CRANFIELD UNIVERSITY

ZHONGTIAN TAI

AIRCRAFT ELECTRICAL POWER SYSTEM DIAGNOSTICS, PROGNOSTICS AND HEALTH MANAGEMENT

SCHOOL OF ENGINEERING
Aircraft Design Programme

MSc
Academic Year: 2009 - 2010

Supervisor: Dr. Craig Lawson
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ABSTRACT

In recent years, the loads needing electrical power in military aircraft and civil jet keep increasing, this put huge pressure on the electrical power system (EPS). As EPS becomes more powerful and complex, its reliability and maintenance becomes difficult problems to designers, manufacturers and customers. To improve the mission reliability and reduce life cycle cost, the EPS needs health management.

This thesis developed a set of generic health management methods for the EPS, which can monitor system status; diagnose faults/failures in component level correctly and predict impending faults/failures exactly and predict remaining useful life of critical components precisely. The writer compared a few diagnostic and prognostic approaches in detail, and then found suitable ones for EPS. Then the major components and key parameters needed to be monitored are obtained, after function hazard analysis and failure modes effects analysis of EPS. A diagnostic process is applied to EPS using Dynamic Case-based Reasoning approach, whilst hybrid prognostic methods are suggested to the system. After that, Diagnostic, Prognostic and Health Management architecture of EPS is built up in system level based on diagnostic and prognostic process. Finally, qualitative evaluations of DPHM explain given.

This research is an extension of group design project (GDP) work, the GDP report is arranged in the Appendix A,

Keywords:

FMEA, DCBR, power converter, DPPM

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AC	Alternating Current
AEA	All-Electric Aircraft
ANN	Artificial Neural Network
APU	Auxiliary Power Unit
AVIC	Aviation Industry Corporation of China
BIT	Built-In Test
BITE	Built-In Test Equipment
BPCU	Bus Power Control Unit
BTB	Bus Tie Breaker
C.G.	Centre Gravity
CBM	Condition-based Maintenance
CBR	Case-based Reasoner
CCAR	China Civil Aviation Regulations
CFDS	Central Fault Display System
CMS	Central Maintenance System
DCBR	Dynamic Condition Based Reasoner
DPHM	Diagnostic, Prognostic and Health Management
EBOM	Electrical Bill of Material
EHA	Electro-Hydrostatic Actuator

ELMC	Electrical Loads Management Centre
EMA	Electro-Mechanical Actuator
EPS	Electrical Power System
FHA	Functional Hazard Assessment
FMEA	Failure Mode, Effects and Analysis
FMECA	Failure Mode, Effects and Criticality Analysis
FTA	Fault Tree Analysis
GCB	Generator Current Breaker
HTRU	High Transformer Rectifier Unit
ISHM	Integrated System Health Management
IVHM	Integrated Vehicle Health Management
LRM	Line Replaceable Modular
LRU	Line Replaceable Unit
LTRU	Low Transformer Rectifier Unit
MEA	More Electric Aircraft
OMS	On-board Maintenance System
PDF	Probability Density Function
PPDU	Primary Power Distribution Unit
RUL	Remaining Useful Life

1 INTRODUCTION

1.1 Research Objectives

The secondary power of the aircraft generally comprises electric, hydraulic and pneumatic, which are off-take from engines. Each kind of power has its own power generation, distribution and control components, making a complex secondary power system. This hybrid secondary power is low cost-efficiency for the considerable power loss in the power conversion and transfer.

In order to improve power efficiency, researchers and designers managed to reduce the types of secondary power to simplify the system architecture and achieve power optimization, by using electric power to replace other power source. The notion of more-electric aircraft（MEA）, and all-electric aircraft (AEA) came to our eyes towards this kind of power optimization. In an MEA, the electric power replaces the hydraulics to power the flight control system, landing gears retraction etc or substitutes the bleed-air system to power the environment control system. The above all drives the increase in electrical power capacity. For instance, the main electric power capacity is about 300KVA in traditional aircraft, such as Airbus A-340, and the number is doubled in new generation aircraft-Airbus A-380， while it even reaches 1M VA in Boeing-787 dream liner! [23]The architecture of Electrical Power Systems (EPS) becomes more and more complex with increasing requirement of power capacity as well as power quality. Therefore, problems rose in terms of these trends, that is, how can we reduce the failure rate of EPS in all the flight phase or mission? How can we detect the fault in EPS in its incipiency? How can we estimate the remaining useful life (RUL) of a major component of EPS? How can the life cycle cost of system be reduced?

To answer these questions, the writer intends to develop a set of health management methods which meet the electrical power system needs. The writer also would:

Choose EPS in Group Design Project (GDP) as the case study.

Consider two subsystems of EPS, that is, power generation, power distribution (primary distribution and secondary distribution).

However, the detection and measurement techniques, for instance, sensors and probes, are less concerned in this thesis, in terms of limited time.

1.2 Diagnostic, Prognostic and Health Management (DPHM) background

“Health management concept is the capability to make appropriate decisions about maintenance actions based on real-time monitoring, diagnosis and prognosis information, available resources, and operational demand.” [19; 24]This concept has been applied in the space shuttles and international space station for years [28], the ground based and monitoring centre could receive and analyse the real-time status data, and then respond. In recent years, the aircraft manufacturers like Boeing and Lockheed-Martin developed the technology respectively [28], aiming to prognostic the failure optimizes the system architecture, reduce redundancies, and integrate the supply chain and lower operational cost.

EPS provide power not only to flight-critical loads, like flight control system, but also to non-essential loads, such as the entertainment system on board. Accordingly, to detect and acquire the operating status of EPS is very important to the whole aircraft. Therefore, the DPHM technology is introduced into the design of the aircraft electrical power system. The DPHM system could monitor and analyse operating status of EPS in real-time, transmitting health data to the ground base. As a consequence, the development of DPHM in EPS is able to further improve system reliability. Till now, health management technology is a novel notion for electrical power system on board, which might provide a new design philosophy and synthesis for the design.

1.3 Methodology

The writer intended to develop generic DPHM methodology in EPS of the aircraft with a design flow as follows.

Firstly, the author chose the architecture of EPS of Flying Crane as a case study, Here, Flying Crane is an advanced jet liner designed in the group design project (GDP) [5].

Secondly, monitoring all the components of EPS is unnecessary, because it will increase complexity and cost. Hence, it is suitable to monitor the key parameters and major components in EPS. The writer conducts functional hazard analysis (FHA) to judge primary functions and hazard levels. Then, the failure modes effect criticality analysis (FMECA) would be performed to rank the failure mode and get the critical components in EPS. Fault Tree Analysis (FTA) is used to ensure no other failures are missed.

Thirdly, the writer will diagnose and prognose the selected components of EPS, by using appropriate diagnostic methods and prognostic approaches to components respectively. Then, build the architecture of DPHM for electrical power system.

Finally, an evaluation will be performed of the DPHM system.

2 Literature Review

2.1 Health management history and trends

Diagnostic, Prognostic and Health Management (DPHM) developed from traditional status monitoring technologies. Over forty years, with the increasing quality and complexity of system and equipment, DPHM has undergone several stages: exterior test of system, Built-In Test (BIT), Testability being an independent subject, proposal and development of synthetically diagnosis, Condition-Based Maintenance (CBM), prognostic and health management (PHM).

2.1.1 From exterior test to Built-In Test (BIT)

Early-generation aircraft relied on manual detection and isolation of problems on the ground [41] .These systems typically were analogue and independent of one another. They needed measurement instruments, such as a voltmeter, to link and measure. In this stage, diagnosis is difficult and the fault is not easy to find. In most circumstance, the operators' experiences played an important role in diagnosis.

After that, some flight-critical systems and equipments, for instance, the avionic system, required to be monitored, so that the operators could be clear about the working status. If there are any faults, the operators could take some actions in time. Therefore, BIT was employed. At first, BIT was used to test a few critical parameters, while the judgement of fault was employed by hand. As aircraft systems became more complicated and integrated, more powerful BIT was required, which led to the appearance of BITE (Built-In Test Equipment) BITE could detect and isolate faults automatically, However, BITE results at that time were often confusing, not reliable, and difficult to use. [41] It was applied to many systems on airplanes such as the Boeing 707, and the McDonnell Douglas DC-8 [18] .

2.1.2 Testability being an independent subject

With the development of exterior test and built-in test, the problem of test design increased. For the complex systems and components, the fault diagnosis needed exterior test and built-in test synthetically. Also, the design of BIT/BITE should be embedded into systems. Hence, the system should consider the testability design. In 1985, United States Department of Defense issued *TESTABILITY PROGRAM FOR ELECTRONIC SYSTEMS AND EQUIPMENTS* (MIL-STD-2165) [1]. In this standard, testability was put on the same level with reliability and maintainability for design requirement of product. It means that testability becomes an independent subject.

2.1.3 Proposal and development of synthetically diagnosis

Synthetically diagnosis was brought forward by National Defense Industrial Association (NDIA) of U.S. [40] In 1983, and it aimed at detecting and isolating all the faults, including incipient faults, effectively in minimum cost. Synthetically diagnosis was not a new technique or combination of techniques, but a process of engineering system. It was performed in the beginning of system design and went through the life cycle of a product. For instance, central fault display system (CFDS) using one display to get the maintenance indications for all the systems on the airplane. Many of the systems on later Boeing 737s, the Airbus A320 family and the McDonnell Douglas MD11, employ this techniques [18].

In the 1990s, the ARINC 624 was developed to solve the problem that a single fault on an airplane could cause fault indications for many systems. Central Maintenance System (CMS) or Onboard Maintenance System (OMS) were used to combine the information of failure or fault from multiple systems and provide extra aid to support condition-based maintenance. For example, systems on the Boeing 747-400 and 777 airplanes monitor certain bus data by using the integrated maintenance system [18].

2.1.4 CBM and PHM introduced

These years, condition-based maintenance (CBM) and prognostics and health management (PHM) emerged to reduce operation and support cost, and bring a revolutionary maintenance concept. The technology mainly focuses on employing various sensors onboard an aircraft, while the monitoring software decodes the output data from these sensors automatically. PHM can estimate the remaining useful lifetime or time to failure of a failing component, and arrange logistics in advance [18] .Aircraft can get maximum availability and capability under this concept. PHM was adopted for the F-35 to keep logistical system minimization and get maximum sortie generation rate with the lowest cost.

At present, the major obstacles lie in the assessment of uncertainty of the remaining useful lifetime or time, and detect the intermittent failure of electric product [40].

2.2 Categories of Diagnosis

Diagnosis, as an important part of health management, detects and identifies an impending or incipient failure condition [18] .A simple diagnostic process could be found in three steps as shown in Figure 2-1:

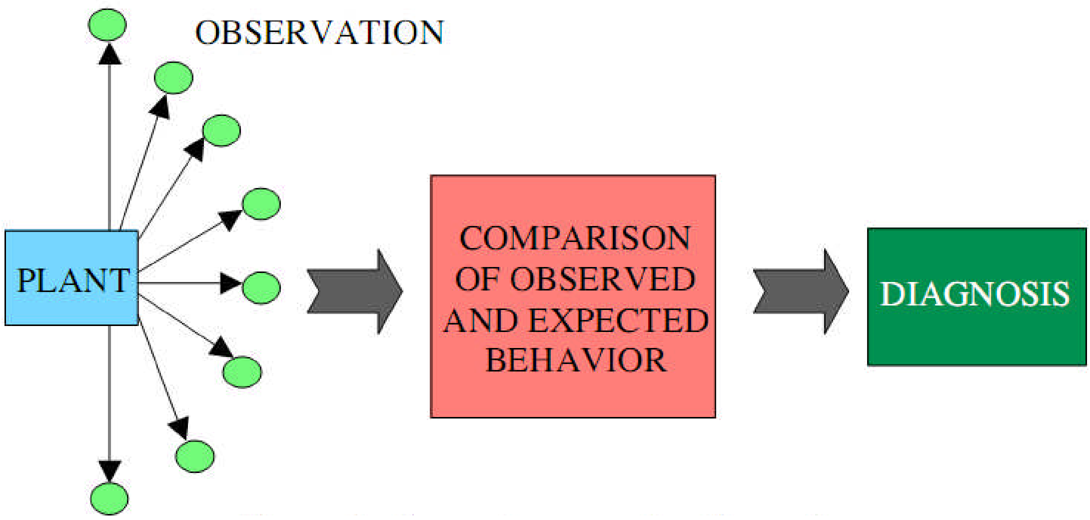

Figure 2-1 Typical diagnostic process [4]

“Many engineering disciplines have specific approaches to model, monitor and diagnose anomalous conditions, Therefore, there is no “one-size-fits-all” approach, which could build diagnostic and health monitoring capabilities for a system” [4]. For fault diagnosis, different authors defined different categories. In ref [11] , fault diagnosis typically separates into two major categories: model-based and data-driven approaches. While according to Robert Milne [35], fault diagnosis can be divided into analytical model-based, knowledge-based and data-driven approaches.

The author adopted a classification suggested by [4] , which includes four categories for intelligent systems:

- Rule-based expert system
- Case-based reasoning system
- Model-based reasoning system
- Learning system

2.2.1 Rule-based expert reasoning system

Rule-based expert reasoning system is a typical artificial intelligence technique. The basic reasoning rule statement is “if-then-else” [29]. These “rules” are particular sentences and the reasoning engine will search for these sentences in the rules that match patterns in the data. The meaning of "if" is "when the condition is true," the "then" represents "adopt behaviour 1" and the "else" implies "when the condition is wrong, adopt behaviour 2” [24]. The maintainers could load a series of rules into the database of a rule-based expert system, from which the diagnostic solutions could be suggested. [4].

2.2.2 Case-based reasoning (CBR) system

Case-based reasoning system is a specific reasoning engine of knowledge solutions which means to use past problem to solve current problems [4]. It employs both qualitative and quantitative algorithms of reasoning by learning from old cases. It uses a case library of maintenance to record all previous incidents, faults, and malfunctions of equipment. This technique works well in the condition of poorly understood problem areas [4] . Figure 2-2 shows the process of CBR.

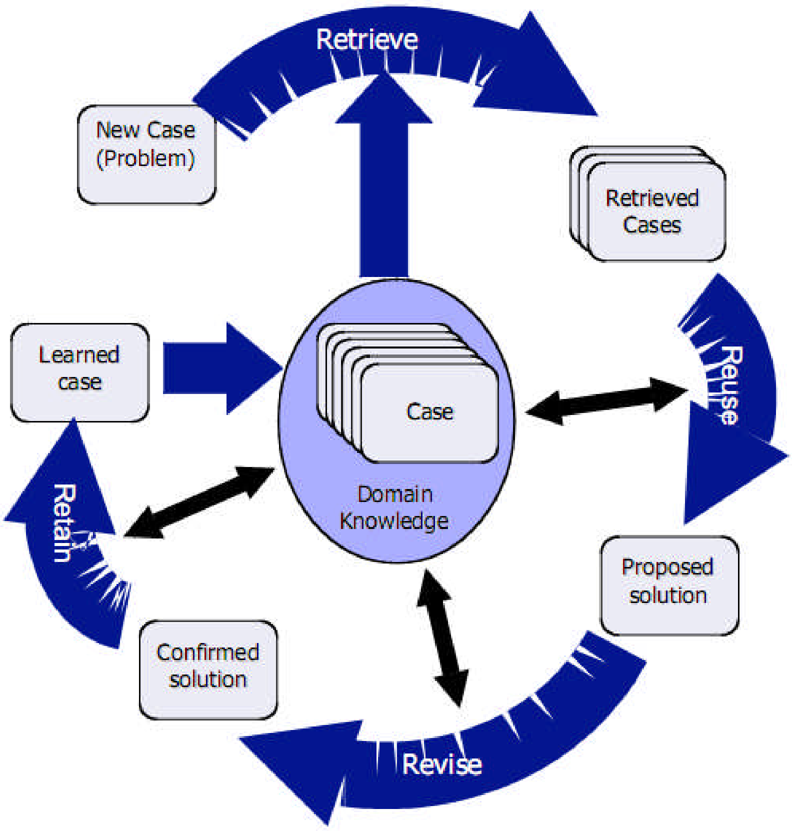

Figure 2-2 The Retrieve-Reuse-Revise-Retain Process [4]

2.2.3 Model-based reasoning system

Model-based reasoning is" a broad category that describes the use of a wide variety of engineering models as the foundation for the knowledge and the techniques applied for diagnoses".[4] This kind of technique has been used in the electrical power system such as Deep Space and Earth Observing applications as diagnostics and troubleshooting [4; 29].

2.2.4 Learning systems

Learning systems are data-driven approaches that are derived directly from routinely monitored system operating data. These data could be calibration, power, vibration, temperature, pressure, oil debris, currents or voltages.

Learning systems depend on the presupposition that the statistical characteristics of the data are stable unless an unanticipated failure event was found in the system [4; 29].

2.2.5 Comparison of diagnostic approaches

The four diagnostic approaches mentioned in above chapters have different advantages and drawbacks respectively, which were developed by manufactures or researchers severally, which focus on application to the particular systems or components. The writer made a comparison with these four methods to choose one of them as a tool for studying EPS.

Approach	**Advantages**	**Drawbacks**	**Application**
Rule based	Increased reliability for diagnostic decision making	Domain knowledge acquisition step translate into rules	Small systems (used by NASA)[4]
Case based	Use past problems in case library to solve current problems	Hard to obtain a completeness of case base	Gas turbine (used by GE)[18]
Model based	Efficient and accurate in linear dynamic systems	Hard to non-linear complex behaviours	Power industry, space craft etc
Learning	Lower dimension of noisy data and provide monitor	Need long term historical data record	Pattern recognition

Table 2-1 Comparison of four diagnostic approaches

2.3 Category of Prognosis

“From the users’ point of view, what distinguishes prognosis from diagnosis is the provision of a lead time or warning time to the useful life or failure and that this time window is far enough ahead for the appropriate action to be taken” [18].

In PHM, the term “prognostics” covers not only the functions of fault/failure detection, fault/failure isolation, but also the functions of enhanced diagnostics, material condition, performance monitoring, and life tracking, rather than just prognostic functions alone[9]. Figure 2-3 shows briefly a set of possible prognosis approaches, which are applied to various systems, besides, the trend of their relative accuracy and implementation cost are also illustrated[18] .Three main approaches are given in the following sections.

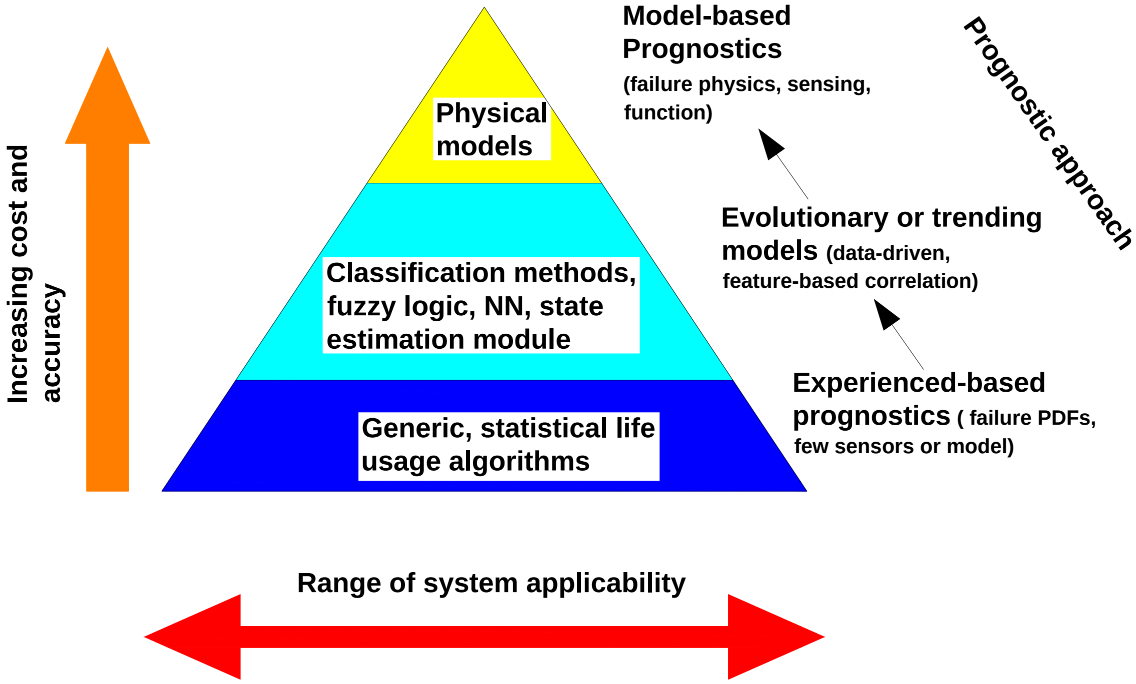

Figure 2-3 Prognostic technical approaches [18]

2.3.1 Model-based prognosis techniques

Model-based approach employs a dynamic model to predict, which includes physics-based models, autoregressive moving-average (ARMA) techniques, Kalman/particle filtering, and empirical-based methods [18] .Model-based approaches have advantages that they make remaining useful life(RUL) estimates even without any measurable events, however, when the related diagnostic data is provided, the model could be adjusted based on this diagnostic data. Figure 2-4 shows a schematic of physics based modelling approach, which belongs to model-based.

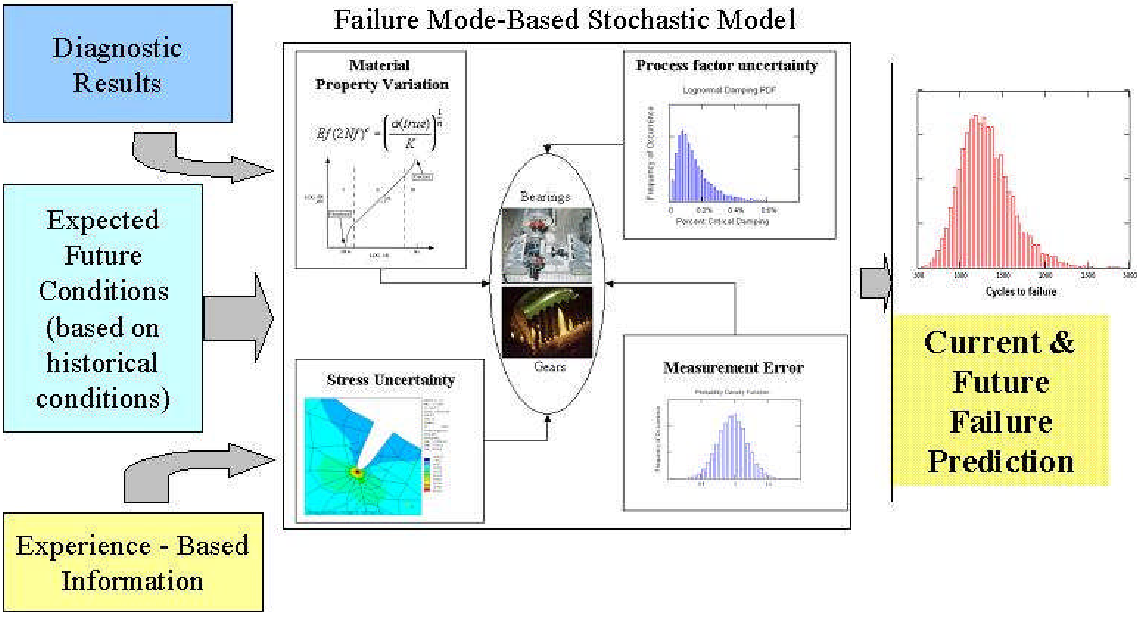

Figure 2-4 Physics-model based approach [31]

2.3.2 Probability-based prognosis techniques

Probability-based approach is effective in the situations of statistical failure points supplied by historical or experimental data. These methods have less information demands than model-based methods, for the reason that the information belongs to various probability density functions (PDFs). [18] These PDFs generally include Bayesian Probability Theory, Weibull Distribution model and Remaining Useful Life Probability Density Function. Figure 2-5 gives an example of Weibull Distribution.

2.3.3 Data-driven prediction techniques

In many cases, the prediction is difficult to get although the historical or statistical data has been obtained. Therefore, the data-driven approach provides a nonlinear network method which is able to be tuned by the fixed formal algorithms so as to get the prediction outputs. “Nonlinear networks include the neural network, which is based on signal-processing techniques in biologic nervous systems, and fuzzy-logic systems, which are based on the linguistic and reasoning abilities of humans.” [18] Figure 2-6 shows a neural-network prognosis schematic.

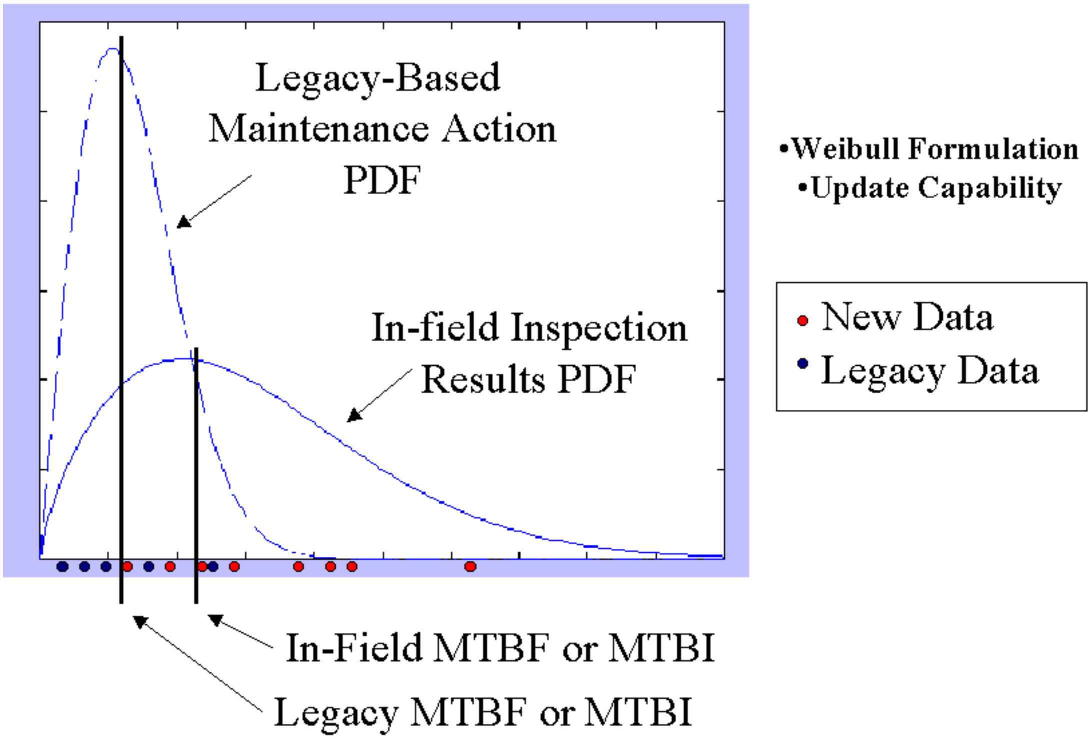

Figure 2-5 an example of probability-based approach [31]

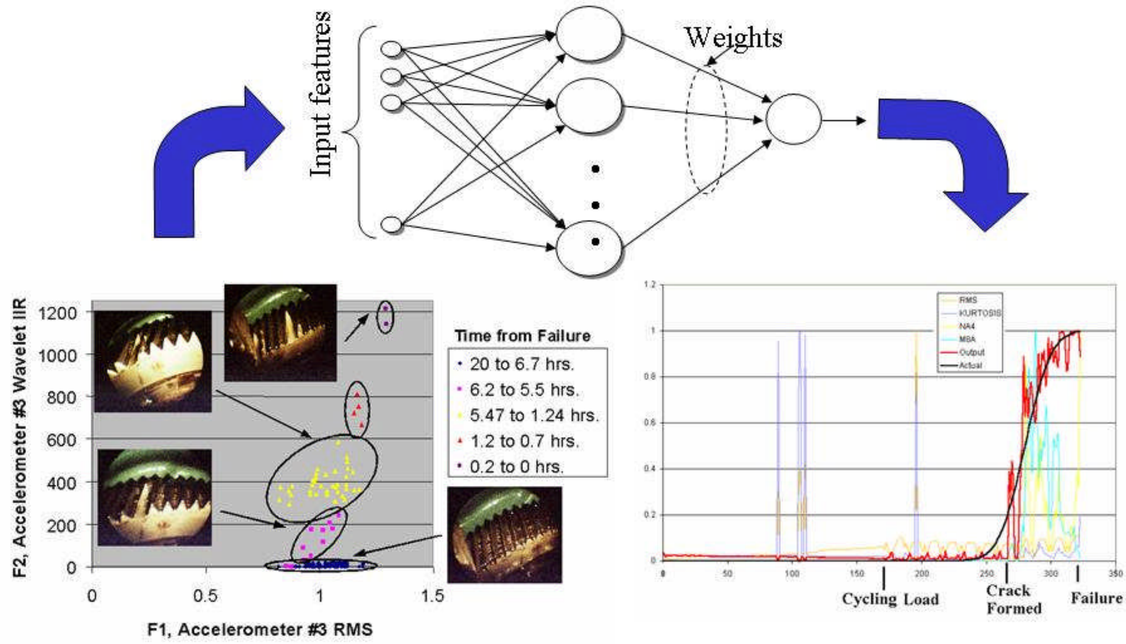

Figure 2-6 Neural-Network schematic [31]

2.4 Typical faults in EPS

In the writer's view, faults in EPS can be divided into three parts, that is, power supply faults, power control and distribution faults and loads faults.

Power supply is composed of main generators, emergency generator, batteries and external power. The winding wore out is a common phenomenon. The Figure-4 shows motor bearing arc failure. Another case is the degradation of batteries [7; 26] .

Power distribution comprises the bus bars, various kinds of contactors, SSPCs and wires, cables, etc. Among them, the wire arc and wire aging is difficult to detect. In fact, the wire ageing and insulated protection was hot topic in recent years [27].

Electrical loads are increasingly complex in MEA, which also raises a challenge to electrical power capacity and reliability. Typical loads are actuators, fuel pump/valves, and fans etc, which are critical in flight.

In this thesis, the faults of loads are not considered due to the short time.

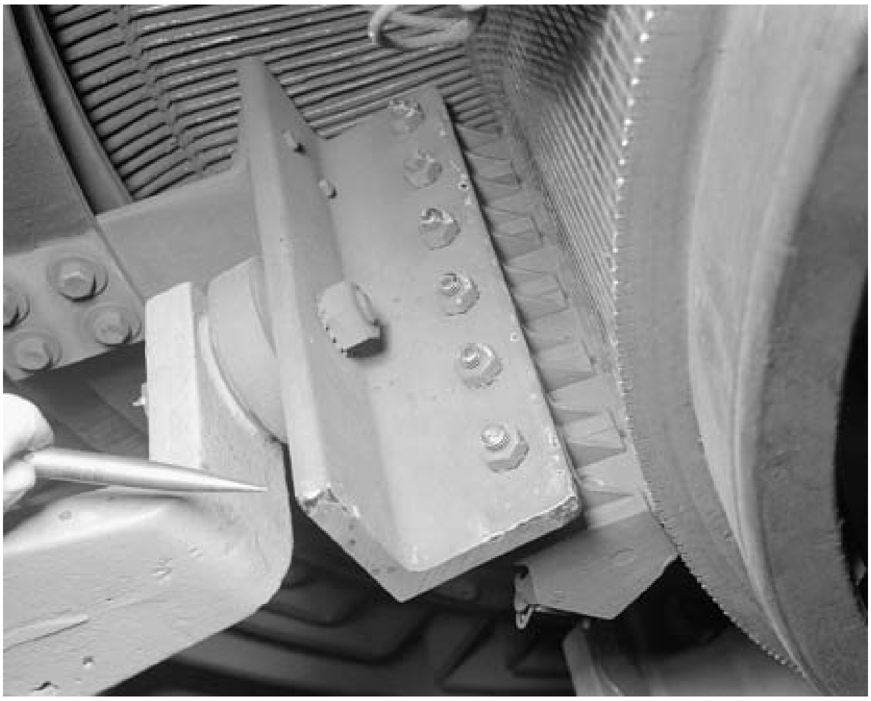

Figure 2-7 Motor and bearing arc caused failure from NASA [2]

3 EPS Selection and Analysis

The Electrical Power System mainly provides power to all the loads in an aircraft, and it becomes more and more complicated with increasing loads requirement these days. Consequently, a highly reliable system with fault/ failure easily detected or isolated is required by designers, manufacturers and customers. A generic health management for EPS would be developed in this thesis, and the writer will choose one kind of EPS in civil aircraft as case study. The EPS of Flying Crane, which is the aircraft designed in the writer's Group Design Project, is selected. Firstly, the architecture of EPS of Flying Crane will be introduced in this chapter, and then detailed analysis from the point of function and safety will be performed, from which the critical components and their associated parameters in EPS would be determined.

3.1 Introduction of EPS of Flying Crane

Flying Crane jet liner is 130-seat mainly for China's domestic market in 2020, and it is designed as an all-electric aircraft. That is to say, electrical power becomes the only kind of secondary power, replacing the work of hydraulics and pneumatics totally [5] .

Flying Crane has a power generation capacity of 500kVA supplied by two 250kVA variable frequency (VF) starter/generators (230V/ 360-720Hz). A 90 KVA VF generator powered by the Auxiliary Power Unit (APU) could switch to work in emergency conditions. A hybrid system is employed in the aircraft, which could provide 230V AC, 270VDC, 28VDC and 115VAC/400Hz operating voltages to drive the various airframe systems. The primary power distribution is supplied by two Primary Power Distribution Units (PPDUs), while, six Electrical Loads Management Centres (ELMCs) transform and manage power to the loads [10] .

The two 90 Ah Ni-Cd batteries make sure that there is no power break. The working status of batteries is monitored and controlled by Battery Charger Rectifier Units (BCRU). The Ram Air Turbine (RAT) will be extended automatically while all the generators fail and provide the crew with sufficient power while attempting to restore the primary generators or carry out a diversion to the nearest airfield [23] .The overall system is fully automatic achieved by utilizing the solid state devices. The mass of wiring in the system is reduced by utilizing the remote power distribution architecture which has been used in B777 and B787, also, reliability and maintainability are improved. The design of EPS in Flying Crane complies with China Civil Aviation Regulations (CCAR)-25.

The author found a few problems in the architecture of EPS in the stage of GDP:

a. The EPS of Flying Crane provides four types of electric power, that is 230V VF AC, 115v 400 Hz AC, 270 V DC, and 28V DC power, which seems complicated.

b. Another question is the mismatching between the ground power supply and main power, which will decrease the reliability of EPS.

To make the architecture simple and clear, the writer discussed with EPS designer, and made a few modifications to the system.

a. The power supply voltage of main start/generators changes from 230v to 115v, which is matching with the voltage of ground power. Hence, the power type becomes 115V VF, 115v 400 Hz AC, 270 V DC, and 28V DC.

b. The ground power input adapts each main AC bus, for the reason that "the aircraft generators are variable frequency (VF) and the frequency of the AC power depends upon the speed of the appropriate engine, the primary AC buses cannot be parallel [23] ."

In the following pages, the architecture of EPS and typical secondary power distribution are represented.

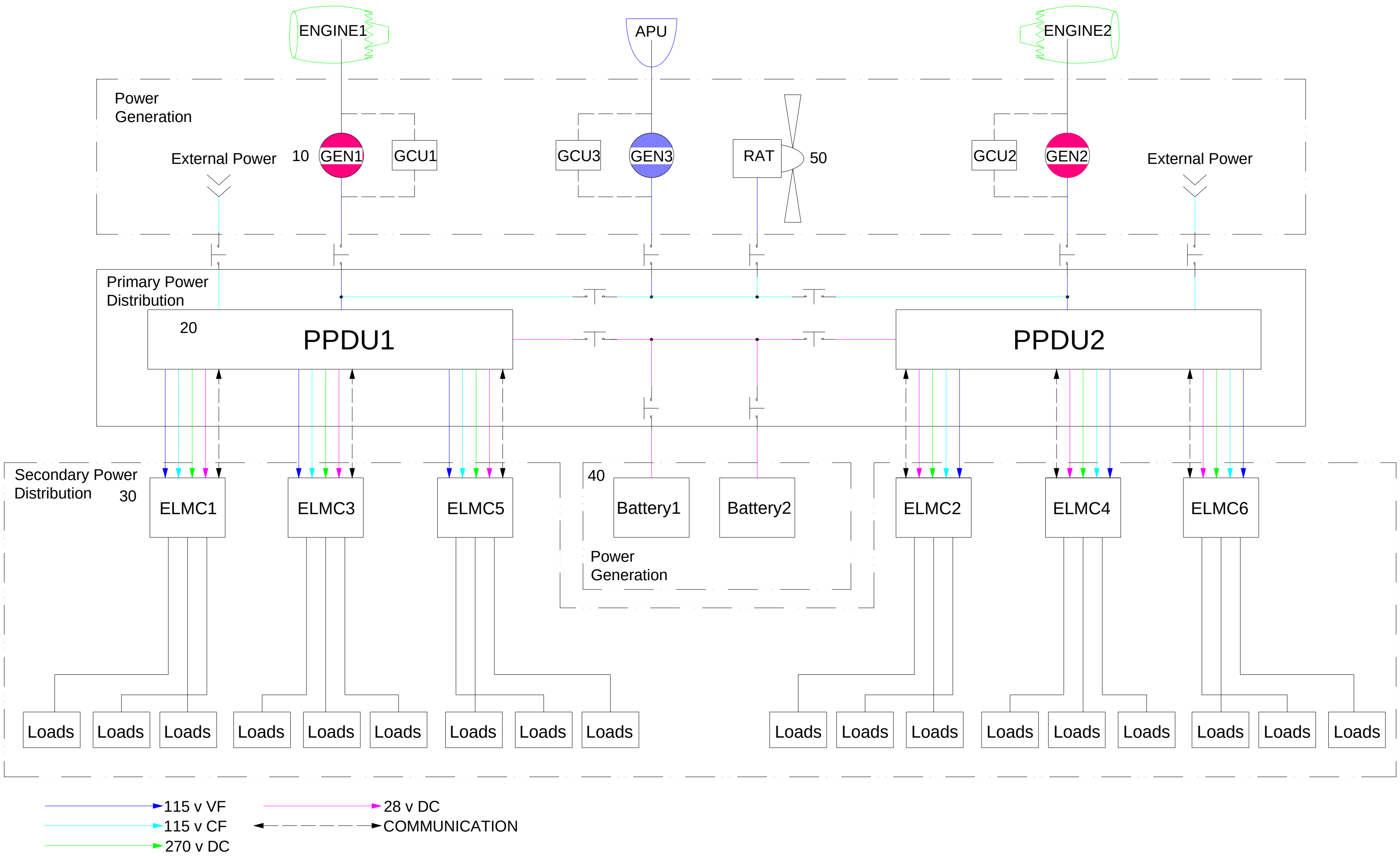

Figure 3-1 Architecture of EPS of Flying Crane [10]

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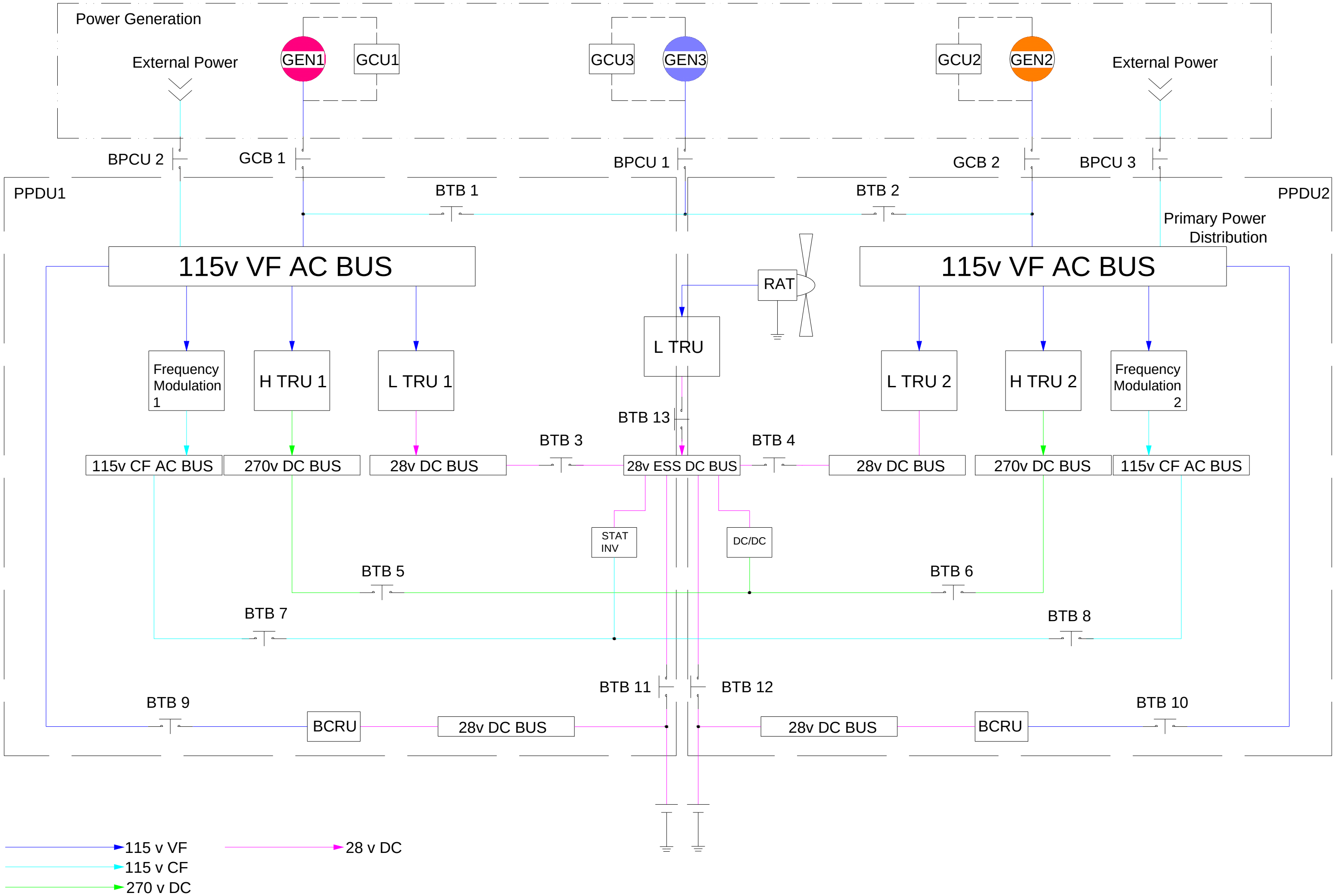

Figure 3-2 Components of PPDU of Flying Crane

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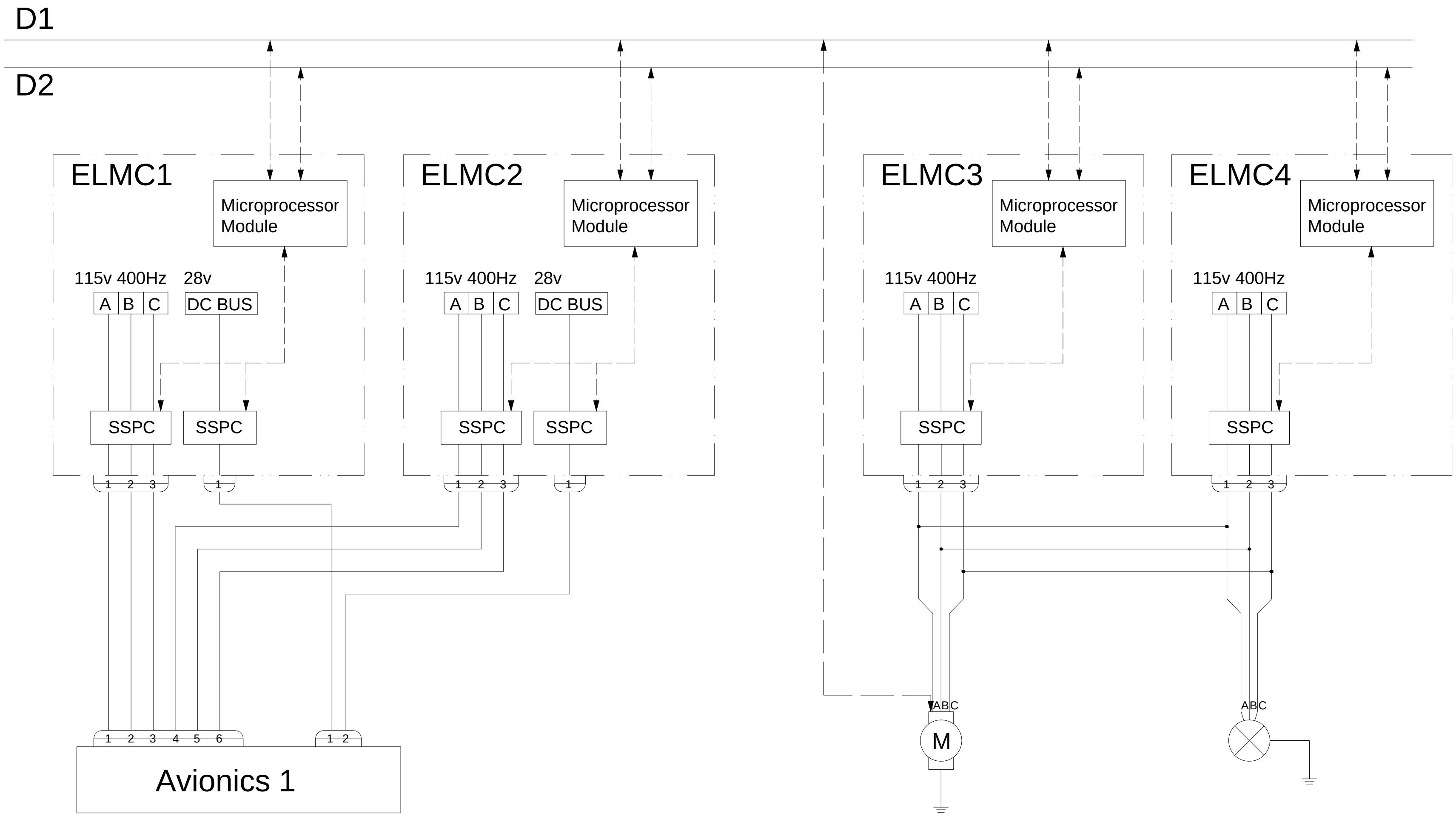

Figure 3-3 Typical Secondary Power Distribution of Flying Crane

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The table below expresses some typical components of the EPS, in the left column, they are power components, whilst in the right column, the loads of EPS are listed.

Item	**Component (Power)**	**Item**	**Component (Loads)**
1	Variable Frequency Generators	**16**	Electric powered actuators
2	APU Generator	**17**	Fuel pump
3	Ram Air Turbine	**18**	Fuel valve
4	Battery	**19**	Fans
5	Generator Control Unit	**20**	Indicated light
6	Generator Circuit Breaker	**21**	Lights
7	Bus Tie Breaker	**22**	Galley
8	Bus Power Control Unit	**23**	Sensors
9	Low/ High Transfer Rectifier Unit	**24**	Probes
10	Frequency Modulator		
11	Static Inverter		
12	Electrical Loads Management Centre		
13	Circuit Breaker		
14	Solid State Power Controller		
15	wires, cables		

Table 3-1 Electrical Bill of Material (EBOM) of EPS

The writer needs to analyse the failures mode effect of the components in left column, so as to isolate specific parts. However, the failure modes effect of loads are not developed in this thesis, because the loads are various and complicated.

3.2 Functional Hazard Assessment (FHA) of EPS

3.2.1 Introduction of FHA

The electrical power system transforms and manages power to different loads in the aircraft, among them; it should provide no-break power to critical loads in the emergency conditions. The design should be compliant with CCAR-25.1351 and be compatible with the aircraft level safety requirements. To meet the safety consideration in CCAR-25.1309, a FHA must be implemented to the EPS, from the results of FHA, all of the catastrophic, hazardous, major and minor failure conditions in EPS design are identified, which the system may cause, or contribute to, by malfunction, failure to operate, or as normal response to any unusual or abnormal external factor [29] .

The FHA provides the data on fault symptoms needed to test the EPS; hence it is helpful to develop the FMECA and other analyses.

3.2.2 Failure Condition Severity and Effect Classifications

To measure the worst potential effect due to the system failure, the airworthiness regulations classify five kinds of failure condition severities according to different effects to aeroplane, flight crew and occupants [16] .

They are:

Category I (Catastrophic)   A failure may cause multiple fatalities and hull loss;

Category II (Hazardous)   A failure may cause severe injury, property damage or mission loss;

Category III (Major)   A failure may cause minor injury, property damage or mission delay or degradation;

Category IV (Minor)   A failure may not cause minor injury or some degree economic loss, but it may result in unscheduled maintenance or repair.
“The safety objectives associated with Catastrophic Failure Conditions, may be satisfied by demonstrating that:

(1) No single failure will result in a Catastrophic Failure Condition; and
(2) Each Catastrophic Failure Condition is Extremely Improbable. “[16]

3.2.3 FHA Summary of EPS

The FHA of EPS of Flying Crane was performed by Mr Chen in the Group Design Project stage, and the writer made some modifications corresponding to the updated architecture of EPS. The detailed results of FHA are listed in Appendix B.

The result of FHA shows that the function of controlling the system independently among different DC channels, and the function of vital and essential loads power supply control will great influence the flight safety and system performance. For the next research work, the table 3.2 list the failure conditions catastrophic and hazardous which are directly from Appendix B.

Failure condition	**Severity**
Loss of 28VDC channels controllability operating independently	Catastrophic
Loss of 270VDC channels controllability operating independently	Catastrophic
Loss of vital loads power supply control	Catastrophic
Loss of Essential loads power supply control	Catastrophic
Loss of 115V/400Hz channels controllability operating independently	Hazardous
Loss of main feeder protection	Hazardous
Loss of Busbar protection	Hazardous

Table 3-2 Summary of EPS function hazardous assessment

3.3 Failure Modes, Effects and Criticality Analysis of EPS

3.3.1 Introduction of FMECA

The objective of an FMECA is to" identify all modes of failure within a system; its first purpose is the early identification of all catastrophic and critical design correction at the earliest possible time." [15] FMECA ranks each failure mode according to a severity classification of failure mode and probability of occurrence. Generally, FMECA could deploy two steps:

a) Failure modes and effects analysis (FMEA)

It is used to analyse the effect of failures on the system and to classify every potential failure by severity so as to obtain the basis of constituting design, improving process and taking compensated measure [16] .

b) Criticality analysis (CA)

Each failure mode is ranked according to the combined influence of severity classification and probability of occurrence, which evaluates generally all kinds of possible effects of failure modes. The CA is an extension or supplement of the FMEA and is based on the FMEA [16] .

3.3.2 Analysis approaches

When FMEA is performed, each single item failure is to be considered the only failure in the system [16] .

According to [16], there are two primary means for accomplishing an FMEA.

One is the hardware approach, which lists hardware items exhaustive and analyzes their possible failure modes.

The other is the functional approach. As each item is designed to carry out several functions, it needs to be recognized and then be classified as outputs. The outputs are listed and their failure modes analyzed.

For complex systems, for instance, electrical power system, a combination of the functional and hardware approaches may be considered.

The CA has qualitative approach and quantitative approaches. The qualitative approach is suitable for the case that the product technology state or failure rate data is not available. The quantitative approach is suitable for the product technology state or failure rate data is sufficient [16] .

3.3.3 FMECA Summary of EPS

FMECA is an interactive process and it goes through the whole design flow, whereas, the FMECA here is based on the preliminary design of EPS. The result of FMECA can be seen in Appendix C.

3.4 Fault Tree Analysis of EPS

3.4.1 Introduction of FTA

"To design systems that work correctly we often need to understand and correct how they can go wrong" (Dan Goldin, NASA Administrator, 2000) [13] FTA identifies, models and evaluates the unique interrelationship of events leading to failure, undesired events/ states and unintended events/ states [29] .The development of FTA is based on the results of FHA and FMECA.

FTA is a top-down approach and it can identify the root causes using a deductive mode (general to the specific). It also provides risk assessment, employing cut sets to get qualitative results and calculating probability to get quantitative results.

3.4.2 Basic Reliability Equations

Generally, the probability of basic event can be calculated to measure the reliability.

Equation 3-1 $R = e^{-\lambda T}$

Equation 3-2 $R + Q = 1$

Equation 3-3 $Q = 1 - R = 1 - e^{-\lambda T}$

Where R= Reliability or probability of success, Q= Unreliability or probability of failure, λ =component failure rate= 1/MTBF, T= time interval (mission time or exposure time)

From the equations, it can be inferred that the longer the mission time or exposure time, the higher the probability of failure; the smaller the failure rate, the lower the probability of failure [12].

3.4.3 Calculation of Top Event Probability of EPS

The calculation of top event probability is in accordance with Boolean algebra; however, the probability of the basic events must be got. The fault tree architecture can be seen in the Appendix D. Through the fault tree architecture, the writer got the relation between the top events and basic events. It shows that no single component leads to a catastrophic event. So the architecture of EPS is reasonable. Table 3-3 expresses the result of failure probability.

Failure condition	**Severity**	**Required Probability**	**Calculated Probability**
Loss of 28VDC power	Catastrophic	10^{-9}	2.18×10^{-19}
Loss of 270VDC power	Catastrophic	10^{-9}	6.3×10^{-13}
Loss of 115V/400Hz AC power	Hazardous	10^{-7}	6.3×10^{-12}

Table 3-3 Results of FTA

From the above table, it can be seen that the calculated result is far smaller than the expected. Although it meets the required probability from FHA, the calculated data is impractical. This reason lies that the writer built the fault tree architecture without consideration of failure rate of the wires and connectors. For the preliminary design phase, these parameters are unobtainable. After consulting safety and reliability designers, the writer took the wires and connectors as a whole event and put them to the higher level, and then got the reasonable failure rate of top events.

3.5 Summary and discussion

The purpose of this chapter is to identify the crucial components to be monitored based on the results of FHA, FMECA and FTA. FHA stands at the system level to analyse the functional hazard. FMECA ranks the failure mode of the components. FTA identifies the root cause of a failure, so that any catastrophic can not be resulting from a single failure. According to these analyses, the components that can lead to hazardous or catastrophic effect are identified. The key parameters and monitoring methods related to these components can be acquired, either. The table 3-5 shows the detailed information of required parameters and components.

Failure condition	Loss of 28v DC power supply		
Severity	**Related component**	**Parameters**	**Detection method**
Catastrophic	two main generators, APU generator, RAT, batteries, BTB, GCB, LTR, ESS TR	voltage, current, control signal	voltmeter, ammeter, CMS
Failure condition	Loss of 270v DC power supply		
Severity	**Related component**	**Parameters**	**Detection method**
Catastrophic	two main generators, APU generator, RAT, BTB, GCB, HTR, ESS TR, DC/DC converter	voltage, current, control signal	voltmeter, ammeter, CMS
Failure condition	Loss of 115v/400HZ power supply		
Severity	**Related component**	**Parameters**	**Detection method**
Hazardous	two main generators, APU generator, STAT INV, BTB, GCB, Frequency Modulator	voltage, current, control signal	voltmeter, ammeter, CMS

Table 3-4 Analysis results of EPS

4 DIAGNOSIS RESEARCH FOR EPS

In this chapter, a diagnostic method will be developed based on the techniques review and selections of critical components of EPS. Firstly, the Case Based Reasoning (CBR) would be chosen from the four diagnostic approaches mentioned in chapter 2. Secondly, the writer discusses the design flow and challenges of this approach. Thirdly, the Dynamic Case Based Reasoning (DCBR) will be introduced into work, and then a diagnostic architecture of EPS will be built based on this technique.

4.1 Chosen of diagnosis approaches

Case Based Reasoning was chosen mainly based on the following comparison of the characteristics between the four diagnostic approaches.

As mentioned before, the Rule-based expert system executes diagnostic tasks from a hypothesis, and uses a conflict set to solve problem. Its advantages include an increase in the availability and reusability of expertise. However, the challenges of Rule-based expert systems include domain knowledge acquisition and the completeness, consistency and correctness of the derived rule base for the complex systems [4] .

The Model-based Reasoning system uses the engineering models from the real and basic products to diagnose the system, and it is very efficient and accurate for a linear dynamic systems. The drawbacks of this technique exist, also. The difficulty of acquiring accurate model for the non-linear and complex system is its bottle neck [4]. Particularly, it is difficult for the writer to build the diagnostic model for the EPS in two months, and the lack of detailed product parameters in preliminary design stage is another reason.

Learning system belongs to the data-driven approaches, and it could identify the special noise caused by faults and provide the monitoring capability. However, learning system needs long term historical data record to improve the precision of diagnosis [4] .

Case-based Reasoning system uses the past problems in the case library to solve the current problems, and it is well suited for poorly understood problem areas. However, the challenge lies in achieving completeness of case base [4] .

According to the above comparison and analysis, the Case-based Reasoning system is suitable for the diagnostic development of EPS. In the following paragraph, the CBR will be discussed detailed and applied into EPS.

4.2 Structure of Case-based Reasoning system

Case-based reasoning system makes use of “4R” to diagnose the faults of the systems, where “4R” is Retrieve, Reuse, Revise and Retain respectively. A new problem is matched against cases in the case library, and one or more similar cases would be retrieved [38]. A solution suggested by the matching cases then would be reused and tested. Generally, the solution probably will have to be revised, unless the retrieved case is a close match, after that, a new case would be retained. Case revision (i.e., adaptation) is often carried out by the managers of the case base [18] .The structure of CBR is illustrated in Figure 4-1.

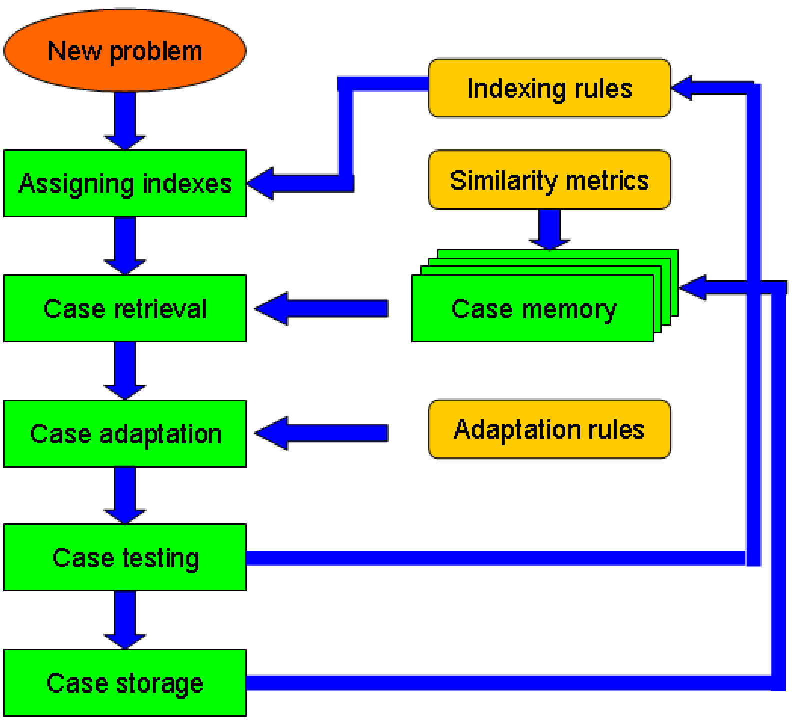


Figure 4-1 The structure of CBR [18]

4.3 Challenges of Case-based Reasoning design

When the writer begins to design the CBR diagnostic architecture for EPS, two major challenges might to be considered, which are coding of cases and adaptation.

4.3.1 Challenge 1: Coding

To describe specific situations, the data to be coded must achieve a desired level of abstraction. Meanwhile, the coding scheme should address both quantitative and qualitative information. A case, therefore, might have the following fields: [18]

• *Condition*. Shows the conditions of components currently

• *Measurement*. Measurements or features of case, they can be qualitative and quantitative

• *Symptom (optional)*. Other unusual observations complementary

• *Component*. Describes the component affected and the related fault

• *Action*. Shows the solutions or actions adopted before

• *Result*. Describes the success and failures associated with the preceding actions

• *Explanation*. Includes an explanation why a particular decision was taken [18]

An example of case combination is shown in Figure 4-2.

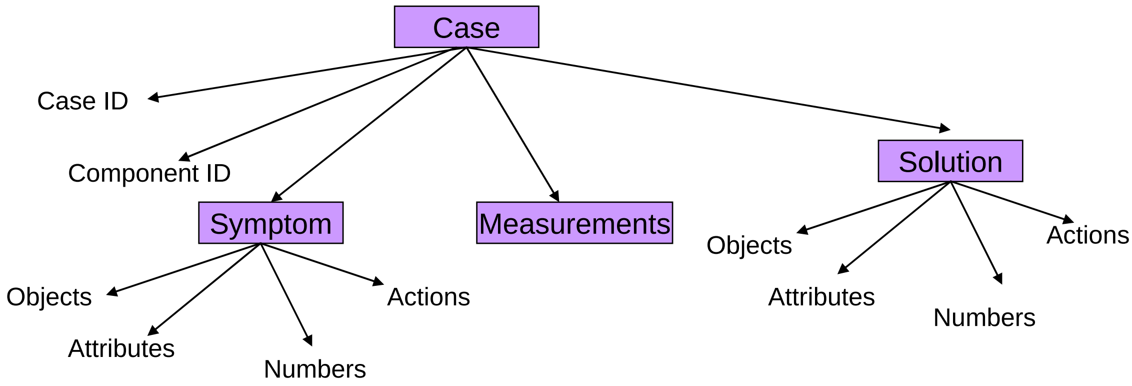
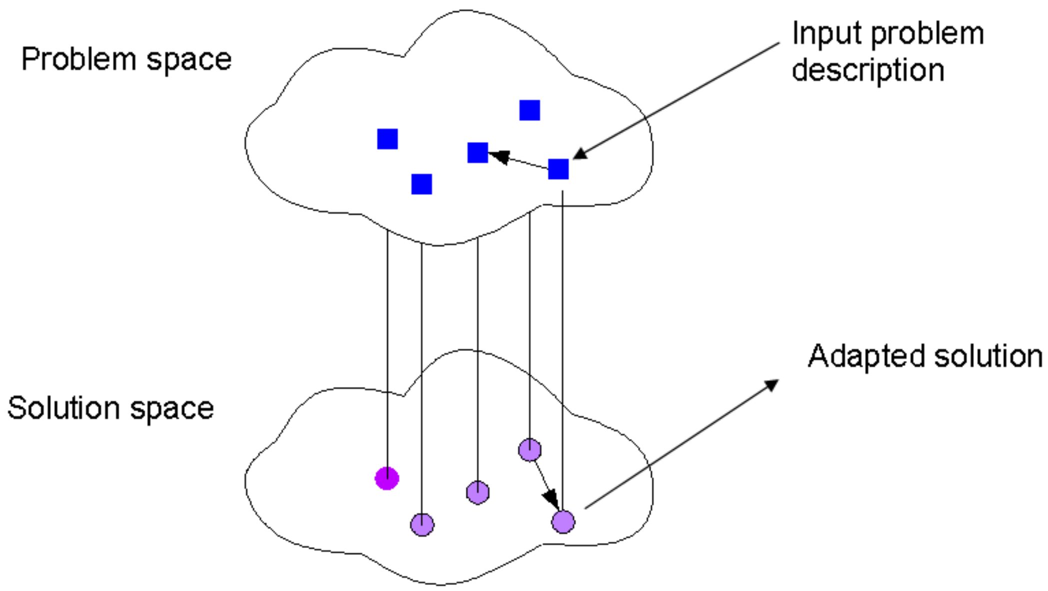


Figure 4-2 a case with its components (Adapted from GEORGE VACHTSEVANOS etc. 2006)

4.3.2 Challenge 2: Adaptation

In many circumstances, a case queried from the case library might not be the exact solution. An adapted solution, therefore, must be created so as to match the input problem description as closely as possible [18] ; Figure 4-3 shows the relationship between the input problem description and adapted solution. In general, there are two methods, interpolation and extrapolation, used to solve these problems.



4-3 A schematic of an adapted solution [18]

4.4 Dynamic Case-based Reasoning

Dynamic Case-based Reasoning (DCBR) is a variation of Case-based Reasoning system, providing continuous updating of cases over time. The core of the “dynamic” feature of the CBR is to store a statistic corresponding to different situations in a single case, and it is difference from those storing several cases of a similar configuration [18] .

The case identification mechanism of DCBR works like this:

When a problem is identified as a case, it will be compared to the old cases in the base. If it is a new case, the data base would update to include this case. Else if the case has some similarity with old cases, the old ones would be updated so as to contain the new features. Figure 4-4 illustrates the operating process of DCBR.

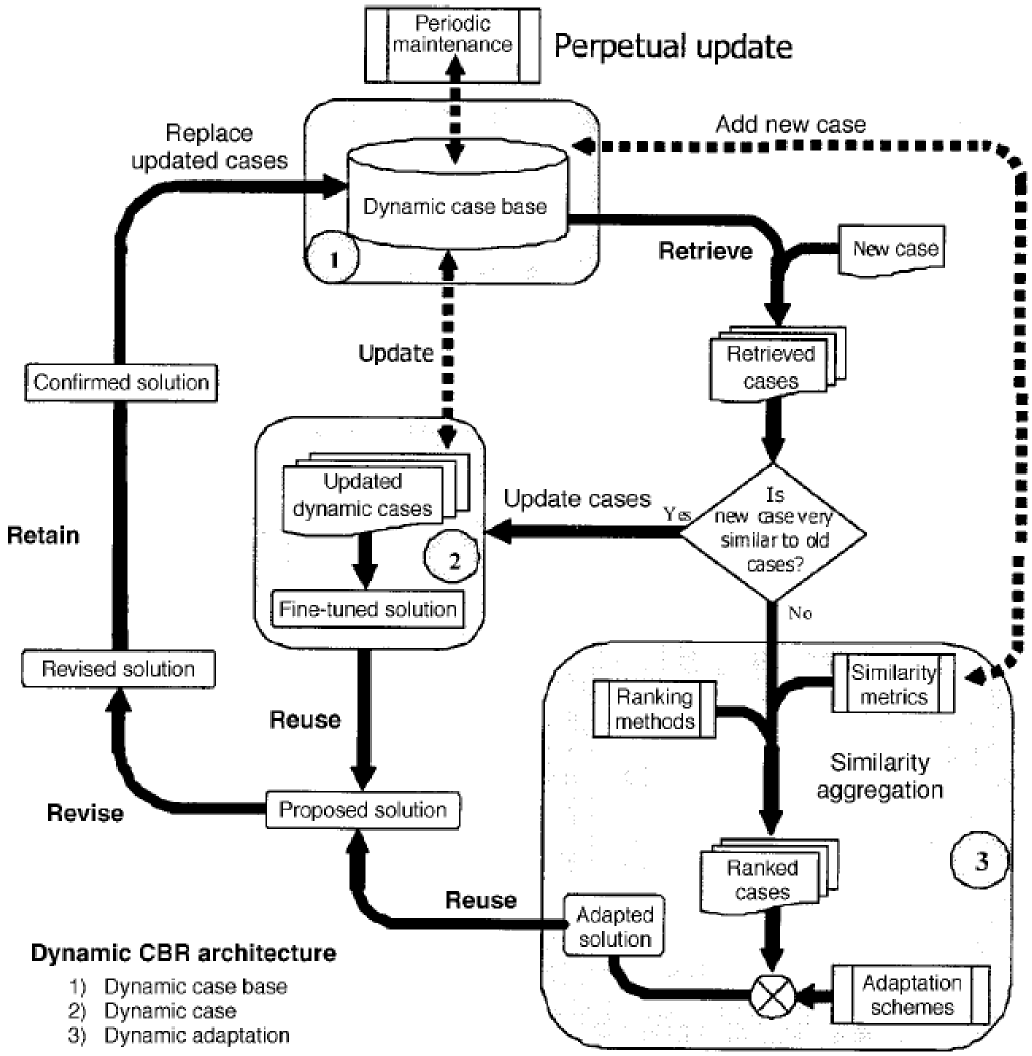


Figure 4-4 Operating process of DCBR [18]

4.5 DCBR Solution for EPS

It is unnecessary to monitor and diagnose everything in the EPS, from the point of economy, what is more, the complex and great numbers of sensors will degrade the reliability of the monitors.

In chapter3, the critical components and related parameters have been discussed and selected. According to these, DCBR techniques will be applied to them. A diagnosis flow chart of EPS is designed by using DCBR methods, as shown in Figure 4-5.

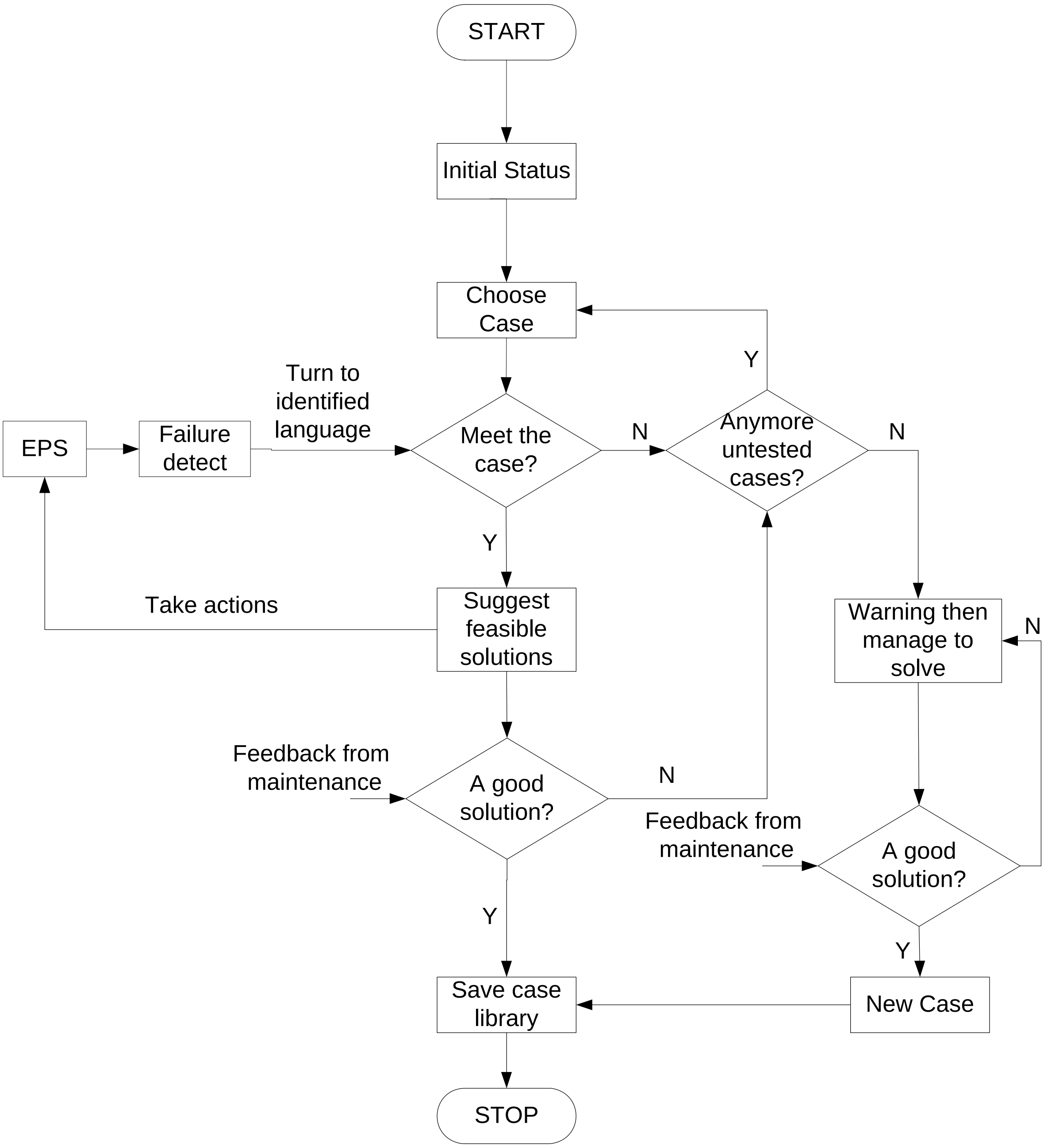


Figure 4-5 Fault Diagnostic flowchart for EPS using DCBR

From the flowchart, the diagnostic process is given as follow:

a. When the EPS operating, the sensors and probes monitor the system and collect the abnormal signals.

b. The abnormal signals would be pre-processed (filter to lower noise and magnify signal) and transformed to a processor unit, called Diagnostic, Prognostic Processing Unit,

c. Before the diagnosis, the collected signal is translated into the specific sentence so as to be identified by the case base. The basic sentences include three types: "A IS B", "A IN B" and "A AT B", which mean A is_type B, A when B, A in_state B respectively [18] .

d. A similar case would be searched for in the case library, in order to match the symptom as closely as possible.

e. Comparing the selected case with symptom, if they match exactly, the case library would suggest a feasible solution, corresponding to the symptom.

f. Else if they are not similar in detail or the feedback from maintenance addresses that the suggested actions fails to solve, another case would be searched for in the library.

g. If there is no case available, a warning will be issued and maintainers must seek expert assistance to make a new solution to solve the problem.

h. When the problem is solved successfully, what has done would be saved as a new case into the library.

4.6 Summary and discussion

Case-based reasoning system uses a case library and adopts an approach of search then answer. The management of case store and retrieve needs a particular coding; besides, to match the input problem closely, a case would be adapted in general. The dynamic case-based reasoner is a variation of CBR, and it uses a statistic to answer the similar problems, which saves the data space and time in searching.

The diagnostic process of EPS has been discussed in the above paragraph, it can be inferred that the operators will maintain the systems frequently in the early stage, because of fewer cases in the library. With growing completeness of the library, less of maintainers' actions would be needed. On the other hand, it is important that feedback from maintenance. The feedback could evaluate the case, increasing/ decreasing the success rate of case.

5 PROGNOSTIC RESEARH FOR EPS

In this chapter, an appropriate prognostic method would be found for EPS. Firstly, the writer discusses the characteristics of three prognostic approaches mentioned in chapter 2. Secondly, the failure symptom of EPS will be analysed. Finally, a feasible prognostic approach would be suggested for EPS.

5.1 Features of Prognostic Methods

Probability-based approach is a basic prognostic method based on a great deal number of historical statistical data, which is constituted by fault points. From the distribution of the fault points, it can infer the stage of high frequency fault in the whole life-cycle of a production. This technique could be applied into many fields for the low cost; however, its accuracy of prognostic is low also, compared to other two solutions.

Data-driven prediction techniques derive from probability-based techniques, and are used in the place where the prediction is hard to make although the fault points exist. Data-driven techniques employ a nonlinear network, for instance, Artificial Neural Network (ANN), to find the subtle relation among the providing data.

Model-based approach builds up models to simulate the working status of a component or system, so that it could predict the RUL of an object based on the simulation results. The prediction accuracy is high relatively, but its cost is high as well. Besides, it is difficult to grasp the functional rule of a specific object; meanwhile, building and updating a model according to the feedback of prediction is troublesome and low efficiency.

5.2 Failure Characteristic of EPS

Electrical Power system is mainly divided into three parts, the power generation subsystem, the power distribution subsystem and the loads. As digital and integrated manufacture techniques develop, the components of EPS tend to be modular and highly integrated. Increasing numbers of EPS products turn to the type of electronic and electro-mechanical.

When the writer uses MTBF to estimate the failure rate of a component, it can be seen that the failure rate is a small constant value. However, in fact, the statistic shows the failure rate of electric/electronic components follow a bathtub curve [14] .The Figure 5-1 shows a bathtub curve, it can be seen that in the early stage and end of life of products, the failure rate is high, while it is low when the component works steadily.

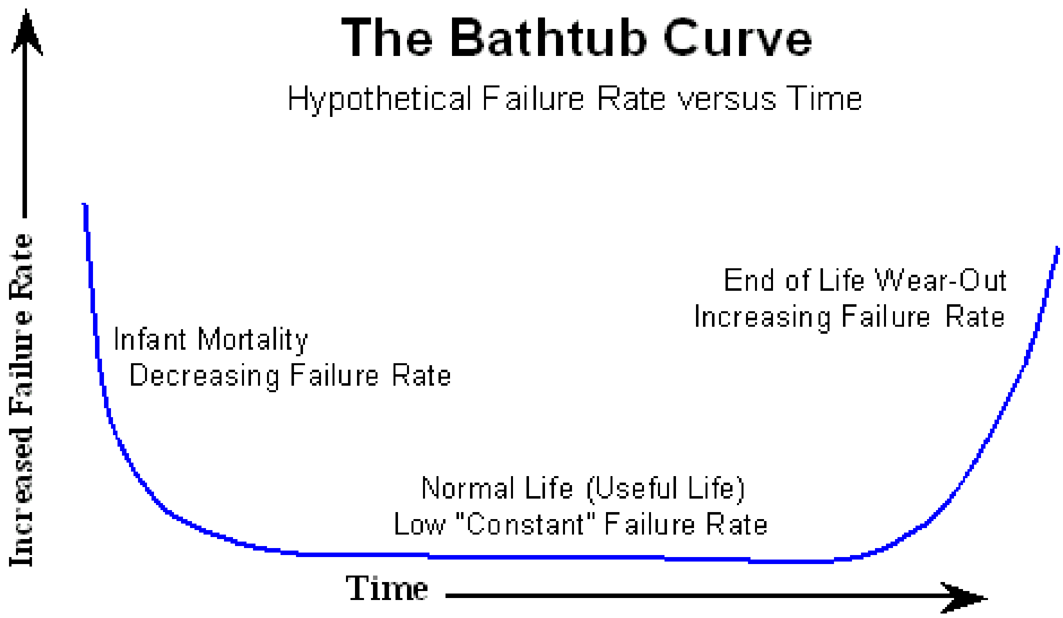


Figure 5-1 The Bathtub Curve [14]

One of the prognostic tasks is to predict the remaining useful life and find the working point that the components work correctly as long as possible before they wear out. After analysis to the EPS, it can be inferred that the useful life of components are related with temperature and humidity, vibrations, shock and power quality. Besides, the extreme operating conditions can exaggerate the probability of system failure.

5.3 Prognostic approach for EPS

The above chapters analyse the features of three prognostic approaches, and failure symptom of EPS. It can conclude that none of the methods is suitable for the entire electrical power system from the point of cost and accuracy. Therefore, the feasible solution would be given at components level, corresponding to the single equipment. Because the time is too short, the writer can hardly give a design flow of prognostic to EPS, but provides some suggestions to predict only.

Item	**Type**	**Detection method**	**Suggested methods**
Variable Frequency Generators, APU Generator, Ram Air Turbine generator	Electro-mechanical	voltmeter, ammeter, CMS	Hybrid methods of model-based and data-driven
Generator Control Units, Bus Power Control Units, Static Inverter, Low Transfer Rectifier Units, High Transfer Rectifier Units , Frequency Modulators ,DC/DC converter, Electrical Loads Management Centres,	Power Electronics	current sensor, voltage senor, thermal sensor,	ANN (one of the data-driven prediction methods)
Solid State Power Controllers, wires and cables , Generator Circuit Breakers, Bus Tie Breakers ,Circuit Breakers	Power Electronics	CMS	model simulation (Model-based)
Batteries	Electrochemistry	current sensor, voltage senor, thermal sensor,	Probability-based

Table 5-1 Prognostic methods to EPS

5.4 Summary and discussion

In this chapter, different prognostic approaches were suggested to the components of EPS, after a qualitative analysis of failure symptoms of the electrical/electric systems.

a. For the electromechanical product, such as generator, a model could be built based on its external characteristic, and adjusted according to the historical or experimental data.

b. As for the power-electronic product, like GCU, it is difficult to model because of complicated internal circuit and limited output parameters. So the artificial neural network is a reasonable method to them. Although the SSPCs and GCBs belongs to power-electronic also, their internal circuits are simple and easily to model.

c. Batteries are electrochemistry, and their remaining useful life could be predicted according to the historical data.

The notion of prognostic techniques has been proposed and then developed for many years, however, the assessment of uncertainty of the remaining useful lifetime or time is still difficult for researchers. A hybrid method using model-based and data-driven approaches to predict seems most accurate at present, but the cost is considerable either. Therefore, a trade-off between the cost and accuracy is considered, which needs the new technology or algorithms to be developed continuously.

6 CASE STUDY

This chapter gives a case study based on the results of diagnostics and prognostics approaches in the last chapters. Firstly, the power converter is chose from EPS, Secondly, the failure modes of power converter are analysed in detail. Thirdly, DCBR techniques apply into the converter. Finally, a conclusion will be given.

6.1 Introduction of Power Converter

DC/DC power converter is used to provide emergency power to the vital loads in Flying Crane. The converter employs the boost circuit theory. Its voltage in input terminal is 28V DC supplied by two batteries, and output voltage is 270V DC to the loads. A simplified internal circuit of DC/DC converter is shown in Figure 6-1.

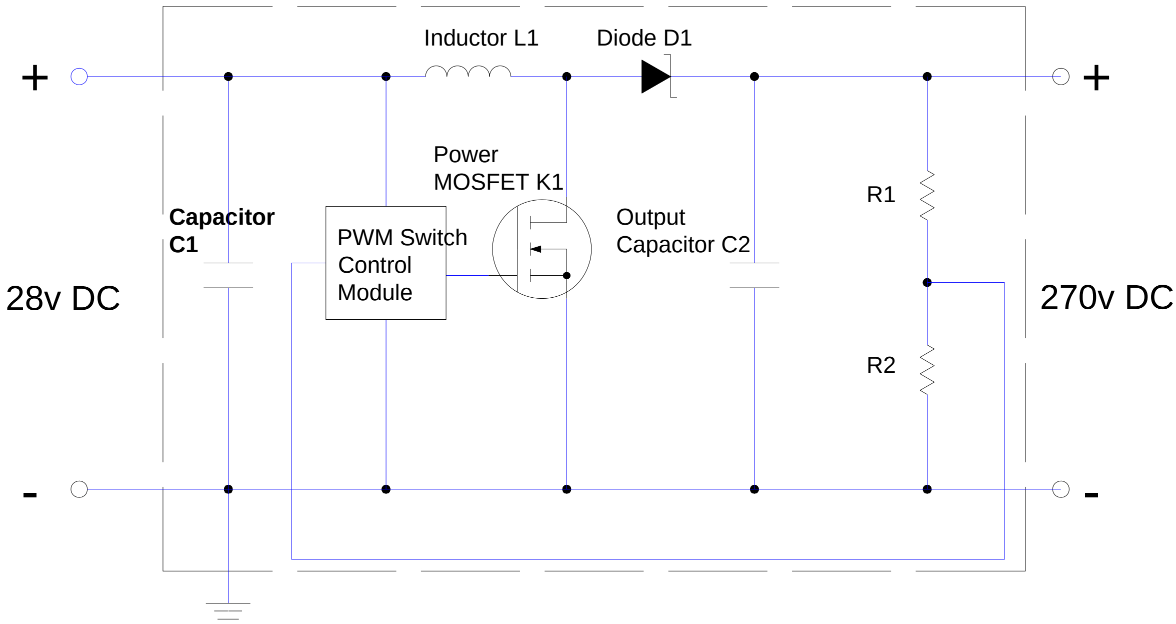


Figure 6-1 Internal circuit of DC/DC boost converter

The main components among the converter are inductor L1, Diode D1, Capacitor C2, Power MOSFET K1 and PWM switch control module. The output wave form is mainly decided by the rated value of L1, capacity of C2, and switch frequency of K1.

The DC/DC power converter is chosen for study, mainly based on the following reason:

From the safety analysis and fault tree analysis, it is found that DC/DC converter plays an important role in power supply to vital loads. If both of main generators and APU generator were out of work, the battery and RAT would switch to the electric network automatically. In this condition, the flight-critical loads, powered by 270V, would be supplied through the DC/DC converter. In case of the converter failure, a catastrophic scene would occur. Hence, diagnostics and prognostics to the converter are necessary.

6.2 Failure modes of Power Converter

To make a detail diagnosis to converter, a failure modes effect analysis was performed to the internal circuit.

Item	**No.**	**Failure**	**high level effect**
Capacitor C1	1a	short circuit	the fuse between the battery and converter would blow
	1b	open circuit	ripple waves increase and outputs degrade
Inductor L1	2a	short circuit	the fuse between the battery and converter would blow
	2b	open circuit	output turn into zero
POWER MOSFET K1	3a	short circuit	output turn into zero
	3b	open circuit	output decrease from 270V to zero
PWM Switch Control Unit	4a	failure	output decrease from 270V to zero
	4b	malfunction	output voltage degrade
Diode D1	5a	short circuit	power return to battery
Item	No.	Failure	high level effect
Diode D1	5b	open circuit	no output
Capacitor C2	6a	short circuit	the fuse between the battery and converter would blow
	6b	open circuit	output equal to input, that is 28v
R1/R2	7a	short circuit	ripple waves increase in output side, the power quality degrade
	7b	open circuit	ripple waves increase in output side, the power quality degrade

Table 6-1 FMEA of DC/DC converter

Comments: Table 6-1 shows the failure modes of internal circuit of converter, and consider briefly the external equipment which links to the converter, too. Actually, the real circuit of converter is more complicated than this.

6.3 Diagnostic Process for Power Converter

According to analysis of the converter circuit, probes are placed in a suitable position to obtain the operating data respectively. The writer adopts DCBR methods to diagnose the converter. The following diagnostic processes base on the hypothesis that all of the probes work correctly and the abnormal signals reflect the actual fault status exactly.

a. Fault signal is detected by sensors (measure current and voltage)
b. The signal is processed by filtering harmonic wave
c. Translate the signal into sentence in particular style that could be identified by processor, called Diagnostic, Prognostic Processor Module (DPPM)

For instance, Inductor L1 is open circuit and the current through L1 is zero, the sensor transfers the signal as “A IS B” style, which is (L1_CURRENT)-IS-(ZERO)

d. Perform similarity assessment by matching the new case with the old ones.

Input	Output	Sim_value
(L1_CURRENT)-IS-(ZERO)	(L1_CURRENT)-IS-(ZERO)	1*0.5+1*0.2+1*0.3=1

“Sim_value=1” means that the fault closely matches the case stored in the library.

e. Confirm the failure case, and suggest corresponding solution by DPPM.

f. Take actions and then receive the feed back from maintainers.

6.4 Prognostic approach

DC/DC power converter belongs to power-electronic equipment, the incipient failure includes: Gate Oxide, hot, Electromagnetic Interference etc. Artificial Neural Network based on data-driven method could be used to predict the Remaining Useful Life of this kind of product.

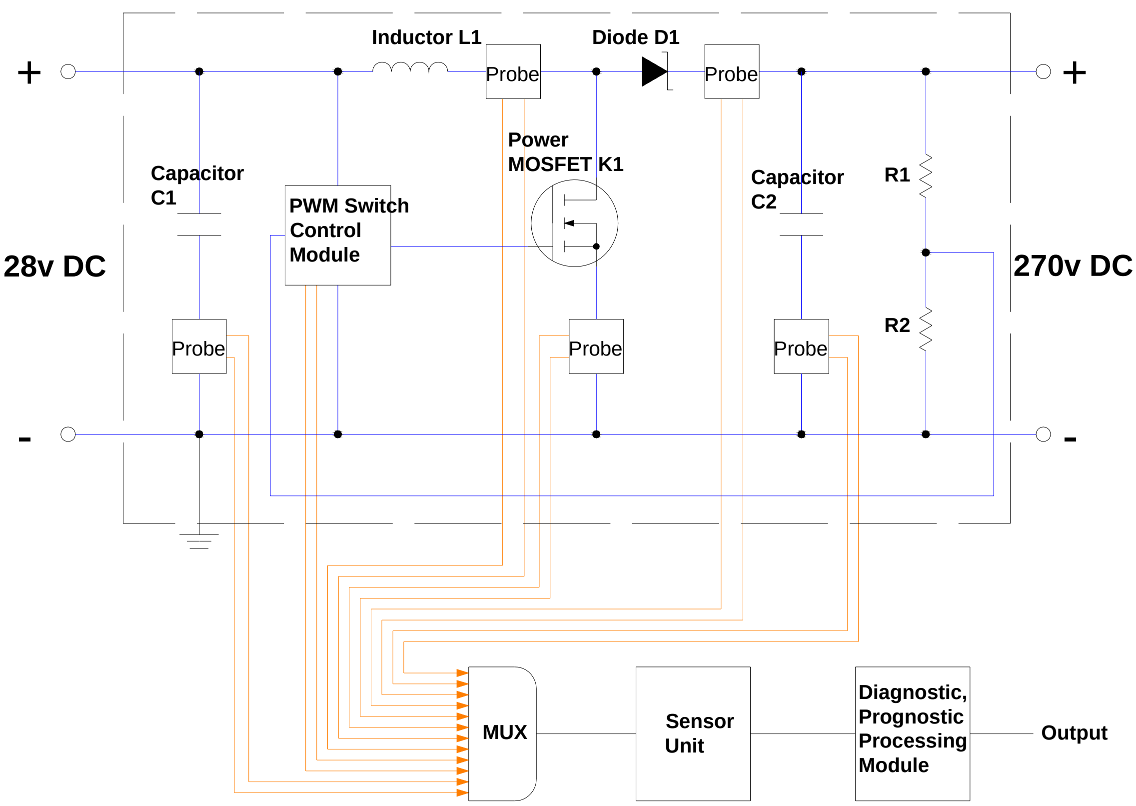


Figure 6-2 DPHM architecture of DC/DC converter

Combined diagnostic and prognostic approach, DPHM architecture of DC/DC converter is given in Figure 6-2.

6.5 Summary and discussion

In this chapter, a detailed DPHM architecture was built up for power converter, as a result, it could qualitatively diagnose the fault or failure successfully according to the case-based approach. Besides, from the sensors' data, the ANN methods could found the rule in converters so as to predict the RUL and detect the incipient fault in time.

In Figure 6-2, six pairs of probes are placed into the converter, and twelve signals are transformed to the processing module. It seems too many probes in a converter, which would degrade the reliability of monitors. With the statistical data updating, the components with high failure rate could be found, thereby, the probes setting could only focus on these components, so as to reduce the quantities of probes or sensors. Usage of smart sensors is another solution to improve the reliability of health management. The smart sensors have ability of auto-calibration and self-configuration, and they synthesise many new techniques such as wireless, opt-fibre etc. to increase the data validity [25] .

7 DPHM ARCHITECTURE OF EPS

In this chapter, an entire architecture of diagnostics, prognostics and health management (DPHM) of EPS will be introduced. Then the working process of DPHM will be discussed as well. Finally, the writer discusses the relationship between DPHM of EPS and health management at aircraft level.

7.1 DPHM Architecture of EPS

The DPHM architecture of EPS is mainly based on the results of diagnosis and prognosis in Chapter4 and Chapter 5. Generally, the DPHM system is divided into two parts, on-board and off-board separately.

7.1.1 On-board system

The on-board system comprises series of sensors, probes, signal processing unit, Diagnosis, Prognosis Processing Module (DPPM), communication interface etc . Actually, some sensors or probes have been integrated into the design and manufacture of components, hence, these components provide monitoring interface to DPPM. Besides, some other components without placing sensors can provide status information to DPPM through the communication data bus.

The on-board system could implement the following works:

a. Monitor the EPS operating status
b. Obtain the anomaly signals and process them
c. Fault diagnosis and suggest solutions
d. Failure record and display
e. Information communication with aircraft level health management

7.1.2 Off-board system

The off-board system contains a ground support centre, which provides support and decision-making.

The off-board system can implement the following tasks:

a. Information communication with aircraft level system
b. Technical assistance for crucial problems
c. Less critical problems solved post flight
d. maintenance record management

Figure 7-1 shows the DPHM of EPS schematic

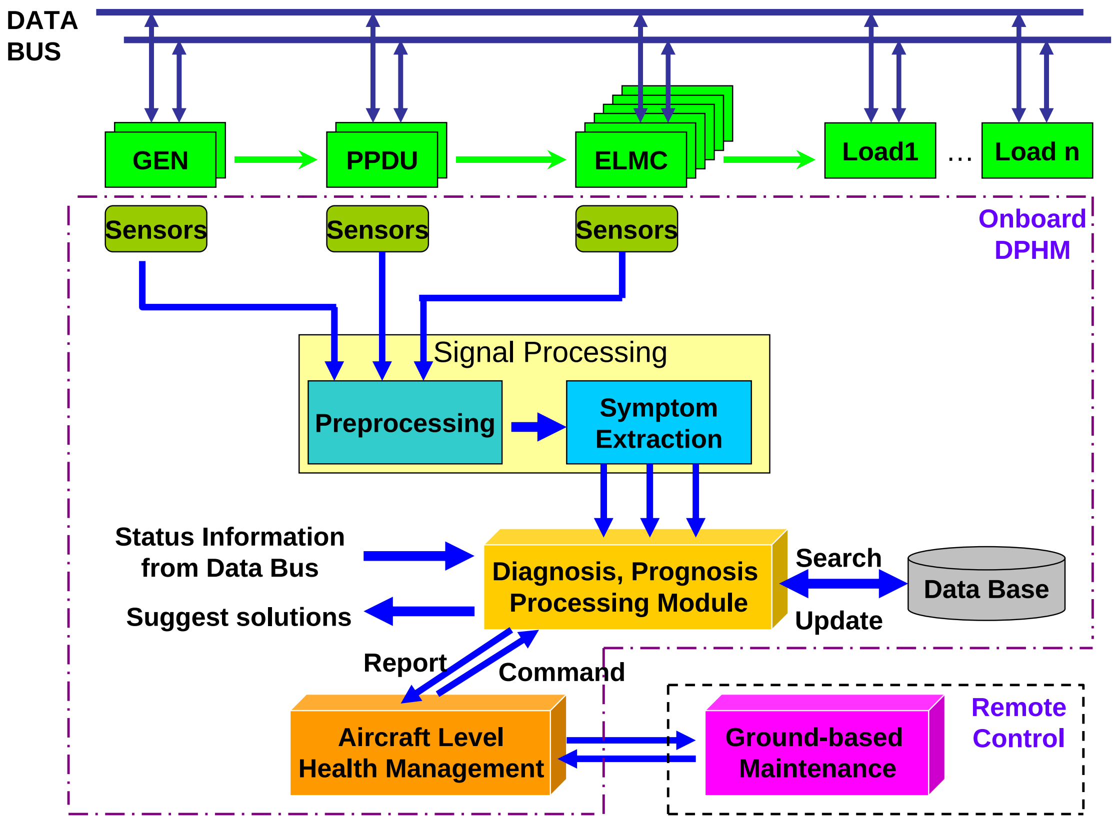


Figure 7-1 Schematic of DPHM of electrical system

7.2 Operating Process

The operating flows of DPHM are as follows:

a. When the EPS works, the sensors monitor the major components working status and collect abnormal signals;
b. The signals would be pre-processed, that is, filtered to reduce noisy;
c. After that, the feature of signal will be extracted and identified;

d. The DPPM judges the type of signals through algorithms, to decide if they are components fault/failure or warning of incipient failure;
e. For the component fault/failure, the DPPM will diagnose the reason, then give a suggestion, diagnosis process needs the assistance of ground base if necessary; on the other hand, for the warning of incipient failure, the DPPM will record the information;
f. All of the actions will be sent to aircraft level health management, which could be checked and traced by maintainers.

7.3 Relationship with aircraft level health management

The above discussion is about the architecture of electrical system health management. Actually, airframe structure and other on-board systems like avionics, fuel system, flight control system and landing gear system etc also need health management. Accordingly, The DPHM of EPS is not an isolated one in the aircraft, and it constitutes a complete aircraft level health management together with other systems. We call this aircraft level health management system Integrated Vehicle Health Management (IVHM) or Integrated System Health Management (ISHM).

The IVHM considers the aircraft as a whole; it combines all the system functions rather than be employed separately on individual subsystems, components and elements. [31] The electrical DPHM will communicate with IVHM by data bus, that is to say, receiving command, and sending results or suggestion.

The end-users press for reducing maintenance costs and increasing reliability and safety of aircraft, all of which accelerate the integration and development of IVHM. The function of DPHM of EPS, as a result, will probably embed into IVHM as a LRM.

8 EVALUATION

In this chapter, a qualitative assessment is performed to evaluate the diagnostic, prognostic and health management methods of EPS, from the aspects of system accuracy, cost and mass.

8.1 System accuracy

To diagnose the fault/ failure of a system correctly and prognose the impending fault/failure of a system accurately is one of the main goals of the DPHM system. Accordingly, a qualitative analysis of system accuracy is employed from the aspects of signal obtained, diagnostic engine and prognostic algorithms.

8.1.1 Validity of signals

It is considered firstly that the obtained signal could reflect the condition of the equipment being monitored as truly as possible. As the signals, in general, are captured by sensors and extracted by signal processor, forasmuch it is critical to ensure the sensitivity of sensors and good condition of processors.

The sensors should keep high reliability to capture and transform monitoring data even in the extreme conditions, for instance, the environment of high temperature, low temperature, vibration and electromagnetic interference etc. These need the sensors to have the capability of self-health assessment. [25] The smart sensors network, therefore, might be applied in the DPHM of EPS, which could ensure the failure rate of sensors much lower than the failure rate of objects being monitored.

The signals will be processed to extract the fault feature after they are acquired by sensors. In this thesis, the signal processing technique is not mentioned. However, the writer would like to say that the signals could be filtered and amplified, to remove the harmonic wave and lower noise. After that, the signals are translated into specific patterns, which the DPPM could identify.

8.1.2 Accuracy of diagnostic engine

The Dynamic Case-based Reasoning method is selected to diagnose the faults of the electrical power system. It matches the new problem to the old case in the case base, which is easy to find the fault reason and take actions for the system maintainers. For the common failure/fault problems, this method is correct and efficient. However, the condition becomes complex if a new problem has no similarity with all of the cases in the database. Besides, in the early time of system operating, there are a few cases in the base. The diagnoses need manual work frequently, which is less efficient. Since the case base grows permanently, less and less manual work is needed. Consequently, from the standpoint of long time works, the DCBR is able to diagnose problems of EPS accurately and correctly.

8.1.3 Precision of prognosis algorithm

For the prognosis of EPS, synthetic approaches are recommended for the different subsystems based on the components features. It is difficult to detect the impending failures and predict the remaining useful life of electronic products precisely till now. Although the writer suggests the prognosis approach corresponding to the features of components, it still could not be believed completely. When the prognosis actions appear, the maintainers or operators should analyse the system status with other tools or methods simultaneously, then make a proper decision.

8.2 Cost Consideration

The cost consideration is an important issue in evaluation of DPHM of EPS. There are many companies and research institutes studying the health management systems these days, but few of the techniques are really applied into the aircraft and most of the projects are still in the test stages. So it is difficult to estimate the expense in the DPHM of EPS. However, the cost of tests and experiments would be considerable before they come into use in the aircraft.

The cost lies in two parts: hardware and software design, the hardware includes sensors, digital processing modules and DPPMs. With the smart sensors appearance, the design of electrical components would consider embedding sensors techniques, which lower the cost of sensors. Meanwhile, the highly integrated aircraft level health management, such as IVHM system, manage health management of the whole aircraft. In this condition, the DPHM module of EPS would be a chip of IVHM system that means the cost will decrease.

8.3 Mass of DPHM system

As mentioned in above paragraph, the sensors could be integrated into the design of components, whilst DPHM module could assemble in the slots of IVHM. The signals could transfer by data bus. Consequently, the actual mass of DPHM system of EPS is lower than the weight be imagined.

9 CONCLUSION

In this thesis, a set of generic diagnostic, prognostic and health management method was developed for the electrical power system in the aircraft. To design this DPHM architecture, a series of tasks were performed as follows:

First of all, the EPS of Flying Crane was chosen as a research object. The writer employed analysis methods such as FHA, FMECA and FTA to distinguish all the failure conditions. From the FHA results, it showed that four functions loss, such as loss of 28V DC channels controllability operating independently, could lead to catastrophic event. Consequently, the components related to these functions were considered particularly. FMECA was conducted after the FHA analysis, most of the components in the EPS were analysed, except the wires, connectors etc and each failure mode of them was ranked according to severity level. Then the writer performed FTA to identify the root causes, which ensured no other failures were left. The results of FTA also showed that no single failure in EPS could lead to a catastrophic event. Based on the results from three analyses, the components monitored were selected, and their failure could be hazardous or catastrophic. Also, the related parameters and detection method were acquired.

Secondly, the case-based reasoning approach was chosen to diagnose EPS failures after comparing four diagnostic methods in detail. The coding and adaptation needed to be considered when a CBR is used. The writer adopted DCBR, a variation of CBR, to the EPS, which retrieve case from data base quickly. The operating process could locate the failure reason correctly and quickly. As the case includes right solutions to the problem, the maintainers can take actions effectively to make the system resume working in a short time.

Thirdly, the writer divided the components of EPS into three types: electromechanical, power-electronic and electrochemistry. The appropriate prognostic approaches were suggested to these components according to their different characteristics. Besides, the selection of prognostic approach was referred to compromises of accuracy and cost.

To demonstrate the diagnostic process, DC/DC power converter was analysed as a case study by using DCBR. It can diagnose the faults in the system reasonably.

Finally, a synthetic architecture of diagnosis, prognosis and health management for EPS was built. The architecture showed a clear relationship between the monitored components, diagnosis module, aircraft level health management and ground-based maintenance.

Above all, an integrated DPHM system was developed for the EPS based on the previous research. It could monitor the operating state of systems in real time, detect and process the abnormal signals, diagnose the fault/ failure reason correctly and quickly, then provide solutions to maintainers, give a warning in time when impending failures occur, and predict the remaining useful life of products. That the failures are detected and solved in a short time could reduce the delay rate of EPS significantly. The DPHM provide a method of maintenance on condition, and prediction before system failure, which reduce the expense of the periodical maintenance and unscheduled maintenance. The design achieved the objective in chapter 1 fully.

The writer did not give detailed prognostic flow to the EPS due to limited time. This thesis did not include the diagnosis and prognosis of wire and connectors because of shortage of data in the preliminary design. The health management of loads was not considered either, for the loads are various and complicated.

There are several problems which should be noticed:

1. How to distinguish the validity of signals in case of sensors’ malfunction or loss of work?

2. In the early time of system operating, there are a few cases in the base. It is probable that the failure occurred could not match the cases in the data base. So, the diagnoses need manual work frequently, which is less efficient.

3. Although a series of prognostic approaches has been suggested to EPS, to predict the remaining useful life of electric/ electronic components precisely is still difficult. Furthermore, a trade-off between the cost and accuracy is considered.

A series of work are recommended to extend in the future:

a. The DPHM of electrical power system should be researched further, and focus on every major component or subsystem, the wires, cables and connectors failure are concerned as well, in order to make the diagnostic work as correctly as possible.

b. The prognostic approaches should be deployed fully. A combination method of modelling simulation and statistical data support should be tested on the EPS.

c. The health management of EPS is not isolated in the aircraft, therefore, the interface with IVHM should be considered.

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APPENDIX A: GDP Report

A1 Fuel System Introduction

A1.1 Introduction to Flying Crane

The Flying Crane is a project developed by the Aviation Industry of China students during the academic year 2008-2009. Flying Crane is aimed primarily at China's domestic air traffic market.

Designed as a next generation airliner to replace the current Boeing 737 and Airbus 320 family, Flying Crane is a medium to short haul, single-aisle, 128-seat (two-class) jet liner with conventional configuration. [6]The design range of Flying Crane is 2,000 nautical miles. Flying Crane has a MTOW of 64,582 kg. The design cruise speed of Flying Crane is M 0.78 at the altitude of 39,000 ft, which equals to 230.15 m/s. In the preliminary phase, the Flying Crane is designed as an "All-Electric Aircraft", whose standard is much higher than the conceptual one. Hence, all the systems of Flying Crane must accommodate with the characteristic of AEA.

In the conceptual phase of Flying Crane, The design fuel capacity is 14,978kg. The fuel tank is made up to wing tanks and central tank. Among them, the central tank is optional, which is different from the conceptual one, for the reason that the data of the wing box is not accurate in that phase, and a mistake of calculating the capacity of the fuel tank had been executed.

In the preliminary phase, the architecture of fuel system was designed according to the requirement from airworthiness management etc. The configuration of fuel tank was given in both 2D and 3D environment, especially the dry bay design which was a highlight in the GDP.

A1.2 Description of the Task

The writer was charge of two systems, Actuation system and Fuel system, which were totally different from his own major, electrical system. Therefore, the writer could not make the fuel system design 100% correct.

A1.3 Designing the baseline fuel system

From the general task, the writer decided the task of fuel system design as follows.

- Estimate power requirements of fuel system
- System safety assessment of fuel system
- Design of fuel tank; consider the dry bay design in case of rotor burst
- System architecture of fuel system
- Mass and size
- System layout within flight vehicle airframe (CAD)
- Airworthiness Compliance Matrix

A1.4 Design Flow

The fuel system design flow in preliminary phase included several major steps. The flow chart of fuel system is illustrated in Figure 1-1.

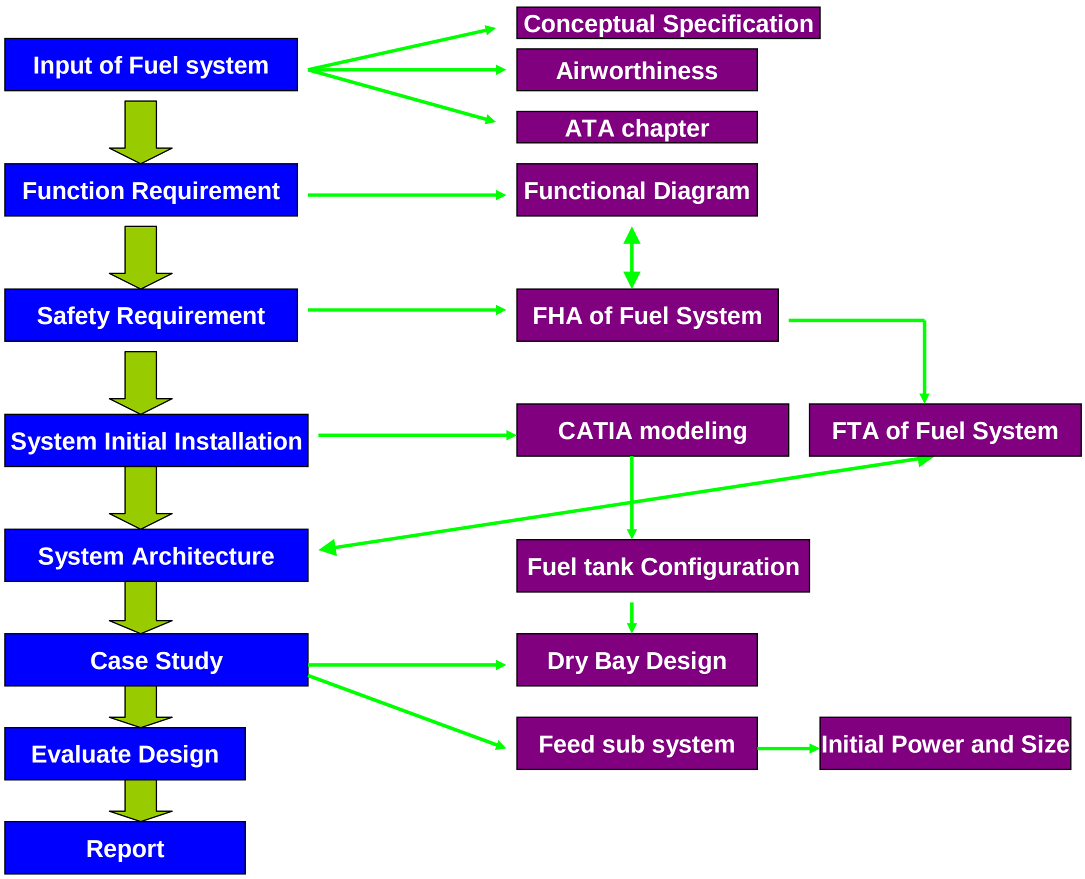


Figure 1-1 Design Flow of fuel system

A2 Requirement of Fuel System

A2.1 Airworthiness Requirement

There are three series of regulations available to the project, the CCAR-25, the FAR-25 and CS-25. As Flying Crane mainly faces to the domestic market in medium to short haul in China, the team made a decision that the airplane must comply with CCAR-25 in the all design phase. The fuel system must comply with CCAR-25 and other related safety regulation as well as other airframe systems.

A2.2 Customer Requirement

The designer needed to take in to account the potential requirement of Airliner companies that the needs of fuel capacity of the Flying Crane's family for a prolong range. Therefore, it is necessary to plan an optional tank for the customer to choose.

A2.3 Functional Requirement

When the writer started to design a system to achieve the needed function, the first thing to do is to make a thorough study of the system so that no component or critical function is left out. [37]It is necessary to have an exhaustive approach to conceive the fuel system; although it can be seen as a compilation of different entities, some of them could in fact be complementary and the analysis that has to precede the beginning of the design should enable optimizing the system, thus saving weight and possibly development, production and operating costs.

The fuel system analysis in the first stages of the project leads to the identification of the following functions:

A primary function: To provide a reliable supply of fuel to engines. These conditions must be met even under the most extreme circumstances defined by the flight envelope.

Another function is to provide the auxiliary power unit with fuel whenever it is required.

A2.3.1 Function of Fuel system deployment

The first functions described in the previous paragraph are major functions; they suppose peripheral functions:

A. storage function: fuel is stored in the aircraft using the fuel tanks during the flight mission. B. refuel/defuel function: the fuel will have to be pumped directly into the tanks of aircraft during the grounded refueling operations. A defuel system will be needed to empty the fuel tank. C. collector function: during the flight, fuel has to be transferred into collector tanks. D. transfer function: given the large quantities of fuel that are needed for the long range civil transport and the tanker version, the progressive consumption of fuel will have an influence on the position of the CG of the aircraft. E. measurement function: In the flight mission, the crew needs to know the amount of fuel as precisely as possible when it is available. f. Venting/Inerting function: throughout the flights, the fuel tanks are being filled and emptied. G. pressurizing function: using an on-board inert gas generation system can suppose two uses. H. fuel dumping function: the CCAR requirements impose performances on the aircraft, particularly in the case of engine failure; it must be required to maintain climbing capabilities as CCAR 25.119 and 25.121(d).

As the previous chapter said, the general fuel system function considers all kinds of cases so as to make no sub-functions left. However, it is unnecessary to choose the entire functional requirement for the project, after the analysis of the characteristics of Flying Crane and the conceptual design specification.[6] For instance, the landing gear system has extra 10% design redundancy which means no fuel jettison in flight during any emergent landing, which has been discussed with landing gear structure designer.

A2.3.2 Functional Hazardous Assessment of Fuel System

It is essential to precede Functional Hazardous Assessment (FHA) of systems in the design phase. In CCAR 25.1309 and the advisory circular, it is specified in detailed about the system failure conditions.

Following the detail guideline in SAE ARP 4754 and CCAR 25.1309, the writer performed the FHA of fuel systems based on the functions in the former chapter.

A2.3.3 Physical and Installation Requirement

In the preliminary design phase, the major concern of physical and installation requirement of fuel system is the fuel tank deployment and installation of the main components, for instance, the pump, the valve etc. So, it needs to cooperative with wing designer about the wing box where the fuel holds in the early stage.

A2.3.4 Maintenance and Reliability Requirement

Maintenance and reliability are critical issues for the customer, and they should be considered in the early stage. To achieve them, the designer needs to take into account the following items,

The simplicity design,

The redundancy design,

The operating environment design,

The human-engineering design,

The lighting protection design (CCAR 25.1316),

The test ability design,

The derating design and

The thermal design

For the fuel system, it should be satisfied this requirement.

A2.3.5 Interface requirement

Interface requirement is a considered issue when the writer designed the fuel system and thought about the relationship between the systems. In the early stage of the preliminary design phase, the system boundary and the Input/Output needed to be discussed among the whole group included system designers and structured designers. With the engineering experience in civil jet design, the whole group referred to the ATA 100 to make it clear the boundary between all the systems. The designer took into account the following situations.

What is the power supply for fuel system?

Which kind of system needs the fuel providing?

Which kind of data needs to be displayed?

After the writer considered the above questions, the relation chart was drawn as follows.

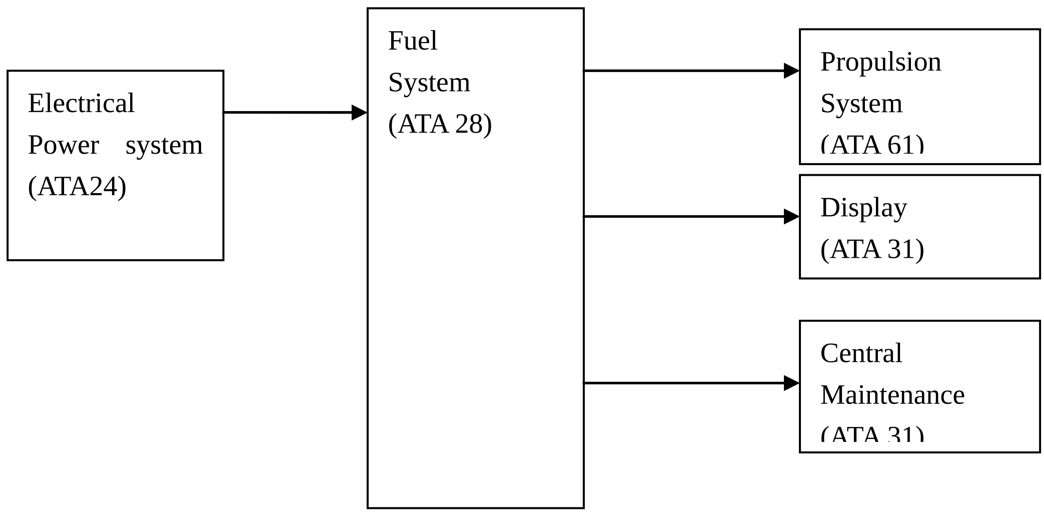


Figure 1-1 the simply relationship between fuel system and others

A3 Fuel Tank Design

The fuel tanks store the fuel which provides to the engines, and the fuel in the wing tank could reduce the loads on the wing. In this phase, the writer needs to design the fuel tank included the shape of tank (consider the rotor burst effects), the volume of the tank individual and totally and the location of the tank, as well as the CG position.

In the former specification of Flying Crane, the roughly configuration of fuel tank was provided as well as the design point of fuel capacity. The fuel tank was composed of five tanks, the right inboard tank, the right outboard tank, the left inboard tank, the left outboard tank and the central tank.

The design point of fuel capacity is 14,978kg, and the maximum mass of fuel is 17,560kg. Therefore, the writer should base on this provided data to design the fuel tank, except for the critical issue to mend it.

The fuel tank design should be compliant with CCAR 25.963 and AMC 20.128, which is the first consideration item on airworthiness regulation.

A3.1 The volume of tanks

The Flying Crane needs to be able to accommodate enough fuel to fulfil its mission safely. 14,978kg is needed according to the performance team.

Concerning the tanker version, the higher the mass of fuel that it can store, the better it would be in terms of off-load capabilities during operations.

Concerning the volume of the tanks themselves, it is relatively difficult to estimate precisely from the CAD models. However space must be left for the structure of the wing, as well as for the different components of the fuel system: pipes, pumps and sealant mainly. Sufficient expansion space must also be left since fuel can be submitted to different pressures and temperatures during the flight. As a result, it is about 8 % of the given CAD model volume that will have to be withdrawn to obtain the truly available volume of fuel. In order to have some flexibility in the CG position when the fuel load is to its maximum, it can be considered that a margin of 10% is required.

Some rapid calculations for the Flying Crane:

Initial required mass of fuel: 14,978 kg.

Equivalent volume in the lower value of the density at 800 kg/m3

$$\frac{14978}{800} = 18.72 m^3$$

Equation 3-1: Minimum volume required in the tanks

This volume should represent 90% of the CAD volume; therefore the CAD volume should be:

$$\frac{18.72}{0.9} = 20.8 m^3$$

Equation 3-2: Minimum volume required on the CAD model

Finding the volume was subject to the dimensions that were chosen for the wing box. Issues arose concerning the position, size and retraction mechanism of the main landing gear. This could have had an impact on the position of the rear spar thereby affecting the volume of the wing box.

After several discussions and cooperative work with wing designer and landing gear designer, the front and rear spars' positions settled to respectively 15% and 65% of the chord.

Moreover, due to the threat posed by the lightning strikes, fuel must not be located on the most outer part of the wings, hence, the outer boundaries located at 16.14 m from the centreline. With a wingspan of 33.48m, that would give a 1.2 m margin for the Flying crane in case of lightning strikes.

A3.1.1 A big issue about volume calculation in 3D environment

When the writer made the decision of the front and rear spars' positions, a problem rose. According to Mr Xue's GDP report of fuel tank configuration in last year, [39] the volume calculations are follows:

$$V_{inboard} = 4.48m^3$$

$$V_{outboard} = 3.84m^3$$

$$V_{central} = 6.77m^3$$

$$V_{total} = (V_{inboard} + V_{outboard}) \times 2 + V_{central} = 23.41m^3$$

Equation 3-3: The volume on the report in last year

While the volume in the new 3D model are follows:

$$V = (V_{inboard} + V_{outboard}) \times 2 = 23.16m^3$$

Equation 3-4: The volume calculation of fuel system

It means that only the volume of wing tank inboard and outboard in new model is satisfied with the design point 20.38m3. When the writer and wing designer checked again, they found that the wing shape used in last year is totally wrong. Therefore, the volume derived from the wing box in last year is incorrect as well. However, it is really good news for fuel system reliability and the safety of Flying Crane.

After the supervisor, Prof. J. Fielding, checked the design and communicate with designers, the group made a decision that the central tank is set as a optional tank for stretch type, whilst the base line fuel systems includes four tanks, the right inboard tank, the right outboard tank, the left inboard tank and the left outboard tank.

Consequently, the total volume of fuel tank is 23.16m3(without considering the rotor burst influence)

A3.2 The configuration of tanks

As the previous said, the total volume has been decided, and then the next step is to confirm the boundary of tank individually.

A3.2.1 The boundary of inboard tank and outer tank

The wing tank starts at No.1 rib and ends at No.19 rib in each side of the wing, and the volume is 23.16 m^3 in together, thereby the task is to decide the boundary between the inboard tank and outer tank.

The GDP report of last year about fuel tank layout gives a reason to decide the wing tank boundary; he said that to shorten the time of refuelling, it is better to make the capacity of two tanks close to insure the refuelling time close. From the reality consideration, the refuelling gap and receptacle on the right wing of the aircraft, and it is no relation with the similar time for refuelling. Hence, it is not a proof to support the boundary dividing.

It is difficult to find some proof or solution to determine the boundary of them, so the writer referred to some similar aircraft, for instance, A320 family, Global express etc.

After discussed with supervisor, the writer decided the boundary based on two proofs. First, for a better stress of structure considering all the components and loads in the wing, such as the engine pylon, the position of boundary sets to the inflexion of the wing. Second, as Michael Niu said. [33]

Considering the proofs above, the boundary sets at No.8 rib, that is to say, the inboard tank occupies from No.1 rib to No.8 rib, the outer tank occupies from No.8 rib to No.19 rib. The volume of inboard tank and outer tank is 8.212 m^3 and 3.368 m^3 in each side respectively.

A3.2.2 The boundary of surge tank

The surge tank occupies at least 2 percent of totally volume.

The surge tank starts at No.19 rib and ends at No.22rib in each side of the wing and the volume is 0.408 m^3 for each side. Therefore, the ratio= $\frac{0.408 \times 2}{23.16} = 3.52\%$, which is suitable for the requirement.

A3.2.3 The layout of wing tank

Following the boundary instruction in previous paragraph, the layout of wing tank illustrated as follows, (only shows the ring wing)

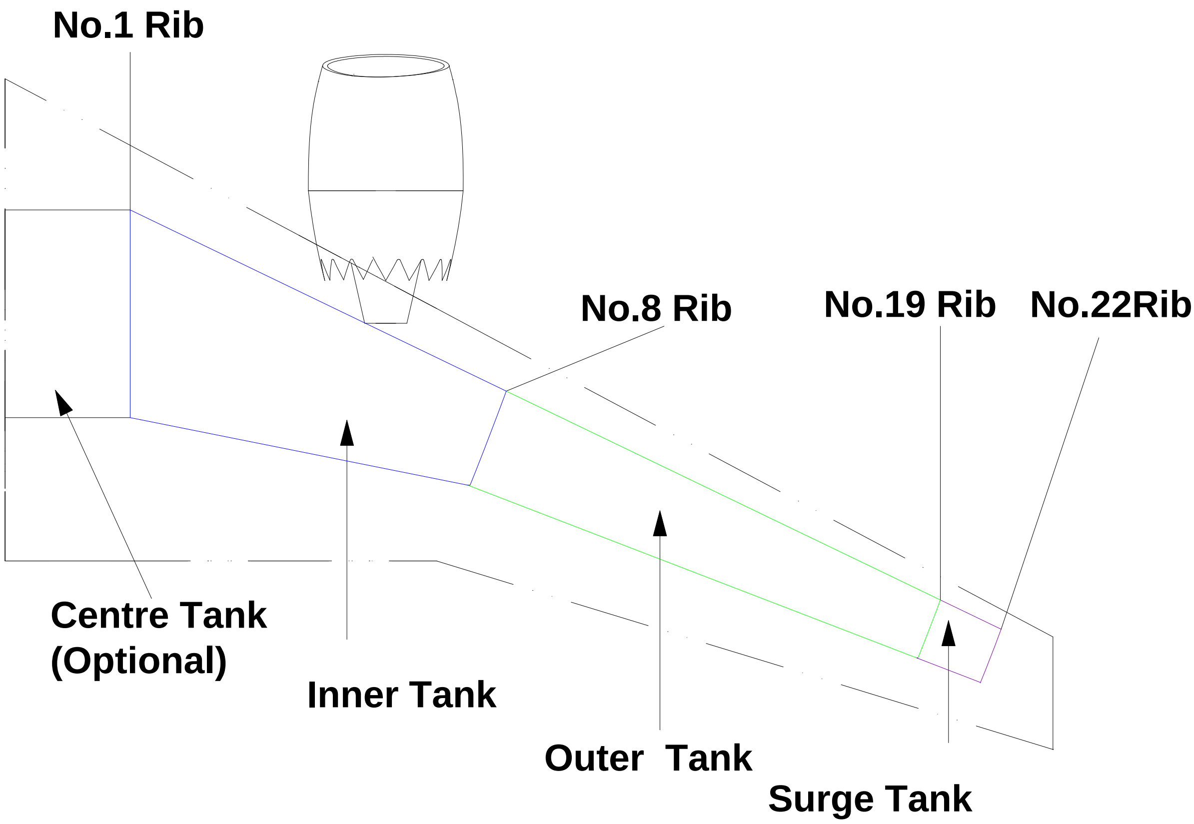


Figure 3-1 the fuel tank layout in right wing [34]

A3.3 Dry Bay Design

The dry bay design is a brand new task, which was distributed to safety designer and fuel tank designer by Prof. J. Fielding. The aim of the design is to minimize hazards caused by uncontained turbine engine and auxiliary power unit rotor failure.

Because there was neither reference nor thesis involved the process of dry bay design, the only proof is AMC 20-128A. It is really a challenging work to design the dry bays for Flying Crane. After a few weeks learning and discussing about the drawing, the designer found a solution finally.

With the wing designer's help, the drawing was managed to finish in CATIA sketch.

Some key point of consideration as follows,

The spread angle of rotor debris is 15 degrees for the composite material wing.

Calculating the first and the last rotor debris spreading trace decides the envelope of all the area of rotor burst influence.

Sweep angle from 15 degree to 0 degree to determine the envelope of penetrating track.

Based on the above consideration, the writer minimized the size of dry bay and it can be seen in Figure 3-4.[17]

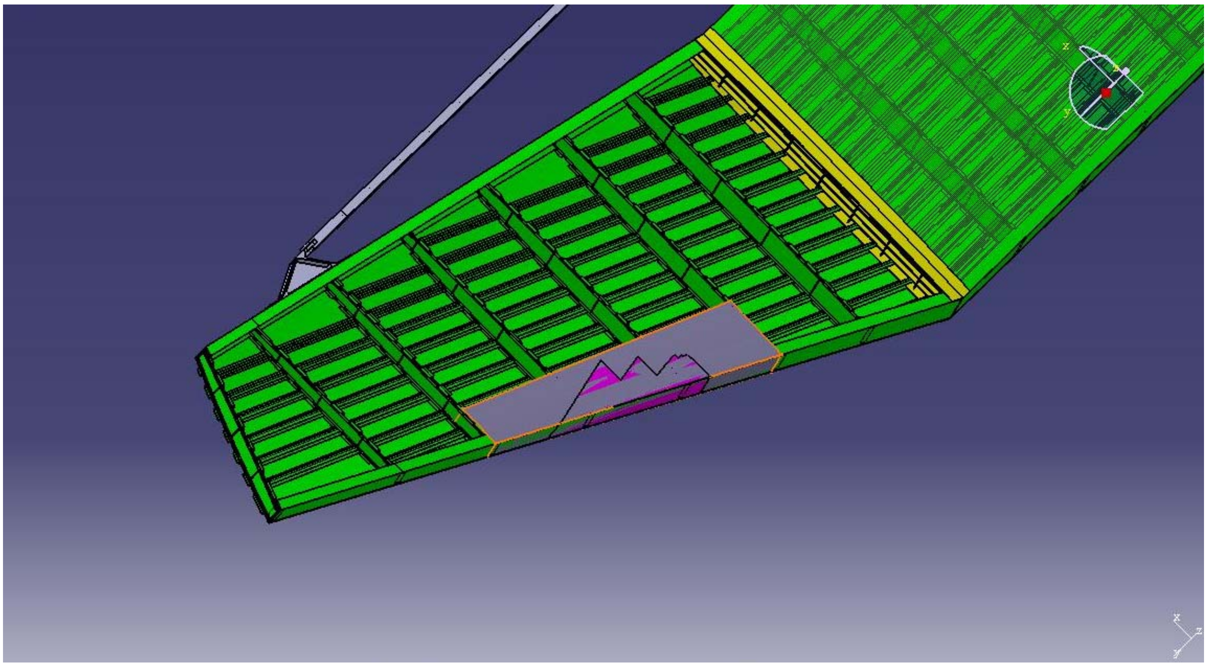


Figure 3-2 Dry bay area in the wing

A4 Architecture of fuel system

Architecture of fuel system is one of the major tasks in the preliminary design phase, and it is the basis of case study. From the design flow chart in chapter 1, it is known that the design process is iterative; therefore, the architecture of system needs to keep update with proceeding in the whole design process.

A4.1 Architecture in first round

When the writer began to design the architecture of fuel system, the first thing was to ascertain the start point of it, that is to say the input of architecture. From the chapter 2, the functions of fuel system was provided, accordingly, the sub-system and/or components in the architecture should realize these functions.

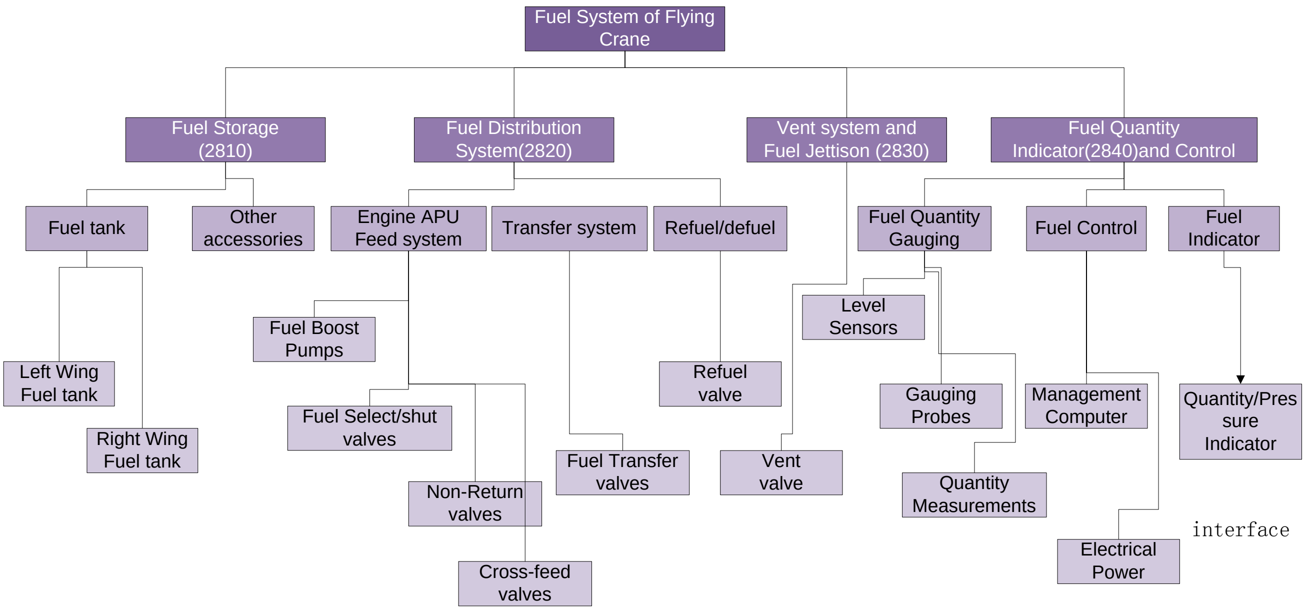


Figure 4-1 sub-system and major components of fuel system

A4.2 Fault tree analysis of fuel system

From the schematic of architecture in first round, the writer could do the fault tree analysis (FTA) of fuel system so as to calculate the redundancy of major components such as, boost pumps, transfer valves etc. One sample of the FTA report illustrated as follows.

A5 Case study-Engine feed sub-system

In the preliminary phase, the writer gave a case study of engine feed subsystem, providing the boost pump product specification; engine feed pipe installation in 3D environment.

A5.1 Power calculated of boost pump

The architecture of engine feed system was introduced in Chapter 4, and the major components of this system are boost pumps, APU pumps, transfer valves, engine feed shut-off valves, APU feed shut-off valves, cross feed shut-off valve, scavenge injector pumps and sensors. In the preliminary phase, it is really difficult to calculate the power requirement and mass of each component, besides, the writer never study the fuel system before the GDP work started. Therefore, the writer gave the power estimation of boost pump as a case study.

The methodology of power estimation is as follows. First, the mechanical power of pump equal to the power of proving fuel flow rate with pressure in the outlet of pump. Second, the writer supposed that the fuel flow rate equal to the fuel consumption rate. Besides, as the fuel feed pressure varied from 10-15 psi, [23]10 psi (68.95KPa) was selected in this study.

The fuel consumption rate derived from performance and flight quality report by Mr.Jia, and the rate in every flight phase could be seen in the table 5-1.

Phase	Time(s)	Quantity of Fuel consumption(Kg)	Consumption Rate(Kg/S)	Fluid (L/min)
Take-off + Clime	657	851	1.295	97.15
Cruise	12770	8875	0.695	52.12
Descent	1199	614	0.512	38.41

Table 5-1 Quantity of fuel consumption in the flight

In the worst-case scenario, one side of the boost pump is out of order in the take-off phase; the pump remaining provides fuel to two engines. So the peak power of the boost pump could be calculated, considering the efficiency of power transfer (η) is 0.5.

$$P_{mech\_pump} = Flowrate \times \Pr essure = 68.95 \times 97.15 / 6000 = 1.116Kw$$

$$P_{elec\_pump} = P_{mech\_pump} / \eta = 1.116 / 0.5 = 2.23\text{K}w$$

Equation 5-1 Calculate power of boost pump

According to the calculated result, the writer found the similar product on the shelf.

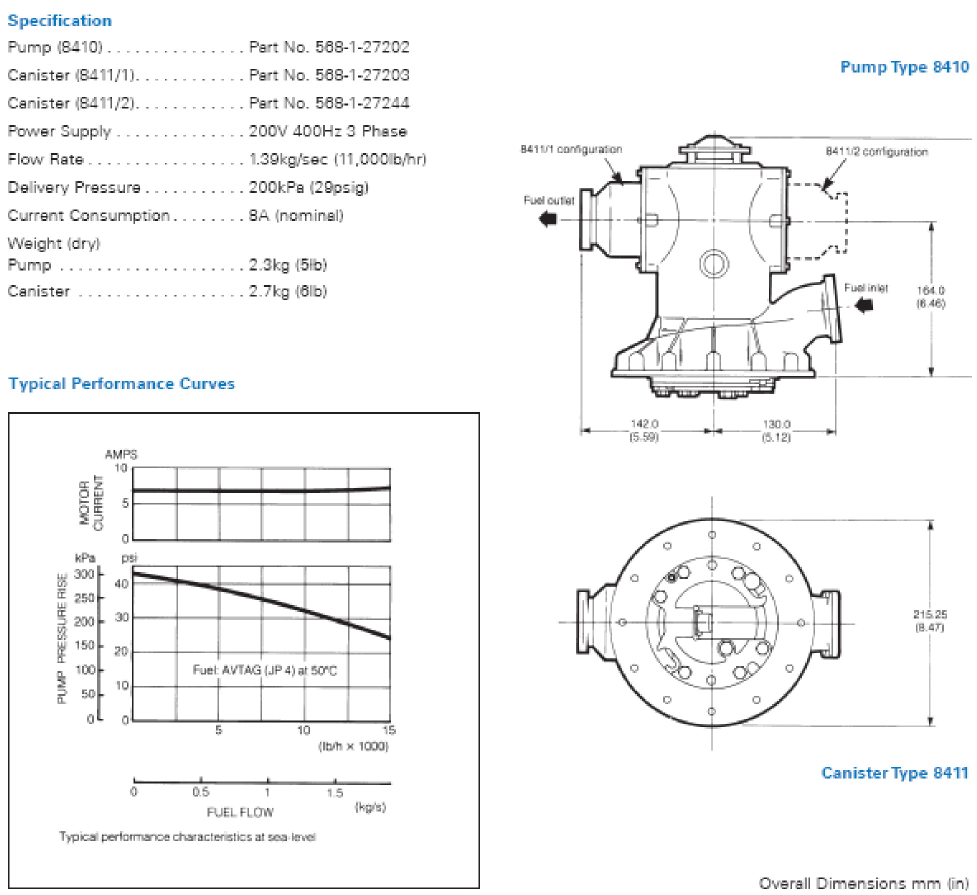


Figure 5-1 Technical characteristic of the booster pump, EATON Aerospace [21; 21]

This pump from EATON Aerospace is used in A320 family, and Flying Crane has a similar capacity with A320 family, which also validated the results of estimate is reasonable.

A5.2 Lay out in CAD

The writer made simply lay out of engine feed in CATIA environment, showing the installation of boost pumps, cross feed shutoff valve, transfer valve and pipes. When the major components were installed in the fuel tank model, the rotor burst influence must be considered.

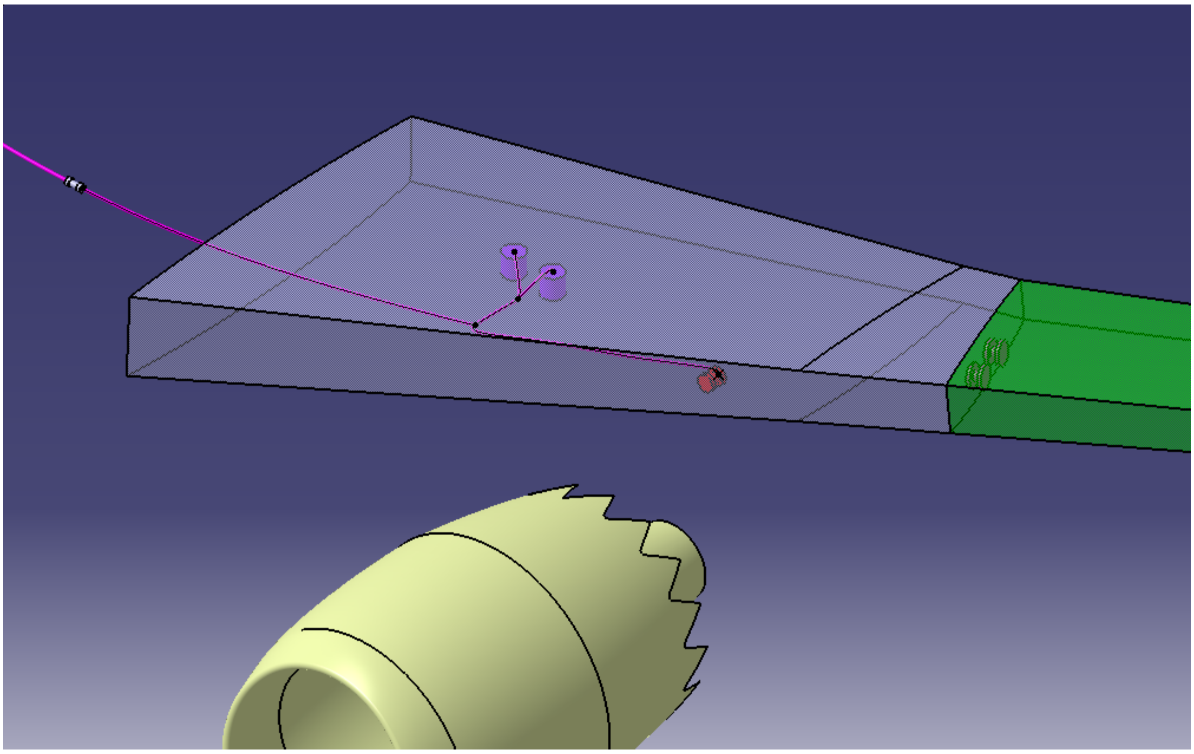


Figure 5-2 Engine feed lay out in CTAIA model

A6 General consideration of fuel system

In the preliminary phase, the mass, CG control is important consideration to the whole aircraft; meanwhile, the power consumption needs to provide to electrical power system for power capacity.

A6.1 Mass, CG estimation

The mass of fuel system is composed of fuel mass and mass of the sub-system and major components, pipe etc.

A6.1.1 Mass, CG of fuel

The design point of the fuel capacity is 14,978kg in specification book, and it needs to distribute the wing tanks to manage the CG position. Besides, the maximum mass of fuel should design to exceed mass of the design point.

For the aviation fuel, the density is between 775 and 840 kg per cubic metre. A value of 800 kg per cubic metre will be used for the calculations relative to the sizing of the fuel system. Therefore, the mass of fuel in each tank could be calculated. The table 5-1 shows the maximum capacity of baseline fuel tank.

	Inner Wing Tank	Outer Wing Tank	Total
Design Volume（m^3）	7.39×2	3.03×2	20.844
Mass（Kg）	5,912.6×2	2,424.9×2	16,675.2

Table 6-1 mass distribution of fuel

	Inner Wing Tank (right)	Outer Wing Tank (right)	Inner Wing Tank (left)	Outer Wing Tank (left)	Total
X（mm）	16314.718	18737.167	16314.718	18737.167	17019.353
Y（mm）	4159.277	10327.986	-4159.277	-10327.986	0
Z（mm）	-821.325	-170.227	-821.325	-170.227	-631.955

Table 6-1 CG position of each fuel tank

A6.1.2 Mass estimate of fuel components

The specification book in last year gave a budget of 987 kg for the fuel system components and pipes in baseline aircraft. The final mass clearly does not match with the budget. First, in a baseline fuel system, the central tank was optional, that is to say, there were few pumps and pipes installed in it. Accordingly, that proportion of components need not include in the total mass. Much of the weight comes from the sealing of the tanks; there are different existing technologies, and the mass of sealant can be expected to be between 50 and 80 pounds per 1000 gallons. [8] An optimistic value of 60 was chosen for the system.

The other main contributor to the weight is the OBIGGS. This system is not compulsory and possibly not taken into account in the mass budget of the specification book. Its mass is in accordance with other OBIGGS currently being studied and developed but the masses of smaller components are not accurate enough and can only be thought of as provisional allowances to develop a component specific to this OBIGGS.

A6.1.3 Fuel consumption curve and fuel management

The CG of fuel consumption is complicated because the fuel left depends on the aircraft’s angle of inclinations, and accelerations in three dimensions. Nevertheless, a first approximation can be calculated using CAD model. The results were obtained by splitting the tanks' volumes into successively smaller volumes with planes parallel to ground level. CATIA can then compute the centre of gravity of a homogeneous volume.

In addition to the attitude of the aircraft, the tanks can also deform with the wing: that was not taken into account in these calculations. The fact that the lift will bend the wing upwards will make fuel flow towards the centreline. On the ground, the mass of the fuel in the tanks will locally decrease the dihedral angle and move the fuel outwards.

A6.2 Power requirement of fuel system

In this phase, the power consumption of fuel system needs to provide to the electrical system designer, for which the power capacity of Flying Crane could be estimated.

The components needed power include the pumps, the valves, the sensors, and electronic control units etc. In this stage, the writer could not do some study deeply in it, so most of power estimation comes from the reference book and product specification from website. However, the power type could be decided depending on the provided data. For the fuel boost pumps and valves, 115vAC power is needed; while28 v DC power could activate the APU boost pump and electronic control unit.

A7 Safety, Maintainability, Reliability

A7.1 Safety of fuel system

Safety is important issue in the design process, and there are three kinds of catastrophic conditions due to the system failure .(the FHA report of fuel system) in this stage, the writer thought of two critical items, the dry bay design in case of rotor burst and on board inerting gas generation system related to fuel tank safety.

The dry bay design was stated in Chapter 3, and according to supervisor's advice, the engine pylon set half metre than the previous one, which lead the dry bay removal from the wing tank totally.

'The centre wing fuel tank is categorised as hazardous; requiring fuel tank inerting due to the temperatures encountered and the proximity to external heat sources of which the air conditioning units represent a significant heat source. '[23]'Left and right wing tanks are usually considered to be nonhazardous, primarily as the fuel contained within is much cooler and the fuel does not suffer from the proximity of hot aircraft components.'[23] In the baseline fuel tank, there are only wing tank in Flying Crane that means the OBIGGS could be an optional systems.

A7.2 Maintainability and Reliability

Maintainability was a major issue in the design of the fuel system. In the engine feed subsystem installation, the writer had considered about it.

The reliability target for the fuel system was 99.85%. [22] The subsystems featured in the fuel system were designed to provide enough reliability to meet this target. The calculation of the total reliability of the system is a function of the number of components, their dependency, and the fact that they have redundancies or not. The team in charge of computing the reliability of the system concluded that it reached a value of 99.88%.

A8 Discussion

A8.1 Fuel pump of HVDC powered

The fuel boost pump used 115 AC power to drive, which was conservative technology compared to 270V DC powered. The reason lies to the safety of fuel tank according to AC 25.981.1C. When the writer designed the fuel tank and components in it, the electrical arc and electrical spark is an important issue of fuel tank safety. The HVDC pump has its advantage in smaller size and less weight; however, the pump drove by 270v DC has much potential hazard than the AC pumps. Although the cable connected to the pump is much lighter and more flexible than the AC one, the protection must be considered seriously in case of wire damaged and aging.

Consequently, the usage of HVDC fuel pump is reasonable unless the technology is matured and has the same level in safety compared to AC pump.

A8.2 Dry bay design

Another issue about fuel tank safety is the influence in case of rotor burst. Although the probability of rotor burst is extremely improbable, the effect of debris penetrating fuel tank or airframe is nearly catastrophic. Therefore, the dry bay design is necessary unless the engine location moving too forward to remove it. There two cases are considering: the large volume of dry bay, the less fuel remains, which is not economic; that the moving forward of engine derates the whole performance of aircraft. So it is trade off in fuel tank design.

A9 Actuation System Introduction

A9.1 Introduction of actuation system

The general introduction of Flying Crane was described in Chapter 1, and the characteristic of this aircraft is all electric which is different from the specification given by AVIC students in last year. The all-electric aircraft has only one kind of second power- electric power, with hydraulic power and pneumatic power removed, which means that the role of providing power to flight control by hydraulic was replaced by electric power. Consequently, the actuator design should meet the new power requirement.

In the next chapters, the writer introduced the actuator design for the primary flight control system, secondary flight control system, landing gear system and so on.

A9.2 Description of task

- Choose the suitable type of actuator for control organism
- Estimate power requirements of actuation system
- Schematic block diagrams of actuation system
- System safety assessment of actuation system
- System architecture of actuation system
- Mass and size
- System layout within flight vehicle airframe (CAD)
- Check interfaces with structure and other systems (CAD)
- Airworthiness Compliance Matrix

A9.3 Compare Hydraulic Powered Actuation system and Electric Powered Actuation system

Hydraulic actuation as traditional powered has number of advantages like it has high output force capability, effective positioning of actuator corresponding to command signal. One of the advantages is that it has high accuracy even in very high force. [3]

However, on the other hand, hydraulic system has many drawbacks, either. First, hydraulic needs sets of accessory gear box of engine, central reservoir for store fluid, and many kinds of pipes and valves, which occupy considerable weight of aircraft. Second, hydraulic systems keep high pressure throughout the flight, resulting in great power loss. Third, fluid leakage and fire hazardous are potential threaten to the whole aircraft. Finally, maintenance hydraulic components cost considerable labour-intensive.

Electrical actuation uses electric power for actuation and removes the accessory gear box, central reservoir and all the pipes, reducing a great deal of mass. Another aspect, actuators powered by electric could active on command, which saves much energy. Finally, the controlling and powered wires are flexible so that they are easily routed in manufacturing, and maintenance becomes easy, either.

A9.4 Review of existing electrical actuator technologies

A9.4.1 FBW actuators

The actuators used with the FBW systems are hydraulically powered and electrically controlled by Flight Control Computer.

A9.4.2 IAP, EHA and EBHA

a) IAP

Integrated Actuator Package is similar to the conventional linear actuator, which operations on demand, summing and feedback, and linkage. The actuator power is provided by a Constant Speed Constant Frequency (CSCF) electrical motor driving a variable displacement hydraulic pump.

b) EHA

Electro-Hydrostatic Actuator is an actuator which is commanded and also powered by electrical systems. A variable speed electric motor is used to drive a hydraulic pump integrated into a hydraulic circuit with components like reservoirs and accumulators as in a typical hydraulic circuit but in smaller space.

The suppression of hydraulic components through the entire aircraft minimizes risks of leakages and consequently, improves the reliability of EHA in comparison with Hydraulic actuators.

c) EBHA

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). The EBHA normally works with the available hydraulic system and switches to electric powered EHA whenever hydraulic system is not available.

A9.4.3 Electromechanical Actuator (EMA)

The EMA “replaces the electrical signalling and power actuation of the electro-hydraulic actuator with an electric motor and gearbox assembly applying the motive force to move the ram. EMAs have been used on aircraft for many years for such uses as trim and door actuation; however the power, motive force and response times have been less than that required for flight control actuation. A major concern regarding the EMA is the Flight Control Systems consideration of the actuator jamming case and this has negated their use in primary flight controls on conventional aircraft. “[23]

A9.4.4 Comparisons of Actuation Technologies

The general comparisons are presented in table 9-1.

Considerations	FBW	IAP	EHA	EBHA	EMA
Power	Centralised hydraulic	High power electric	High power electric	Centralised hydraulic/ High power electric	High power electric
Control	Low power electric	Low power electric	Low/High power electric	Low/High power electric	Low/High power electric
Efficiency	normal	normal	high	normal	normal
Complexity	normal	normal	normal	very complex	very complex
Thermal	low	high	high	normal	high
Environmental	Fluid leak and disposal	Fluid leak and disposal	Fluid leak and disposal	Fluid leak and disposal	Negligible
actuator size	small	small	slightly bigger	big	moderate
Jamming	No	No	No	No	Yes

Table 9-1 comparison among different types of actuators

The advantages and disadvantages are clearly shown in this table, and towards the more-electric aircraft even all-electric aircraft, the central-hydraulic power has been removed, which means the FBW are not suitable for Flying Crane. Accordingly, the writer used the electrical powered actuator which detailed choosing was introduced in the Chapter 11.

A10 Requirement of Actuation system

A10.1 Airworthiness Requirement

Flying Crane mainly faces to the domestic market in medium to short haul in China, so the aeroplane must compliance with CCAR-25 in the all design phase. The actuation system must compliance with CCAR-25 and other related safety regulation as well as other airframe systems.

A10.2 Functional Requirement

The first step of actuation system is to decide the scope and boundaries with other systems, because the actuation system has close relation with them. The writer discussed with flight control system designer about the boundary and then decided that the control command and power function belongs to the flight control system, while the actuation system performs the function of controlling the surface.

The actuator needs to drive the active component to perform normal flight function, and they are follows,

Primary flight control surfaces:
Ailerons
Elevators
Rudders
Secondary flight control surfaces:
Slats
Flaps
Spoilers
Tail horizontal stabilizer
Nose landing gear steering
Landing gears retraction
Landing gears break

Thrust reversal

Cargo doors

Functional Hazardous Assessment was given according to the different control parts playing roles in the whole aircraft.

A10.3 Maintenance and reliability Requirement

Maintenance and reliability are critical issues for the customer, and they should be considered in the early stage. To achieve them, the designer needs to take into account the following items,

The simplicity design,

The redundancy design,

The operating environment design,

The human-engineering design,

The lighting protection design (CCAR 25.1316),

The test ability design,

The derating design and

The thermal design.

A11 Actuation architecture

The Actuation architecture on A340 was conventional centrally powered hydraulic channels, whist the one in A380 was more advanced with removing one hydraulic channel and increasing two electric channels. Towards the Flying Crane, an all electric aircraft serving in 2020s, these two kinds of architecture seem not suitable. Accordingly, all actuators powered by wire were needed, a 3E type was chosen for the Flying Crane.

The structure designers have worked out the control surfaces of Flying Crane as follows.

Control surfaces	Number of surfaces	Title
Elevators	2	Left and right elevator
Ailerons	2	Left and right aileron
Rudder	1	one piece of rudder
Slats	10	Left and right, inboard, middle, outboard slats
Flaps	4	Left and right, inboard, outboard flaps
Spoilers	12	Left and right, inboard, middle, outboard spoilers
Tail plane(Horizontal Stabilizer)	1	Tail plane

Table 11-1 Control surface of Flying Crane

Similar to three kinds of hydraulic power, blue, green and yellow in A340, the Flying Crane used three channels of electric power to ensure the redundancy. The writer communicated with electrical power distribution designer about the type of power distribution. After several times hesitated and turn over, the power supplied to actuators was decided, and all the actuators were powered by Electrical Loads Management Centre (ELMC) 4, 5, 6 separately, and they were corresponding to the blue, the green and the yellow one. The relationship between the actuators and power distributions could be seen in Figure10-3.

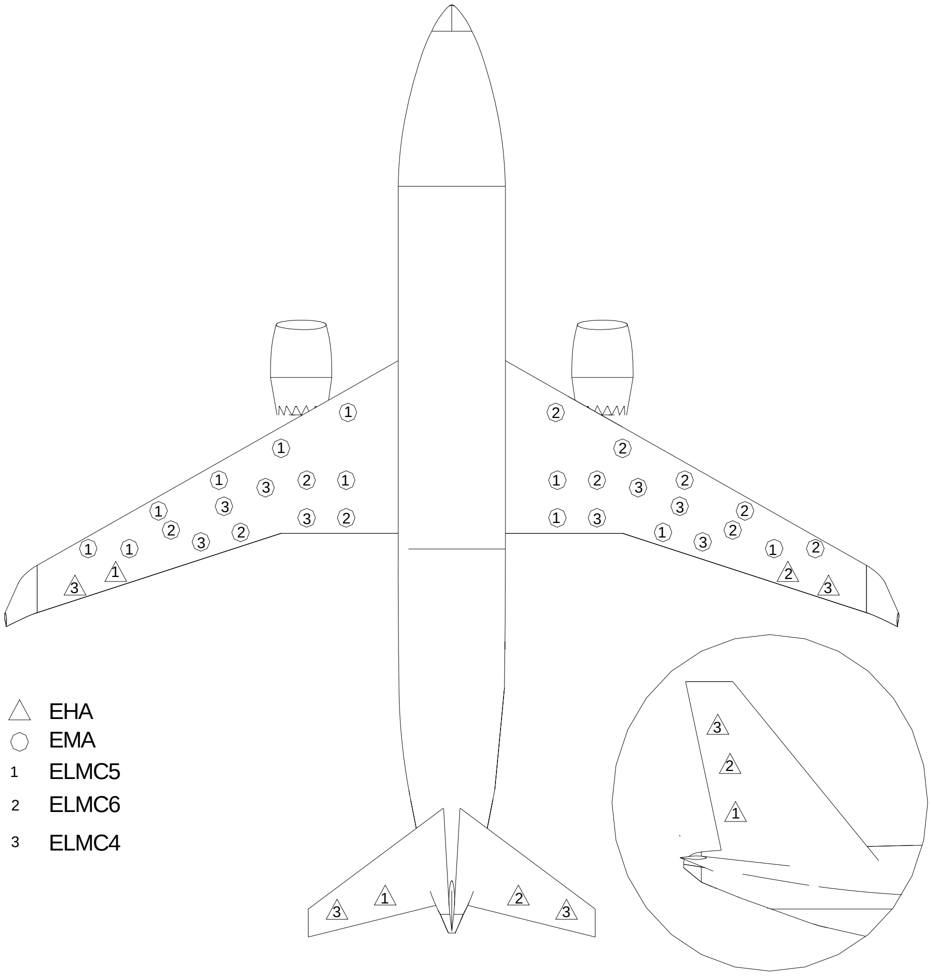


Figure 11-1 Actuator location and power distribution

Control parts	Numbers of Actuator	ELMC 4	ELMC 5	ELMC6
Elevators	4	Left and right outboard	Left outboard	Right outboard
Ailerons	4	Left and right outboard	Left and right inboard	
Rudder	3	Upper	Middle	Lower
Slats	10	Parallel connection	Parallel connection	

Flaps	8	Left and right inboard No.1, outboard No.1	Right inboard No.2,outboard No.2	Left inboard No.2,outboard No.2
Spoilers	12	Left and right No.3, No.4	Left and right No.1, No.6	Left and right No.2, No.5
Tail plane (Horizontal Stabilizer)	1	Tail plane		
Nose Wheel steering	1			
Landing gear retraction	2			
Landing gear break	24			

Table 11-2 Actuator powered distribution of Flying Crane

A11.1 Fault Tree Analysis for actuation system

When the Fault Tree Analysis (FTA) was performed for the actuation system, it was difficult for analysis, because the actuation system in this stage has close relation with flight control system and overlaps each other, which means repeated work would be done for the analysis. The writer discussed with Mr. Lv, flight control system designers, and decided to perform the FTA of flight control system and Actuation system together [30].

A12 Selection and Initial Sizing actuator on Flying Crane

A12.1 Selection of EHA

The principle of EHA was already discussed in Chapter 9; EHAs were employed for Primary Flight Control system and Landing Gear system control and Nose Wheel Steering of Flying Crane. The details of EHA are as given bellow:

A12.1.1 Operating system pressure

As mentioned before, the EHAs use a local hydraulic loop, and the size of the actuator mainly depends on the operating pressure of system. The higher pressure is operating, the miner diameter of actuator gets.

Generally, the operating hydraulic pressure is 3000 psi in civil aircraft, while in the advanced aircraft, for instance, A380 and Boeing 787, a 5000 psi system is used. In the conceptual specification of Flying Crane, it is 5000 psi in hydraulic system. The application of higher pressure has absolutely advantage-reduces the size as well as weight of actuator. However, the higher pressure has some disadvantage, such as thicker wall, more server condition for sealing rings, more chances of leakage etc.

Compared the advantage with drawbacks, the writer chose 5000 psi as operating pressure, for the reason that, the size and weight of actuator is more important especially in limited space, and the technology of higher pressure is developing to eliminate the disadvantage, including leakage.

A12.1.2 Type of EHA

a. Double acting Linear Actuator[36]

Double-acting linear actuator is the most commonly used for all types of applications and it includes two fluid chambers formed due to piston. The rod could retract and extend on either side of piston through hydraulic pressure. Effective working area of the rod side of the piston is less than that of the other side, therefore, the double-acting cylinders retract faster than they extend, and exert less force on the retraction stroke.

b. Rotary Vane Actuator (EHA)

Rotary vane actuator comprises a shaft with vanes and cylinder with opposite type of vanes on internal diameter. The hydraulic pressure driven by a motor can act in either direction of vanes for rotation. Vanes experience the differential pressure and withstand it in order to maintain the integrity of the actuator.[20]

In most cases of actuation on Flying Crane the number of vanes was two, because multiple vanes could magnify the mechanical torque as compared to a single vane actuator for the same input in hydraulic pressures.

The end-casing is attachment between the cylinder of actuator and the hinge of aircraft. They also help in completely enclosing the cylinder thereby preventing any leakages. The flange helps in transferring the torque loads to the control surface.

The rotary actuator controls the surfaces of aileron, rudder and elevator.

A12.1.3 Selection of EMA

As described in previous section, the EMAs were employed in secondary flight control system. There are two kinds of EMA used in Flying Crane, and they are Lead Screw Actuator and Planetary Roller Screw Linear Actuator.

Now days, in most of the EMA's planetary roller screw is used because of its ability to withstand at high speed and acceleration and also having capacity to handle heavy load for a long time.[20]

A12.1.4 A case study of sizing actuator

All of the actuators used in Flying Crane need to be sizing, and the methods are different for various kinds of actuator. In this section, the writer gave a case study of sizing the main landing gear actuator.

The actuator employed in main landing gear is linear EHA. In fact, it is really difficult to calculate the size, the writer referred to the supervisor to find an acceptable solution.

The actuator is designed by considering the followings:

The hydraulic force of actuator is capable to withstand the proof actuation load.

The cylinder wall thickness can meet the burst pressure requirement which is 2.5times nominal pressure.[3]

All other components are capable to handle ultimate load without any fracture failure and also return to original working conditions after the load has been removed.

Some information of landing gear is shown on the table12-1.

Actuator stroke	167.000	mm
Pressure	5000	psi
Time taken	4.000	s
Stall Load	136.923	KN
Arm	130.000	mm
Moment	17.800	KNm

Table 12-1 Some load data of Main Landing Gear

1. Overall Actuator Length:

$$L_{act} = Stroke + Fitting + Sealing + 0.5Stroke$$ [32]

Fitting is the cylinder length of holding the rod, and sealing is the overlap and redundancy between rod and cylinder. Considering the fitting is 0.18m, and the sealing is 0.01m, $L_{act} = 0.167 + 0.18 + 0.01 + 0.167 \times 0.5 = 0.44m$

2. Euler Buckling Load $P_{load} = \frac{\pi^2 EI}{L^2_{act}}$, P_{crit} is stall load, and consider

   E=200,000 MN/㎡, so, $I = \frac{P_{load} L^2_{act}}{\pi^2 E} = 1.347 \times 10^{-8} m^4$

3. Actuator Rod diameter $I = \frac{\pi r_{rod}^4}{4}$, so $r_{rod} = \left(\frac{4I}{\pi} \right)^{\frac{1}{4}} = 0.0111m$

4. Effective Piston Area $P_{\max} \left(A_{cyl} - A_{rod} \right) = P_{load}$, as shown in table $P_{\max}$ =5000psi, so

   $\left(A_{cyl} - A_{rod} \right) = 0.0052987m^2$ (A_{cyl} is the inner cross area of cylinder, while

A_{rod} is the cross area of rod)

5. $r_{cyl} = \left[\frac{\pi r^2_{rod} + \left(A_{cyl} - A_{rod} \right)}{\pi} \right]^{\frac{1}{2}} = 0.043m$

6. Cylinder thickness The actuator had to cater for 2.5 times the nominal system pressure, but that they were often design to cater for 4 times.

$P_{burst} = \frac{\delta}{2} \left[\frac{r_0^2 - r^2_{cyl}}{r_0^2} \right], \delta = 445MN/m^2$ so $r_0 = 0.069m$

7. Thickness of cylinder, $r_{thickness} = r_0 - r_{cyl} = 0.026m$

A13 Power Requirement and Mass of Actuation System

A13.1 Some factors about power requirement

When the writer estimated the power of actuators, some factors needed to be considered. First, time is critical for it has close relation with power. Second, the efficiency of power transfer, that is, mechanical power to hydraulic power and hydraulic power to electrical power is necessary to thought of. Third, the power needed is various during the flight, so the peak power and the average power is main consideration for an actuator.

A13.2 Power Estimation

A13.2.1 Power requirement of Nose Landing Gear

The landing gear retraction and deployment occurred during the Take-off and Landing phase in a flight. Therefore, the calculations for power requirement are based on the ratio of work done to time taken for actuation.

The writer needed to find the peak power and average power of actuator. The clues were found that the force reached the maximum value when the actuator fully retracted, while the retraction is a process with acceleration and deceleration.

Force (kN)=	67.708
Stroke(mm)=	283
Time Taken (s)=	4

It was necessary to discretized time and Force to acquire the force-velocity curve.

F(KN)	v(mm/S)	T(S)
0	0	0
6.6708	37.73	0.4
13.3416	75.46	0.8
20.0124	94.33	1.2

26.6832	94.33	1.6
33.354	94.33	2
40.0248	94.33	2.4
46.6956	94.33	2.8
53.3664	75.47	3.2
60.0372	37.73	3.6
66.708	0	4

Table 13-1 Force and velocity variation with Time

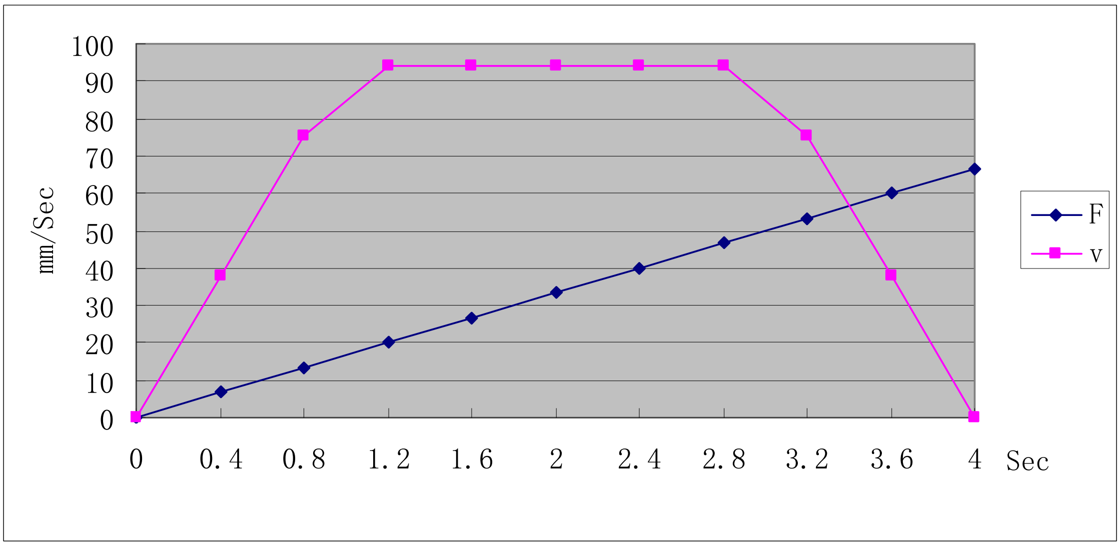


Figure 13-1 Force and velocity variation curve

From the curve, the peak mechanical power was got.

Peak Mechanical Power=4.4 kW

Peak Hydraulic Power= Peak Mechanical Power/ η(pump)

Peak Electrical Power= Peak Hydraulic Power/ η(hydro to electro)

Where, *η(pump)*=0.75, *η(hydro to electro)*=0.9

Therefore, the *Peak Electrical Power*=6.52kW

The *Average Electrical Power=F×Stroke/time*=3.56kW

A13.2.2 Power requirement of Rudder

The actuator used in rudder is rotary EHA, and its feature could be found in the table.

Deflection(left)	20	°
Deflection(right)	20	°
Actuator Stroke	48.400	mm
Angular Stroke	0.349	radians
Time taken	2.000	s
Average Velocity	0.175	radians/s
Stall Load	66.895	KN

Table 13-2 Feature of Rudder

The writer referred the method used in MRT-7 to estimate the power requirement. First, time and Force was discretized to acquire the force-velocity curve. It shows in Table13-3.

	D.F	Time(s)	Angular V (rad/s)	Torque(KNM)	Theoretical Power(Kw)
1000	0	0	0.35	0	0
1400	0.1	0.2	0.315	1.392	0.438
1800	0.2	0.4	0.28	2.749	0.770
2200	0.3	0.6	0.245	4.039	0.990
2600	0.4	0.8	0.21	5.230	1.098
3000	0.5	1	0.175	6.291	1.101

3400	0.6	1.2	0.14	7.198	1.008
3800	0.7	1.4	0.105	7.927	0.832
4200	0.8	1.6	0.07	8.462	0.592
4600	0.9	1.8	0.035	8.787	0.308
5000	1	2	0	8.897	0

Table 13-3 Velocity and torque various with time

As noticed in the first column of Table 13-3, the extreme values for Theoretical Power are Zero, which can not be possible.

The writer followed MRT-7 design, and gave a solution to calculate the Power as indicated in the second column of Table 13-4.

Power= T.P + (0.1+[D.F]×0.5) × (T.P)max

Where, T.P = Theoretical Power,

D.F = Discretization Factor

(T.P) max=Maximum Theoretical Power.

It can be seen that the peak power is 1.486

KW. The average power is 1.486 × 0.369=0.55 kW. (0.369 is the ratio of Avg/Peak)

Peak Hydraulic Power= Peak Mechanical Power/ η (pump)

Peak Electrical Power= Peak Hydraulic Power/ η (hydro to electro)

Where, η (pump)=0.75, η (hydro to electro)=0.9

Therefore, the Peak Electrical Power=2.21kW

The Average Electrical Power=0.815 kW.

Theoretical Power(Kw)	Power(Kw)
0	0.11
0.438	0.606
0.770	0.990
0.9904	1.265
1.098	1.428
1.101	1.486
1.008	1.448
0.832	1.328
0.592	1.143
0.308	0.913
0	0.661

A13.2.3 Power requirements for each actuator

The overall power requirements for the actuation system of Flying Crane were listed in the Table 13-4.

Actuation Electrical Power Requirements				
Control parts	No. of actuators	Maximum power of one actuator (kW)	Maximum Power of Controlling (kW)	Average Power of Controlling (kW)
Ailerons	4	1.76	3.52	1.30
Elevators	4	3.4	6.8	2.51
Rudder	3	2.2	4.4	1.62
Flaps	8	1.15	9.2	3.39
Slats	10	1	10	3.37
Spoilers	12	1.5	18	6.64
Horizontal Stabilizer	1	1.56	1.56	0.57
Main Landing Gear retraction	2	8.46	16.92	6.24
Nose Landing Gear retraction	1	6.52	6.52	2.40
Main Landing Gear breaking	24	1.14	27.4	27.4
Nose Wheel Steering	1	3.8	3.8	3.8

Table 13-4 Actuation Electrical Power Requirements

A13.3 Mass estimation

In the preliminary design phase, it is really difficult to estimate the mass of actuators. The writer gave a roughly estimation of mass according to the methods in MRT-7 and A-8.

A13.3.1 Mass Estimation of EHA

The EHA comprises pump, accumulator, reservoir, and actuator control electronic (ACE), in which the ACEs of flight control surfaces, were considered by Mr. Lv (Flight Control system designer.)

The Power to Weight ratio for electrical motor driven pump is 0.5 kW/ kg, and it can be used for mass estimation.[3]

Mass of accumulator and reservoir was considered as 3 kg.

The ACE used for the Landing gear retraction and steering was considered to be 1.5 kg, while the ACE for landing gear break was 1 kg.

Main Landing Gear retraction actuator:

Mass of rod= $\pi r^2_{rod} \times L_{rod} \times \rho$ (ρ is the density of material)

Mass of cylinder= $\pi\left(r_0^2 - r_{cyl}^2\right) \times L_{cyl} \times \rho$ (ρ is the density of material)

Serial No.	Components	Mass (kg)
1	Rod	1
2	Cylinder	13
3	Motor- pump	16.9
4	Accumulator and reservoir	3
5	ACE	1.5
6	Additional mass including sensors	1
	Total	36.4

Table 13-5 Mass of main landing gear retraction actuator

A13.3.2 Mass Estimation of EMA

When estimating the mass of EMA, followings were considered

Mass of motor = 3 kg

Mass of Gear set and universal joint= Mass of rod

Mass of end connection= Mass of rod/2

A13.3.3 Mass Estimation of Actuation system

Table 13-6 was created by following the above explained procedure on each of the major actuator in the system.

Actuator	Mass of actuator (kg)	No. of actuators	Total Mass (kg)
Ailerons	13.25	4	53
Elevators	17.6	4	70.4
Rudder	15.85	3	47.55
Flaps	7.3	8	58.4
Slats	7	10	70
Spoilers	8	12	96
Horizontal Stabilizer	29	1	29
Main Landing Gear retraction	36.4	2	72.8
Nose Landing Gear retraction	31	1	31
Main Landing Gear breaking	4	24	96
Nose Wheel Steering	25	1	25

Table 13-6 Mass estimation of actuation system

APPENDIX B: Electric Power System Functional Hazard Assessment

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-1	Control system runs independently among different 115 v/400Hz channels	24-4A-1-1a	Loss of 115V/400Hz channels operating independently controllability	G1,G2	1. If this function loss, it will result in all 115V/400Hz power channel loss in the event of any channel fault, as a result, **all** the 115/400Hz loads will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-1	Control system runs independently among different 115 v/400Hz channels	24-4A-1-1b	Loss of 115V/400Hz channels operating independently controllability	T1, F1,F2,F3,F4,L1	1. If this function loss, it will result in all 115V/400Hz power channel loss in the event of any channel fault, as a result, **all** the 115/400Hz loads will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. The occupant might be injured due to lose pressure in cabin or hard landing.	Ⅱ			

Flight Phase				Failure Condition Classification
Ground	Takeoff	In Flight	Landing	
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent F4 Approach F5 Other (Describe)	L1 Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-2	Control system runs in parallel among different 115 v/400Hz channels	24-4A-2-1a	Loss of 115V/400Hz channels operating in parallel controllability	G1,G2	1. If this function loss, it will prevent power transfer from the normal channel to the power loss channel, as a result, part of the 115V/400Hz channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-2	Control system runs in parallel among different 115 v/400Hz channels	24-4A-2-1b	Loss of 115V/400Hz channels operating in parallel controllability	T1, F1,F2,F3,F4,L1	1. If this function loss, it will result in all 115V/400Hz power channel loss in the event of any channel fault, as a result, part of the 115V/400Hz channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅲ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-3	Control system runs independently among different 28VDC channels	24-4A-3-1a	Loss of 28VDC channels operating independently controllability	G1,G2	1. If this function loss, it will result in all 28VDC power channel loss, as a result, all the 28VDC loads will loss lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-3	Control system runs independently among different 28VDC channels	24-4A-3-1b	Loss of 28VDC channels operating independently controllability	T1, F1,F2,F3, F4,F5,L1	1. If this function loss, it will result in all 28VDC power channel loss, as a result, all the 28VDC loads will lose power, the airplane will lose control. The reason is that the entire vital loads power requirement is 28VDC. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. The occupants will suffer the severe injury due to lose of airplane controllability.	Ⅰ			

Flight Phase				Failure Condition Classification
Ground	Takeoff	In Flight	Landing	
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent F4 Approach F5 Other (Describe)	L1 Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-4	Control system runs in parallel among different 28VDC channels	24-4A-4-1a	Loss of 28VDC channels operating in parallel controllability	G1, G2	1. If this function loss, it will result in all 28VDC power channel loss, as a result, part of the 28VDC channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-4	Control system runs in parallel among different 28VDC channels	24-4A-4-1b	Loss of 28VDC channels operating in parallel controllability	T1, F1,F2,F3, F4,F5,L1	1. If this function loss, it will result in all 28VDC power channel loss, as a result, part of the 28VDC channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅲ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-5	Control system runs independently among different 270VDCchannels	24-4A-5-1a	Loss of 270VDC channels operating independently controllability	G1, G2	1. If this function loss, it will result in all 270VDC power channel loss in the event of any channel fault, as a result, all the 270VDC loads will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-5	Control system runs independently among different 270VDC channels	24-4A-5-1b	Loss of 270VDC channels operating independently controllability	T1, F1,F2,F3,F4,L1	1. If this function loss, it will result in all 270VDC power channel loss in the event of any channel fault, as a result, all the 270VDC loads will lose power, Therefore, the aircraft will lose control. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. The occupants will suffer the severe injury due to lose of airplane controllability.	I			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	
G1 Aircraft Static with System Operating	T1 Takeoff	F1 Climb	F4 Approach	L1 Landing	Ⅰ Catastrophic
G2 Taxi		F2 Cruise	F5 Other (Describe)		Ⅱ Hazardous
		F3 Descent			Ⅲ Major
					Ⅳ Minor
					Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4A-6	Control system runs in parallel among different 270VDCchannels	24-4A-6-1a	Loss of 270VDC channels operating in parallel controllability	G1, G2	1. If this function loss, it will result in all 270VDC power channel loss in the event of any channel fault, as a result, part of the 270VDC channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅳ			
24-4A-6	Control system runs in parallel among different 270VDC channels	24-4A-6-1b	Loss of 270VDC channels operating in parallel controllability	T1, F1,F2,F3,F4,L1	1. If this function loss, it will result in all 270VDC power channel loss in the event of any channel fault, as a result, part of the 115V/400Hz channels will lose power. 2. Flight crew will take the relative procedures to deal with this circumstance. 3. None	Ⅲ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	
G1 Aircraft Static with System Operating	T1 Takeoff	F1 Climb	F4 Approach	L1 Landing	Ⅰ Catastrophic
G2 Taxi		F2 Cruise	F5 Other (Describe)		Ⅱ Hazardous
		F3 Descent			Ⅲ Major
					Ⅳ Minor
					Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4B-1	Vital loads power supply control	24-4B-1-1a	Loss of vital loads power supply control	G1, G2	The aircraft will lose all the vital loads power supply. The flight crew work load will be increased to cope with this emergency circumstance. No effect on the passengers	Ⅳ			
24-4B-1	Vital loads power supply control	24-4B-1-1b	Loss of vital loads power supply control	T1, F1,F2,F3,F4,F5, L1	The aircraft will lose all the essential loads power supply. The aircraft will loss controllability The flight crew work load will be excessively increased to cope with this emergency circumstance All the occupants might be seriously injured due to lose controllability of the aircraft.	Ⅰ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4B-2	Essential loads power supply control	24-4B-2-1a	Loss of Essential loads power supply control	G1, G2	The aircraft will lose all the essential loads power supply. The aircraft will loss some functions. The flight crew work load will be increased to cope with this emergency circumstance. All the occupants might feel discomfort due to some function loss of the aircraft.	Ⅳ		24-4B-2	
24-4B-2	Essential loads power supply control	24-4B-2-1b	Loss of Essential loads power supply control	T1, F1,F2,F3,F4,F5, L1	1、The aircraft will lose all the Essential loads power supply. The aircraft will significantly lose function. The flight crew work load will be increased to cope with this emergency circumstance All the occupants might feel discomfort due to some function loss of the aircraft.	Ⅰ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-4B-3	Non-Essential loads power supply control	24-4B-3-1a	Loss of Non-Essential loads power supply control	G1, G2,T1,F1,F2,F3, F4,F5,L1	1、The aircraft will lose all the Non-essential loads power supply. As a result, The aircraft will loss some functions, but it will not effect on aircraft performance. No effect on crew No effect on passengers	Ⅴ			
24-4B-3	Load circuit protection	24-4B-3-1b	Loss of load circuit protection	G1, G2	1、If this function loss, it will effect on other loads operating which is located on the same busbar. As a result, it would result in the busbar loss. 2、Flight crew work load will be increased 3、No effect on occupants	Ⅳ			24-4B-3

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-5A	Load circuit protection	24-5A-1	Loss of load circuit protection	T1, F1,F2,F3,F4,F5, L1	1、If this function loss, it will effect on other loads operating which is located on the same busbar. As a result, it would result in the busbar loss. Because the loads on this busbar are mixed. If the vital loads are affected, It will threaten flight safety. 2、Flight crew work load will be significantly increased 3、No effect on occupants	Ⅲ			24-5A
24-5B	Main feeder protection	24-5B-1a	Loss of main feeder protection	G1, G2	If this function loss, it will result in large amount of power loss even cause fire due to high current level. Flight crew work load will be significantly increased Occupants might be injured due to fire, but the majority occupants can survive by emergency evacuation.	Ⅲ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-5B	Main feeder protection	24-5B-1b	Loss of main feeder protection	T1, F1,F2,F3,F4,F5, L1	If this function loss, it will result in large amount of power loss even cause fire due to high current level. Flight crew work load will be significantly increased Occupants might be injured due to fire.	II			
24-5C	Busbar protection	24-5C-1a	Loss of Busbar protection	G1, G2	If this function loss, it will result in large amount of power loss even cause fire due to high current level. Flight crew work load will be significantly increased Occupants might be injured due to fire, but the majority occupants can survive by emergency evacuation.	III		24-5C	Busbar protection

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	I Catastrophic
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	II Hazardous III Major IV Minor V No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-5C	Busbar protection	24-5C-1b	Loss of Busbar protection	T1, F1,F2,F3,F4,F5, L1	1、If this function loss, it will result in large amount of power loss even cause fire due to high current level. Flight crew work load will be significantly increased Occupants might be injured due to fire.	Ⅱ		24-5C	Bus bar protection
24-6A	System internal data(signal)exchange	24-6A-1a	Loss of inter data transfer	G1, G2	If this function loss, it will significant effect on electric power distribution capability, as a result the vast range of airframe systems will be affected. Flight crew work load will be increased 3. No effect on occupants.	Ⅳ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	Ⅰ Catastrophic
G1 Aircraft Static with System Operating G2 Taxi	T1 Takeoff	F1 Climb F2 Cruise F3 Descent	F4 Approach F5 Other (Describe)	L1 Landing	Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect

Function Ref.	Function Description	Failure Condition Ref.	Failure Condition (Hazard Description)	Flight Phase	Effect on Aircraft/Occupants 1. Aircraft 2. Crew 3. Cabin occupants	Classification	Validation	Verification	Remark
24-6A	System internal data(signal)exchange	24-6A-1b	Loss of inter data transfer	T1, F1,F2,F3,F4,F5, L1	If this function loss, it will significant effect on electric power distribution capability, as a result the vast range of airframe systems will be affected. Flight crew work load will be significant increased Occupants will feel uncomfortable due to some systems function loss.	Ⅲ			
24-6B	External data exchange	24-6B-1	Loss of external data exchange	G1, G2,F1,F2,F3,F4, F5,L1	If this function loss, electric power system function can not be provided to flight crew immediately, but since system operate automatically, and EPS operates well or not can reflected indirectly by other system information. So no effects on aircraft operational capability.	Ⅴ			

Flight Phase					Failure Condition Classification
Ground	Takeoff	In Flight		Landing	
G1 Aircraft Static with System Operating	T1 Takeoff	F1 Climb	F4 Approach	L1 Landing	Ⅰ Catastrophic Ⅱ Hazardous Ⅲ Major Ⅳ Minor Ⅴ No Safety Effect
G2 Taxi		F2 Cruise	F5 Other (Describe)		
		F3 Descent			

APPENDIX C: FMEA of EPS

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
11	Generator1	Failure of excitation	Flight	low power factor	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Interphase short circuits	Flight	loss power output	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Single phase to earth fault	Flight	unbalanced power output	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Inter turn faults	Ground and Flight	Generator failure	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Earth fault on rotor	Flight	increased vibration in the alternator	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
12	GCU1	Circuit fault (open or short)	Flight	GCU1 failure	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
13	Generator2	Failure of excitation	Flight	low power factor	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
13	Generator2	Interphase short circuits	Flight	loss power output	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Single phase to earth fault	Flight	unbalanced power output	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Inter turn faults	Ground and Flight	Generator failure	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
		Earth fault on rotor	Flight	increased vibration in the alternator	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
14	GCU2	Circuit fault (open or short)	Flight	GCU2 failure	decrease the supply voltage and the stability of the system	Reduce power supply capability	Major
15	Generator3 (APU)	Failure of excitation	one or two main generators failure	low power factor	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous
		Interphase short circuits	one or two main generators failure	loss power output	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
15	Generator3 (APU)	Single phase to earth fault	one or two main generators failure	unbalanced power output	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous
		Inter turn faults	one or two main generators failure	Generator failure	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous
		Earth fault on rotor	one or two main generators failure	increased vibration in the alternator	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous
16	GCU3	Circuit fault (open or short)	one or two main generators failure	GCU3 failure	decrease the supply voltage and the stability of the system	Reduce emergency power supply capability	Hazardous
17	GCB1	Failure to trip due to an open or short in the trip circuit wiring	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to trip due to an open or short in the trip coil	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
17	GCB1	Failure to trip due to mechanism problem	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to clear by mechanism problem	Flight	Failure to clear	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to clear by dielectric	Flight	Failure to clear	over current and make generators paralleled	Decrease the stability of the system	Major
18	GCB2	Failure to trip due to an open or short in the trip circuit wiring	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to trip due to an open or short in the trip coil	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to trip due to mechanism problem	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to clear by mechanism problem	Flight	Failure to clear	over current and make generators paralleled	Decrease the stability of the system	Major

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
18	GCB2	Failure to clear by dielectric	Flight	Failure to clear	over current and make generators paralleled	Decrease the stability of the system	Major
19	BPCU1	Failure to trip	Flight	Failure to trip	over current and make generators paralleled	Decrease the stability of the system	Major
		Failure to clear	Flight	Failure to clear	over current and make generators paralleled	Decrease the stability of the system	Major
110	BPCU2	Failure to trip	Ground	Failure to trip	over current	over current	Minor
		Failure to clear	Ground	Failure to clear	Over current	Over current	Minor
111	BPCU3	Failure to trip	Ground	Failure to trip	over current	over current	Minor
		Failure to clear	Ground	Failure to clear	Over current	Over current	Minor
201	Frequency Modulation1	Overfrequency/ underfrequency	Flight	Overfrequency/ underfrequency	decrease power quality	Decrease the stability of the system	Minor
		Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power quality	Decrease the stability of the system	Minor
		Internal circuit open	Flight	Loss power	decrease CF AC power capability	Reduce power capability	Minor

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
201	Frequency Modulation1	Internal circuit short	Flight	Loss power	decrease CF AC power capability	Reduce power capability	Major
202	Frequency Modulation2	Overfrequency/ underfrequency	Flight	Overfrequency/ underfrequency	decrease power quality	Decrease the stability of the system	Minor
		Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power quality	Decrease the stability of the system	Minor
		Internal circuit open	Flight	Loss power	decrease CF AC power capability	Reduce power capability	Minor
		Internal circuit short	Flight	Loss power	decrease CF AC power capability	Reduce power capability	Major
203	H TRU1	Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power quality	Decrease the stability of the system	Minor
		Internal circuit open	Flight	Loss power	decrease 270v DC power capability	Reduce power capability	Minor
		Internal circuit short	Flight	Loss power	decrease 270v DC power capability	Reduce power capability	Major
204	H TRU2	Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power quality	Decrease the stability of the system	Minor

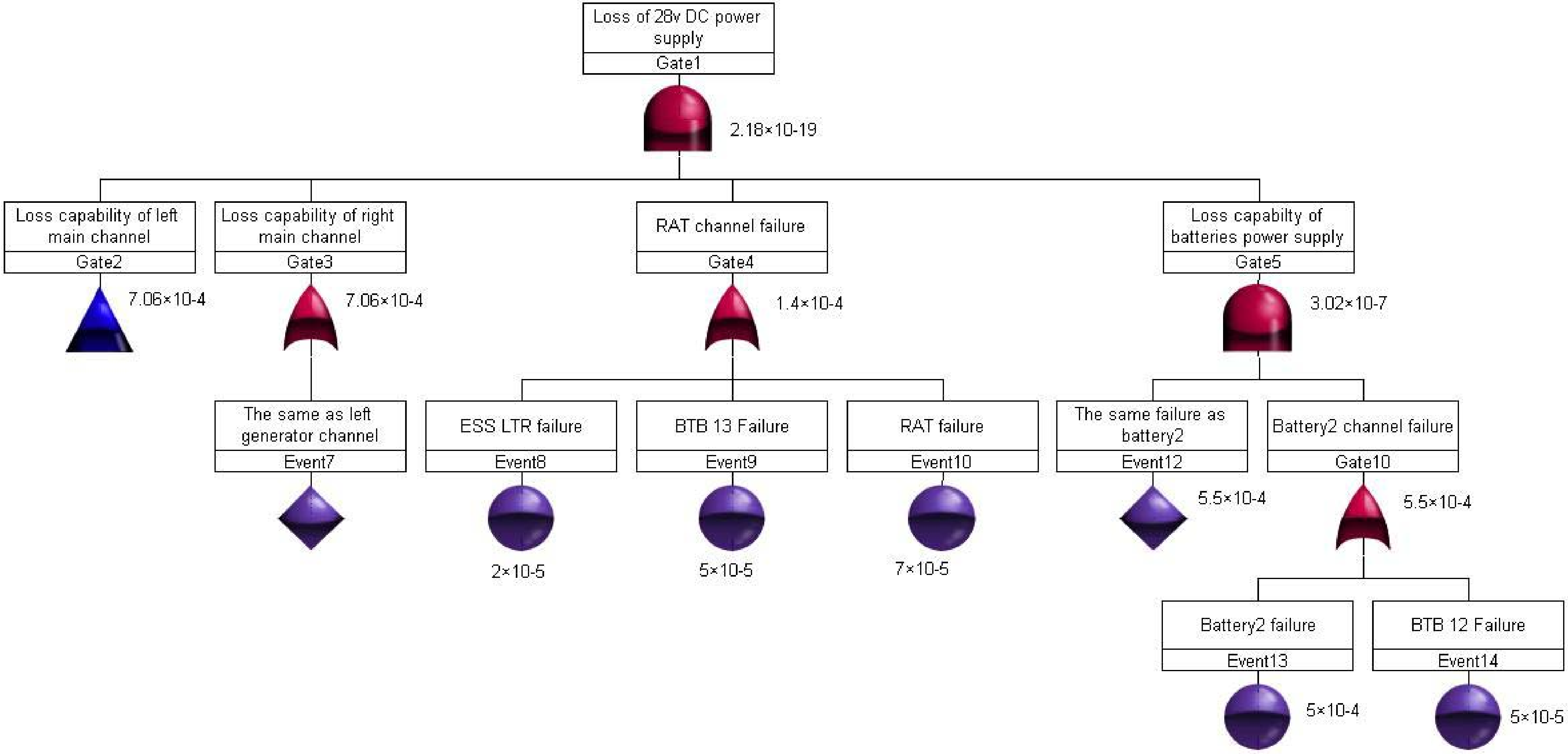
Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
204	H TRU2	Internal circuit open	Flight	Loss power	decrease 270v DC power capability	Reduce power capability	Minor
		Internal circuit short	Flight	Loss power	decrease 270v DC power capability	Reduce power capability	Major
205	L TRU1	Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power 28v DC quality	Decrease the stability of the system	Minor
		Internal circuit open	Flight	Loss power	decrease 28v DC power capability	Reduce power capability	Minor
205	L TRU1	Internal circuit short	Flight	Loss power	decrease 28v DC power capability	Reduce power capability	Major
206	L TRU2	Overvoltage/ undervoltage	Flight	Overvoltage/ undervoltage	decrease power 28v DC quality	Decrease the stability of the system	Minor
		Internal circuit open	Flight	Loss power	decrease 28v DC power capability	Reduce power capability	Minor
		Internal circuit short	Flight	Loss power	decrease 28v DC power capability	Reduce power capability	Major
207	BTB1	Fails to open due to mechanism problem	Flight	Fails to open	possibly connect APU to Generator 1	None	Minor

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
207	BTB1	Fails to close due to mechanism problem or internal circuit failure	Flight with Generator 1 failure	Fails to close	Disconnect APU Generator to main AC Bus	Decrease emergency power capability	Hazardous
208	BTB2	Fails to open due to mechanism problem	Flight	Fails to open	possibly connect APU to Generator 2	None	Minor
		Fails to close due to mechanism problem or internal circuit failure	Flight with Generator 2 failure	Fails to close	Disconnect APU Generator to main AC Bus	Decrease emergency power capability	Hazardous
209	BTB3	Fails to close due to mechanism problem or internal circuit failure	Flight with LTRU1 1 failure	Fails to close	decrease 28v DC power capability	Reduce power capability	Major
210	BTB4	Fails to close due to mechanism problem or internal circuit failure	Flight with LTRU2 failure	Fails to close	decrease 28v DC power capability	Reduce power capability	Major
211	BTB5	Fails to close due to mechanism problem or internal circuit failure	Flight with HTRU 1 failure	Fails to close	decrease 270v DC power capability	Reduce power capability	Major

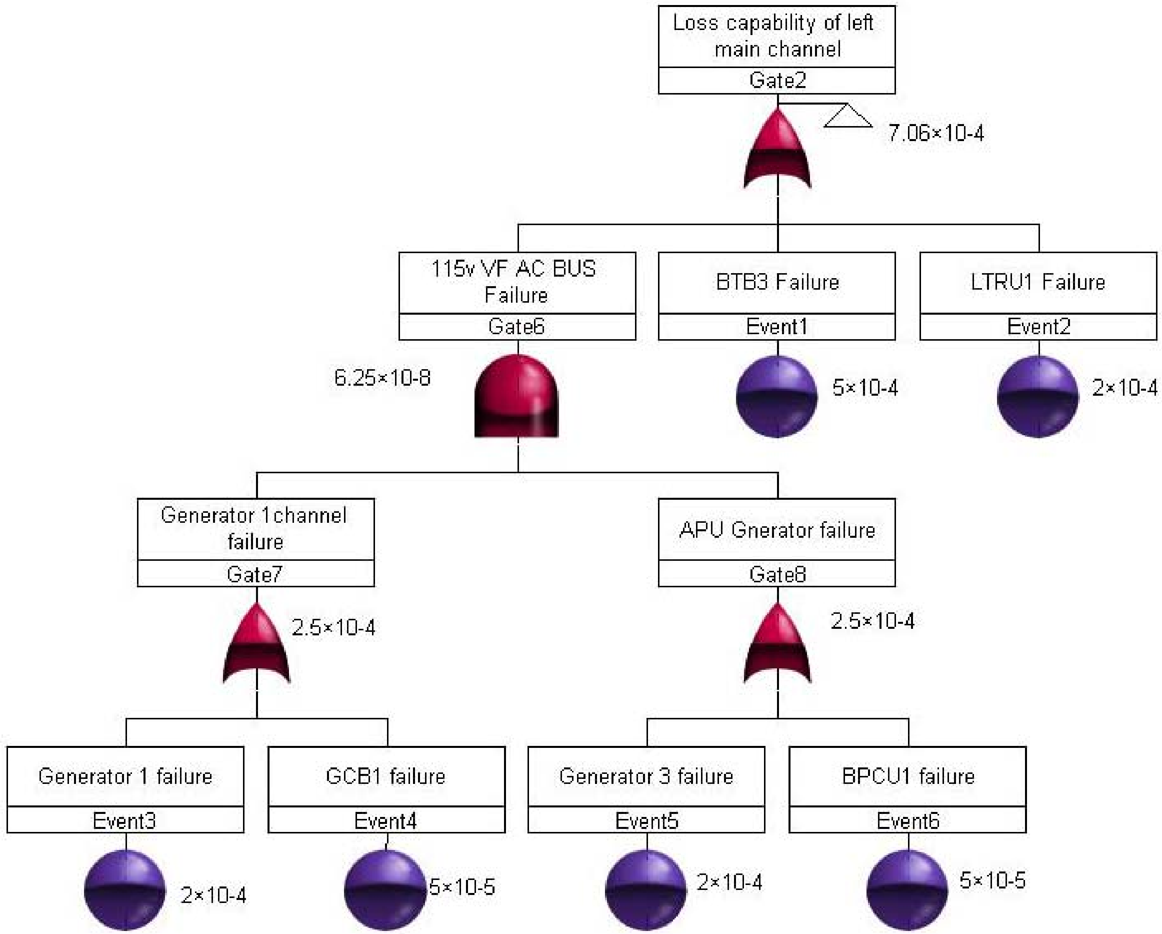
Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
212	BTB6	Fails to close due to mechanism problem or internal circuit failure	Flight with HTRU 2 failure	Fails to close	decrease 270v DC power capability	Reduce power capability	Major
213	BTB7	Fails to close due to mechanism problem or internal circuit failure	Flight with Frequency Modulation 1 failure	Fails to close	decrease 115v CF AC power capability	Reduce power capability	Major
214	BTB8	Fails to close due to mechanism problem or internal circuit failure	Flight with Frequency Modulation 2 failure	Fails to close	decrease 115V CF AC power capability	Reduce power capability	Major
215	BTB9	Fails to close due to mechanism problem or internal circuit failure	Ground and Flight	Fails to close	loss capability to charge Battery1	None	None
216	BTB10	Fails to close due to mechanism problem or internal circuit failure	Ground and Flight	Fails to close	loss capability to charge Battery2	None	None
217	BCRU1	Open or short in internal circuit	Ground and Flight	BCRU1 failure	loss capability to charge Battery1	None	None
218	BCRU2	Open or short in internal circuit	Ground and Flight	BCRU2 failure	loss capability to charge Battery2	None	None

Identification number	Item	Failure Modes and Causes	Flight/Mission Phase	Local Effect	Next Higher Effect	End Effect	Severity Level
219	BTB11	Fails to close due to mechanism problem or internal circuit failure	Flight with main generator failure	Fails to close	Reduce capability to power ESS DC Bus	Reduce vital power capability	Hazardous
220	BTB12	Fails to close due to mechanism problem or internal circuit failure	Flight with main generator failure	Fails to close	Reduce capability to power ESS DC Bus	Reduce vital power capability	Hazardous
221	ESS LTRU	Internal circuit open or short	Flight with generators failure	ESS LTRU failure	Reduce capability to power ESS DC Bus	Reduce vital power capability	Hazardous
222	BTB13	Fails to close due to mechanism problem or internal circuit failure	Flight with main generators failure	Fails to close	Reduce capability to power ESS DC Bus	Reduce vital power capability	Hazardous
223	STAT INV	Internal circuit open or short	Flight with generators failure	STAT INV Failure	loss capability to power 115v CF AC BUS	Reduce emergency power capability	Hazardous
224	DC/DC	Internal circuit open or short	Flight with generators failure	DC/DC failure	loss capability to power 270v DC BUS	Reduce emergency power capability	Hazardous
31	Microprocessor Module1	Malfunction	Flight	Malfunction	decrease reliability of system	Reduce power supply capability	Major

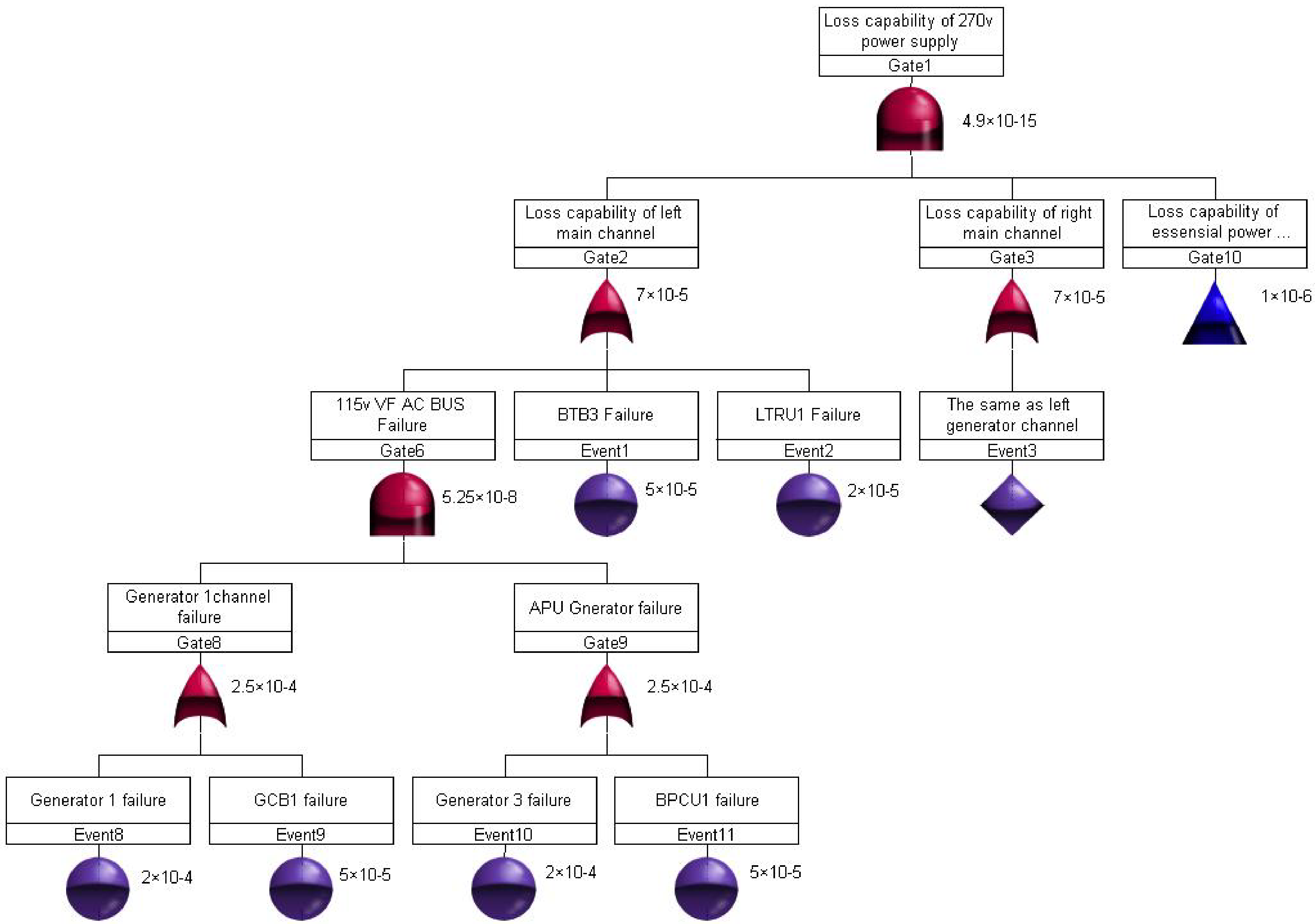
APPENDIX D: FTA OF EPS



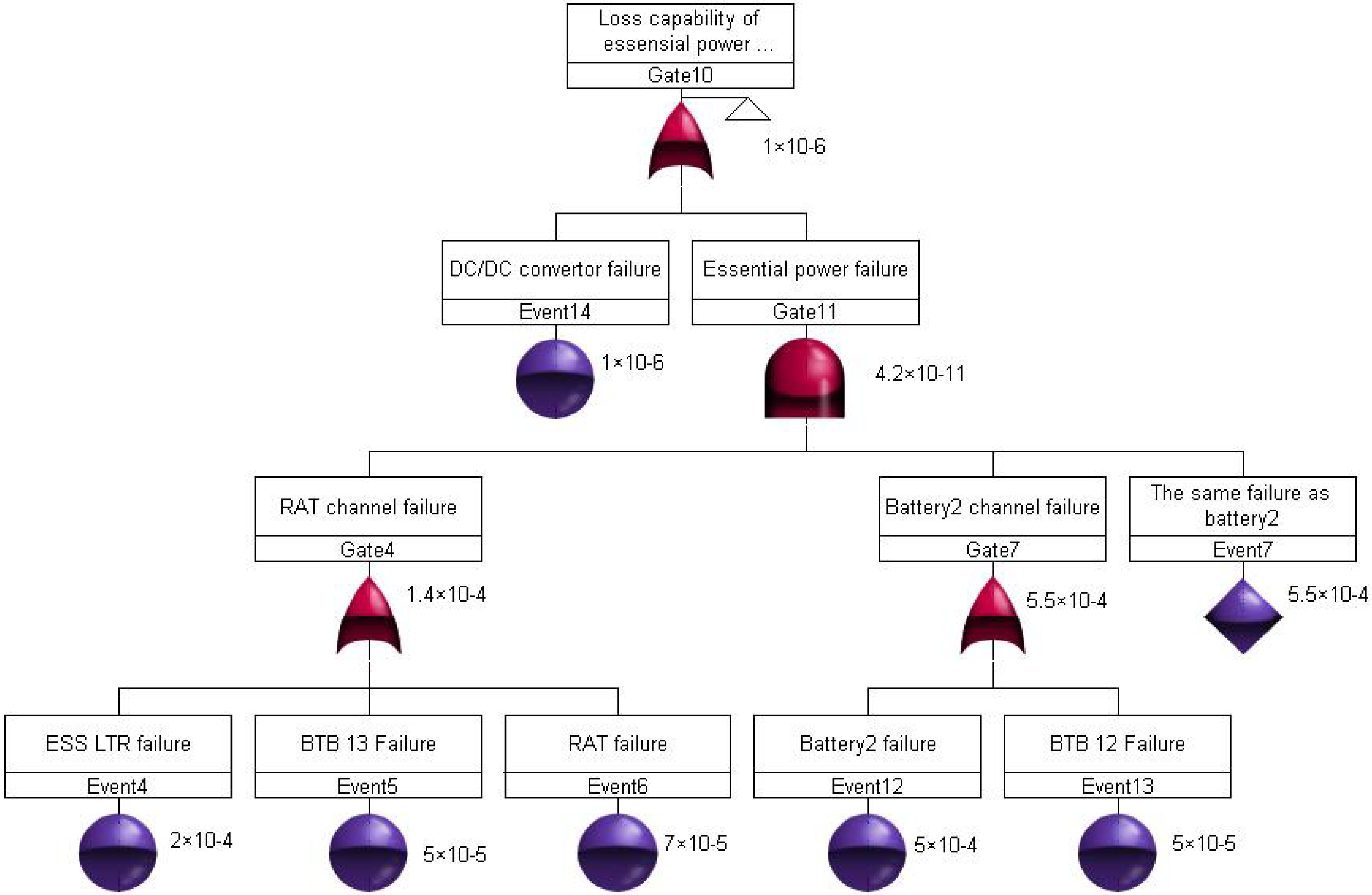
Loss power supply of 28V DC (1)



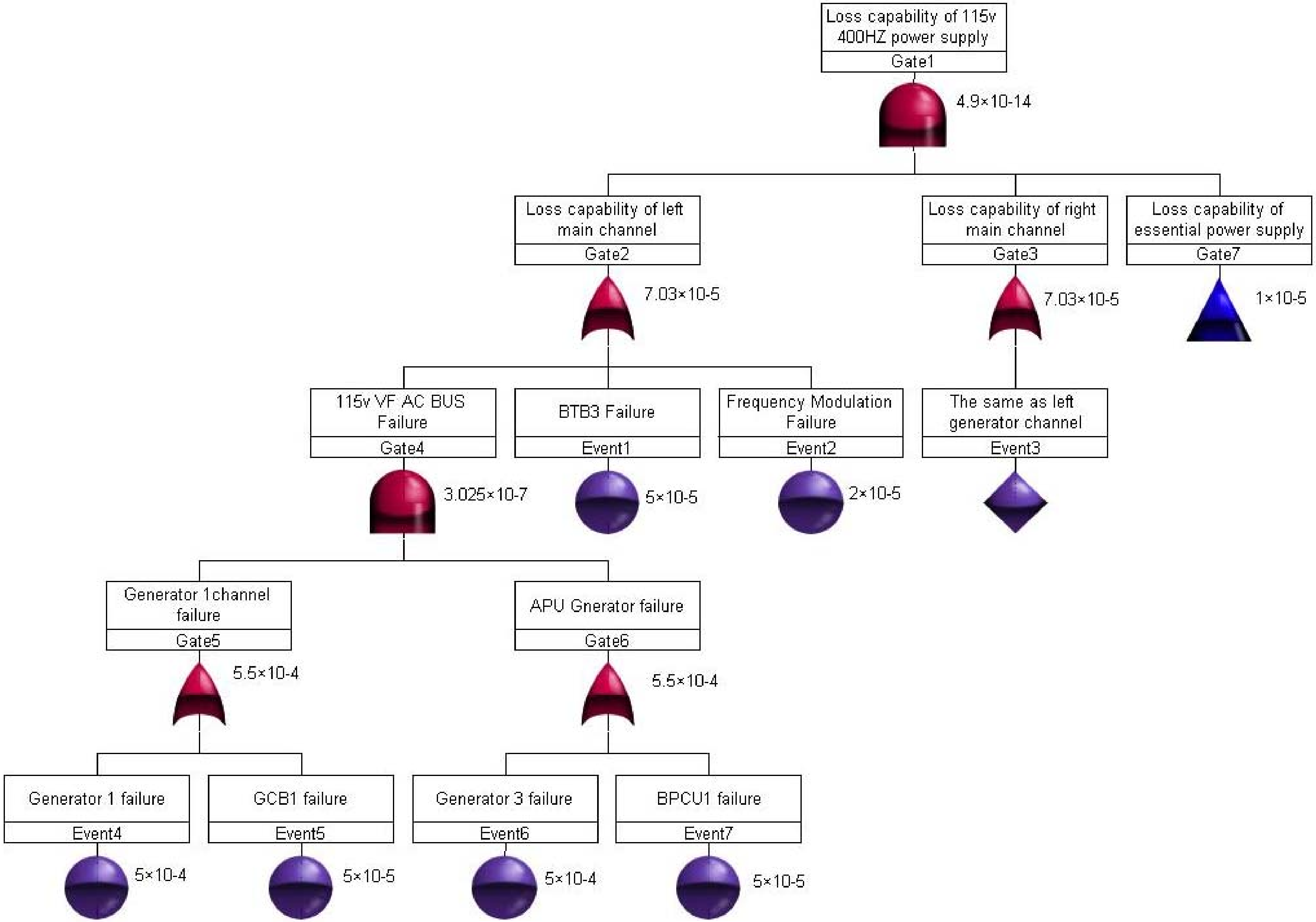
Loss power supply of 28V DC (2)



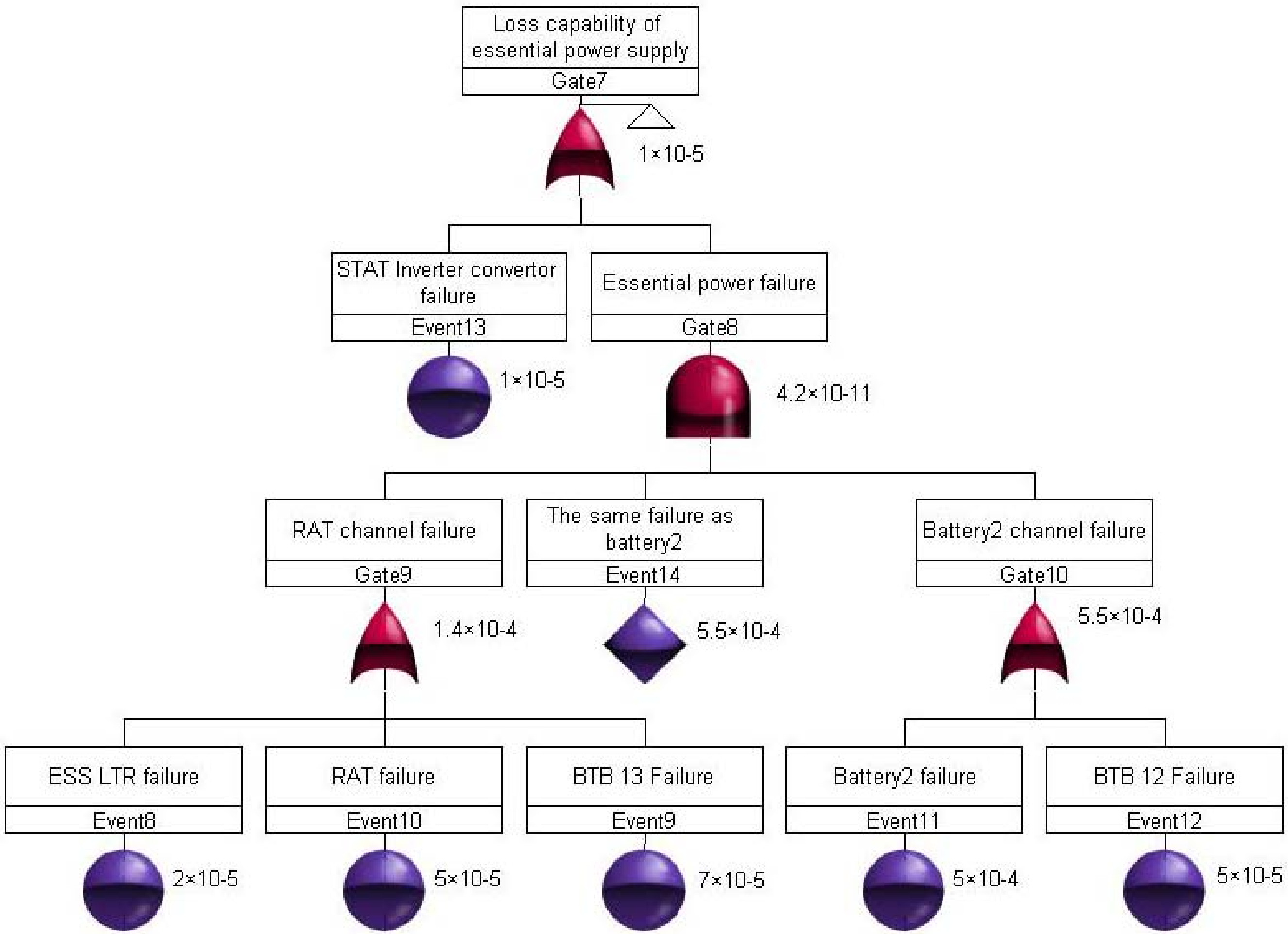
Loss power supply of 270V DC (1)



Loss power supply of 270V DC (2)



Loss power supply of 115V/400HZ AC (1)



Loss power supply of 115V/400HZ AC (2)