Design and Development of Multi-Lane Smart Electromechanical Actuators

The unavoidable element in the development of flight control systems (to date) has been in hydraulic actuators. This has been the case primarily because of their proven reliability and the lack of alternative technologies. However, the technology to build electromechanically actuated primary flight control systems is now available, which may mark the end of the hydraulic actuation system—a significant step for the development of future “all-electric” aircraft.

Design and Development of Multi-Lane Smart Electromechanical Actuators describes design concepts in electromechanical actuators by considering an actuator that has the capability to drive the aerodynamic and inertial loads of an aileron control surface similar to that of the Sea Harrier. It provides the necessary theoretical background to design smart multi-lane electromechanical actuation systems, and provides a general methodology that engineers (electrical, mechanical, mechatronic, aerospace or chemical) will find useful.

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The Institution of Engineering and Technology
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Chapter 1
Introduction

Early aircraft controls were totally manually operated, that is, the forces required to move control surfaces were generated by the pilot and were transmitted by cables and rods. As aerodynamics and airframe technology developed and speeds increased, the forces required to move control surfaces increased, as did the number of surfaces. Thus, to provide the extra power required, hydraulic technology was introduced. The pilot’s manual inputs were used to control the flow of hydraulic fluid to cylinders that subsequently moved the surfaces.

With further developments in aerodynamic and airframe technology and the arrival of airborne computers came the need for stability and control augmentation, which led to further developments in electrohydraulic actuation systems. In these systems, the pilot utilises flight control computers to electrically control hydraulic valves that in turn control the fluid flow. The control authority of computers was initially maintained well below that of the pilot to permit overcoming erroneous control inputs.

Today, with the advent of statically unstable aircraft, pilots are only able to control their aircraft with the assistance of complex and fast flight computers. This has led to the fly-by-wire concept, where the flight control surface hydraulic actuators are controlled entirely by electrical inputs. Here, the pilot input is sensed electrically and the mechanical control system is eliminated (i.e. rods or cables are eliminated).

The unavoidable element in the development of flight control systems (to date) has been in hydraulic actuators. This has been the case primarily because of their proven reliability and the lack of alternative technologies. However, the technology to build electromechanically actuated primary flight control systems is now available. Motors developing the required power at the required frequencies are now available with the use of high-energy permanent magnetic materials and compact high-speed electronic circuits. Therefore, development of this technology may mark the end of the hydraulic actuation systems, which are the last major non-electrical elements in the modern-day aircraft. This is considered to be an important step for the development of the future ‘all-electric’ aircraft. The purpose of the ‘all-electric’ aircraft concept is the consolidation of all secondary power systems into electric power. By eliminating the hydraulic and pneumatic secondary power systems, the aircraft benefits from improved maintainability and reliability, reduced life cycle costs, improved flight readiness and efficient energy use.
In recent years there has been much interest in the ‘all-electric’ aircraft, and its supporters have emphasised the serious consequences of hydraulic fluid loss on the aircraft safety. Furthermore, they highlighted the weight and space disadvantage of a centralised hydraulic power distribution system, especially when there are large distances between the primary power source and the actuators. Most experts agree that electric surface actuation systems will only show a weight-saving advantage if the hydraulic system is removed completely from the aircraft. There is, therefore, a strong case for the study of all-electric power actuators, their performance and integrity.

An enabling technology that took place in parallel with these developments was the development of brushless dc motors, which are relatively recent additions to electrical drives. The concept of brushless dc motors was first developed in 1964 by the National Aeronautics and Space Administration (NASA). The term ‘brushless’ is used to indicate that the motor is electronically commutated by sensing the rotor position, eliminating the brushes and commutators with the potentially dangerous sparking that is associated with conventional dc motors.

Although these types of motors were introduced in the early sixties, only recent developments in solid-state devices (for rotor position sensing and controlling armature input power) and rare earth magnetic materials have contributed to their availability and wide spread use, both for aerospace and ground applications. The need for the development of brushless dc motors was urged by some for the advantages they enjoy over the brush-type motors. However, it must be emphasised that brushless dc motors will not totally replace the brush-type motors, since each type has its place in the range of motor applications.

Traditionally more than a channel of operation has been used to meet the stringent integrity requirements for aircraft control surface actuators. In fact, aircraft with powered flying controls would be fitted with at least two independent hydraulic power supplies driving a tandem hydraulic ram.

Fly-by-wire systems pose more difficult problems since they must be designed for failure-survival, thus corresponding designs of surface actuators are commonly found to have up to four lanes of parallel ‘first stage’ hydraulic actuation driving a duplicated or triplicated hydraulic second ‘power stage’.

Although electric power actuation is commonly used for slow-acting aircraft controls such as trim motors, flaps and slats, so far it has found almost no application, except in demonstrator aircraft, for driving primary aircraft controls because of the speed of operation required. However, hydraulic controls have the advantage of high power to weight ratio, thus they can easily achieve high speeds, and are commonly used in fast acting aircraft controls.

Actuator manufacturers currently hold the opinion that purely electric power actuators will eventually be developed to replace electrohydraulic actuators in the next few decades. To make this possible, there is a need to advance various essential areas of technology, including: power electronics, motor technology, thermal design, gearbox design, layout or architecture, control systems, failure monitoring and detection techniques, reliability and integrity.
This book describes design concepts in electromechanical actuators by considering an actuator with four lanes of actuation that has the capability to drive the aerodynamic and inertial loads of an aileron control surface similar to that of the Sea Harrier, with two lanes failed. Each lane of the multi-channel actuator contains its own dedicated microprocessor/s to perform control and comprehensive monitoring tasks. The mathematical models of the actuator (in its lumped and three-phase representations) and the acting loads will be derived from first principles. Furthermore, failure development, detection and isolation methods that are generally applicable to multi-lane electric surface actuators will also be explained.

The overall design has to meet both local actuator and global aircraft system performance requirements. Actuator requirements include: identify the control surface and the loads over the full flight envelope; maximum rotary output; minimum output rate; operation bandwidth; first nuisance disconnect probability; minimum damping of position servo; failure transients envelopes (which will affect the aircraft response). The aircraft system conditions that must be met include: the control surface geometry, aircraft speed range, maximum aileron authority limits at different aircraft speeds, maximum aircraft manoeuvre; maximum allowable bank angle and roll rate following first (and any other permitted following) failures.

The structure in which the design will be presented is highlighted next, starting with Chapter 2 where relevant technologies (to multi-lane electric actuators) will be presented.

Chapter 3 will present and verify basic equivalent circuits for the brushless dc motor.

Chapter 4 will present control system design steps, with the load mathematical modelling across a defined flight envelope.

Alternative methods of output consolidation with associated monitoring methods will be presented first in Chapter 5. A Simulation Graphical Monte Carlo (SGMC) method is then presented to set the thresholds on the Monitoring-Voting Devices (MVDs) within the Fault Detection and Isolation (FDI) System. The general methodology of its implementation to scheduled and unscheduled threshold settings will be presented.

Chapter 6 will present hardware cross monitoring in torque and velocity summing architectures, by considering lumped servo models.

Chapter 7 will present both hardware and digital cross monitoring to the torque summed architecture by considering three-phase servo models and lumped digital models.

Chapters 6 and 7 will present statistical variations in lane disparities for potentiometers, tachometers and motor currents. These are inherent random disparities due to random internal variations in the motor parameters and feedback signals from feedback devices, which were produced from samples of 1500 tests for different aileron deflections and during different flight cases.

Appendix 1 lists simulation tests (from Chapter 6) in hardware cross monitoring.

Appendix 2 lists simulation tests (from Chapter 7) in digital cross monitoring.
Over the years, aircraft flight control systems have evolved from being totally manually operated and generated to (what is commonly known today as) wire driven, hence the expression fly by wire. Fly by wire technology is where human or computer generated desired flight controls movements are represented by electronic signals that feed and operate (hydraulic or pneumatic) valves to drive the control surfaces, thus enabling on-board computers to perform functions at frequencies beyond pilot’s abilities. With recent developments (in high-energy permanent magnetic materials and compact high-speed electronic circuits) it was made possible to extend this technology to the near future realisable concept of all-electric aircraft.

It is the aim of this chapter to present state of the art developments in electrohydraulic, electromagnetic and electromechanical actuators that led the way and influenced actuation designs for the future all-electric aircraft development.

2.1 The all-electric aircraft concept

The purpose of the all-electric aircraft concept is the consolidation of all secondary power systems into electric power. By eliminating the hydraulic and pneumatic secondary power systems, the aircraft benefits from improved maintainability, reduced life cycle costs, improved flight readiness and efficient energy use. This prompted numerous studies into electromechanical primary flight control actuation, which is considered a major milestone in the development of the all-electric aircraft, and led to the development of the electromechanical actuator systems, survivability, vulnerability and fault tolerance test programmes that addressed problems and benefits of the concept in the next generation fighters [1].

With advances in rare earth magnetic materials, solid-state power switching devices and high voltage dc power control, electromechanical actuator systems were made possible. However, concerns over the safety of flight mission reliability and combat survivability will have a major impact on the flight control hardware and architecture chosen for future aircraft. Retaining undiminished performance after failures places an enormous burden on the flight control system, and this capability is attained only at considerable cost and complexity. To meet these demands requires investigation into electromechanical actuator systems’ survivability and reliability [1].
Electromechanical actuators have advantages over hydraulic actuators, not because they are better actuators in a conventional comparison sense, but because of the changes they allow in the total secondary power system of the aircraft, which means less weight, no hydraulic fluid leakage and consequently a reduction in fire hazards, less complexity and cheaper installation and maintenance costs [2].

Studies by the authors in [3] showed the need to reduce the susceptibility (in US Air Force aircraft) to hydraulic fluid-related fires. They tabulated hydraulic fluid-related fire mishaps over the period 1965–1983 and found that:

- Over 1970–1975, 90 hydraulic fluid-related fire mishaps mounted a total cost estimate of $100 million. Significant reduction was noted over the period 1976–1979, which rose again over the period 1980–1983.
- Over the period 1965–1979, 37 per cent of the hydraulic fluid fires were in cargo aircraft.

To increase aircraft survivability, the report recommended:

- The development of a truly non-flammable hydraulic fluid (which increases the weight).
- Urged further development in seals and equipment compatible with both high pressure and non-flammable fluids.

The study concluded that future aircraft losses (due to inadequate maintenance procedures or on-board equipment and system failures) must be reduced, which is an indirect request to investigate alternatives to hydraulic systems for the future aircraft.

The above studies pointed out that electromechanical actuators have more benefits when compared with hydraulic actuators. Although hydraulic actuator systems possess high force/weight ratio and high reliability, hydraulic systems still suffer from pressure transients, leakages, fire hazards, survivability in battle field scenario, and high installation and maintenance costs. However, the studies also showed that hydraulic actuators were more reliable than electromechanical actuator systems, due to the high probability of mechanical gearbox failure. In fact, acceptance of electromechanical actuators for primary flight control surface applications will undoubtedly rely on the acceptance of the gears in the mechanical transmission [1].

Holmdahl [4] described a single-channel electromechanical actuator that was designed by Boeing and was installed on the inboard spoilers of the NASA-owned Quiet Short-Haul Research Aircraft, which was completed in June 1982 and has undergone flight tests since then. The units were designed to fit the installation space and to replace a hydraulic actuator. The electromechanical actuator comprised of a samarium-cobalt permanent-magnet motor manufactured by Inland Motors of Radford Virginia and a gear-ballscrew assembly (designed by Plessy Dynamics, Hillside, New Jersey). Holmdahl reported that the electromechanical actuator performance and response matched that of the original hydraulic actuator on the inboard spoiler and that of the outboard spoiler. The report also listed the further payoffs that this system offered, such as energy conservation, life cycle cost, operational readiness/dispatch reliability, reduction in aircraft weight, reduction in system development and
test efforts, improved survivability and the consolidation of energy/power sources into one electrical system. Furthermore, the report also emphasised the need for further investigations in performance after failure, tolerance and isolation, in order to achieve the above payoffs. It raised the importance of correct architecture (velocity versus torque/force summing) implementation and stated that a mix of the two types will be required in most aircraft applications. However, the choice between the two summing techniques is to a great extent, dependent upon failure considerations and failure mode characteristics. One consideration is that a velocity-summed actuator tends to fail open, zero output torque, leaving the control surface unrestrained. Whereas a torque-summed actuator tends to fail locked at the point of failure. Therefore, a surface such as a rudder, stabiliser or a canard that is flutter-prone tends toward torque summing, whereas spoilers tend to be candidates for velocity summing.

The reliability and redundancy study in the Holmdahl report revealed that the primary concern about the reliability of electromechanical actuators is the probability of a jam or structural failure in the gearbox section. A requirement that was established early in the Grumman programme, which stated that the electromechanical actuator systems reliability should be at least equal to that of the dual-tandem hydraulic actuator, so that, the probability of a critical failure be no greater than 10E-4 per flight hour [5]. The author also suggested that a fail-operational/fail-safe requirement would usually be acceptable in flight-critical applications, thus, the likely configuration of an electromechanical actuation systems would be a four-motor design that meets full performance requirements with three motors operating and has a degraded performance (get home capability) with two motors operating.

2.2 Electromechanical actuator development

This section describes key published programmes that gave basic design comparisons between electromechanical and hydraulic multi-lane actuators.

One of such programmes is the Lockheed-Georgia and Sundstrand Corporation teamed together with the USAF Flight Dynamics Laboratory. The programme was started in 1982 to develop flight worthy primary flight control electromechanical actuator hardware, evaluate its impact on flight control system design and demonstrate its capabilities in flight [6–9]. The programme described a dual-motor driving a single ball screw through a torque summing gearbox electromechanical actuator (Figure 2.1). The aircraft chosen to demonstrate the actuator was the C-141, and the surface to be controlled was the left-hand aileron. The actuator was intended to replace the existing hydraulic actuator on the aileron.

The controller was housed in the cargo bay of the aircraft with cabling to the actuator running along the inside of the wing’s trailing edge. The controller was to perform the functions of motor control, actuator position control, cross-channel equalisation and failure detection. The primary control input to the controller was the position error signal, which was generated by the mechanical feedback linkage and sensed by a Rotary Variable Differential Transformer (RVDT). The controller
drove the motor such as to maintain the position error signal as near to zero as possible. The voltage to the motor was controlled by pulse width modulating an internal 270 V dc power supply. The resultant average voltage was switched by six-switch inverter to the three-phase motor stator windings. The inverter switching sequence was determined by sensing the rotor position. For each rotor position, a unique switch sequence was selected from a custom programmable memory.

To minimise the force fight between the two motors, the controller equalised the current to each motor by regulating the output voltage. The current to each motor was monitored, and the difference between them connected to each motor’s control loop so as to reduce that difference. The authority of this equalisation was minimised to reduce the impact of any component failure in its circuit. The controller performs several fault management functions, and all the actuator position sensors (including the motor commutation resolver) were monitored for failure. The operation of the control electronics was monitored by comparing the actual control parameters with those of a model that is incorporated into each channel. The balance between each motor was also monitored, so that if currents were to differ beyond a pre-set limit, the failed channel is shut down. The actuator was also monitored for over-speeding and over-travel, and when a failure was detected,

Figure 2.1 Block diagram of the electromechanical actuator developed by Lockheed-Georgia, Sundstrand Corporation and USAF Flight Dynamics Laboratory
the faulty channel was isolated and the cockpit enunciators were illuminated. Furthermore, the input power, bridge voltages and the temperature were monitored to provide an additional level of servo protection.

Thompson [8] and [9] shared reporting the success of the Lockheed-Georgia flight tests and reported that the electromechanical actuator operation was smooth and trouble free with no electromagnetic interference problems. It was also reported that there were no thermal problems and that the actuator ran a few degrees cooler than the hydraulic actuator in the other wing. Furthermore, power consumption during extensive system exercises was considerably less than expected, and the actuator drew a maximum current of 12.5 A. Moreover, the aircraft performance was identical to that of a standard C-141, and the pilot comments were 100 per cent favourable, indicating that the electric system performance was identical to its hydraulic counterpart.

The Sundstrand programme tests also reported that (in 1986) the C-141 underwent six sorties of 9–12 h duration flight tests [9]. In these tests, the electromechanical actuator system was operated in both dual- and single-channel modes throughout 9,000–41,000 ft and 80–390 kts flight envelopes. It was also reported that the controller had operated successfully after detecting a sensor failure that led to a shutdown of the faulty channel.

The other interesting programme that addressed electromechanical actuation and servo power conditioners was that conducted by Virginia Polytechnic [10–12]. The studies presented the development of a discrete time model to simulate a samarium cobalt type permanent magnet brushless dc motor. The model was developed as part of an overall discrete time analysis of the dynamic performance of electromechanical actuators. This was conducted as part of a prototype development, built for NASA Johnson Space Centre as a prospective alternative to the hydraulic actuator used in the shuttle orbiter applications. The mathematical modelling included the interaction between the machine and its transistorised power conditioner unit, where excellent correlations were reported between the numerical simulation and the actual hardware experimental tests.

2.3 Electromagnetically summed actuators

Another unique and very interesting programme was reported by Pond and Wyllie [13]. The authors described a magnetic torque summed electromechanical actuator that comprised of a rare earth permanent magnet brushless motor with quad redundant windings. Despite the reported encountered problems, the design was a brave and interesting attempted to increase reliability by removing the gearbox assembly.

2.4 Electrohydraulic actuators

Multi-lane actuators were also investigated by Dowty Aerospace [14]. The author compared the application of Electro-Hydraulic-Servo-Valves (EHSV) actuator with
Direct Drive Valves (DDVs) actuation systems. Although an Electro-Hydraulic actuator system was described, architecture and the redundancy concepts remain of interest to state of the art actuators development. The programme investigated the Experimental Aircraft Project Actuator (shown in Figure 2.2), which was controlled by four computers, with each computer taking the error between demanded and measured positions to signal an EHSV. The actuator position was measured from four LVDTs within the actuator. The actuator operated from two hydraulic systems to provide redundancy, where each hydraulic system is associated with two EHSVs. This arrangement gives basic survivability of continued operation after one electrical and one hydraulic failure or two electric failures.

Each EHSV controlled a miniature actuator, and the group of the four actuators drove a tandem main control valve with local position control loop. The survival rule quoted previously means that the system has to continue to function on a single electrohydraulic channel, detecting and rejecting failures at the EHSV level, thus adding considerable complexity. The position of the EHSV spool position is measured by an LVDT and the measurement is compared against a valve computer model. In the event of error, a signal is generated to switch off a fail-safe solenoid valve, thus blocking the EHSV outputs and putting the failed valve actuator into a bypass condition so that it imposes no further constraint on the operation of the good lane.

The author then described the DDV actuator, which is much simpler (Figure 2.3). In this configuration the computer signals are flux summed at a single linear motor which directly drove the tandem main control valve. Again a position loop is closed.
by four LVDTs. In this case the loss of a hydraulic supply does not impact the number of electrical lanes. Here, monitoring was achieved by measuring individual motor currents and comparing them to expected values.

Finally, the author described a simpler solution to the variation to the EAP solution, that is, the development of EHSVs with an integral bypass mode which is automatically selected after failure. This gives the benefit of the use of four separate control devices which give a better ballistic survivability when compared to DDV actuators. Despite this, the author favoured the implementation of DDVs for the significant benefits they offer for multiplex actuators and better reliability for simplex actuators. However, size works against DDV actuators, hence the author emphasised on the need for further investigations in their development, so that this technology would be even more widely applicable.

Figure 2.4 shows the electrohydraulic system of the B-2 advanced technology bomber. In this system, the four aircraft computers compute and generate commands to the actuator (in response to sensor inputs and guidance commands) and perform fault detection and isolation as well as provide control communications with the remaining avionics system [15]. The Flight Control Actuation System (FCAS) contains the flight control actuators and a quadruplex Actuator Remote Terminals (ARTs). The ARTs are distributed on the airframe so that a set of four redundant ARTs operated on each side of the aircraft. Each ART contains loop closure electronics for all flight control actuators on its own side of the airframe.
Figure 2.4  The B-2 flight control actuation system architecture block diagram
The actuator of each of the 11 primary flight control surfaces contains a Direct Drive Servo-Valve, a single piston and numerous quadraplex transducers.

Two hydraulic systems powered each actuator, with a mode control switch valve to select one of the three (primary, alternate and bypass) operating modes. At least two identical actuators drive each control surface and if there were sufficient failures to cause an actuator to bypass, the other actuator will continue to control the surface, albeit with reduced hinge moment capability. The hydraulic system is designed so that each of the four independent systems is available to every flight critical surface.

The FCAS has the capability to detect failures and reconfigure, utilising judicious monitors to detect failures. Each ART has numerous monitors that continuously measure the health of central ART functions, such as power supply voltages, central processing functions and dozens of other monitors. When a failure is detected, the offending ART is disabled. Monitors that detect failures within individual branches employ cross-channel comparison, comparing a local signal with the voted value of that signal from all branches (transmitted via a cross-channel data link). If the difference between the local signal and the voted value is greater than a predetermined threshold for longer than a programmed interval, then the signal is declared failed and its branch is shut down. The voter excludes failed signals (in the voting process) for the remainder of the flight, unless the failed signal is cleared.

### 2.5 The book approach

This book is concerned with the development of a smart electromechanical actuator that utilises brushless dc motors. Two architectures will be presented, the velocity- and torque summed architectures. Two motor models will be used in the designs, one is based on the McCormick and Electro-craft lumped model, and the other is the three-phase Taft model [16–19]. These two models will be used to address the designs issues; however, other more complex models are also available in the literature for the reader, such as those in [9, 10, 11, 20, 21].

Initially, the McCormick model is implemented for its simplicity to initially examine the actuator, and the effect of any lane disparities due to internal parameter variation and external transducer fluctuations for both velocity and torque summing architectures. The three-phase Taft model is then implemented to examine the torque ripple effect on lanes disparity.

While the three-phase model is used in hardware cross monitoring, the lumped and the three-phase models will be used in digital cross monitoring. A Monte Carlo Method will be utilised to evaluate scheduled and unscheduled threshold values, so that the system will have a maximum false alarm rate of 1.0E-4. Designs will be assessed by examining the effect of failure transients on the actuator and its impact on the aircraft response.

It is the aim of this book that the reader develops clear design strategy in high integrity systems.
Chapter 3

Modelling the brushless dc motor

Motors used for space applications have to meet certain special requirements, such as high reliability and very long (seven to ten years) unattended operational life under hard vacuum. They have to operate over a wide temperature range and should be capable of withstanding space radiation during operational life, vibration and shocks during launch and out gassing of the materials used. Also, they should have low power consumption and obviously minimum weight and volume. These requirements impose constraints on lubrication, bearings and materials used, therefore, the concept of a non-contacting system is preferred, hence, brushless dc motors were first developed in 1964 by NASA [22]. The term brushless is used to indicate that the motors are electrically commutated by sensing the rotor position, thus eliminating the brushes and commutator that are found in the conventional dc motors. Although they were introduced in the early sixties, only recent developments (in solid-state devices and rare earth magnetic materials) have contributed to the availability and wide spread of their use, both for aerospace and ground applications.

The need for their development was due to some of the advantages they enjoy over the brush-type motors, however, it must be emphasised that the brushless dc motors will not in all likelihood completely replace the brush-type motors, since each type has its place in the range of motor applications. Brushless dc motors enjoy advantages that are traditionally associated with the conventional dc motors, such as better efficiency, response time and linearity. They also enjoy the following additional advantages:

1. There are no brushes to wear or limit rotational speed. The wear factor may be important in applications that require high reliability for the military/aerospace markets or in situations where the motor installation makes it extremely difficult to replace brushes. Similarly, the absence of commutator bars precludes the problem of the commutator bar oxidation; therefore, brushless dc motors can sit idle for years with no loss of performance. Also, there are environments where brush arcing is very hazardous such as fuel pumps, grinders of materials that produce explosive dust, or spraying with flammable materials. Moreover, unlike the brush-type (where sliding contact against the commutation bars limits the maximum practical speeds of the brush-type motors), the brushless-type speeds is limited by the commutation electronics; therefore, motors with speeds up to 80,000 RPM have been built. Furthermore, the elimination of brushes also eliminates certain forms of radio frequency interference (RFI).
2. The winding configuration of a brushless dc motor allows the windings to be more effectively cooled. The stator may be cooled by conduction through mounting flange that is equipped with cooling fins or may have a water or cooling jacket.

3. High energy product magnet materials, such as samarium cobalt or Neodymium-Iron-Boron compounds, allow for a smaller rotor diameter than in conventional brush-type motor. This means lower inertia, faster acceleration and an enhanced control system.

However, the disadvantage in using brushless dc motors is the increase in the complexity of the drive circuits which means a definite increase in cost. Although the reliability of the brushless dc motor itself is higher than that of the brush type, the reliability of the drive electronics may be reduced due to the increase in the overall system complexity.

A brushless dc motor has the same physical laws that govern the conventional brush-type dc motor, but the primary difference is the inverse relationship of the rotating and stationary elements. The conventional dc motor has a stationary magnetic field and a rotating armature, while the brushless dc motor has a rotating permanent magnet assembly and a stationary armature winding, as shown in Figure 3.1. Commutation of electric current in the stationary armature is done by switching on the appropriate windings (by solid-state amplifiers) as a function of rotor position. Hence, brushless dc motors are also known as electronically commutated motors [17, 22].

There are a variety of devices, such as optical, inductive and capacitive transducers, magnetoresistors and Hall Effect generators for rotor position sensing [17, 22]. The optical arrangement consists of a disk attached to the rotor, an optical source and a photosensitive device. The light reflected by the rotating disk falls on the photosensitive device to generate the switching signal. Generally, inductive-type sensors are not used because they require complex circuits for excitation and demodulation. Also, the moving magnetic field may inhibit self-starting capability.

![Figure 3.1 Cut-away view of brushless and conventional dc motor assemblies](image-url)
The capacitive transducers also require complex circuits and hence are not suitable. Magnetoresistors do not have polarised output. The Hall Effect generator develops a polarised voltage depending on the control current and magnetic field passing through it. The unique property of polarised output means that only two sensors are required. The Indium and Antimony Hall effect semiconductor devices are very small in size and have higher sensitivity. As Hall Effect sensors have high reliability, low power consumption and don’t require additional rotating disk and complex circuitry, they are widely used.

The following analysis was developed by the authors in [16, 17, 23]. This knowledge will be used in the development of the brushless dc motor lumped mathematical model in the actuator.

3.1 The lumped mathematical model

3.1.1 Elementary magnetics

A current carrying wire placed in a magnetic field (with the current flow perpendicular to the direction of the field) will cause a force to be exerted between the field producing element (permanent magnet, in this case) and the wire. This force is a cross product of the field strength, the wire length and the current in the wire.

\[ \mathbf{F} = \mathbf{I} \times \mathbf{L}_w \times \mathbf{B} \]

where

- \( \mathbf{F} \) Force vector
- \( \mathbf{I} \) Current vector
- \( \mathbf{B} \) Flux density vector
- \( \mathbf{L}_w \) Displacement vector in the direction of the current \( \mathbf{I} \)

The direction of the force depends on the orientation of the magnetic field and the direction of the current in the wire. If the wire is disconnected from the current source and either it or the field was moved in a direction perpendicular to the other, a voltage may be measured across the ends of the wire. The magnitude of the voltage, \( \mathbf{E} \), depends on the velocity, \( \mathbf{v}_o \), with which the wire is moved through the field, or

\[ \mathbf{E} = \mathbf{B} \times \mathbf{L}_w \times \mathbf{v}_o \]

Therefore, the force on the conductor in the magnetic field can be controlled by changes in \( \mathbf{B} \), \( \mathbf{L}_w \) or \( \mathbf{I} \). In motor design, these are affected by the permanent magnetic material circuit design \( \mathbf{B} \), the length and number of active conductors \( \mathbf{L}_w \) and the total current \( \mathbf{I} \). Similarly, the generated voltage is controlled by changes in \( \mathbf{B} \), \( \mathbf{L}_w \) and the rotor speed \( \omega_m \).

The above knowledge may be applied by considering the toroidally wound two-pole brushless dc motor shown in Figure 3.2. The conductors are shown wrapped around the stator. \( \mathbf{L}_w \) from the previous discussion is the length of the active conductors (number of turns times the length \( \mathbf{L}_w \)), the \( \mathbf{B} \) is the magnetic flux density in the magnetic circuit, which is comprised of the North and the South
permanent magnets and the toroidal core. If current is applied to the conductors then $\mathbf{F}$ is imparted tangentially to the rotor and torque results in the direction of the arrow. If the current source is removed and the rotor was to be allowed to rotate, an alternating voltage will appear across the terminals as shown in Figure 3.3. The alternating voltage results from the change in magnetic polarity as the rotor turns, where each coil sees alternately a North and then a South Pole. The amplitude and frequency of the voltage are dependent on the speed of rotation.

3.1.2 Commutation and the back emf waveform

One problem with the above design is that there is no way to make the motor rotate for a complete revolution. That is, as the current is applied in the direction shown in Figure 3.2, the motor will rotate to a point where the torques from the North and South fields are balanced, and no amount of current in the winding will cause torque. Thus the developed torque obtained for a given current is not constant as the rotor position is varied between two points where the field is balanced. However, intuition indicates that the amount of torque available for a given current should increase to some maximum at a midway between the two points where the
fields are balanced. Therefore, to make the motor rotate through a complete revolution, we need to commutate the current in the windings as a function of rotor position. This means changing the direction of current in the coil at the proper time. In the brush-type motors this is done by the arrangements of brushes and commutation bars. For brushless dc motors, electronic switches and rotor position sensors are used.

Figure 3.4 schematically shows a typical brushless motor wound with three phases and the voltages seen between the phases (when the motor was driven as a generator at constant speed). The motor is wound to provide overlapping, sinusoidal three-phase voltages, electrically spaced by 120°. The North/South balance for each winding occurs where the voltages go through zero and reverse polarity. It is the method and type of winding as well as the geometric and physical characteristics of the rotor and stator that create the sinusoidal shape of the terminal voltage, the back emf (BEMF) of the motor. The torque produced by a motor with a given winding and physical geometry is directly related to the voltage it produces.
when the rotor is externally driven, or when the motor is used as a generator. In fact, the motor torque constant, $K_T$, and the motor voltage or BEMF constant, $K_E$, are equal when expressed in Nm/A and V/rad/s respectively.

This applies not only to the motor constants, but also to the wave shape throughout the commutation cycle. In other words, if the BEMF waveforms are viewed as a function of rotor position on an oscilloscope (when a constant current is applied to the motor), the torque as a function of rotor position will vary in a similar manner. Therefore, there is a logical way to decide when to commutate a brushless motor. In the discussion above it was found that commutating at the zero crossing of the BEMF waveform is not a good place to start since there is no resultant torque no matter how much current is injected into the phase. Also, it was found that peak torque for a running motor is achieved at the peak of the BEMF waveform. Hence if the motor is to run smoothly between commutation cycles, then the commutation zone should take place at the points of the centre of the peak BEMF waveform, as shown in Figure 3.5. This will provide equal sharing of motor phases in the process of producing torque, but will introduce variations of 50 per cent in $K_T$ [16, 17].

![Torque ripple as a function of rotor position with constant current input and commutations at the positive half of the BEMF](image)

**Figure 3.5** Torque ripple as a function of rotor position with constant current input and commutations at the positive half of the BEMF.
This might be considered acceptable in some applications such as ventilators and pumps. To improve this variation of torque during commutation, or the torque ripple, commutation frequency is doubled by using the negative and the positive half of the BEMF, as shown in Figure 3.6. This will yield a 13 per cent ripple. For a three-phase motor, this is considered to be the best that can be achieved when considering the BEMF waveform.

Further improvements to reduce the torque ripple may be achieved. One way is to increase the number of phases of the motor, thus increasing the number of commutation cycles per revolution. This means using smaller and smaller portions of the peaks of the BEMF waveform and thereby limiting the torque ripple excursion. This method is similar to that of increasing the number of commutation bars of the brush-type motor. The penalty in applying this to brushless dc motors is the increased number of transistor switches and commutation frequency.

A second method, and one that is more common, is to modify the shape of the BEMF waveform so that it is more trapezoidal (i.e. flat during the period of commutation), as shown in Figure 3.7. The ripple of this type of motor is much less than a motor with a sinusoidal BEMF waveform. In practice, the BEMF waveform cannot be made precisely flat due to the armature reaction, but torque ripple of 5–6 percent is achievable.
3.1.3 Model development

The lumped model is a simplified mathematical representation, thus it is very useful in examining designs at the early stages. It is possible to assume that the brushless dc motor has the same mathematical model of that of the brush type if the following is assumed [16, 17]:

- Commutations are properly implemented using rotor position sensors and semiconductor switches, so that the brushless dc motor is commutated to follow the trapezoidal BEMF waveform.
- The values of $K_T$ and $K_E$ were assumed to be constant during commutation.
- Losses in semiconductor switches were assumed to be minimum and are neglected.

Thus, the brushless dc motor may be represented by the equivalent circuit shown in Figure 3.8. From this circuit the mathematical model may be derived as follows:

$$V = I \cdot R + L \frac{dI}{dt} + K_E \cdot \omega$$  \hspace{1cm} (3.1)

where

- $R$  Resistance of a phase winding
- $L$  Inductance of a phase winding

Figure 3.7  Commutation with trapezoidal waveform
V Terminal voltage

$K_E$ Voltage constant of a phase winding over the conduction angle induced back EMF

$I$ Total of phase currents

$\omega$ Motor shaft angular velocity

The above relationships hold well for the common brushless motor structures, although there are lower order effects due to mutual inductance between windings, overlapping conduction angles and unequal rise and fall times of current due to differing charge and discharge paths. The dynamic equation for the brushless dc motor when coupled to a load is given by

$$K_T * I = (J_m + J_L) \frac{\partial \omega}{\partial t} + D * \omega + T_f + T_L$$  \hspace{1cm} (3.2)

where $K_T$ Torque constant of the motor winding

$J_m$ Motor moment of inertia

$J_L$ Load moment of inertia

$D$ Viscous damping coefficient

$T_f$ Friction torque

$T_L$ Load torque

In brushless dc motors $T_f$ is small, usually only due to bearing drag, the viscous damping coefficient is also very small, and both items can usually be ignored in dynamic performance calculations.

Equations 3.1 and 3.2 may be rewritten as follows:

$$\dot{I} = \frac{1}{L} [V - I * R - K_E * \omega]$$  \hspace{1cm} (3.3)

$$\dot{\omega} = \frac{T_o}{(J_m + J_L)}$$  \hspace{1cm} (3.4)

where $T_o = K_T * I - T_L$ (Net Torque)

Equations 3.3 and 3.4 represent the lumped mathematical model of the brushless dc motor, which doesn’t exhibit any torque ripple effects behaviour.