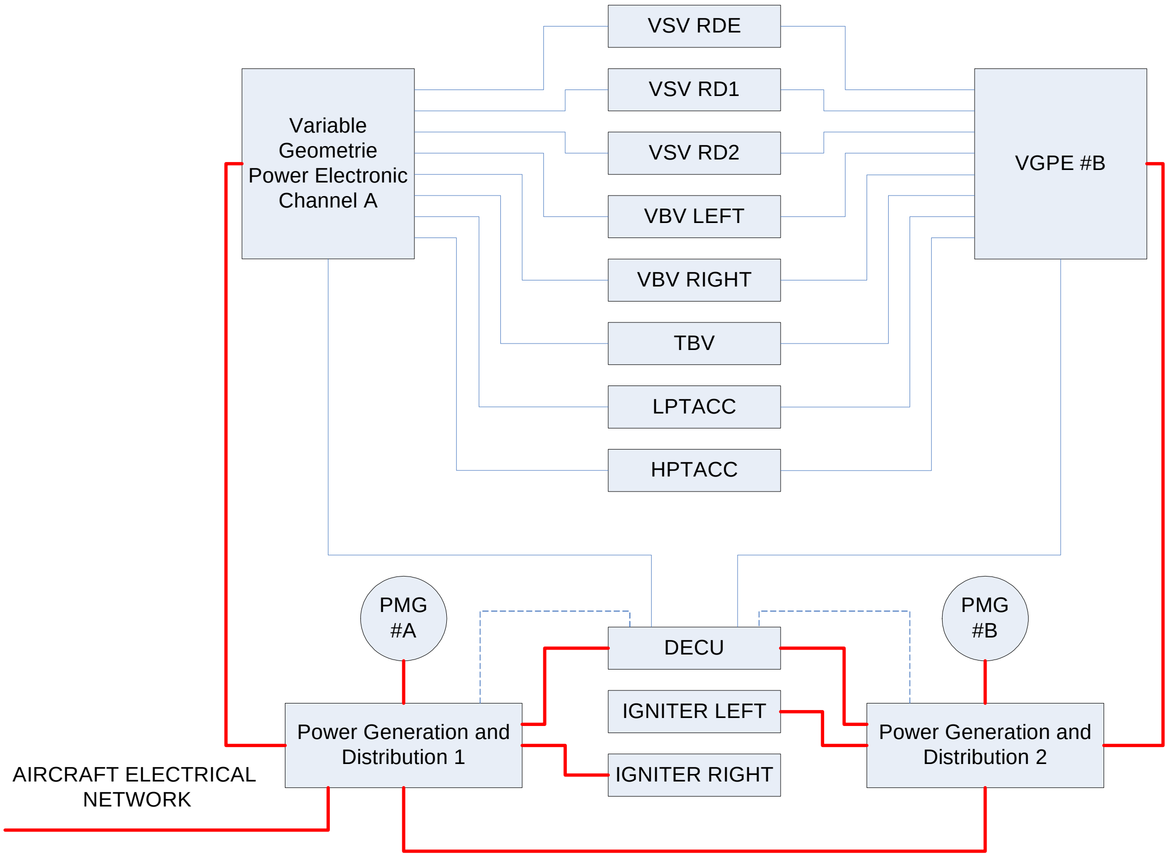


Figure 5. General Architecture for Engine Control. System for *More Electric Engine (MEE)*

Each channel may provide all engine functions without restrictions. The FADEC is powered by the aircraft electrical systems, and in most modern aircraft it uses power from a separate generator connected to the related engine and can operate with aircraft power when engine is below 12% speed or not running. In addition, there is a separate hydro-mechanical back-up (or, hot spare) system which can provide basic control of the engine functions without optimizing them.

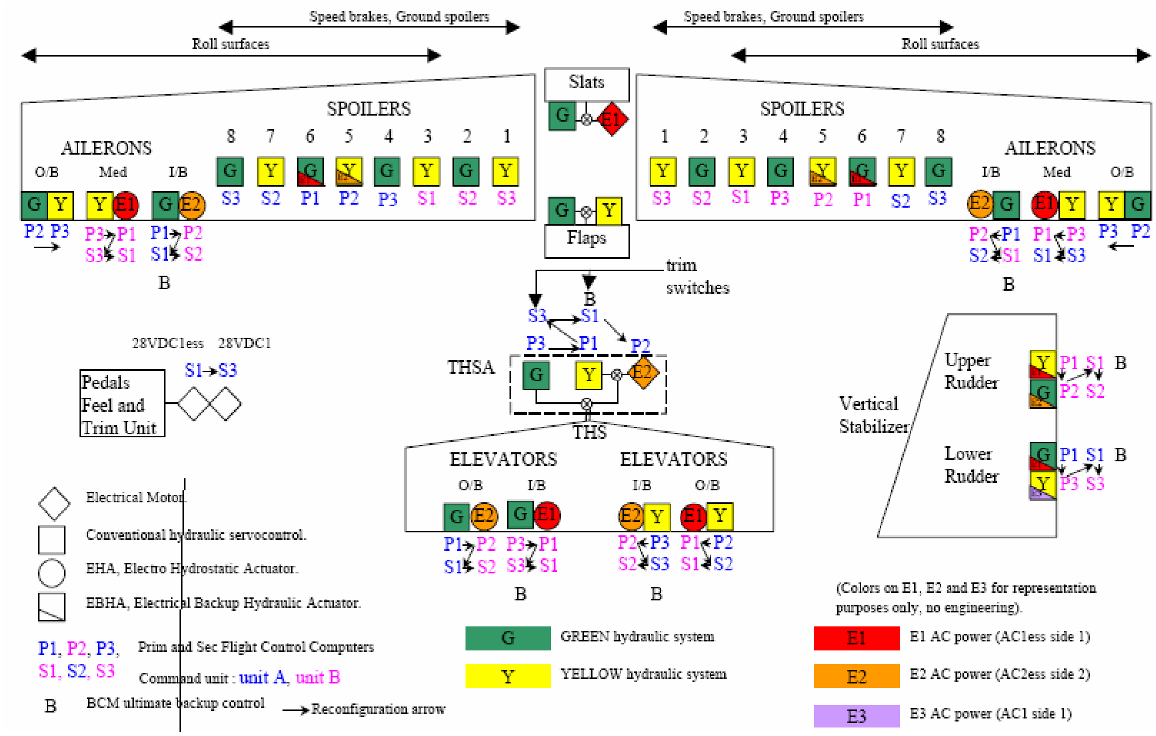
A. Actuation Control Power Source & Electric Actuation

The FBW systems require a continuous control of the aircraft control surfaces under all operating conditions. Current reliability of the conventional central hydraulic system providing a power to the hydraulic actuators over long length of the airplane requires three independent hydraulic systems. The Flight Control Computers (FCC) and Actuator or Remote Servo Control Electronics (ACE/RE) controlling hydraulic actuators also require a non-interruptible electrical power source to ensure the control of the actuators under the possible failure of various aircraft systems. This resulted in three independent hydraulic and three independent electric power sources some times referred as 3H/3E configuration. The advantage of the use of electrical powered actuator results in reduction of the redundancy of the hydraulic systems from 3 to 2 or, even 1. Such systems configurations are sometimes referred as 2E/2H configuration that can meet safety requirements as shown in Figure 6 for A380. This elimination of one hydraulic system with hydraulic lines running across the aircraft provides significant weight and volume savings with improvement in reliability at aircraft level while the individual weight of the electric powered actuator may be higher than hydraulic powered actuators with similar output power ratings.

The electric powered actuation draws the electric power on demand, proportional to the load requirements, unlike hydraulic system dissipating significant power through pressure drop across control valve. These advantages result in improvement in performance with possible energy optimization resulting in lower life cycle cost and reduced overall weight at aircraft level.

Civil transport aircraft typically use two actuators per primary surface as shown in Figure.3 with hydraulic power distributed from three hydraulic systems due to the reliability of the current generation hydraulic power system. These hydraulic power lines carrying hydraulic power need to be segregated with enough spacing to ensure that any common mode failure such as engine burst etc would not cause a simultaneous failure of more than one actuator. Electric powered actuation offers replacement of the heavy and often inflexible hydraulic tubing with more easily routable electric cables.

Engine control system uses fluid (fuel) based actuators powered by engine driven fuel pump. Its replacement by Electric Actuation also helps to improve safety and reliability as fluid based actuators suffer due to leakage problems with seals etc at high temperature operation. It also helps to reduce or, eliminate engine driven accessories gear box as we move forward with More Electric Engine



Courtesy~ Airbus

Figure.6: Electric Actuation A380 2E/2H System

II. Electrical Powered Actuation System

Electric powered actuation shown in Figure. 7 may be classified as:

(i) Electro-Mechanical Actuator (EMA) with mechanical transmission translating rotary power drive motion into either rotary hinge line motion through rotary gear box or, linear using ball or roller screw mechanism,

(ii) Electro Hydrostatic Actuator (EHA) with hydraulic transmission driven by variable speed motor driving fixed displacement pump also shown in Figure. 8

(iii) Integrated Actuator Package (IAP) with fixed speed electric motor driving a variable displacement pump driving hydraulic transmission moving the control surfaces.

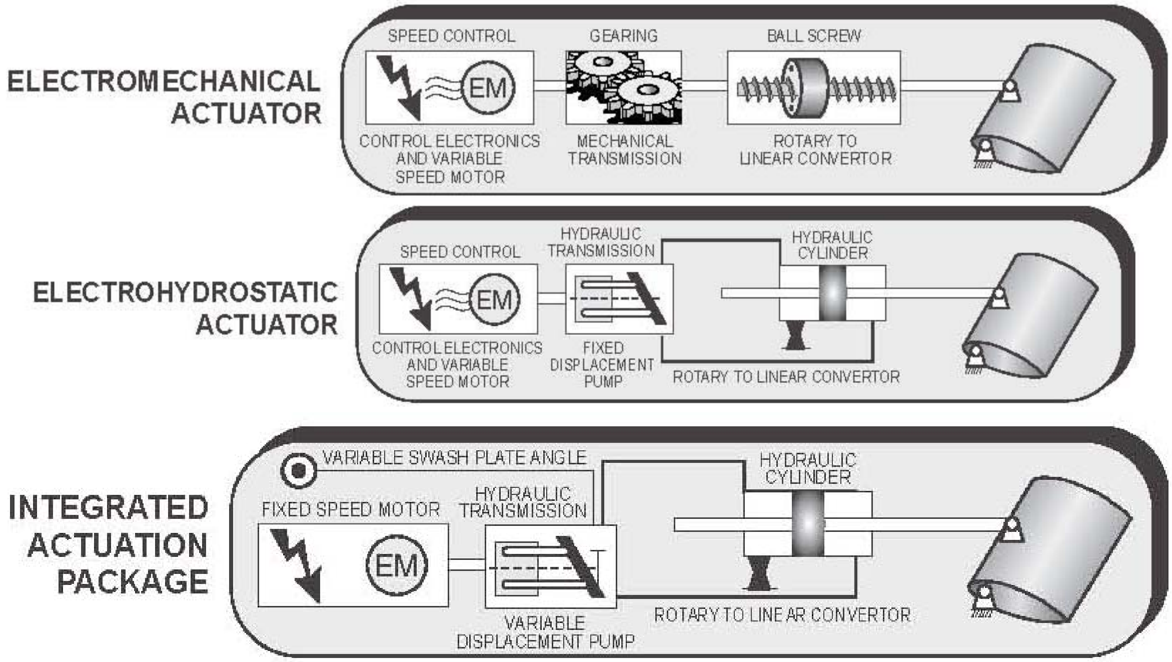
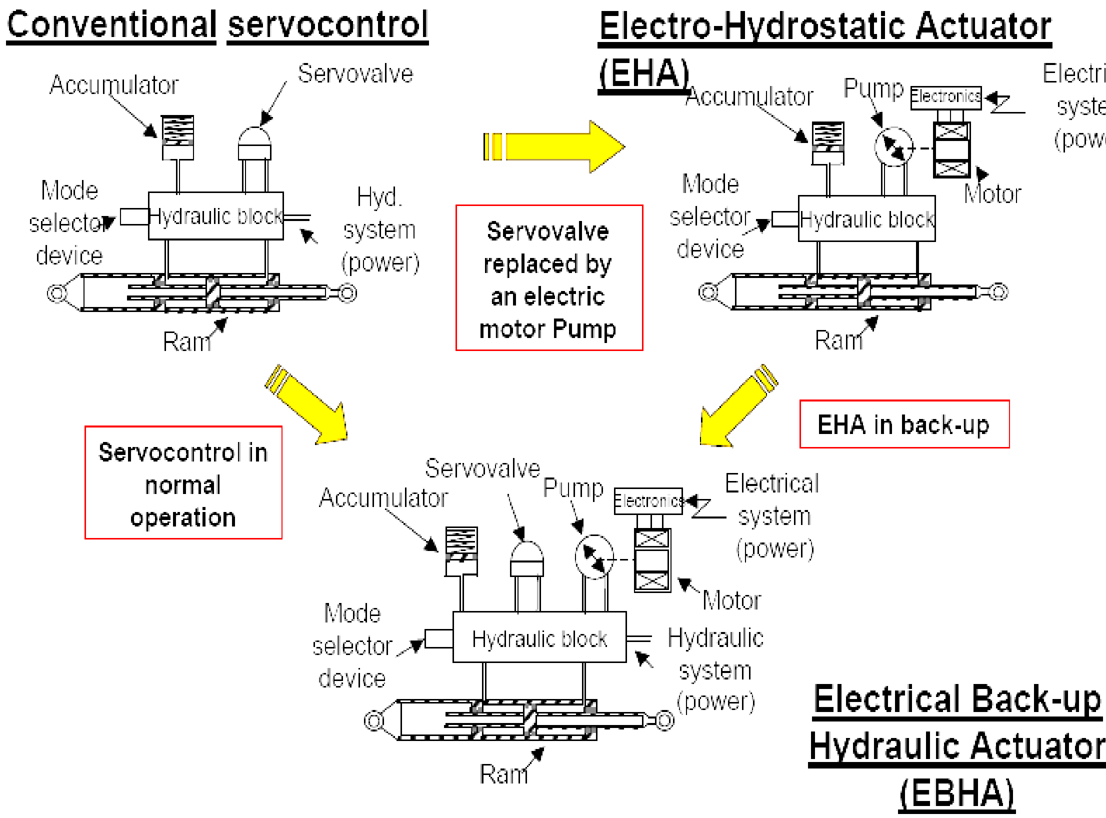


Figure 7. Electric Actuation System Architecture

EHA technology provides an electrically powered actuator with proven hydraulic transmission/powered control which is ‘time proven’ immune to jamming concerns. The current generation EMA technology suffers from concerns for jamming etc and this has prevented its use in primary flight actuation systems in current generation ‘manned’ aircraft. However, there are many initiative to develop ‘jam tolerant’ to ‘jam immune’ EMAs involving the use of high power density electric motors providing direct drive etc and such designs and products should be available in the near future for next generation aircraft. IAP technology suffers from high losses due to constant speed operation of the pump and results in lower reliability systems.

Conventional hydraulic actuators have continuously flowing hydraulic fluid passing through an external heat exchanger for the removal of the heat from the actuator. Self contained EHA suffer from their inability to remove the heat generated in the small volume by its motor/pump and the power electronics. The high temp operation of EHA with added components including electric drive and the pump results in lower reliability of the EHA compared to conventional servo control hydraulic actuator. Since most of the aircraft still requires hydraulic power for heavy loads such as landing gears etc, it was considered advantageous to use this source of power combined with electric power. This led to the design of a hybrid configuration incorporating hydraulic servo valve with EHA called Electrical Back-Up Hydraulic Actuator (EBHA) shown in Figure 8. The EBHA normally works with the available hydraulic system and switches to electric powered EHA whenever hydraulic system is not available.



Courtsey~Airbus/SAE

Figure 8. Electro Hydrostatic Actuator (EHA) with Electric Motor Control Electronics Unit (EMCU) controlling motor speed/torque for actuator control

B. Electric Actuation System Elements:

The electric actuators could be based upon a ‘mechanical transmission’ or, ‘electromagnetic coupled transmission’ elements translating electric drive power to drive the load for EMA vs. ‘hydraulic transmission’ for EHA/EBHA. Figure 9 show a more detailed overview of the elements for such electric actuators. A common element between EMA and EHA/EBHA is Electric Drive which includes:

(i) Electric Motor converting electric power into mechanical power to drive the transmission elements usually based upon a permanent magnet brushless technology.

(ii) Electronics Motor Control Unit (EMCU) providing variable speed control of the motor for required torque to provide necessary power to the transmission system.

The electric drive is coupled to the transmission system which converts power conversion from electric drive's output of rotary power to the linear or, rotary output power as needed to drive flight control surface i.e., load as required. The key elements for the electric actuator systems for EMA as well as EHA are shown in Figure 8:

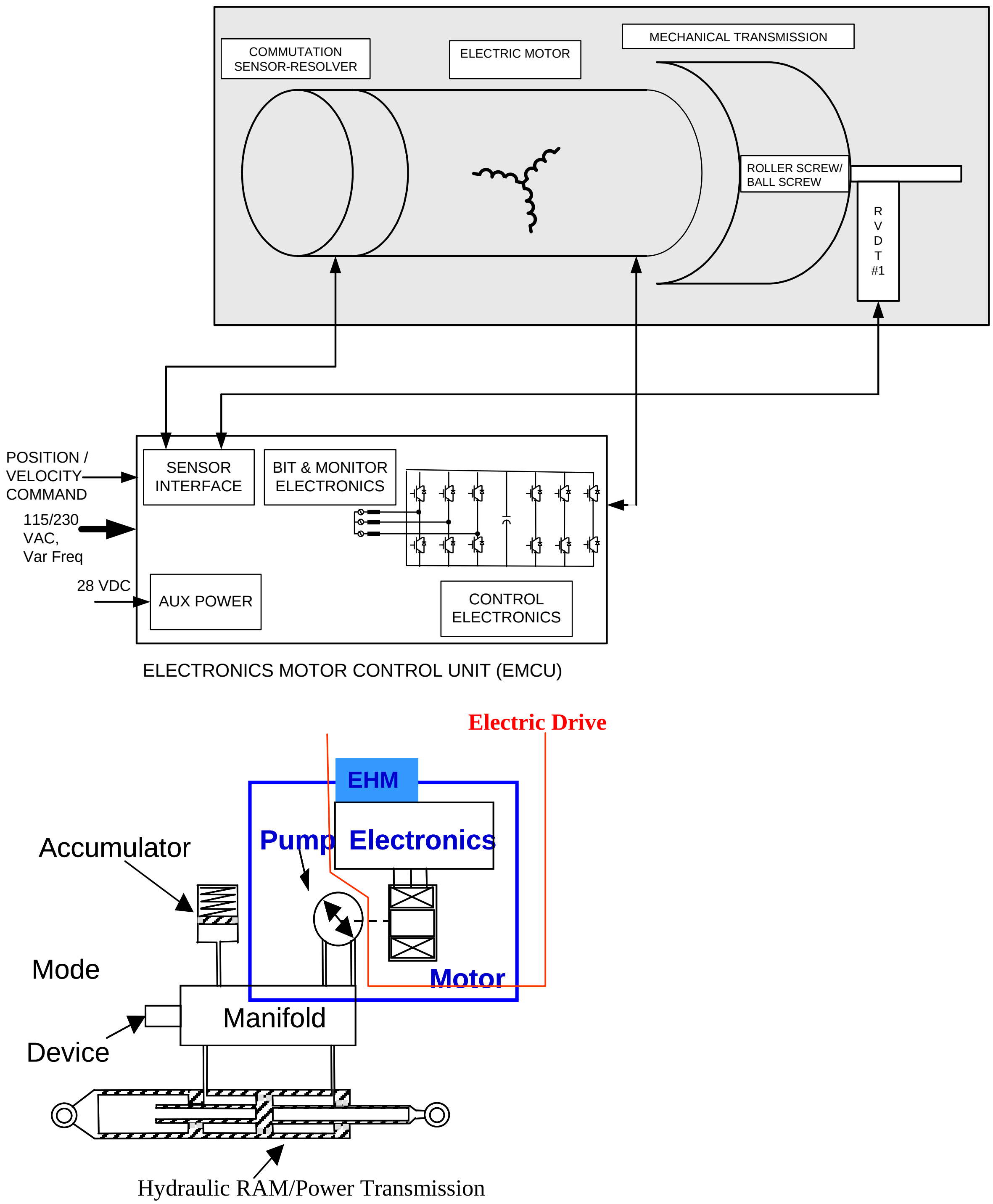


Figure 8: Key Elements of Electric Drive/Actuator: EMA & EHA: EHM includes Electric Drive & Pump

1. Electric Motor Technology

The key elements of the electric motor include a stator with electric coil windings generating rotating magnetic field, rotor with permanent magnets and bearings to allow the rotational motion of the rotor with minimum friction. The current generation systems typically use permanent magnet brushless motors. These motors with either trapezoidal back electro-motive force (emf) or, sinusoidal back emf are sometimes called permanent magnet brushless dc (PM-BLDC) or permanent magnet ac synchronous (PMAC) motors. These motors include permanent magnet on the surface of the rotor, and a rotating magnetic field is generated in the stator coils by the electronics motor control unit (EMCU), which turns the rotor in synchronization with the rotating field. A cross section of such motor is shown in Figure 10.

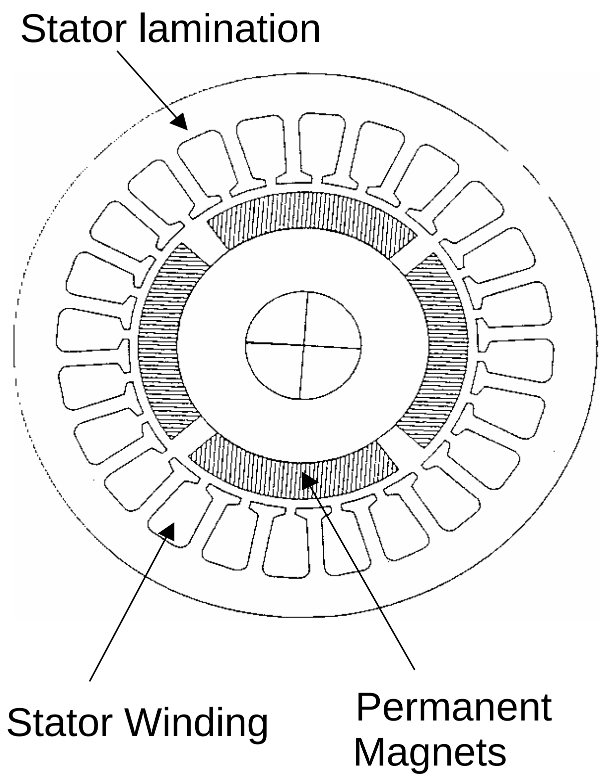


Figure 9: Cross-section of Permanent Magnet Motor

These surfaces mounted permanent magnet motors suffer from saliency issues with the rotor magnetic field and eddy current losses in steel sleeve used to ensure that the magnets remain fixed at the surface while motor is operating at high speed. This led to the development of Interior Permanent Magnet (IPM) motors where magnets were mounted inside the rotor. The control of such motors had been more difficult as well the cost has been higher limiting the applications of such motors in wide spread use for flight control.

Concerns on possible jamming of the gear train limit the use of EMA today. Current efforts are underway to make direct drive electric motors capable of driving transmission mechanism directly with high torque at lower speed. Such designs have been demonstrated with higher magnetic density involving different configuration of permanent magnets, and the coil windings are being tested for aircraft applications.

There had been efforts in parallel to develop fault tolerant windings to allow electric motor operation after a single point failure of a coil. This has led to the design of dual redundant windings driven by two independent inverters. These designs have been commonly used and illustrated in Figure. 10.

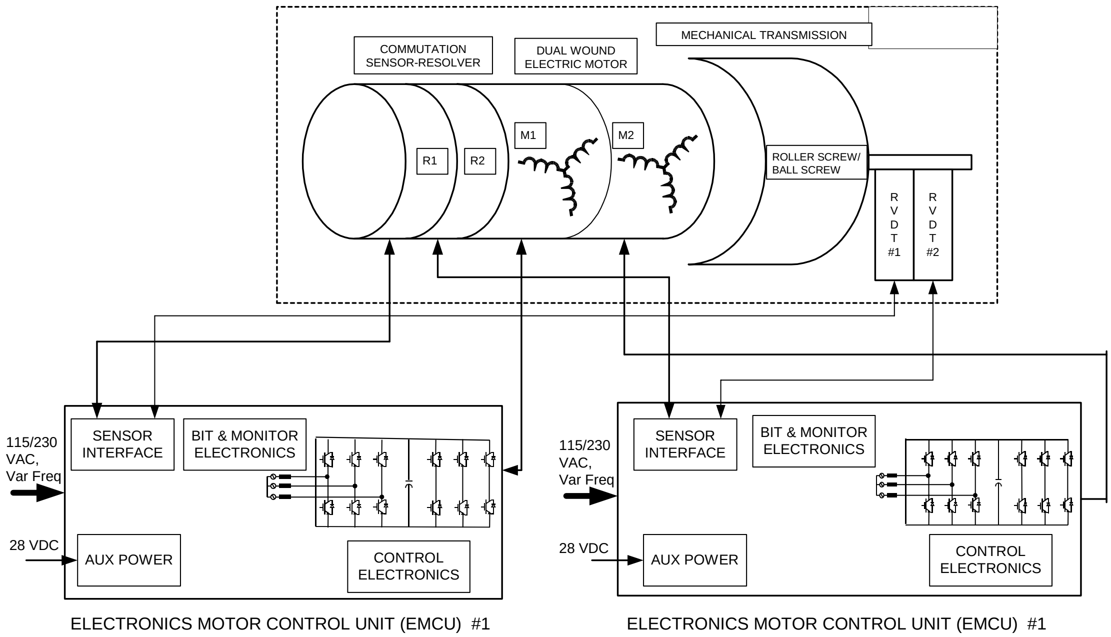


Figure. 10: Dual Wound Stator Based Electric Actuator

The industrial applications were looking into more cost effective applications and these led to another topology of multiphase motor driven by multiphase inverter with independent legs controlling each phase of the motor as shown in Figure 11.

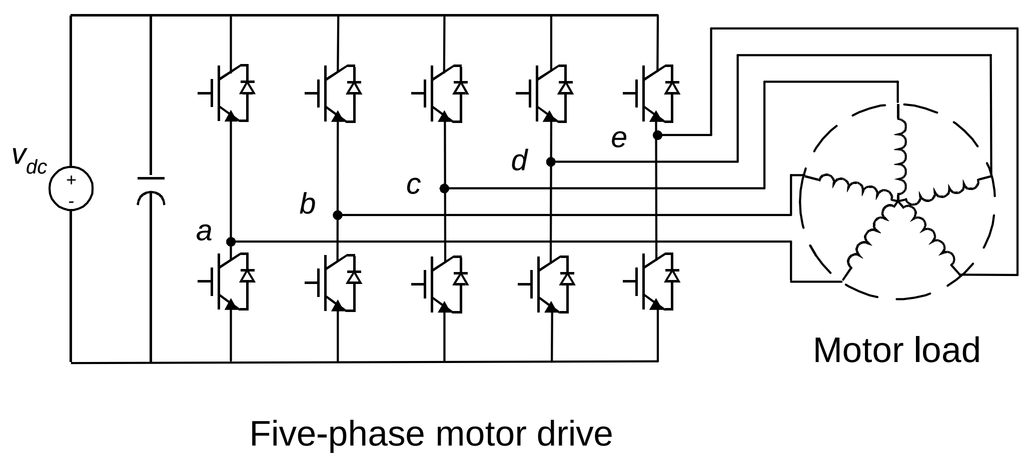


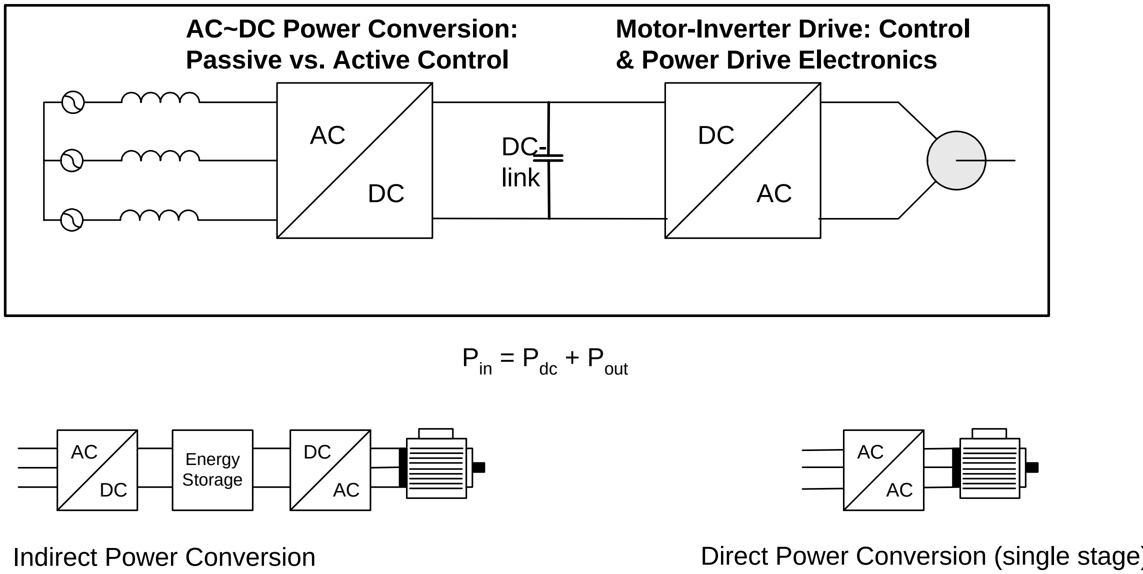
Figure. 11: Multiphase Motor Driven by Multiphase Inverter requiring less hardware with similar or higher fault tolerance

2. *Electronic Motor Control Unit (EMCU)*

EMCU controls electronic motor to provide required speed/torque needed to drive load or transmission mechanism. Its functions include:

(i) Input Power Interface & Conversion for Motor Inverter Drive
(ii) Generating Control Signals for Inverter to drive the electric motor with appropriate speed/torque
(iii) Power Drive Electronics (PDE) to drive Motor Phase Current by efficient high speed switching control

A block diagram of the EMCU is shown in Figure. 12.



Courtesy. U of Nottingham

Figure. 12: Block Diagram of EMCU

Current generation transport aircraft generate and provide 115/230VAC with variable frequency (380~890Hz) ac power. Many military aircraft convert ac power into 270VDC by rectification and provide DC power at the cost of added rectification and heavier as well as costlier power control-switching equipment for load management The AC power could be converted to DC for variable frequency ac to drive motor inverter is usually done by either AC~DC Conversion using multi-pulse transformer rectifiers or active front end.

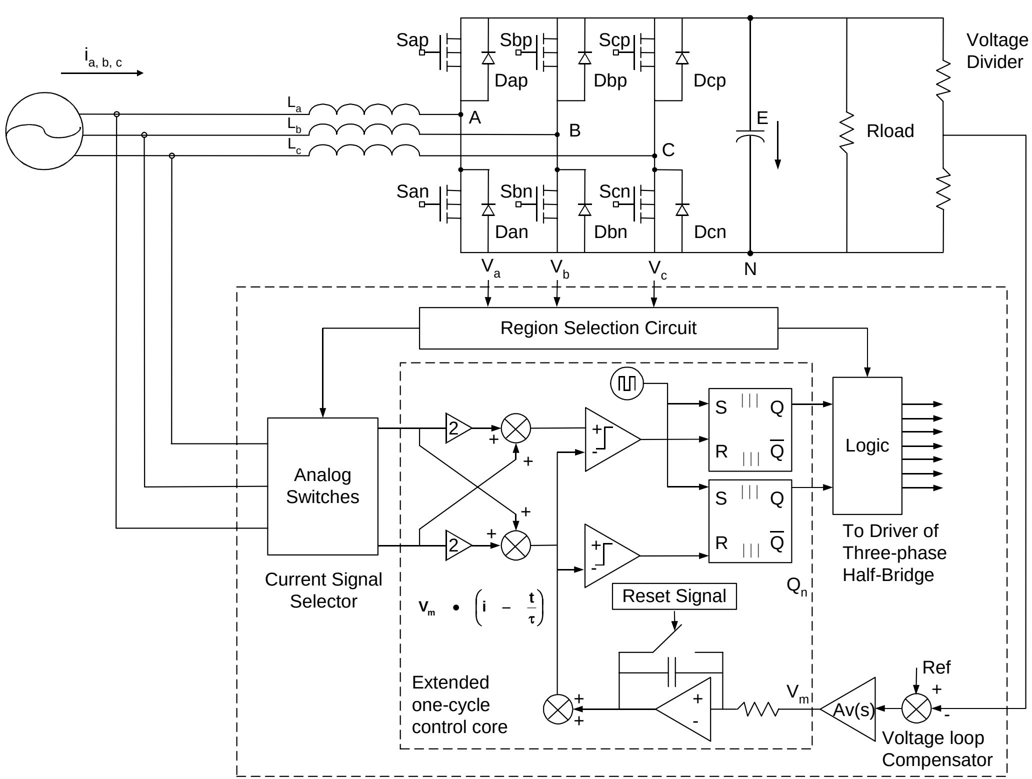


Figure. 13: Active Front End for AC-DC Power Conversion using 'One Cycle Control'

The direct AC~AC conversion using matrix converters has also been demonstrated and under consideration for future use. Active front-end control provides flexibility in control of the input power harmonics, which are becoming a larger problem in newer generation aircraft. Some schemes for such implementation are shown below:

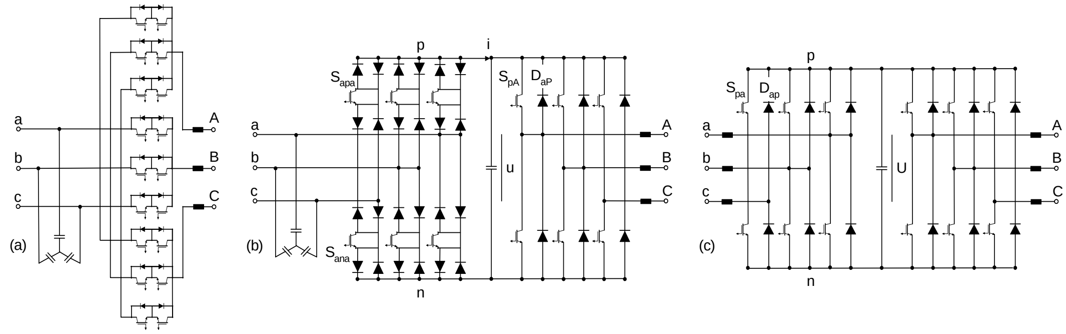


Figure. 14: AC~AC Direct Power Conversion with Matrix Converters, Reduced Switch & Back-Back Converter

The control function within EMCU is responsible for driving either type of electric motor by generating appropriate waveforms for PDE as shown below:

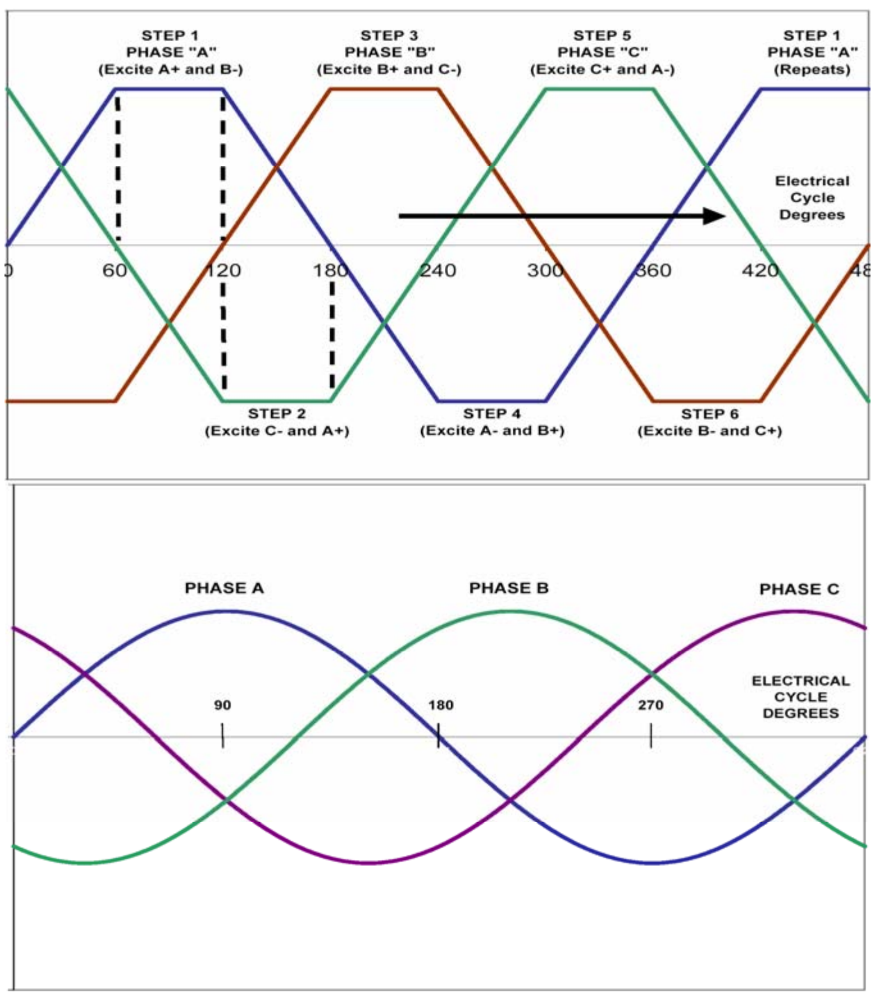
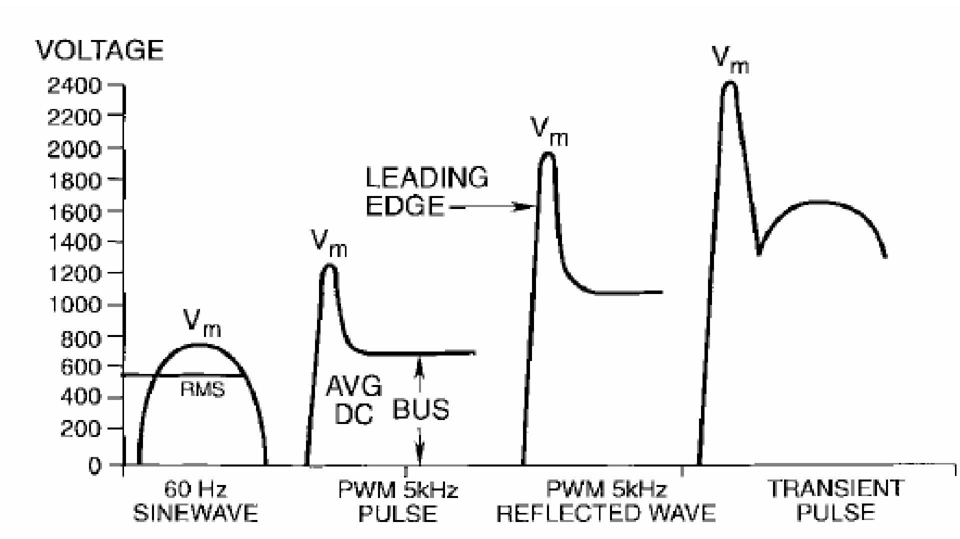


Figure. 15: Trapezoidal/BLDC or, Sine/PMAC Drive Control of Motor

In traditional designs, it had been observed that while BLCD provides higher peak torque, it results in higher level of torque ripple compared to Sine drive with PMAC motors. The Sine drive can be implemented by either using conventional PWM or Space Vector Modulation (SVM), which provides added voltage to get more current/torque from motor.

Power drive electronics (PDE) provides drive signals to the motor windings with required current for adequate torque. It is currently based upon Power MOSFET or, IGBT. The selection of FET or, IGBT/BJT is dependent upon overall power ratings, operating voltages and the efficiency of the conversion. FET as well as IGBT allows high speed switching operation for the pulse width modulation of the motor current to minimize power dissipation with FETs technology for lower power/voltages and IGBTs for higher power/voltages. However, this picture is changing and FETs are becoming available for higher ratings.

PDE had traditionally been implemented within EMCU but with more emphasis on distributed architectures, the PDE may be installed with the motor and could be controlled by remote control electronics. This distributed architecture provides heavy weight power cable weight savings from EMCU to the actuator as well reduce the impact of the reflected wave within motor winding as observed:



Typical voltage stress levels for sinewave and PWM.

Figure, 16: Reflected wave resulting in higher voltages in motor windings with PWM driven motor control.

The installation of the PDE elements within motor had been performed earlier for ground applications by Nottingham University as shown below and would require the use of high temp power electronic switches using upcoming SiC or, other wide band gap devices.

Figure. 17: Motor End Plate Mounted PDE

3. Transmission System

MA currently employ gear reduction to obtain high torque at lower speeds coupled to a ball or roller screw based rotary to linear motion translation mechanism. The transmission system needs special considerations for flight critical applications such as flight and engine controls where jamming is a major concern and performance is very demanding.

It seems that for real advantages of EMA design, maximum use should be made of the elements where the probability for jamming is minimized such as direct drive high torque/low speed electric motor. The advances in magnetic gearings could also help in future but the current torque available limit their use in relatively small power applications.

EHA uses hydraulic transmission powered by local hydraulic pump driven by electric drive. This is a traditional hydraulic actuator configuration well proven over time. In addition to the jamming immunity, hydraulic transmission may be operated like a traditional hydraulic actuator in case of failure providing blocking, bypass or damped mode of operation as needed.

III. Current Challenges in Electric Actuation

The challenges in the electric actuation technology that are common to EMA as well as EHA include:

(i) Thermal Design & Power Management
(ii) Performance-Load/Speed Curve
(iii) Noise & Ripple in Control System, High Power PWM Noise & with Sensor Noise

In addition, EMA technology presents challenges to possible failures in jamming and mechanical disconnect apart from significant performance degradation after each failure.

C. Thermal Design Issues

Thermal power dissipation has been a challenge for electric actuation due to higher electric motor dissipation for holding load condition as well high ambient temp of 90 deg C for engine actuation with its peak temp. ranging from 160~180 deg C. This requires special winding and coil insulation material for electric motors and special design approach for electronics in engine environment.

The dissipation of high power in small volume does not leave many options for such challenges. The current technology for common fluids and seals limit the temp to 275 deg F for reasonable and practical life. While there is progress in electronics as well as elastomer materials for high temp operations, another possible way to reduce the temp rise is to look at the newer design approaches for minimizing the losses. For example, electronic power switching element used with PDEs are Si based Diodes and IGBTs. Now with commercial availability of wide band gap devices such as SiC based diodes, the losses in EMCU could be reduced by approx. 30%. The main switching element IGBT may also be replaced by SiC based JFET currently being sampled and should help to reduce power dissipation by more than 50% for all next generation controllers/EMCU. Typical characteristics for SiC based devices are shown in Figure. 18:

	V_F @1A	Turn-on loss 800 V 1 A Inductive RT	Turn-off loss 800 V 1 A Inductive RT	T_{max}	DC drive current
SiC BJT	0,52 V	0,03 mJ	0,006 mJ	pack. limited	10 mA
Si IGBT	2,25 V	~ 0,05 mJ	~ 0,05 mJ	150 °C	0

Figure 18: Low Loss SiC based Power switch

There is also a need for design optimization at overall design level of the electric actuator for reduced power dissipation without impacting the weight. In addition, optimal control approach may be integrated with the motor control function to reduce heat dissipation by minimizing energy dissipation.

D. Performance-Load/Speed Curves

Military aircraft typically require primary flight control actuator with 10 in/sec speed at no torque and 120 deg/sec rate with 5 Hz bandwidth. These requirements for civil aircraft can reduce to 60 deg/sec rate and bandwidth of 3 Hz. Typical actuator load requirements are shown below in Figure. 19:

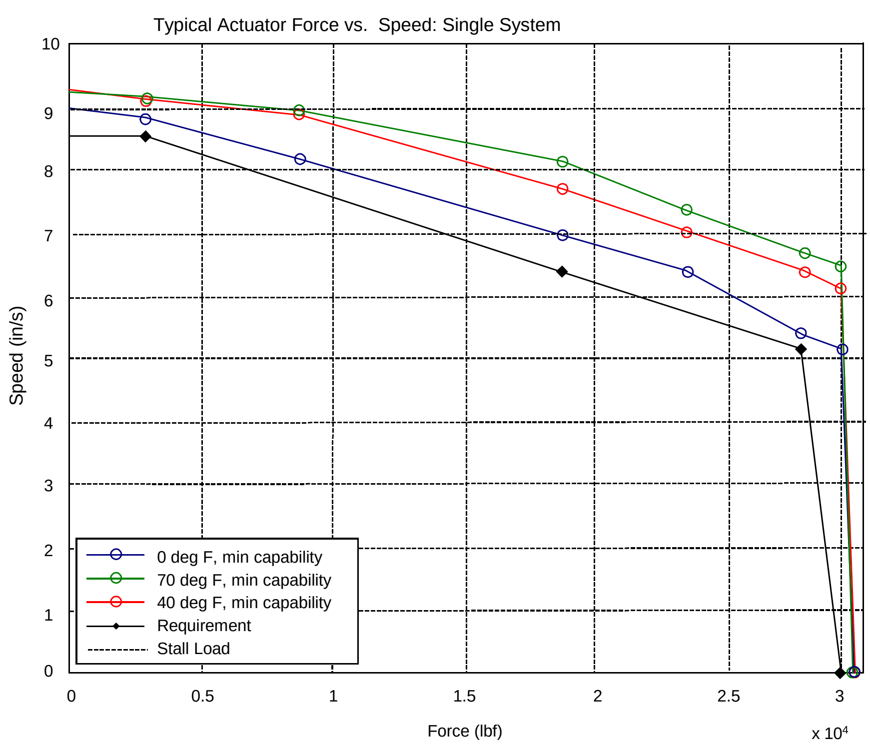


Figure. 19: Typical Load/Speed Curve for Actuator

The hydraulic actuators powered by 3000 psi~5000psi provide enough power with reasonable weight/volume. The electric actuator with acceptable weight/volume is not able to provide adequate power in current designs as shown in Figure. 20.

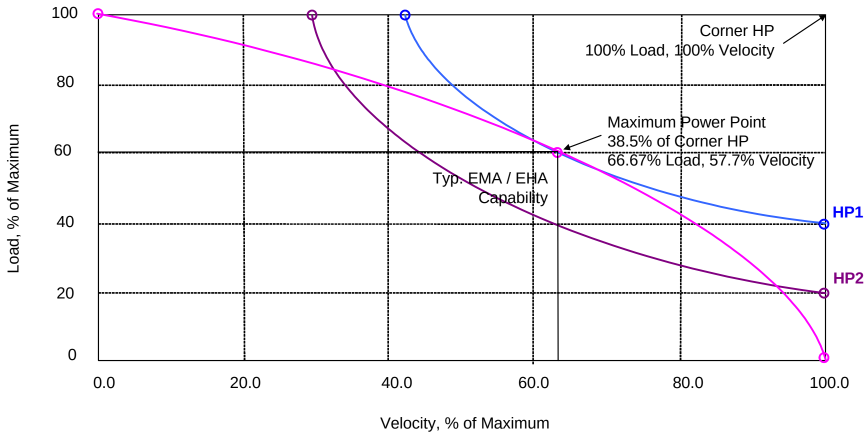


Figure 20: Actuator Sizing Power Point

The primary flight control actuator requiring higher bandwidth with load translates to high acceleration for inertial load of the control surface. The amount of the current that can be used to drive the electric motor becomes a limiting factor considering max current density etc.

E. High Power PWM & Noise in Sensor Feedback Signal

The high power PWM signals from currently EMCU located PDE get coupled to other low level signals such as sensor feedback below the loop bandwidth becomes a part of the normal control signal as the system is unable to differentiate it and the actuator system attempts to respond to it resulting in added duty cycle and increased losses.

One of the options for eliminating the sensor noise is to develop a sensor-less control approach for motor control. Such observer based motion controllers will also reduce cost and offer improved reliability. A scheme for self-sensing control by injecting a high freq ac signal taking advantage of built in saliency of the rotor is shown in Figure. 21.

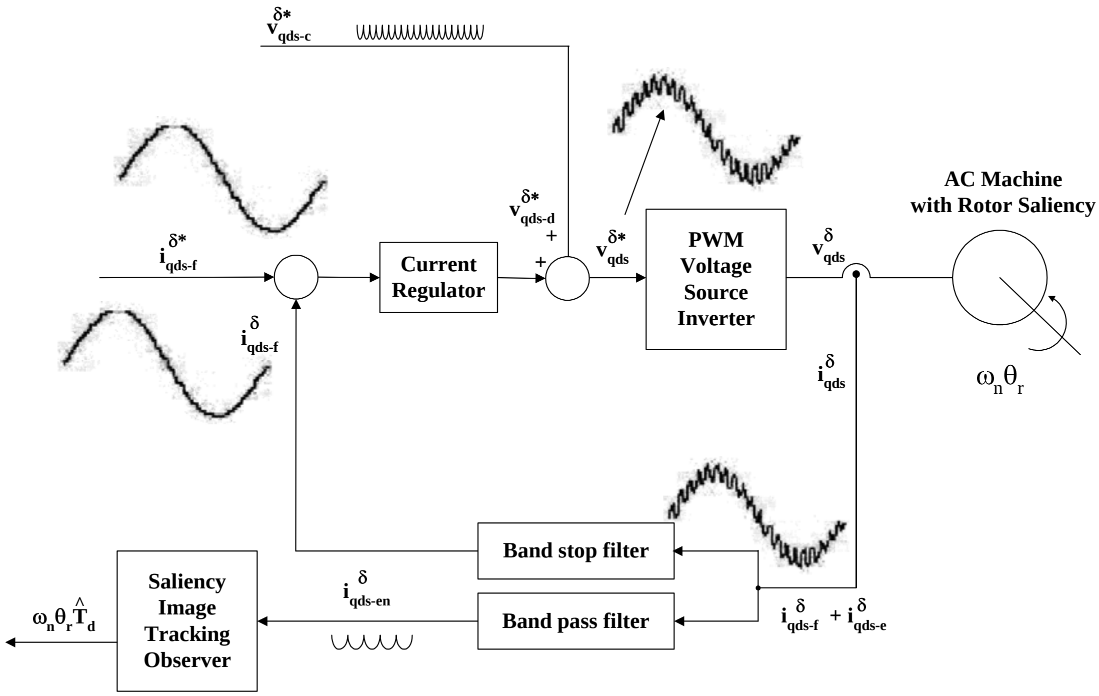


Figure. 21: Sensor-less Control of Electric Drive

F. EMA & Redundancy

The EMA use torque or velocity summing as a means for coupling dual redundant mechanical drive inputs, it allows certain type of faults to be tolerated. The structural redundancy tolerant to mechanical jamming and disconnect can be done by position summing of dual load path.

The jam tolerant design in addition to the requirements for auto-centering in case of failures require additional sensors and mechanisms to be added to the EMA design and these mechanisms add to their own failure modes and affect overall reliability of the system. These have been some of the limitations for a practical use of the EMA in primary flight control system of a manned aircraft.

Health monitoring technology involving stress/wear-out issues in rotating elements and predicting possible upcoming failure is being investigated. The 'physics of failures' are being establishes and these technologies could be used to make a more deterministic possible failures as they get more matured. It could be used for reducing some of the redundancy in actuation elements.

IV. Electric Power Interface

The electric drives require electric power and the question of 270VDC power vs. 115/230VAC power with variable frequency need to be reviewed from aircraft level considerations to ensure overall cost effective solution.

There is significant concern on the electric power system to be free of any common cause or, generic error, which may result in loss of all electric systems with a single failure. It is usually advantageous to provide 'raw' electric power for flight critical systems to avoid such design errors as providing a fault tolerant electric power system for all the aircraft may not be a cost effective solution.

Many of the current FBW systems have electric power inputs from dedicated Permanent Magnet Generators (PMGs) which provide electric power with very high integrity. FBW makes use of this raw power and converts it to the voltages required. In addition, there is Ram Air Turbines (RAT) for emergency power system. A combination of aircraft electric power from multiple aircraft buses, PMGs, RAT etc. provides virtually non-interrupted power for flight and engine controls.

Electric power generation for engine actuation control needs some special considerations including its redundancy and fault tolerance. A possible architecture is shown in Figure. 22.

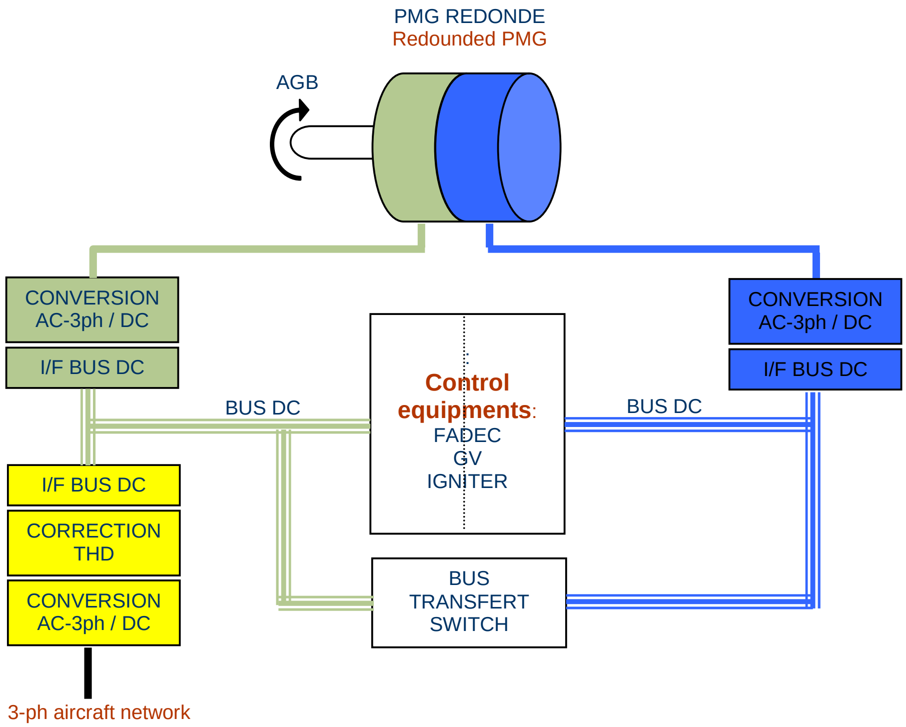


Figure 22: Power Generation Architecture

The electric power system may be designed to allow the use of regenerative converters, which could put regenerative power back on the electric bus instead of dissipating it in the EMCU. This would allow energy optimization and limit the thermal temp rise in the EMCU. Matrix converter is one such topology and needs further evaluation.

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4. Y.C.(Bob) Yeh
Unique Dependability Issues for Commercial Airplanes Fly-By-Wire Systems
SAE Automatic Control & Guidance System Meeting, 2006