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Design of an Active Fault-Tolerant Control System with transducer Faults for attitude control of rigid Spacecraft

MSc Thesis for the Partial Fulfillment of Master of
Science in Electrical Automation and Control Technology Management

By,

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Design of an Active Fault-Tolerant Control System with Transducer Faults for attitude control of rigid Spacecraft

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DECLARATION

I hereby declare that the work presented in this thesis, " Design of an Active Fault-Tolerant Control System with Transducer Faults for attitude control of rigid Spacecraft**,**" is my own original work, that it has not been submitted for a degree at this or any other university, and that all sources of materials used in this thesis work have been fully acknowledged.

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Date of submission: _________________

This thesis has been submitted for examination with my approval as a TVTI advisor

Dr. Saravanakumar Gurusamy	_____________	______________
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Thesis on

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APPROVED BY THESIS ADVISOR COMMITTEE

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Dr. ESkinder Anteneh   --------------------------   29/08/22

Name of Chairperson   Signature   Date

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ABSTARCT

Instability and a reduction in control performance are mostly caused by faults and failures in components of the system. In order to increase the control system's tolerance against faults and failures, fault-tolerant control (FTC) techniques have been developed recently. This thesis investigates an active FTC for rigid spacecraft attitude control subject to transducer faults, parameter uncertainty and external disturbance. The designed fault detection observer is capable of detecting the transducer fault without false alarms brought on by outside disturbances, model uncertainties, and the designed fault estimation is to estimate the overall effects of the fault and system states. Introduces an auxiliary variable that uses an indirect fault identification approach in order to estimate accurately and reduce the effect of faults or failures. Once the fault has been identified with a certain level a backstepping controller with fault tolerance, reconstruction efficiency is adapted with nonlinear virtual control input in order to effectively compensate the detected actuator and sensor faults , It has a high degree of accuracy for stabilizing the attitude angle and angular velocity even with actuator saturation restrictions and fault estimation errors. As a result, the proposed FTC method's closed-loop system of a faulty rigid spacecraft attitude control is proved by using the Lyapunov theory. Numerical simulation is used to show that the suggested AFTCS is effective at detection, identification of faults and controller redesign for managing actuator and sensor defects in attitude systems control. Comparing the backstepping controller with the suggested fault-tolerant controller reduces the settling times of the attitude roll, pitch and yaw angle by 67.6%, 5.9% and 95.9 %, respectively, and reduces the settling times of the angular velocity x, y and z-axis by 58.6%, 45% and 104.3 %, respectively, and in terms of rising time the suggested fault-tolerant controller smaller rising time. So, it is confirmed that the performance of the system and effectiveness greatly improved by the proposed FTCS.

Key words*: Transducer, Uncertainty, Fault detection and identification, Fault-tolerant control, Saturation*

Table of Contents

LIST OF FIGURES

LIST OF TABLE

ACRONYMS

ACS	Attitude Control System
AFTCS	Active Fault Tolerant System
BC	Backstepping Controller
EKF	Extended Kalman Filter
EKF	Extended Kalman Filter
FDD	Fault Diagnosis and Detection
FDI	Fault Detection and Identification
FTC	Fault Tolerant Control
FTCS	Fault Tolerant Control System
IMM	Interacting Multiple Model
KF	Kalman Filter
LMI	Linear Matrix Inequality
MRP	Modified Rodriguez Parameters
PCA	Principal Component Analysis
PLS	Partial Least Square
RW	Reaction wheels
SMC	Sliding Mode Control
SMO	Sliding Mode Observers
VCS	Virtual Control Signal

CHAPTER ONE

INTRODUCTION

A machine or vehicle built to go through space is called a spacecraft. For earth observation, space colonization, transportation of people and freight, planetary exploration, telecommunication, meteorology, military uses and navigation, all over the world countries are growing more reliant on space technologies. Technology of the present and quality control methods are commonly used throughout the design and production of spacecraft on earth to prevent and eliminate probable faults. However System failures or unusual behavior are unavoidable for spacecraft after extended operation in the harsh space environment with its fluctuating temperatures and high radiation. The spacecraft attitude control system has 32% of the faults, according to a statistical analysis of 156 on-orbit spacecraft problems from 1980 to 2005, with the gyroscope having the highest failure rate. As a result, researching sensor and actuator failure diagnosis methods are essential [1].

The spacecraft attitude control system places an increasing emphasis on accuracy and reliability. However, when there are actuator faults, moment-of-inertia uncertainty, reaction wheel friction, and space-based environmental disturbances, and even actuator saturation and sensor faults, the dynamics of the rigid spacecraft are very nonlinear. The performance of the attitude control is affected as a result of these uncertainties and disturbances, especially when there are actuator and sensor faults.

It is well known that ageing and the hostile space environment can cause spacecraft components to malfunction. The attitude control performance of the spacecraft system may deteriorate if the issue cannot be handled online and instantly. This would result in the whole scheduled task failing. Even a little fault can result in severe performance degradation or mission failure. To maintain overall closed-loop stability and control performance and a type of control system that can automatically accommodate component defects is called AFTCS[2]–[4].

One of the key aspect that must be considered in attitude control design is fault tolerance ability. The attitude controller should be strong enough to handle these faults, uncertainties and disturbances at the same time. A controller must be capable of accepting potential faults and maintaining desired system performance in order to increase system reliability and safety. Fault-tolerant control (FTC) techniques are a strategy to guarantee the reliable operation of systems in the presence of unwanted faults.

Generally one of the biggest challenges in developing spacecraft attitude control systems is consistently achieving accurate and reliable attitude stabilization. It is crucial to Design a stable attitude control system for a spacecraft with high control accuracy because it must be fault tolerant to unknown faults and insensitive to external disturbances [5]–[7]. The reliability of spacecraft attitude control systems can be increased reliably by using fault diagnostic and fault tolerant control (FTC) technology, as is well known [8]–[10].

Several nonlinear control schemes, including backstepping control [11]–[13], SMC [14], [15], PD controller [16], [17], have been presented for ACS design in the absence of actuator defects.

1.1 Types of faults

An unavoidable flaw in a system's design or parameter progressively lowers its closed-loop performance or even causes the system to stop functioning (failure). faults can be categorized based on their location as plant, sensor and actuator[18].

As shown in Figure 1, faults can be classified into three classes depending on their location.

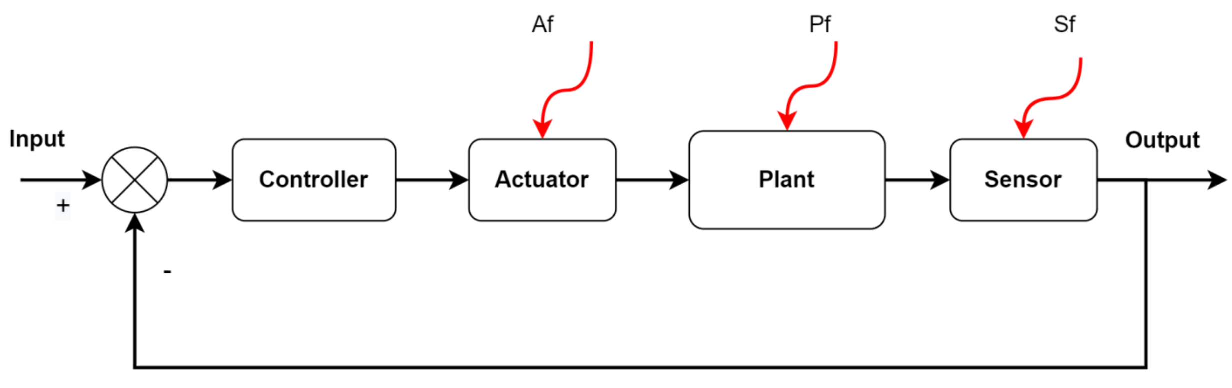

Figure 1. 1 Fault classification based on location

- **Sensor Faults (Sf)**: Plant characteristics are unaffected; however, there are significant inaccuracies in the sensor readings.

- **Plant Faults (Pf)**: The system's dynamic input-output properties are influenced by plant faults.
- **Actuator Faults (Af)**: Plant attributes remain unaffected, but the controller's impact on the plant is disrupted or changed.

Actuator and sensor faults can be grouped into the following categories according to how they are modeled and added to the system: (1) additive faults, and (2) multiplicative faults, based on how the system is designed and introduced. Reaction wheels used for attitude control are frequently related with additive faults. In addition to age, temperature, and other factors affecting friction between the stator and the rotor may cause generated response torque to be higher or lower than expected. Due to stuck control surfaces, malfunctioning actuators may also continually produce reaction torque (reaction wheel) or forces (thrusters). Multiplicative faults are those that are modeled as a sudden change in the nominal control action. Another type of multiplicative fault is actuator deterioration and the sensor faults are loss of accuracy and bias.

According to their time characteristics faults are also categorized as follows:

- **Abrupt faults**: Damage to the hardware immediately results in abrupt faults. This causes the actuator and sensor to fail completely, and such issues continue until the defective item is fixed or changed.
- **Incipient faults**: (drift-like, caused by the motor's wear and tear) represent gradual changes in the actuator's and sensors performance, frequently brought on by aging. In attitude control system issues is requiring prompt reconfiguration of aged actuators, tolerating early errors is critical.
- **Transient faults**: These are the result of an actuator and sensor temporarily going wrong. The term "intermittent faults" refers to temporary faults that occur repeatedly.

1.2 Fault-tolerant control

By reducing the risk of failures and faults, we can increase the reliability and safety of control systems by designing FTC approaches. An FTC system [18]–[21] is a control system that can detect and remedy fault impacts in parts of the system while maintaining the stability of the system and overall performance. Fault-tolerant control systems that rely on fault data are classified as active or passive FTC. A FTC system that does not depend on inaccurate data to control the system is called a passive FTC.

Over the past three decades, major research in FTCS increased need for safety, dependability, and durability, and survivability in industrial processes and aviation system [3]. Passive and active FTC approaches can be used to categorize the current design methodologies for FTC. The controller can tolerate only a small number of failures in the passive FTC. No online fault data is necessary for this method. As a fixed controller, the passive FTC can be quickly installed. The anticipated flaws can be comprised for. Passive FTC, on the other hand, has the following characteristics and a very limited fault tolerance capability:

- Extremely robust to anticipated or premeditated defects.
- Use redundant hardware, such as several actuators and sensors, etc.
- More conservatism is displayed by the controller.
- Only at the expense of reduced nominal performance is it possible to provide robustness to some faults.

It is important to note that the most controllable systems may only have a minimal amount of physical redundancy. Due to cost or practical limitations, it is impossible to change or expand the hardware configuration. These might restrict the use of passive FTC to manage certain kinds of systems. Fortunately, leveraging the use of physical and analytical system redundancy, and the resources available to handle unexpected faults, another sort of FTC approach, namely active FTC, might be devised for these instances. A general schematic diagram for AFTC is shown in Figure 1.2

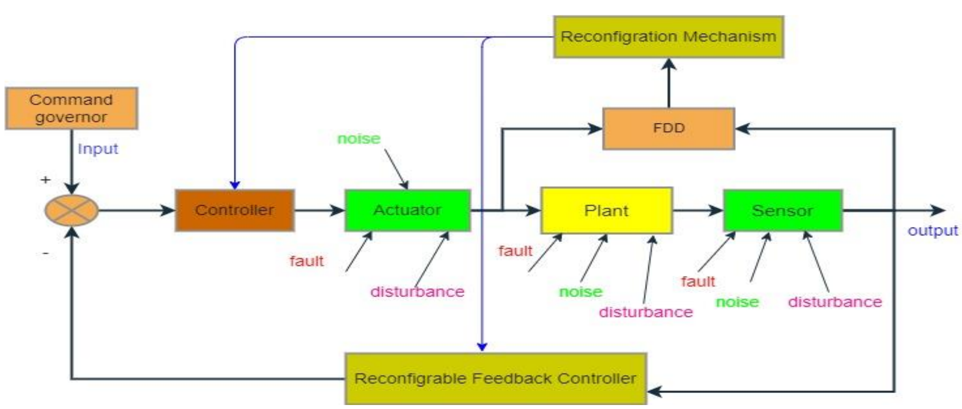

Figure 1. 2 *Block diagram of active FTC system*

Instead of reacting passively to fault events, active FTC uses a reconfiguration mechanism to maintain stability and acceptable performance. To offer the most recent information about the actual changes that the system status has caused and to online modification of the control law, both require a Fault Detection and Identification (FDI) [22] algorithm. In order to prevent misunderstanding, this study uses the term FDI for fault detection and isolation. FDD will be used when FDI also includes the fault identification function. Active FTC systems tolerate a gradual reduction in total system performance in the face of problems and entail a large degree of on-line FDD [23], real-time decision making, and controller reconfiguration. The following characteristics are generally present in Active FTC systems:

- Make full use of analytical redundancy.
- Utilize a reconfigurable controller and the FDD algorithm.
- In the event of a fault, accept reduced performance.
- Minimize conservatism.

It should be noted that a passive FTC design simply includes a fixed FTC architecture, whereas a reconfigurable feedback controller, a controller reconfiguration mechanism, and FDD are all components of AFTC architecture. FDD and a FTC or control law with a controller reconfiguring method ought to be used to construct active FTC architectures. For passive FTC architectures, the only thing that needs to be created is a FTC. The most frequent task to be completed in active and passive FTC systems is fault-tolerant controller design.

1.3 Statement of problem

Creating a spacecraft attitude controller that can sustain performance and stability in the face of actuator failures, sensor failures, external disturbances, uncertainties and control input saturation is one of the difficult issues in the field of aerospace engineering. From the difficult issues, sensor failure and uncertainty are degreed the performance of attitude control of spacecraft. If actuators, sensors, or other system components fail, a traditional feedback control design for a complex system may produce an inadequate performance, or even instability. In order to compensate for sensor failure and uncertainty failures while sustaining desired stability and performance characteristics, new techniques to control system design have been developed to address these limitations. These kinds of control systems are frequently referred to as FTCS.

1.4 Objectives

1.4.1 General objectives

- To designs an active fault-tolerant control system for spacecraft attitude control for actuator faults, sensor faults and model uncertainties.

1.4.2 Specific objective

- To design FDD to detect faults and estimate the total fault effects.
- To design AFTC by using backstepping controller with a nonlinear virtual control to compensate for the faults in the system.
- To maintain closed-loop stability of a faulty rigid spacecraft attitude control by using Lyapunov theory.
- Numerical simulation carried out with the aid of MATLAB software was used to show the performance and efficacy of the proposed FTCS design.

1.5 Scope of the thesis

In this thesis, an AFTCS for spacecraft attitude movement vulnerable to external disturbances, uncertainty, sensor (star tracker and gyroscope) and actuator faults (reaction wheel) are illustrated. FDD and fault-tolerant controllers are presented in this work. The FDD system incorporates fault identification and fault detection are used to acquire the actuator and sensor fault characteristics, such as the occurrence of the fault, the moment when the problem comes, and the degree of the defect. The FTC is made to handle actuator and sensor defects while maintaining control performance based on the fault information from the FDD system. Finally, the proposed system is simulated through MATLAB software.

1.6 Limitation of the thesis

In this thesis some types of sensor, actuator in attitude control of spacecraft, normal controller design are not considered. Attitude control problem for rigid spacecraft can be classified in to the stabilizing problem and the tracking problem, so in this thesis the tracking problem is not addressed.

1.7 Significance of the thesis

The core element of FTCS described in this thesis is that by continuing to deliver acceptable performance in the face of actuator and sensor failures, disturbance and uncertainties. Active fault-tolerant control approaches for attitude control of spacecraft have the following advantages it increases spacecraft lifespan, improve its autonomy and reliability, guarantee the control performance of spacecraft and compensate the occurred faults.

1.8 Methodology

This thesis Describes the dynamic and kinematics modelling of the spacecraft attitude control system for actuator /sensor faults, external disturbance, model uncertainty and actuator saturations with mathematical equations. After modeling this thesis has been implemented by the following procedures

- The system develops fault detection and estimation
- The system develops fault tolerance for the occurrence of faults.
- closed-loop system of a faulty rigid spacecraft attitude control is proved by using the Lyapunov theory
- Numerical simulation carried out with the aid of MATLAB software

1.9 Organization of Thesis

The remainder of this thesis is arranged in the following manner.

This thesis outlines in the following five chapters.

Chapter one: Presents introduction of the research work, types of faults, Fault-tolerant control definition, and fault analysis of spacecraft, statement of the problems, objectives, scope and significance of the research.

Chapter two: Focused on introducing FDD and FTC for spacecraft attitude control with the review of the FDD method (Model -based FDD and Data based FDD) and spacecraft attitude FTC with their methodologies. These methods are adaptive control, SMC and Control allocation based attitude FTC and lastly, the research gap and the main contribution of this paper.

Chapter three: Describes the dynamic and kinematics modelling of the spacecraft attitude control system for actuator /sensor faults, external disturbance, model uncertainty and actuator saturations with mathematical equations. After modelling, the system develops fault detection and estimation, and fault tolerance for the occurrence of faults. It also describes assumptions that are used in Fault tolerant control system (FTCS) design.

Chapter four: Presents the controller design for the proposed active fault tolerant control system by using a backstepping controller.

Chapter five: Presents results and discussion for rigid spacecraft with numerical simulation for the proposed controller with graphs.

Chapter six: conclusion, recommendations and future works are presented in this chapter.

CHAPTER TWO

LITERATURE REVIEW

2.1 INTRODUCTION

The FDD typically has three elements when developed for a ACS of spacecraft or other industrial systems [22]. Defect detection is the initial step in the process, which is determines when a problem has occurred in the system. As a result, a fault isolation method there must be created to identify the issue's nature, location, and affected component. Following that, a fault's Identification or diagnosis is required to determine the value and degree of the detected fault. Numerous studies have been carried out to create FDD techniques for the ACS of spacecraft [8]. Fig. 2 depicts a more detailed types of the current FDD techniques for attitude control of spacecraft. Model-based FDD and Data-based FDD are two subcategories of those methods.

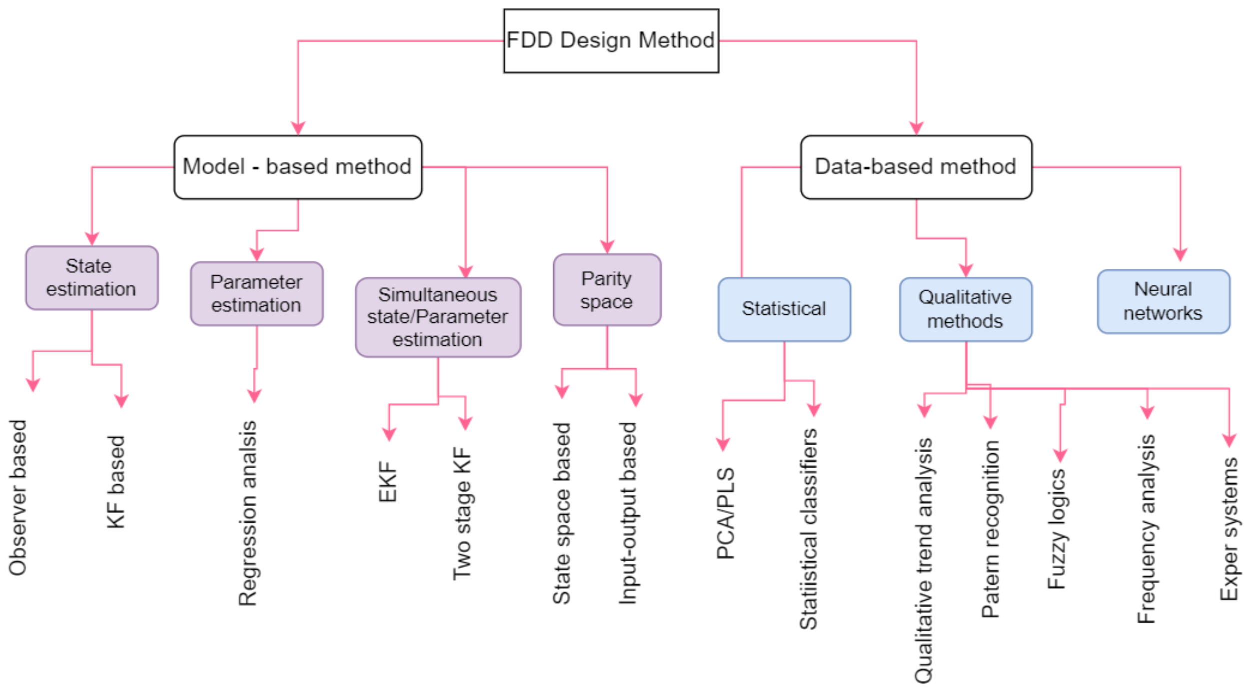

Figure 2. 1 Types of FDD systems for attitude control of spacecraft[23].

2.2 Review of the spacecraft FDD

2.2.1 Model-based FDD Methodologies

Model-based FDD techniques have been widely established and used in spacecraft research over the past few decades because the most control strategies are model-based. The issue of reliable fault isolation and detection for failures influencing a satellite system's thrusters called Mars Express was dealt with in [8]. Errors in attitude measurement, uncertainty, and outside disturbance were taken into account. By using H∞/H2 filters, two model-based FDD techniques were created in [24]. The issue of failure diagnostics for microscope thrusters was addressed in the occurrences of measurement lags, sensor mismucht, measurement noises, and even outside disturbances. For microsatellites to identify and rebuild sensor defects, a state-space-based FDD was suggested in [25].

Due to the complexity of equipment and systems expanding along with the development of new spacecraft, defects are becoming more and more common. A model-based FDD technique with simultaneously estimate of states and parameters is developing into another popular strategy for spacecraft attitude control FDD and FTC design. To correctly estimate reaction flywheel failures, a two-stage Kalman filtering approach was mentioned in [26]. A multiplicative factor was used to model the faults. An FDD based on the Extended Kalman Filter (EKF) for Interacting Multiple Model (IMM) for a class of nonlinear systems was introduced [27]. When sensors and actuators fail in spacecraft, this method has been effectively used. An IMM-based FDD was established in [28] to detect actuator faults under various faulty conditions caused by temperature changes and anomalies, and the loss of torque and current effectiveness in each of the reaction wheels connected to the satellite's three axes. A new EKF-based approach for FDD of gyro faults and attitude sensor problems on spacecraft was reported in [29].

Along with the previously mentioned model-based FDD techniques, using model-based observer for FDD designs and attitude system estimation for spacecraft states have received significant attention from the academic aerospace sector. A conceptual framework for adapting Adaptable flying control was investigated in the work of [30] with the implementation of an observer based FDI. However, the created FDI is only capable of estimating consistent damage to control effectors and is unable to accurately recreate faults. Using the approximation power of a neural network, Talebi introduced a nonlinear observer based on actuator dynamics to accomplish FDD of reaction wheel defects [31]. An recurrent neural network-based observer was researched in [32] to find faults in spacecraft actuators.

By using sliding mode control [33], [34] or adaptive control theory, an observer can be developed robustly in the presence of parametric uncertainty. These observers are frequently referred to as sliding mode observers (SMO) and adaptive observers, respectively [35]–[37]. The fundamental benefit of adopting adaptive observer or SMO is that they are impervious to parid unknown inputs, parameter fluctuations, and outside disturbances. For spacecraft FDD, numerous adaptive observers and SMO techniques have been put forth as of late [37]. For the satellite attitude control system, a reliable actuator failure diagnosis technique was examined in [38]. Gyro drifts, system uncertainties, and disturbances were taken into account. By creating an adaptive observer with uncertain input, the goal of FDD online was accomplished. An SMO algorithm was created in [39] to produce residual signals for a satellite that uses four reaction wheels to operate its attitude control subsystem arranged in a tetrahedron shape. After that, a fault-finding adaptive observer for actuators was developed.

A FDD based on SMO approach was put forth in [40] as a residual generator to find the abnormalities and defects using reaction wheel dynamics. In [41], the design of SMO for the Mars Express satellite's thruster and gyro detection and diagnosis of faults was investigated. Another recent study [41] established an adaptive observer and SMO to handle the satellite attitude control system's sensor failure detection and diagnostics. With no reconstruction error, the sensor defects were properly estimated.

2.2.2 Data Driven-based FDD Strategies

The current model-based FDD techniques are capable of identifying and diagnosing spacecraft component problems, but they heavily rely on the physics model. High resolution models of spacecraft and its components might not be available due to the increasing complexity of spacecraft and its equipment. It would be impossible for the model-based FDD approaches to succeed. Therefore, FDD design without a model is a crucial problem that requires attention. Model-free FDD design applications have been dominated by data-driven based FDD methodologies as a response to this difficulty. Comprehensive summaries of data driven-based FDD are provided in survey studies by Yin et al [42]–[44].

There have recently been research published in the literature on data-driven based FDD for spacecraft ACS's. SVM classifier was used to suggest a data-driven FDD strategy in [45]. A one-versus-one SVM classifier was then displayed to identify sensor or actuator failures of the satellite ACS after the PCA approach was initially employed to create residual for classification. In [46], a different data-driven FDD with a dynamic neural network was developed. Identifying whether a reaction wheel bus voltage fault, current loss fault, or temperature fault had occurred was the main goal, as well as determining which actuator was at fault, was accomplished. To improve fault diagnosis in multivariable systems, a regression-based fault reconstruction technique was created in a related study [46].

The shortcomings of model-based FDD can be rectified by the aforementioned data-driven-based FDD. However, a significant amount of online and offline data may be required for the deployment of data-driven FDD [47]. Due to the on-board computer's restricted data handling capacity, this would not be feasible for an on-orbit spacecraft. Additionally, for data-driven based FDD, the issue of FDD design with poor data quality needs to be further addressed. Actually, noise is present in all measurements of every spacecraft component, which can result in data that is unreliable. By designing FDD with faulty data, a lesser level of diagnostic efficacy will be attained.

2.3 Review of fault-tolerant spacecraft attitude control

The common task in active FTC and passive FTC technique is fault resistant controller architecture. Another important effort in the construction of the attitude FTC system for spacecraft is the law of attitude fault tolerant control [48], [49]. Therefore, the section will examine the most recent work on the FTC design component of spacecraft attitude active and passive FTC [50] system.

FTC regulations should allow for the tolerating of spacecraft component faults while guaranteeing satisfactory performance. The development of the attitude FTC has undergone a number of significant advancements, including, which have been inspired by contemporary control theory. The issue of automatic attitude stabilization for flexible and rigid spacecraft, for instance, was taken into consideration in [49]. On feedback linearization control, the method was founded. A novel FTC law was created in [51] to accommodate control moment gyros errors in order to achieve agile attitude control.

The ability of the controllers to manage uncertainty in the system and outside disturbances must be ensured when constructing fault-tolerant attitude controllers for spacecraft. Otherwise, it will result in poor control performance. Since those two issues have been addressed, attitude FTC design has received increasing attention. The following analysis focuses on the three strategies that are most frequently used to build attitude FTC law: adaptive control, SMC, and control allocation.

2.3.1 Designing Attitude FTC with Adaptive Control

Modern control theory's adaptive control technique [52] has been extensively used to manage unpredictable system parameters. Its capacity to estimate such parameters online is what causes this. Taking advantage of this benefit, numerous spacecraft investigations have used adaptive control to construct attitude FTC law. As [30] describes the design of an adaptive controller for a flight control system. A tethered satellite was given a proposed adaptive attitude FTC stabilization control law. According to the research in [53], it is possible to use adaptive control to successfully execute an attitude-tracking movement for a rigid spacecraft with failed thrusters. Input constraint, uncertain inertia factors, and even outside disturbance were specifically addressed, and were of great relevance to this work.

For flexible spacecraft, the author of [54] developed an FTC technique as to adhere to required attitude. With regard to the recommended scheme's online estimation of the outer boundaries on disturbances and model parameters, different types of reaction wheels defects were successfully corrected by using an adaptive approach. An adaptive attitude controller was created in a recent study [55] by fusing adaptive control and fuzzy control. Two adaptive update laws were proposed to estimating ambiguous parameters and uncertainty and disturbances in the system, and actuator defects were discussed. An adaptive passive FTC controller was created in [56] for the spacecraft formation flying controls that is vulnerable to actuator failure. The use of an FDD technique was not necessary for the implementation of this controller. An adaptive FTC law was created in order to execute attitude tracking maneuvers in order to account for disturbances and actuator faults. The works in [57], [58] can be further cited for more current developments of Designing FTC laws using adaptive control methods.

2.3.2 Sliding mode-based FTC techniques

In recent years, SMC [59] has come to be recognized as a powerful method for regulating ambiguous systems having strongly nonlinear dynamics with coupling. Numerous SMC algorithms were created across the literature and used to construct the FTC for spacecraft attitude because they have a number of benefits, including quick response times and lack of sensitivity to uncertain parameters and outside disturbances [60], [61]. For spacecraft attitude stability issues with an actuator outage fault, passive and active sliding mode control rules have been proposed [62]. When arriving at satellite formations in flight in [81], the terminal SMC was employed, and actuator defects were brought into consideration in the simulation. Although the stability study was not done when there was degrade and a stuck fault. By taking into account the issue of a rigid spacecraft rapidly reorienting itself under dynamic uncertainty and external disturbances, the result from that work was further developed in [63]. In [64], a FTC for attitude based on SMC was introduced. Even with a slow-varying, uncertain satellite mass distribution and many rotating solar flap failure situations, the attitude remained stable. Even though a terminal SMC strategy for performing the rest-to-rest movement with finite-time convergent was developed in [65], only the degradation of actuation efficacy fault was taken into consideration. Air bearing test beds were used to evaluate the controller's performance on the ground. A smaller satellite's attitude FTC law was created in a more recent work [66] using non-singular terminal SMC.

2.3.3 Control Allocation-based Attitude FTC

Redundant components, such as redundant actuators and sensors, are continually designed into the attitude control system of spacecraft to increase reliability. When an actuator develops a problem, the other the ability of active actuators to combine to create a desired amount of control power, enabling the spacecraft to execute the specified movements. The method of control allocation to provide the necessary control attempt is not original because of actuator redundancy [67]. However, static allocation utilizing the pseudo-inverse value of the actuator distribution matrix is the most widely used allocation method [68]. To fully utilize the remaining control power, however, this strategy is ineffective. For overactuated control systems as a result, various efficient dynamic control allocation strategies were suggested [69], [70].

Numerous control allocation strategies have been developed to accommodate actuator faults in conjunction with real-world engineering. By resolving a linear programming issue, a safe attitude controller was put forward by [71].A cost function for the algorithm was established as a quantity relating to the fuel usage for a maneuver. Additionally, attitude maneuvers performed in normal and single-point of failure conditions were used to demonstrate this allocation-based attitude control. For a unique spacecraft: Mars entry vehicles, In [50], a nonlinear fault-tolerant adaptive controller was developed to address the issue of tracking control of the trajectory entrance interface to parachute release. A hybrid linear dynamic control allocation approach for creating the best possible control effort needed to follow the intended trajectory was supplied by the proposed nonlinear control law. By using redundant actuators to account for the decrease in efficiency and time-varying defects, a challenging issue of spacecraft FTC design was researched in [72].

It was suggested to use an unique observer-based fault identification technique with an online control assignment methodology. In [73], a FTC allocation technique for attitude tracking systems for overactuated spacecraft was developed. Faults in the actuator were taken into account. Without modifying the controller, distribute the virtual control signals to the remaining actuators, a control allocation strategy with a trade-off between robustness and performance were developed. Faults in the actuator were taken into account. To transfer the virtual control signals to the rest actuators without changing the controller, Control allocation with a performance/robustness trade-off approach was created. Even when the closed-loop system was vulnerable to disturbances and potential defects, it was assured to be uniformly bounded in the end. A robust control allocation strategy was created in a different study [74] when an actuator fails or is misaligned, for rigid spacecraft attitude stabilization.

2.4 Summary of literature review

The majority of past work has been focused on

- AFTCS design for ACS for spacecraft with actuator faults and external disturbances.
- FTC design for spacecraft attitude control based on BC applying a linear virtual control signals
- To estimate actuator faults for each individual rather than total faults

The main things to consider in this paper are actuator faults, sensor fault, parameter uncertainty, external disturbances by using active FTCS design for spacecraft attitude control. An observer for fault detection is first supplied to identify the fault quickly and clearly specifies the threshold value for the detection residual with the intention of potential actuator and sensor defects. Angular velocity and the total number of actuator and sensor defects were correlated, and then an auxiliary parameter is introduced. The overall fault effects affecting performance of attitude control are calculated using an indirect fault identification approach. Finally, a BC applying a nonlinear VCI is provided to compensate for actuator and sensor failures and guarantee closed-loop stability by applying the Lyapunov theory.

When compared to earlier studies about the FTC system of the rigid spacecraft, the main contributions of this study are stated as follows:

- Rather than accounting for each fault separately, estimate the overall effect of the multiplicative and additive sensor faults. This reduces the complexity of the structure, calculations, and design process compared to the result in [75], improving its applicability in real-world applications, exceptionally where computational power and onboard memory are constrained.
- Comparatively to the traditional BC [12] and [11] using a linear virtual control input, the proposed active FTC controller is based on BC with a nonlinear VCS,and is capable of maintaining that the attitude angles and angular velocities asymptotically converge to zero even with actuator and sensor faults.
- This thesis also takes into account the external disturbance, and model uncertainty while designing fault-tolerant controllers.

CHAPTER THREE

MATHEMATICAL MODEL OF SPACECRAFT AND OVERVIEW OF FAULT TOLERANT CONTROL

3.1 INTRODUCTION

Any spacecraft attitude control system's mathematical model is comprises its attitude dynamics and kinematics. Any on-orbital spacecraft's attitude control system is built around it. In engineering, the Euler angles of attitude are typically used to characterize spacecraft attitude. The theoretical analysis uses other techniques, including the attitude rotation matrix, the MRPs, and the unit quaternion. These attitude description techniques each have unique characteristics [86]. The performance of the system and the design of the attitude controller are also significantly impacted by the attitude dynamics. External disturbances and uncertain inertia must be included in order to produce realistic attitude dynamics.

Establishing an attitude model requires careful consideration of the actuator's uncertainty. The control performance will significantly deteriorate if it is neglected. Therefore, analysis of the actuator uncertainty mechanism and its impact on control accuracy is urgent.

3.2 Coordinate frames

Figure 3.1 shows this frame's inertial f_i (X_I, Y_I, Z_I), the reference frame for orbit f_O (X_O, Y_O, Z_O), and the body-fixed frame f_b (X_B, Y_B, Z_B) For the rigid spacecraft attitude control, coordinate systems are used. The picture f_i, It originates in the earth's natural center and is utilized to calculate the spacecraft's orbital location. The spacecraft's center of mass serves as the origin of the frame f_O, which rotates about the Y_O axis with respect to f_i. The roll axis X_O is chosen to be in the direction of flight, the pitch axis Y_O is chosen to be perpendicular to the orbital plane, and the yaw axis Z_O is chosen to be pointing in the direction of the Earth. The axes of the frame f_b, whose origin is the same as tha t of f_O, are parallel to the main Inertia axis.

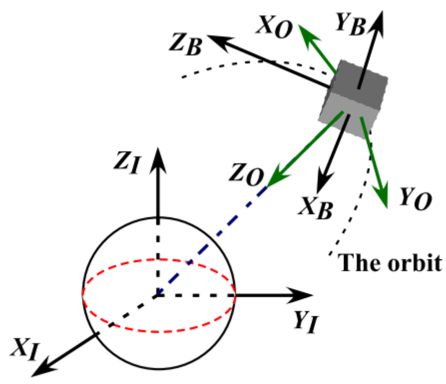

Figure 3. 1 Description of the coordinate reference frames for the ACS for spacecraft[76].

3.3 Attitude kinematics

In this portion, the attitude control systems of the spacecraft in a circular orbit are described using Euler's moment equations. A rigid spacecraft's attitude kinematic equation by Euler angles is described by [77]

$$\begin{bmatrix}\omega_x\\ \omega_y\\ \omega_z\end{bmatrix} = \begin{bmatrix}1 & 0 & -sin\theta\\ 0 & cos\phi & sin\phi cos\theta\\ 0 & -sin\phi & cos\phi cos\theta\end{bmatrix}\begin{bmatrix}\dot{\phi}\\ \dot{\theta}\\ \dot{\psi}\end{bmatrix} - \omega_0\begin{bmatrix}sin\phi sin\theta sin\psi + cos\phi cos\psi\\ cos\phi sin\theta sin\psi - sin\phi cos\psi\end{bmatrix} \tag{3.1}$$

where ω_x, ω_y, and ω_z stand for the three halves of the inertial angular velocity vector ω respectively, ϕ , θ and ψ denoted by roll, pitch, and yaw attitude angles. $\omega_0 = \sqrt{\frac{\mu_e}{r}}$ In this equation, ω_0 is spacecraft orbital rate, r is the distance from the Earth's center to the satellite's center of mass, and μ_e is the Earth's gravitational constant.

When the small Euler angle is taken into account, the equation (3.1) is approximated as follows

$$\begin{bmatrix}\omega_x\\ \omega_y\\ \omega_z\end{bmatrix} = \begin{bmatrix}\dot{\phi} - \omega_0\psi\\ \dot{\theta} - \omega_0\\ \dot{\psi} + \omega_0\phi\end{bmatrix} \tag{3.2}$$

3.4 Spacecraft Attitude Dynamics

Mathematically, the kinematic and kinetic equations describe the rotational equations of rigid spacecraft. The orientation of a spacecraft is described in this section using unit-quaternion format since it is suitable for onboard real-time computation and reflects all spacecraft orientations worldwide [78]. The parameter uncertainty in mass inertia is also caused by the significant torque imbalance effects brought on by the main engine's deployment and the bias modeling residual error [17]. [89] provides the dynamic equation of a rigid spacecraft with respect to the uncertainty of the inertia matrix expressed in terms of unit-quaternions.

$$(J + \Delta J)\dot{\omega}_b = -S(\omega_b)(J+\Delta J)\omega_b + Du + d \tag{3.3}$$

$$\dot{Q} = \frac{1}{2}\begin{bmatrix} S(q) + q_0 I_3 \\ -q^T \end{bmatrix} \omega_b \tag{3.4}$$

Where $J \in R^{3x3}$ represents the symmetric inertia matrix in positive-definite, △J is the parameter uncertainty, the vector $\omega_b \in R^3$ is the body's in its fixed frame, angular velocity $Q = [q_1\ q_2\ q_3\ q_0]^T = [q^T\ q_0]^T \in R^4$ is the unit-quaternion indicating how he body-fixed frame B is oriented in relation to the inertial frame N, $u \in R^n$ is used to indicate the control torque generated by n actuators around the body axis, D∈ R^{3xn} is the matrix for the actuator arrangement, $d \in R^3$ is the environmental disturbances. Observes that the unit-vector quaternion's and scalar parts, q and q_0, respectively, satisfy $q^T\ q + q_0^2 = 1$, and For any vector $x \in R^3$, the matrix $S(x) \in R^{3x3}$ denotes a skew-symmetric matrix. $S(\omega_b)$ is a skew−symmetric matrix that takes the shape of:

$$S(\omega_b) = \begin{bmatrix} 0 & -\omega_3 & \omega_2 \\ \omega_3 & 0 & -\omega_1 \\ \omega_2 & \omega_1 & 0 \end{bmatrix}$$

3.5 Actuator Fault Models

Control torques can be produced for attitude manoeuver or stabilizing missions using systems for propulsion, magnetic torques, momentum exchange mechanisms, also solar torque. Thruster systems, reaction wheels (RW), and control moment gyros are the three primary sources of torques used to control spacecraft attitude. The presence of RW faults will consequently result in decreased performance and potential destabilization of the closed-loop system because it is the most efficient actuator for precise pointing. Therefore, examining the fault method and the mathematical model of RWs are the primary topics of discussion in this part.

Reaction wheels are frequently used as spacecraft attitude control actuators. For instance, the reaction wheel, which is a flywheel coupled to a motor, may experience problems owing to poor lubrication, ageing, minor failures, and increased friction in the electronics, drive motor, bearing, and power supply. The RW is a type of precision equipment, according to [80], [81], which is particularly vulnerable to the abnormal elements. There are primarily four types of faults:

- **Decreased reaction torque**: Some circumstances, such as an increase in the a bearing problem, friction torque between the stator and rotor, or a decrease in driving current, have an impact on how quickly the RW rotor speed changes. As a result, the ratio of the real torque to the command control torque decreases.
- **Increased bias torque**: The RW must maintain constant wheel speed and not generate any torque while the disturbance is ignored and the attitude controller's command torque is zero. As a result of a change in bearing temperature or insufficient bearing lubrication, the RW will still endure a deviation moment.
- **Continuous generation of reaction torque**: Using the command control torque as a base, an additional torque is generated when the RW body accelerates or decelerates as a result of a voltage bus failure or an intermittent motor current fault.
- **Failure to respond to control signals**: Due to a problem with the drive circuit, the motor, or the electricity, the RW can be unable to respond to control commands. After that, the wheels rotational speed gradually declines or nearly stays the same, producing zero torque.

These errors could have a multiplicative or additive impact on the actuator output. The reaction wheel may respond more slowly, perform less effectively, or possibly completely fail if one of these faults develops.

Reaction wheels are subject to the two main failure types listed below for the spacecraft attitude system:

- **Loss of partial effectiveness**: The wheel speed change rate may be smaller than the nominal value because of increased stator-rotor friction, poor bearing lubrication and minor failures, and reduced motor torque. The output torque of the reaction wheel generated as a result can be lower than the controller's commanded torque.

- **Bias fault**: The wheel might not be able to hold its speed due to changes in coulomb friction and viscous friction of the bearings induced by ageing, time-varying temperature, and lubrication, which could cause the wheel to gradually accelerate or decelerate. As a result, even though the commanded attitude control toque is zero, a low bias torque is generated.

Let $u_c \in R^n$ represent the command control torque as a vector. Modelling shows that there is a relationship between the torque applied to the spacecraft under command and the actual torque. The block diagram proposed active FTC system for actuator faults is shown in Figure 3.2.

$$u = (I_n - E)u_c + u_a \tag{3.5}$$

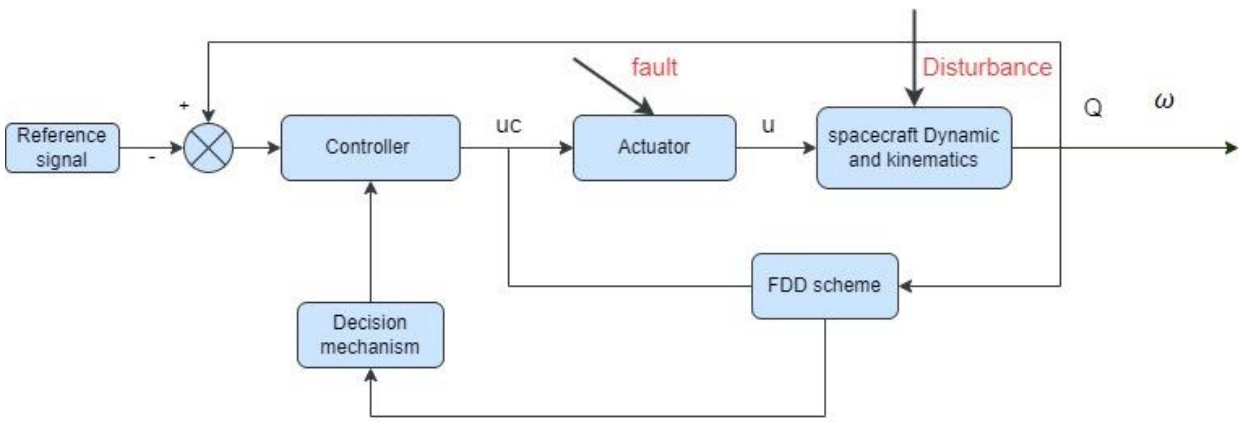

Figure 3. 2: *Block diagram for actuator FTC*

where the diagonal components of the matrix $E = diag\{e_1, e_2, \dots, e_n\} \in R^{n}_{xn}$ reflects of the actuator loss of effectiveness and satisfies $0 \le e_i \le 1, i \in \{1,2,\dots,n\}$. Aware of that $e_i = 0$ shows that the actuator is healthy, $e_i = 1$ imply that the actuator has completely failed. Aware of that $0 < e_i < 1$ suggests that the actuator is partially losing its effectivcy. The additive bias fault is represented by the variable u. The attitude dynamics taking actuator faults into consideration is expressed as follows when the fault model equation (3.5) is substituted into equation (3.3):

$$(J + \Delta J)\dot{\omega}_b = -S(\omega_b)(J + \Delta J)\omega_b + Du_c + f + d \tag{3.6}$$

Where $f = -DEu_c + Du_a$ indicates the total impact of the fault on the system.

The FTCS design makes use of the following assumptions in this thesis are from [82].

Assumption 1: The condition $\underline{J} \leq \|J\| \leq \bar{J}$, where $\underline{J}$ and $\bar{J}$ are two positive constants, is satisfied by the inertia matrix J. The Euclidean norm and its induced norm are denoted by the letter $\|.\|$.

Assumption 2: The effects of gravity, magnetic forces, aerodynamic stiffness, and the pressure of solar radiation on the environment are actually limited. As a result, $\bar{d}$ must be positive in order for $\|d\| \leq \bar{d}$. to exist.

Assumption 3: It is a given that the overall fault effects represented by f satisfy $\|\dot{f}\| \leq \phi$ with ϕ being a constant.

Remark 1: In this study, as stated in Assumption 3, designers take into consideration the slow-varying fault, also known as the incipient fault, which is one of the frequent faults seen in actuators. For instance, a reaction wheel's bias torque may gradually grow due to faults induced on by operation wear and tear, lubrication, ageing, temperature, or lubrication [92].It should be emphasized that the size of the additive fault u_a is restricted, at the very least, by the physical capabilities of the actuators, and cannot be arbitrarily huge.

3.6 Sensor fault model

The many sensor types utilized in attitude determination methods are displayed in this section. Gyroscope, sun sensor, earth sensor, star tracker, and magnetometer are among the several types of sensors. One of the most critical parts of any spacecraft is the attitude determination system. Its purpose is to determine how the spacecraft is oriented in relation to a reference system. During mission modes, star tracker and rate gyroscope measurements are almost always used to estimate the satellite attitude. However, there are also collection modes that make use of magnetometers and sun sensors. Theoretically, attitude and angular rate determination could only be done using one of these two sensors (star tracker and gyroscope). This paper considers star tracker and gyroscope sensors. The block diagram proposed active FTC system for sensor faults is shown in Figure 3.2.

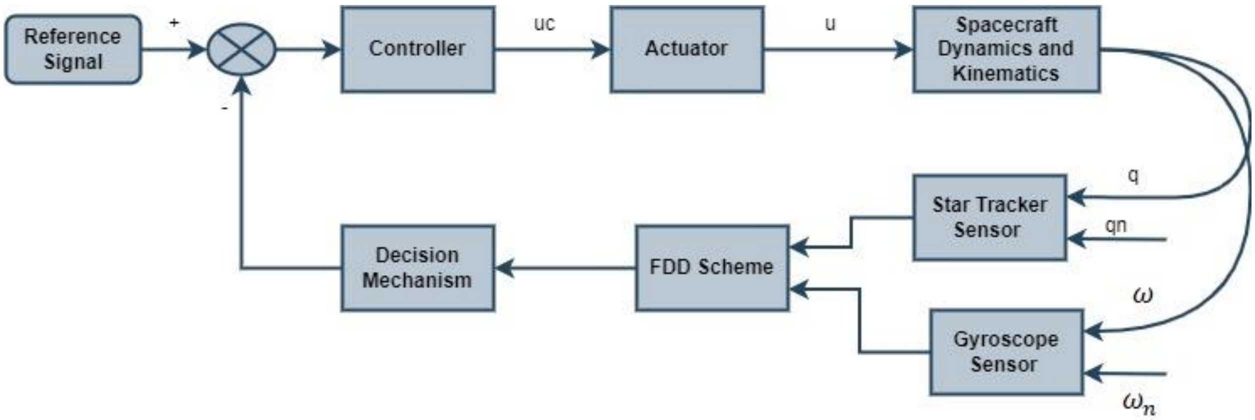

Figure 3. 3 *Block diagram for actuator FTC*

3.6.1 Gyroscope Sensor Mode

Gyroscopes are sensors that measure angular rates in relation to an inertial frame of reference. The model for gyroscope sensor measurement is represented by [83]

$$\begin{bmatrix}\omega_{xm}\\ \omega_{ym}\\ \omega_{zm}\end{bmatrix} = f_G\begin{bmatrix}\omega_{x}\\ \omega_{y}\\ \omega_{z}\end{bmatrix} + \omega_n + \mathrm{b} \tag{3.7}$$

Where f_G is installation matrix, ω_{xm}, ω_{ym} and ω_{zm} are measured value of angular velocity, ω_x, ω_y and ω_z are actual value of angular velocity, ω_n is angular velocity affected by a white noise and b is sensor bias.

3.6.2 Star Tracker sensor model

The most popular type of attitude sensor is a star tracker. A star tracker is an optical tool that uses photocells or a camera to take measurements of the satellite's orientation in relation to distant (fixed) stars. A star tracker's output is an estimated Euler's angle or quaternion, which relates the body's orientation to the inertial frame. This have added random measurement noise to the star tracker, and the model can be represented as follows [83]

$$q_m = q_n \otimes q_s \otimes q \tag{3.8}$$

where q_m is the measured quaternion representing the satellite attitude, q_s is the quaternion transformation from the body to sensor frame, and q_n is the quaternion measurement noise .

3.7 Fault Detection and Identification

In this section, fault identification and fault detection methods are developed in order to obtain fault information with a sufficient level of accuracy.

3.7.1 Fault Detection

The fault detection observer is developed as follows, based on the attitude dynamics in equation (3.3):

$$(J + \Delta J)\dot{\omega}_{b,d} = -S(\omega_b, d)(J + \Delta J)\omega_{b,d} + Du_c + \Lambda\big(\omega_b - \widehat{\omega}_{b,d}\big) \tag{3.9}$$

Where $\Lambda \epsilon R^{3x3}$ is a matrix with a positive gain, and $\widehat{\omega}_{b,d}$ is the angular velocity estimation ω_b for fault detection. Using . $\widetilde{\omega}_{b,d} = \omega_b - \widehat{\omega}_{b,d}$ as the velocity estimation error, it is determined that

$$(J + \Delta J)\dot{\widetilde{\omega}}_{b,d} = -S(\omega_b)(J + \Delta J)\omega_b + S(\widehat{\omega}_b, d)(J + \Delta J)\widehat{\omega}_{b,d} - \Lambda\widetilde{\omega}_{b,d} + f + d \tag{3.10}$$

Assumption 4: Equation (3.10)'s the nonlinear term $-S(\omega_b)(J + \Delta J)\omega_b + S(\widehat{\omega}_b, d)(J + \Delta J)\widehat{\omega}_{b,d}$ is assumed to be known and to meet the Lipschitz requirement with regard to $\widetilde{\omega}_{b,d}$, that is,

$$\left\| -S(\omega_b)(J + \Delta J)\omega_b + S(\widehat{\omega}_b, d)(J + \Delta J)\widehat{\omega}_{b,d} \right\| \leq \ell_g \left\| \widetilde{\omega}_{b,d} \right\| \tag{3.11}$$

where `ℓ_g is the Lipschitz parameter.

Remark 2: The real angular velocity must be continuous and within a certain range for spacecraft that are useful and may perform a variety of scientific objectives. The attitude controller, for instance, attitude controllers in [84], [85], is intended to guarantee attitude stability and angular velocity constraint. As a result, the estimates of the bound angular velocity must also be bound. Since the Lipschitz requirement in Assumption 4 is rational, it is possible to determine the Lipschitz constant beforehand using the limits of angular velocity and spacecraft inertia. [86] makes a similar assumption for the Lipschitz condition as it relates to attitude control.

The residual for fault detection is then generated using the estimation error $\widetilde{\omega}_{b,d}$. The following Theorem states the decision-making process for detecting actuator and sensor faults in [82].

Theorem 1: When the estimation error $\widetilde{\omega}_b$ in equation (3.10) exceeds the $\xi_{dt} = \frac{\bar{d}}{\lambda_{min}[\Lambda]-\ell_g}$ defined threshold, the existence of an actuator fault is determined. In other words, the actuator defect is detected if condition $\|\widetilde{\omega}_{b,d}\| > \xi_{dt}$ is met.

Proof. Take into account a candidate for Lyapunov in the form of

$$V = \frac{1}{2}\widetilde{\omega}_{b,d}^T J \widetilde{\omega}_{b,d} \tag{3.12}$$

Equation (3.12) coupled with (3.10)'s time derivative is given as

$$\begin{aligned} \dot{V} = \widetilde{\omega}_{b,d}^T &\left(-S(\omega_b)(J+\Delta J)\omega_b + S(\widehat{\omega}_b, d)(J+\Delta J)\widehat{\omega}_{b,d} - \Lambda\widetilde{\omega}_{b,d} + f + d\right) \\ &\leq \left(\lambda_{min}[\Lambda] - \ell_g\right)\|\widetilde{\omega}_{b,d}\|^2 + \bar{d}\|\widetilde{\omega}_{b,d}\| \\ &+ \widetilde{\omega}_{b,d}^T f \end{aligned} \tag{3.13}$$

The matrix's smallest eigenvalue is indicated by the letter $\mathrm{d}_{min}[.]$. The total fault effect is $f = 0$. If the system doesn't have an actuator defect, meaning that the defect has no influence on the system.

The previous inequity is transformed into this situation.

$$\dot{V} \leq -\left(\lambda_{min}[\Lambda] - \ell_g\right)\|\widetilde{\omega}_{b,d}\|^2 + \bar{d}\|\widetilde{\omega}_{b,d}\| \tag{3.14}$$

It is obvious that $\dot{V} < 0$. Assuming the fault detection observer's initial condition is set to meet $\|\widetilde{\omega}_{b,d}\| > \frac{\bar{d}}{\lambda_{min}[\Lambda]-\ell_g}$, it is obtained that $\widehat{\omega}_b, d(0) = \omega_b(0)$ continuously, proving that the error $\|\widetilde{\omega}_{b,d}\| \leq \frac{\bar{d}}{\lambda_{min}[\Lambda]-\ell_g}$ is, under a fault-free assumption, upper bounded by a constant.When a fault, $\|f\| \neq 0$,, occurs, it is seen that

$$\dot{V} \leq -\left(\lambda_{min}[\Lambda] - \ell_g\right)\|\widetilde{\omega}_{b,d}\|^2 + \left(\bar{d} + \|f\|\right)\|\widetilde{\omega}_{b,d}\| \tag{3.15}$$

Similar analysis is conducted as in the fault-free situation, and it results in

$$\left\|\widetilde{\omega}_{b,d}\right\| \leq \frac{\bar{d} + \|f\|}{\lambda_{min}[\Lambda] - \ell_g} \tag{3.16}$$

It suggests that in a fault-free case, $\left\|\widetilde{\omega}_{b,d}\right\|$ may be greater than its maximum value. As a result, $\xi_{dt} = \frac{\bar{d}}{\lambda_{min}[\Lambda] - \ell_g}$ can be chosen as the fault detection threshold. If $\left\|\widetilde{\omega}_{b,d}\right\| > \xi_{dt}$ is seen, the fault is detected. The proof has become complete.

Remark 3: A necessary requirement for fault detection is provided by Theorem 1. There must be a defect in the system if the requirement in Theorem 1 is valid. The opposite of this statement, however, could not always be true. In particular, despite the possibility of actuator malfunctions in attitudinal movement, inequality $\left\|\widetilde{\omega}_{b,d}\right\| > \xi_{dt}$ might not hold.

3.7.2 Fault Estimation

This thesis must first identify the fault before determining its size and time characteristics. In order to simplify the estimation algorithm and reduce the amount of memory and onboard processing power needed, to estimates the overall fault consequences in this paper instead of the individual fault itself. An indirect fault detection approach is suggested to locate the total actuator defects f in equation (3.6). First, adding an auxiliary variable [87]

$$\psi = f - G(J + \Delta J)\omega_b \tag{3.17}$$

where $G \in R^{3x3} > 0$ is a matrix of gain. ψ 's time derivative gives us

$$\dot{\psi} = \dot{f} - G(-S(\omega_b)(J + \Delta J)\omega_b + Du_c + G(J + \Delta J)\omega_b + \psi + d) \tag{3.18}$$

The indirect fault estimator is stated as follows, assuming $\hat{\psi}$ and $\hat{f}$ are the estimates of ψ and f, respectively:

$$(J + \Delta J)\,\dot{\widehat{\omega}}_{b,i} = S(\widehat{\omega}_{b,i})(J + \Delta J)\widehat{\omega}_{b,i} + Du_c + \hat{f} + L(\omega_b - \widehat{\omega}_{b,i}) \tag{3.19}$$

$$\dot{\psi} = -G\hat{\psi} - G(-S(\hat{\omega}_{b,i})(J + \Delta J)\hat{\omega}_{b,i} + Du_c + G(J + \Delta J)\hat{\omega}_{b,i} \tag{3.20}$$

$$\hat{f} = \hat{\psi} + G(J + \Delta J)\hat{\omega}_{b,i} \tag{3.21}$$

Where $L \epsilon R^{3x3}$ is a gain matrix with a positive definite and $\hat{\omega}_{b,i}$ is the fault identification angular velocity estimation. In order to make ensuring the estimations fall to the proper values, the term containing $\omega_b - \hat{\omega}_{b,i}$ in equation (3.19) serves as a feedback input. Define $\tilde{\omega}_{b,i} = \omega_b - \hat{\omega}_{b,i}$,$\tilde{\psi} = \psi - \hat{\psi}$ and $\tilde{f} = f - \hat{f}$, accordingly. The error mechanism is developed as

$$(J + \Delta J)\dot{\tilde{\omega}}_{b,i} = -S(\omega_b)(J + \Delta J)\omega_b + S(\hat{\omega}_{b,i})(J + \Delta J)\hat{\omega}_{b,i} + \hat{f} + d - L\tilde{\omega}_{b,i} \tag{3.22}$$

$$\dot{\tilde{\psi}} = \dot{f} - G\tilde{\psi} + G\big(S(\omega_b)(J + \Delta J)\omega_b - S(\hat{\omega}_{b,i})(J + \Delta J)\hat{\omega}_{b,i}\big) - G^2(J + \Delta J)\tilde{\omega}_{b,i} \tag{3.23}$$

$$\tilde{f} = \tilde{\psi} + G(J + \Delta J)\tilde{\omega}_{b,i} \tag{3.24}$$

The outcome for identifying faults is defined in the preceding based on the previous error systems.

Theorem 2: If a matrix L and a constant $\mu > 0$ both exist as such, given the attitude dynamics in (3.6) with Assumptions 1-4, then

$$\begin{bmatrix} L - G(J + \Delta J) - \left(\ell_g + \frac{\varepsilon}{2} + \frac{\varepsilon}{2}\mu {\ell_g}^2\right) I_3 & * \\ \frac{1}{2}(\mu(J^T + \Delta J^T)(G^2)^T - I_3) & \mu G - \frac{\mu}{\varepsilon} G G^T \end{bmatrix} > 0$$

The indirect fault identification method suggested in equation (3.19)–(3.21) ensures that the overall fault estimation error $\tilde{f}$ exponentially converges to an invariant set containing the origin for a given matrix $G > 0$ and constant $\varepsilon > 0$.

Proof. The result of premultiplying equation (3.22) by $\tilde{\omega}_{b,i}^T$ is

$$\tilde{\omega}_{b,i}^{T}\,(J+\Delta J)\tilde{\omega}_{b,i}^{T} \tag{3.25}$$

$$\leq \tilde{\omega}_{b,i}^{T}\,(J+\Delta J)\tilde{\omega}_{b,i}^{T}\left(L-\mathrm{G}(J+\Delta J)-\left(\ell_g+\frac{\varepsilon}{2}\right)I_3\right)\hat{\omega}_{b,i}$$

$$+\tilde{\omega}_{b,i}^{T}\hat{\psi}+\frac{1}{2\varepsilon}\bar{d}^2$$

Where $\tilde{\omega}_{b,i}^{T}d \leq \frac{\varepsilon}{2}\tilde{\omega}_{b,i}^{T}\hat{\omega}_{b,i}+\frac{1}{2\varepsilon}\bar{d}^2$ is used as an inequality.

Additionally, in accordance with equation (3.23) and Assumption 4, this have

$$\tilde{\psi}^{T}\;\dot{\tilde{\psi}} \leq \tilde{\psi}^{T}\left(\mathrm{G}-\frac{1}{\varepsilon}\,GG^{T}\right)\tilde{\psi}-\tilde{\psi}^{T}\,G^{2}\,(J+\Delta J)\hat{\omega}_{b,i} \tag{3.26}$$

$$+\frac{\varepsilon}{2}{\ell_g}^{2}\tilde{\omega}_{b,i}^{T}\hat{\omega}_{b,i}+\frac{\varepsilon}{2}\phi^{2}\frac{\varepsilon}{2}\bar{d}^{2}$$

Where the following inequalities

$$\tilde{\psi}^{T}\dot{f} \leq \frac{1}{2\varepsilon}\tilde{\psi}^{T}\bar{\psi}+\frac{\varepsilon}{2}\phi^{2} \tag{3.27}$$

$$-\tilde{\psi}^{T}Gd \leq \frac{1}{2\varepsilon}\tilde{\psi}^{T}GG^{T}+\frac{\varepsilon}{2}\bar{d}^{2} \tag{3.28}$$

$$\tilde{\psi}^{T}G\big(S(\omega_b)(J+\Delta J)\omega_b-S(\hat{\omega}_{b,i})(J+\Delta J)\hat{\omega}_b\big) \tag{3.29}$$

$$\leq \frac{1}{2\varepsilon}\tilde{\psi}^{T}GG^{T}\tilde{\psi}+\frac{\varepsilon}{2}{\ell_g}^{2}\tilde{\omega}_{b,i}^{T}\hat{\omega}_{b,i}$$

are utilized. They now create a candidate for Lyapunov as

$$V=\frac{1}{2\varepsilon}\tilde{\omega}_{b,i}^{T}(J+\Delta J)\hat{\omega}_{b,i}+\frac{\mu}{2}\tilde{\psi}^{T}\tilde{\psi} \tag{3.30}$$

where $\mu > 0$ is an architectural constant. Along with (3.25) and (3.26), the time derivative of V is provided as

$$\dot{V} \leq -\widetilde{\omega}_{b,i}^T \left(L - G(J + \Delta J) - \left(\ell_g + \frac{\varepsilon}{2} + \frac{\varepsilon}{2}\mu {\ell_g}^2\right) I_3\right) \widehat{\omega}_{b,i} \tag{3.31}$$

$$- \mu\widetilde{\psi}^T \left(G - \frac{1}{\varepsilon} G G^T\right) \widetilde{\psi}$$

$$- \widetilde{\omega}_{b,i}^T (\mu(J^T + \Delta J)(G^2)^T - I_3)\widetilde{\psi} + \frac{\varepsilon}{2}\mu\phi^2 + \frac{\varepsilon}{2}\mu\bar{d}^2$$

The previous inequality also has the following effects:

$$\dot{V} \leq \left[\widetilde{\omega}_{b,i}^T \widetilde{\psi}^T\right] P \left[\widetilde{\omega}_{b,i}^T \widetilde{\psi}^T\right]^T + \sigma \tag{3.32}$$

Where $P = \begin{bmatrix} L - G(J + \Delta J) - \left(\ell_g + \frac{\varepsilon}{2} + \frac{\varepsilon}{2}\mu {\ell_g}^2\right) I_3 & * \\ \frac{1}{2}(\mu(J^T + \Delta J^T)(G^2)^T - I_3) & \mu G - \frac{\mu}{\varepsilon} G G^T \end{bmatrix}$

and $\sigma = \frac{\varepsilon}{2}\mu\phi^2 + \frac{\varepsilon}{2}\bar{d}^2 + \frac{1}{2\varepsilon}\bar{d}^2$. Should the matrix P be positive definite, i.e., $P > 0$, they will have

$$\dot{V} \leq -\varsigma V + \sigma \tag{3.33}$$

Where $\varsigma = \frac{2\lambda_{min}[Q]}{\max\{(J+\Delta J),\mu\}}$. Equation (3.33) indicates that $\widetilde{\omega}$ and $\widetilde{\psi}$ are uniformly bounded, which is in accordance with the Lyapunov stability theory [88]. Assume that $S_{(\widetilde{\omega},\widetilde{\psi})}$ is an invariant set.

$$S_{(\widetilde{\omega}_{b,i},\widetilde{\psi})} = \left\{ (\widetilde{\omega}_{b,i},\widetilde{\psi}) \middle| \frac{(J + \Delta J)}{2} \left\|\widetilde{\omega}_{b,i}\right\|^2 + \frac{\mu}{2}\left\|\widetilde{\psi}\right\|^2 \leq \sqrt{\frac{\sigma}{\varsigma}} \right\} \tag{3.34}$$

and the additional set that it corresponds to is denoted as $\widetilde{S}_{(\widetilde{\omega}_{b,i},\widetilde{\psi})}$. They can conclude that $\dot{V} \leq 0$ if$(\widetilde{\omega}_{b,i},\widetilde{\psi}) \epsilon\, \widetilde{S}_{(\widetilde{\omega}_{b,i},\widetilde{\psi})}$ by once more using equation (3.33). It follows that the estimating errors $(\widetilde{\omega}_{b,i},\widetilde{\psi})$ converging to the invariant set $S_{(\widetilde{\omega}_{b,i},\widetilde{\psi})}$ exponentially at a rate higher than $e^{-\varsigma t}$. Furthermore, the fault estimation error $\tilde{f}$ likewise exponentially converges to an invariant set in light of equation (3.24), which shows that the fault estimate error $\tilde{f}$ is a mixture of $\widetilde{\omega}_{b,i,}$ and $\tilde{\psi}$. This completes the proof.

Remark 4: Larger estimator gains L and G cause estimation errors $\widetilde{\omega}_{b,i,}$ and $\tilde{\psi}$ to converge more quickly, as shown by equations (3.22)–(3.24). However, the restriction in Theorem 2 prevents G from being infinitely large.

Remark 5: Due to the demands of the mission, fault identification time is restricted in practical aerospace engineering. However, it's possible that the fault estimate won't capable of determine the exact value within the given time frame. As no action is taken to mitigate the effects of the defect during the fault identification phase, a longer fault identification process may result in system instability or serious performance degradation[21]. To address this problem, They develop a decision mechanism in which the fault estimation accuracy is determined by an identification threshold ξ_{it}. If $\|\widetilde{\omega}_{b,i,}(t)\| + \|\hat{f}(t) - \hat{f}(t-T)\| < \xi_{it}$ with sampling time T, to determine that the fault diagnosis process is finished and switch from the normal controller to the fault-tolerant controller. Otherwise, the fault is said to have been incorrectly estimated successfully. Noting that longer identification times are required but with improved fault estimation accuracy the smaller the threshold is chosen. In fault-tolerant control design, the control accuracy is further impacted by the fault estimation accuracy.

3.8 Fault tolerant control

Once faults have been properly estimated, a FTC should be presented to compensate for fault impacts and restore performance utilizing the estimated fault information. Sliding mode control technique is one of the promising and effective design approaches for enhancing the resilience of the fault-tolerant controller (see, for instance, [89], [90]). Additionally, if the commanded control exceeds the maximum torque that the actuator is capable of producing, certain unexpected control actions or actuator damage may result [91], particularly if the ACS has actuator faults. Actuator saturation limitations and fault estimate mistakes are therefore incorporated into the design of the FTC.

Assumption 5: The control torque produced by the actuator, which is presumable to be limited by,

$$|u_{ci}| \leq u_{max}, \qquad (i = 1,2,\dots,n) \tag{3.35}$$

where u_{max} is the average saturation value across all actuators. Let Δf represent the estimating errors, which are defined as $\Delta f = f - \hat{f}$. After successful fault identification, it is acceptable to assume that the estimation errors are restricted [75], i.e., $\|\Delta f\| < \delta$ with a fixed $\delta > 0$. The attitude dynamics in equation (3.6) are reformulated as follows after taking into account "the fault estimation errors and remembering actuator saturation constraint" ($|u_{ci}| \leq u_{max}$) in Assumption 5.

$$(J + \Delta J)\dot{\omega}_b = -S(\omega_b)(J + \Delta J)\omega_b + Du_c^{sat} + \hat{f} + \Delta f + d \tag{3.36}$$

whereu_c^{sat} is the restricted control input that the controller has commanded. Define $D^+ = D^T(DD^T)^{-1}$ as satisfying $DD^+ = I_3$ to be the pseudoinverse of matrix D. There is a finite constant ε_0 such that $\|D^+\| \leq \varepsilon_0$ exists because D is the assembled matrix determined by the design of the spacecraft. The estimation error Δf impacts control performance similarly to external disturbances, as shown in equation (3.36). In order to increase control performance and robustness, we must include it in a FTC.

To develop the FTC, the next Lemma 1 is also utilized.

Lemma 1: If a positive constant β is chosen for $x\epsilon[-1,1]$ in order to satisfy $\beta \geq 1.5574$, this have

$$-\alpha x\, tan^{-1}(\beta x) \leq -\alpha x^2 \tag{3.37}$$

where α is a positive constant.

Proof of lemma 1

To prove Lemma 1, They initially take into account the scenario in which $x\epsilon$ [0,1].. For $x\epsilon$ [0,1], the inequality $-\alpha x arctan(\beta x) \leq -\alpha x^2$ equals $arctan(\beta x) \geq x$. My objective is to demonstrate $f(x) \geq 0$ by defining a function $f(x) = arctan(\beta x) - x$.

By taking $f(x)$'s derivatives with regard to x, we obtain $\frac{df(x)}{dx} = \frac{-\beta^2 x^2 + \beta - 1}{1 + \beta^2 x^2}$.Then, They have $\lim_{x\to 0^+} \frac{df(x)}{dx} = \beta - 1 > 0$ and $\lim_{x\to 1^-} \frac{df(x)}{dx} = \frac{-\beta^2 - 1 + \beta}{1 + \beta^2} \leq \frac{-\beta}{1+\beta^2} < 0$ if $\beta \geq 1.5574$. Following that, for $x\epsilon$ (0,1), The further derivative is the second found as $\frac{d^2 f(x)}{dx^2} = \frac{-2\beta^3 x}{(1+\beta^2 x^2)^2} < 0$, implying that f(x) is absolutely concave. The minimum of $f(x)$ is attained at $f(0)$ or $f(1)$ since $f(x)$ is absolutely concave for $x\epsilon$ (0,1), $\lim_{x\to 0^+}\frac{df(x)}{dx}$, and $\lim_{x\to 1^-}\frac{df(x)}{dx} < 0$. With $\beta\ \geq\ 1.5574$ in consideration, it is simple to validate $f(1) = \arctan(\beta) - 1 \geq 0$. $-\alpha xarctan(\beta x) \leq -\alpha x^2$ is established as the result for $x\epsilon$ [0,1]. The result also holds when $x\epsilon$ [0,1] since $\alpha xarctan(\beta x)$ and αx^2 are even functions.This complete the proof

CHAPTER FOUR

CONTROLLER DESIGN

The controller was added to the control system, the output began to move more rapidly in the direction of the input, and the error value dropped. The spacecraft in orbit cannot function without attitude control. Attitude spacecraft control’s main objective is to accurately and precisely position the spacecraft's body. Using backstepping controllers, fault tolerant attitude control against actuator and sensor faults is examined for the rigid spacecraft attitude dynamics. In this study normal controller design is not taken into consideration. To get the actuator and sensor fault description, having the presence of the fault, the moment which the error occurs has occurred, and the size of the problem, the FDD technique involves fault detection and fault diagnosis. Based on the fault information from the FDD system, the fault-tolerant controller is created to account for actuator and sensor problems and maintains performance of the controller. The general structure of the attitude control system for proposed AFTC is shown in figure 4.1, and the controllers used in this paper are provided as follows.

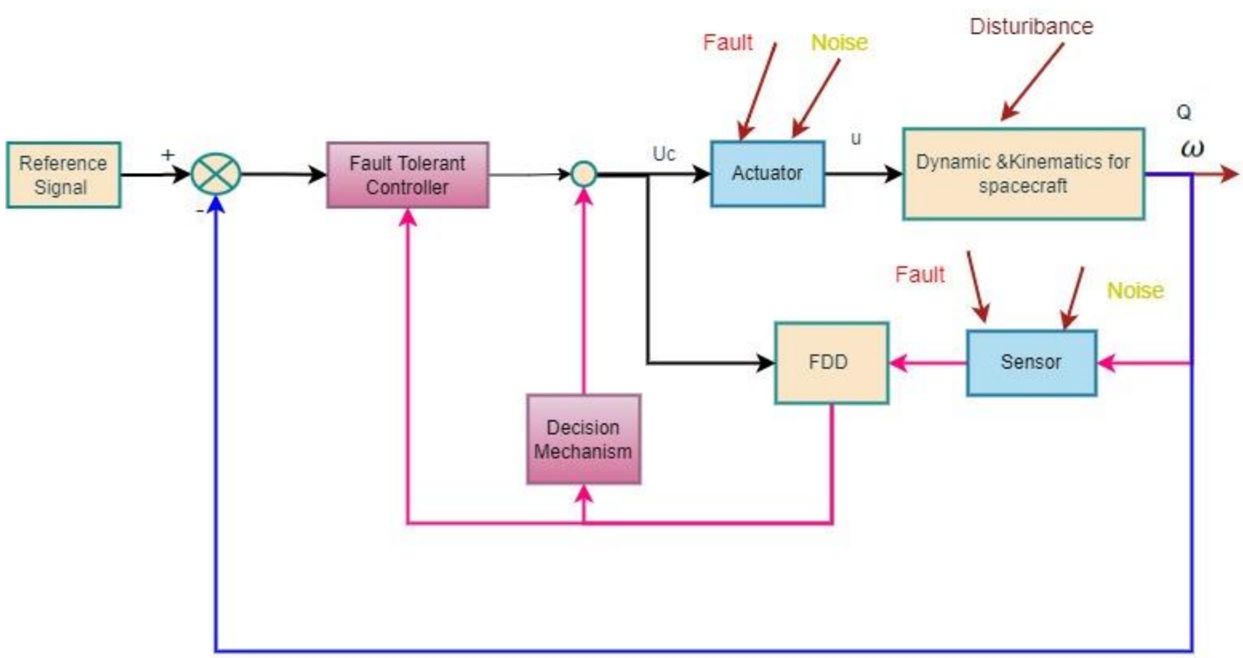

Figure 4. 1 The overall proposed AFTCS for attitude control system structure

4.1 Backstepping controller

Integrator backstepping is a technique used for the fault-tolerant controller design. The angular velocity ω_b is viewed as a virtual control signal input for the kinematics subsystem in equation (3.4), designed as

$$\omega_c = -\alpha tan^{-1}(\beta q) \tag{4.1}$$

Therefore, the error in virtual velocity between ω_b and ω_c is determined as

$$\mathrm{S} = \omega_b - \omega_c = \omega_b + \ atan^{-1}(\beta q) \tag{4.2}$$

The nonlinear virtual control input provided by equation (4.1) eliminates several faults, such as excessive demand for torque at the start of the movement and slow movement when the system's state is almost at the equilibrium [94], in compare to research's of [11], [12] with a linear virtual control input.

selecting a candidate for the Lyapunov function as

$$V_1 = k_0 q^T q + k_0 (1 - q_0)^2 \tag{4.3}$$

The derivative of V_1 is satisfied by applying Lemma 1.

$$\dot{V}_1 \leq k_0 \alpha \|q\|^2 + k_0 \alpha q^T s \tag{4.4}$$

It is obvious that $\dot{V}_1 \leq 0$ if S =0.

After that, the dynamics attitude with regard to the error in virtual velocity S is determined as follows

$$\begin{aligned} (J + \Delta J)\dot{\mathrm{S}} &= (J + \Delta J)\dot{\omega}_b + \alpha\beta(J + \Delta J)(I_3 + \beta^2 \Xi_q)^{-1}\dot{q} \\ &= -k_0 q + F(.) + D u_c^{sat} + \hat{f} \end{aligned} \tag{4.5}$$

where the definition of the nonlinear term $F(.)$ is

$$\begin{aligned} F(.) &= k_0 q - S(\omega_b)(J + \Delta J)\omega_b + \Delta f + d \\ &+ \frac{1}{2}\alpha\beta(J + \Delta J)\left(I_3 + \beta^2 \Xi_q\right)^{-1}(S(q) + q_0 I_3) w_b \end{aligned} \tag{4.6}$$

$\Xi_q = diag(q_1^2, q_2^2, q_3^2)$. According to Assumptions 1-2, we have

$$\|-S(\omega_b)(J+\Delta J)\omega_b\| \le (\bar{J}+\Delta\bar{J})\|\omega_b\|^2 \tag{4.7}$$

$$\|k_0 q+\Delta f+d\| \le k_0+\delta+\bar{d} \tag{4.8}$$

$$\left\|\frac{1}{2}\alpha\beta(J+\Delta J)(I_3+\beta^2\Xi_q)^{-1}(S(q)+q_0 I_3)w_b\right\| \le \frac{1}{2}\alpha\beta(\bar{J}+\Delta\bar{J})\|\omega_b\| \tag{4.9}$$

The three inequalities mentioned above make it evident that

$$\|F(.)\| \le (\bar{J}+\Delta\bar{J})\|\omega_b\|^2+\frac{1}{2}\alpha\beta(\bar{J}+\Delta\bar{J})\|\omega_b\|+k_0+\delta+\bar{d} \le h\Omega \tag{4.10}$$

The variable h is supposed to be unknown in situations where Where $h = max\left\{(\bar{J}+\Delta\bar{J}), \frac{1}{2}\alpha\beta(\bar{J}+\Delta\bar{J}), k_0+\delta+\bar{d}\right\}$, $=\|\omega_b\|^2+\|\omega_b\|^2+1$, and δ may be challenging to obtain. Therefore, h cannot be employed directly in the construction of a FTC.

The following assumption is presented first before they provide the FTC details.

Assumption 6: The following inequality exists

$$\frac{u_{max}}{\varepsilon_0} \ge h\Omega+\|\hat{f}\|+\varrho_0 \tag{4.11}$$

Where ϱ_0 is a tiny parameter.

Remark 6: According to assumption 6, even in the case of serious defects, the functional actuators can provide enough torque to allow the spaceship to make the necessary movement. When the controller design takes the actuation saturation into account, similar assumptions are also observed in [21], [56], and [89].

The fault-tolerant controller can now be provided as of this moment.

$$u_c^{sat} = -\frac{u_{max}}{\varepsilon_0} D^+ sat[\Gamma(.)s] \tag{4.12}$$

$$\Gamma(.) = k+\frac{s^T\hat{f}}{\|s\|^2+\varepsilon_1^2}+\frac{\hat{h}\Omega}{\|s\|+\varepsilon_2} \tag{4.13}$$

When $\varepsilon_2 = \frac{\nu}{\Omega}$ has a little positive constant ν, k and ε_1 are two positive constants, A definition of the function $sat[\Gamma(.)s]$ is

$$sat[\Gamma(.)s] = \begin{cases} \frac{s}{\|s\|}, & if\ \|s\| \geq \frac{u_{max}}{\varepsilon_0 \Gamma(.)} \\ \frac{\varepsilon_0 \Gamma(.)s}{u_{max}}, & if\ \|s\| \leq \frac{u_{max}}{\varepsilon_0 \Gamma(.)} \end{cases} \tag{4.14}$$

The expression for the adaptation law of $\hat{h}$ is

$$\dot{\hat{h}} = -c_1 \hat{h} + \frac{c_2 \Omega \|s\|^2}{\|s\| + \varepsilon_2} \tag{4.15}$$

where c_1 and c_2 are two constants that are positive. Despite being piecewise, the function $sat[\Gamma(.)s]$ described in equation (4.14) is continuous throughout, including at the point $\|s\| = \frac{u_{max}}{\varepsilon_0 \Gamma(.)}$. As a result, equation (4.15)'s suggested FTC is also persistent.

Theorem 3: Considering the dynamics and kinematics of attitude outlined in equations (3.3) and (3.4) when input saturation and actuator faults are accounted. In the event that the FTC created in equations (4.12) to (4.15) is utilized following successful fault identification, The virtual velocity errors and attitude tends to a tiny invariant set encompassing the origin, and the closed-loop system is uniformly eventually bounded stable.

Proof. It is necessary to discuss two situations in the context of equation (4.16) in order to demonstrate the stability of closed-loop with FTC saturated system in equation (4.12) with adaptive law in equation (4.15).

Case I: $\|s\| \geq \frac{u_{max}}{\varepsilon_0 \Gamma(.)}$, In equation (4.12), the controller changes into

$$u_c^{sat} = \frac{u_{max}}{\varepsilon_0} D^+ \frac{s}{\|s\|} \tag{3.16}$$

Because of $\|D^+\| \leq \varepsilon_0$, $\|u_c^{sat}\| \leq u_{max}$ is obvious. In turn, this results in the actuator saturation constraint in equation (3.35) being met. The Lyapunov candidate is being considered as

$$V_2 = V_1 + \frac{1}{2} s^T (J + \Delta J) s \tag{4.17}$$

Then, it follows that

$$\dot{V}_2 \le k_0 \omega_b^T q_e + s^T \left\{ -k_0 q + F(.) - D\left(\frac{u_{max}}{\varepsilon_0} D^+ \frac{s}{\|s\|}\right) + \hat{f} \right\} \tag{4.18}$$

$$\le k_0 \alpha \|q\|^2 - \frac{u_{max}}{\varepsilon_0} \|s\| + h\Omega\|s\| + \|\hat{f}\|\|s\|$$

With respect to Assumption 6's inequality, they have

$$\dot{V}_2 \le k_0 \alpha \|q\|^2 - \varrho_0 \|s\| < 0 \tag{4.19}$$

Case II: $\|s\| \le \frac{u_{max}}{\varepsilon_0 \Gamma(.)}$, In equation (4.13), the controller changes into

$$u_c^{sat} = -\Gamma(.) D^+ s \tag{4.20}$$

Other candidate for Lyapunov is selected as

$$V_2 = V_1 + \frac{1}{2} s^T (J + \Delta J) s + \frac{1}{2c_2} \tilde{h}^2 \tag{4.21}$$

For equation (4.20), the time derivative is

$$\dot{V}_2 \le k_0 \alpha \|q\|^2 + s^T \left(F(.) - \left(ks + \frac{s^T \hat{f} s}{\|s\|^2 + \varepsilon_1^2} + \frac{\hat{h}\Omega s}{\|s\| + \varepsilon_2} \right) + \hat{f} \right) \tag{4.22}$$

$$+ \frac{1}{c_2} \tilde{h}\dot{\hat{h}} \le -k_0 \alpha \|q\|^2 - k\|s\|^2 - \frac{c_1}{2c_2} \tilde{h}^2 + \eta$$

Where $\eta = \frac{\varepsilon_1}{2}\|\hat{f}\| + \nu h + \frac{c_1}{2c_2} h^2 < \infty$ replaces $\|s\|^2 + \varepsilon_1^2 \ge 2\varepsilon_1 \|s\|$ and $\varepsilon_2 = \frac{\nu}{\Omega}$. The previously mentioned inequality means that

$$\dot{V}_2 \le \gamma V_2 + \zeta \tag{4.23}$$

In which $\gamma = \min\{k_0 \alpha, \frac{k}{(J+\Delta J)}, c_1\}$ and $\zeta = \frac{\varepsilon_1}{2}\|\hat{f}\| + \nu h + \frac{c_1}{2c_2} h^2 + k_0 \alpha (1 - q_0^2) < \infty$. Since all internal signals are bounded, the closed-loop system is eventually bounded stable. In addition, it is obvious that $\dot{V}_2 < 0$ if $\|q\| > \sqrt{\frac{\eta}{k_0 \alpha}}$ or $\|s\| > \sqrt{\frac{\eta}{k}}$ according to equation (4.22). Thus, this thesis demonstrate that the error in virtual velocity S and attitude q are completely stable to $S_q = \{q | \|q\| \le \sqrt{\frac{\eta}{k_0 \alpha}}\}$ and $S_s = \{| \|s\| \le \sqrt{\frac{\eta}{k}}\}$,respectively. The proof has been finished.

Remark 7: The parameter's rate of increase $\hat{h}(t)$ (the increasing phase of C if $\hat{h}(0\ )$ has a low starting value is primarily determined by the positive constant c_1, according to the adaptive law in equation (4.15), while the decreasing the parameter's rate $\hat{h}(t)$ (the steady phase of $\hat{h}(t)$ is adjusted by the negative constant of $\hat{h}(t)$. The design parameters c_1 and c_2 should be chosen as tiny constants to ensure that the of $\hat{h}(t)$ varies smoothly.

Remark 8: By modifying the design parameters k_0, α, k, ε_1, and ν, the stability of closed-loop and control performance can be guaranteed. Stability analysis shows that bigger k_0, α, and k may lead to a faster rate of convergence and reduced steady-state errors. The final error sets also indicated that lowering design parameters k_0 and ν in η might further minimize steady-state errors. Although they are also employed to reduce control chattering, ε_1 and ν cannot be too tiny.

CHAPTER FIVE

RESULT AND DISCUSSION

This section demonstrates that the proposed design is capable of detecting and isolating the fault as well as compensating for the effects of faults on the performance of the system by demonstrating how an actuator, sensor, and parameter uncertainty fault in the ACS should degrade the performance of the spacecraft. Numerical simulation is carried out to a rigid spacecraft to show the efficiency and performance of the suggested FTCS design and used MATLAB program to perform. Numerical simulation is carried out by rigid spacecraft system equations (3.2),(3.3), (3.6), (3.7),(3.8) , (4.1), (4.12) to (4.15) and the parameter that are used in this paper are the rigid spacecraft's inertial uncertainty $\triangle J$ and inertia matrix J are chosen as $\Delta J = diag\{\sin(0.3t)\,, 2\cos(0.2t)\,, \sin(0.1t)\}kgm^2$ and $J = \begin{bmatrix} 32.6 & 8.9 & 12.3 \\ 8.9 & 34 & 13.5 \\ 12.3 & 13.5 & 37.2 \end{bmatrix} kgm^2$ [79].

It is assumed that the external disturbance is given by

$d(t) = [-0.005\sin(t)\,; 0.005\sin(t)\,; -0.005\sin(t)] \quad N.m$ [92]. Four reaction wheels are arranged in a pyramid structure to supply controlling torques for attitude in the simulation, and the a matrix of actuator distribution is

$$D = \begin{bmatrix} -1 & -1 & 1 & 1 \\ 1 & -1 & -1 & 1 \\ 1 & 1 & 1 & 1 \end{bmatrix}.$$

The maximum control torque is selected at 0.2 N.m due to the reaction wheel's physical limitations. It is assumed that the spacecraft has its initial angular velocity $\omega_b(0)$ and initial attitude $Q(0)$ are given by

$$\omega_b(0) = \begin{bmatrix} 0.005 \\ 0.006 \\ 0.004 \end{bmatrix} \quad \text{and} \quad Q(0) = \begin{bmatrix} -0.5 \\ 0.3 \\ -0.4 \\ 0.7071 \end{bmatrix}.$$

The sensor parameters are gyro measurement bias is taken to be $1^0/h$ on each axis, they also take into account the mounting The attitude and angular velocity sensor misalignments are $0.005^0$and 0.1^0, accordingly, in light of the constrained manufacturing tolerances. The actuator parameters that are used in this paper are the first and second reaction wheels in the simulation encounter a loss of effectiveness (e_i) fault at $t = 5\ s$ with $e_1 = 0.6$ and $e_2 = 0.2$, whereas the second and third reaction wheels experience an additive bias fault (u_{ai}) at $t = 100\ s$ with $u_{a2} = -0.03$ and $u_{a3} = -0.04$.

The gain matrix of the fault detection observer in equation (3.8) is chosen as follows to detect the actuator fault $\Lambda = 5I_3$. Additionally, $\zeta = 0.002$ is selected for the fault detection threshold. Design parameters $G = 0.5I_3$, $\varepsilon = 5$, and $\ell_g = 3$ are chosen for the defect identification technique in (3.18) to (3.20). In[82], sequentially resolving the inequality in Theorem 2 yields

$$L = \begin{bmatrix} 16.7621 & 0.2 & 0 \\ 1 & 21.2621 & 0.5 \\ 0.5 & 1 & 24.2621 \end{bmatrix} \text{ and } \mu = 0.2333.$$

The control gains of the adaptive FTC in equation (4.13) to (4.16) are selected as $\alpha = 0.2$, $\beta = 1.8$, $k = 100$, $\varepsilon_1 = 0.1$, $\nu = 0.01$, $c_1 = 0.01$, and $c_1 = 0.1$ to tolerate actuator and sensor faults. $\hat{h}(0) = 0.1$ is chosen as the adaptive parameter's initial value. In order to make sure the threshold is appropriate for fault detection, they set the simulation's fault detection threshold as $\zeta_{dt} = 0.002$, Which was determined by trial and error. It have also set a threshold of 0.2 N.m for the normal controller's magnitude in order to satisfy the actuator saturation limitation.

The FTC clearly makes compensates for the impacts of actuator and sensor defects, especially when additive bias faults occur. The integral back-stepping controller with a LVS [11] is used for comparison with proposed controller. With the exception of the linear virtual control input $\omega_c = -\alpha q$ replacing the nonlinear virtual control input $\omega_c = -\alpha tan^{-1}(\beta q)$, the traditional backstepping controller is built with similar framework as the proposed FTC.

The fault detection residual, as shown in Fig. 5.1, is below the threshold ζ before occurring a fault but clearly rises beyond the threshold after actuator and sensor defects at 5s and 100s occur. It has been noted that 6.5s and 100.5s, have additive bias faults and partial loss of efficacy faults are detacted. The identification threshold in this test was set to $\zeta_{it} = 0.002$. Remark 5 states that if condition $\|\widetilde{\omega}_{b,i}(t)\| + \|\hat{f}(t) - \hat{f}(t-T)\| < \zeta_{it}$ is met, they change from the normal controller to the fault-tolerant controller.

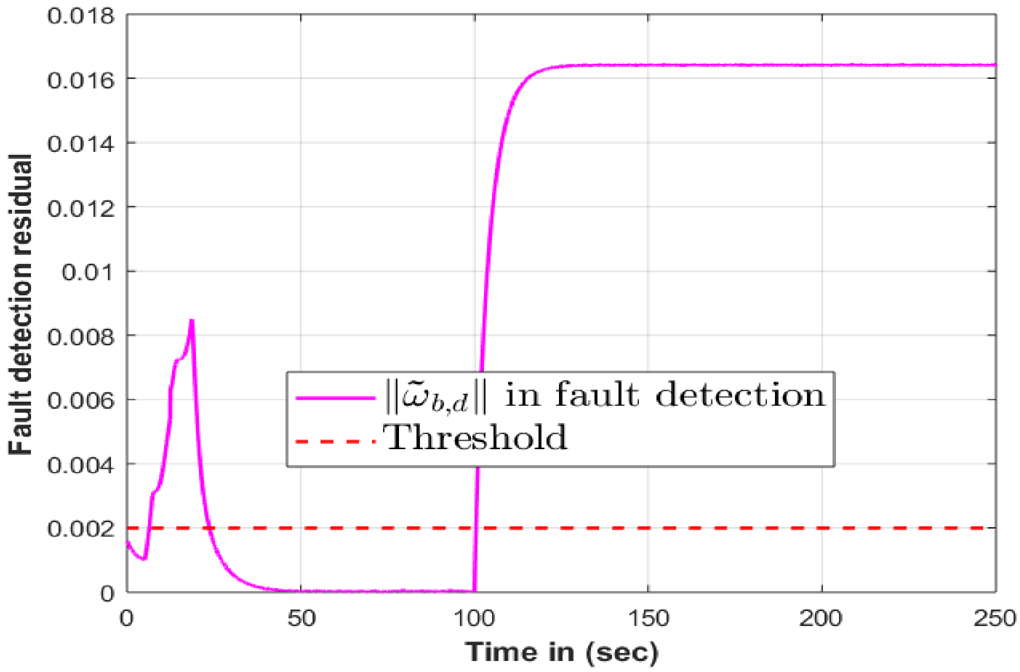

Figure 5. 1: Fault detection time response of the AFTCS under consideration when a fault occurs

The estimation error for the angular velocity and the fault eventually nears to a point close to zero, as shown in Figs. 5.2 and 5.3. Observes that a slight rise in the fault and angular velocity estimate errors after 100s, which is caused by the emergence of additive faults. After 100s, the suggested fault-tolerant controller continues to demand a non-zero control torque to correct for the additive defects, as depicted in Fig. 5.4.

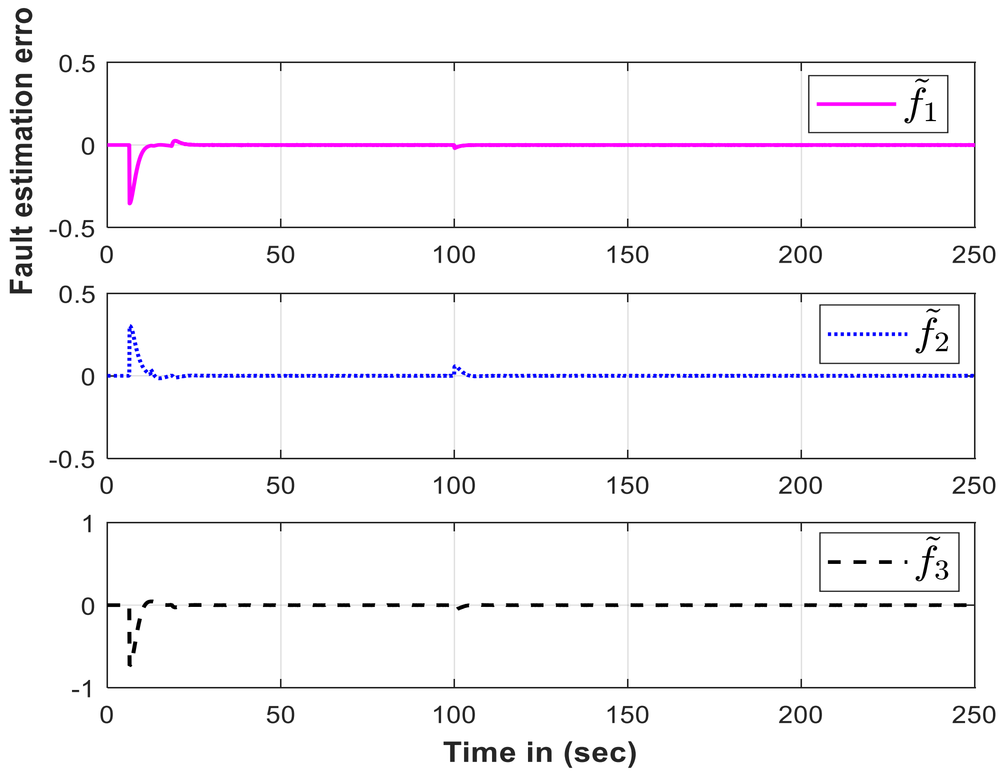

Figure 5. 2 *Fault estimation error time response of the AFTCS under consideration when a fault occurs*

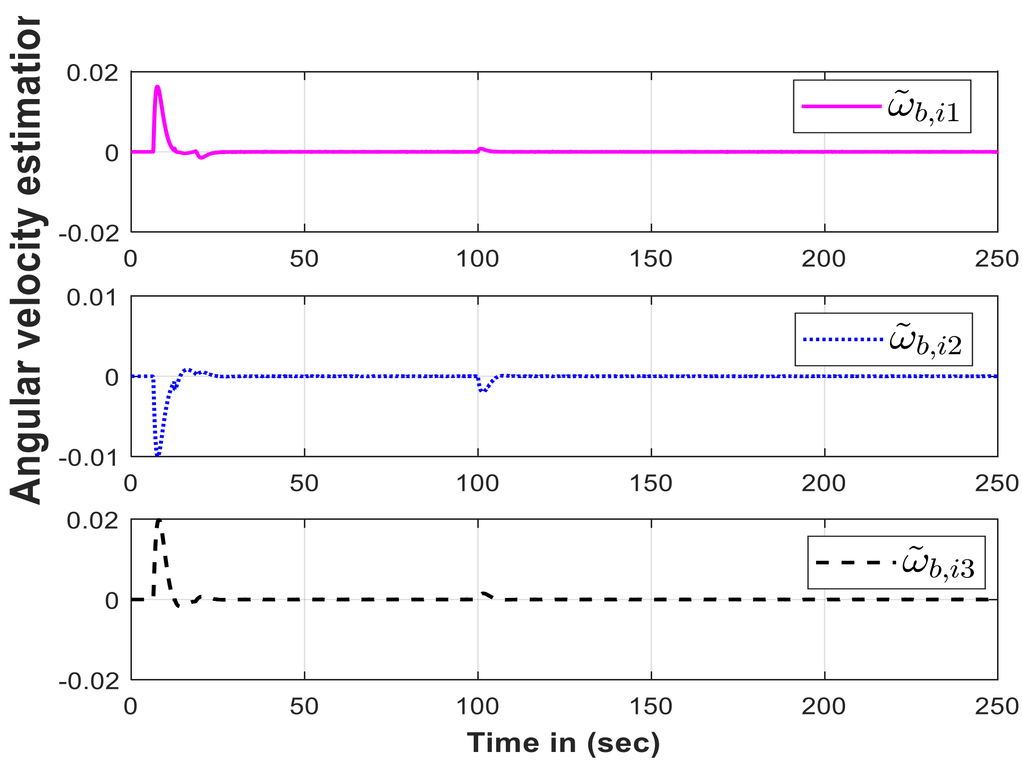

Figure 5. 3 Angular velocity estimation error time response of the AFTCS under consideration when a fault occurs.

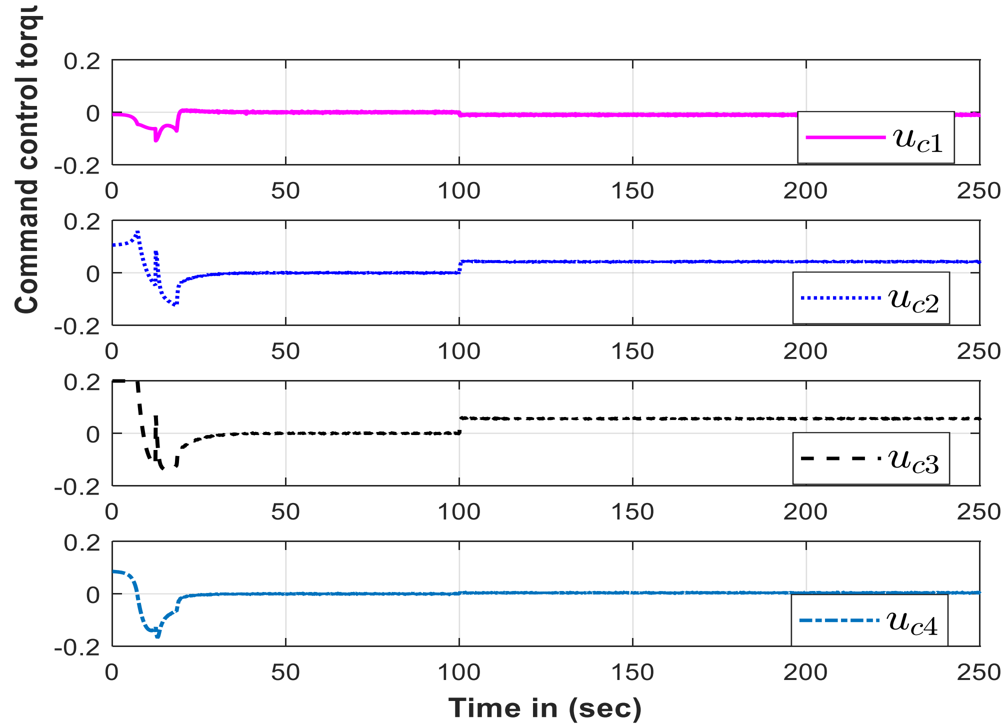

Figure 5. 4 *Commanded control torque time response of the AFTCS under consideration when a fault is occurs.*

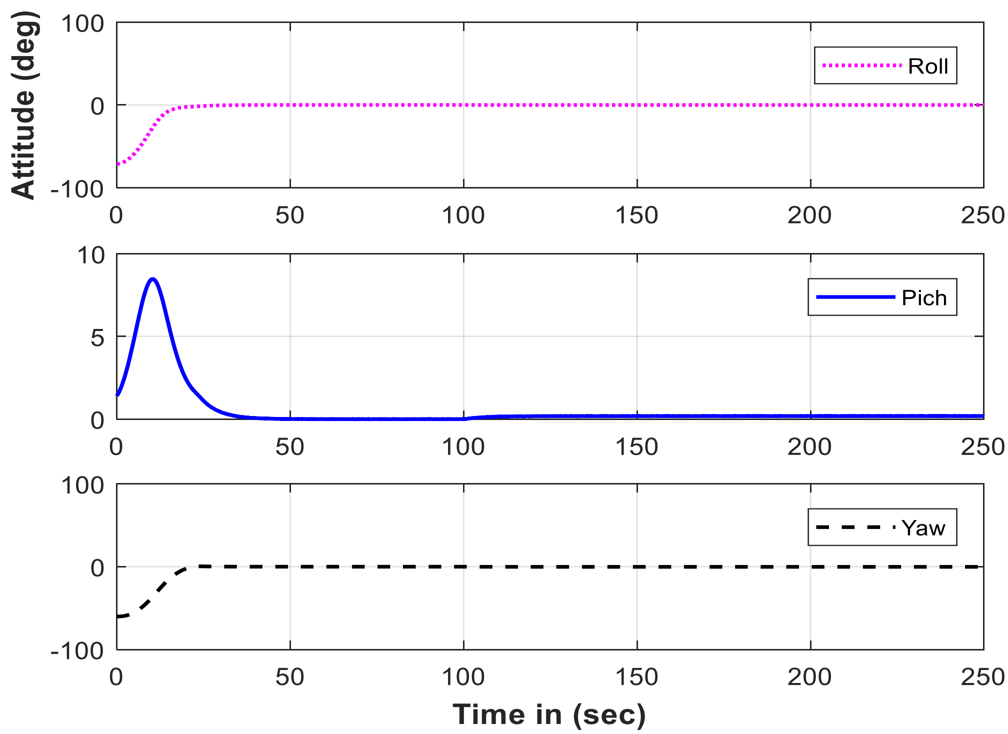

Figure 5. 5 *Euler angle time response of the AFTCS under consideration when a fault occurs.*

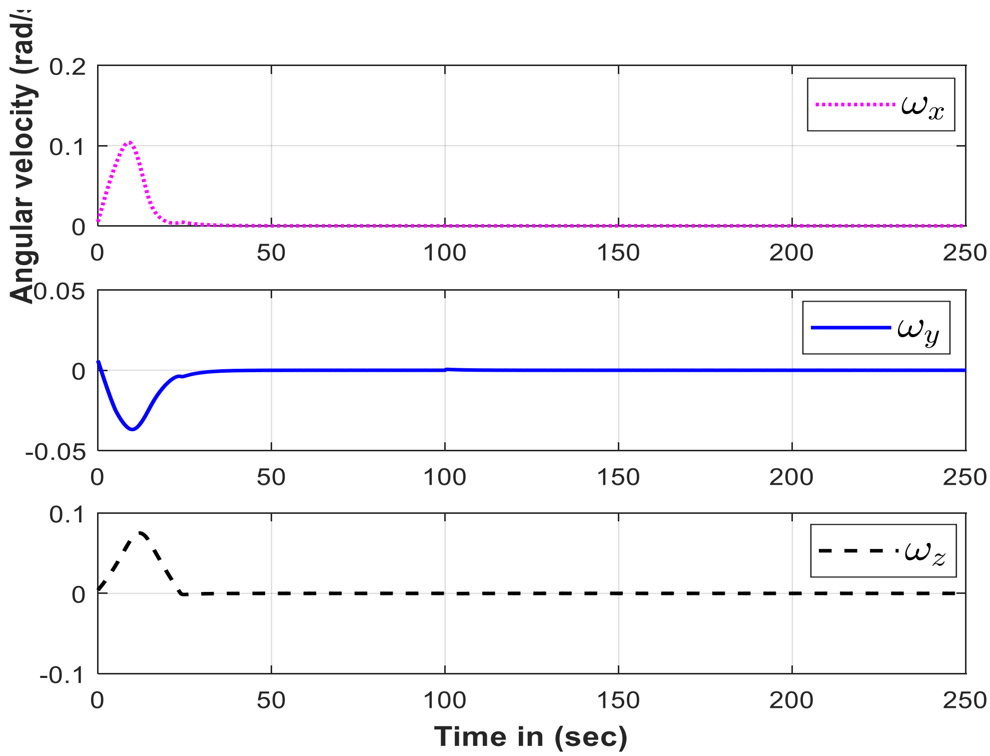

Figure 5. 6 *Angular velocity time response of the AFTCS under consideration when a fault occurs.*

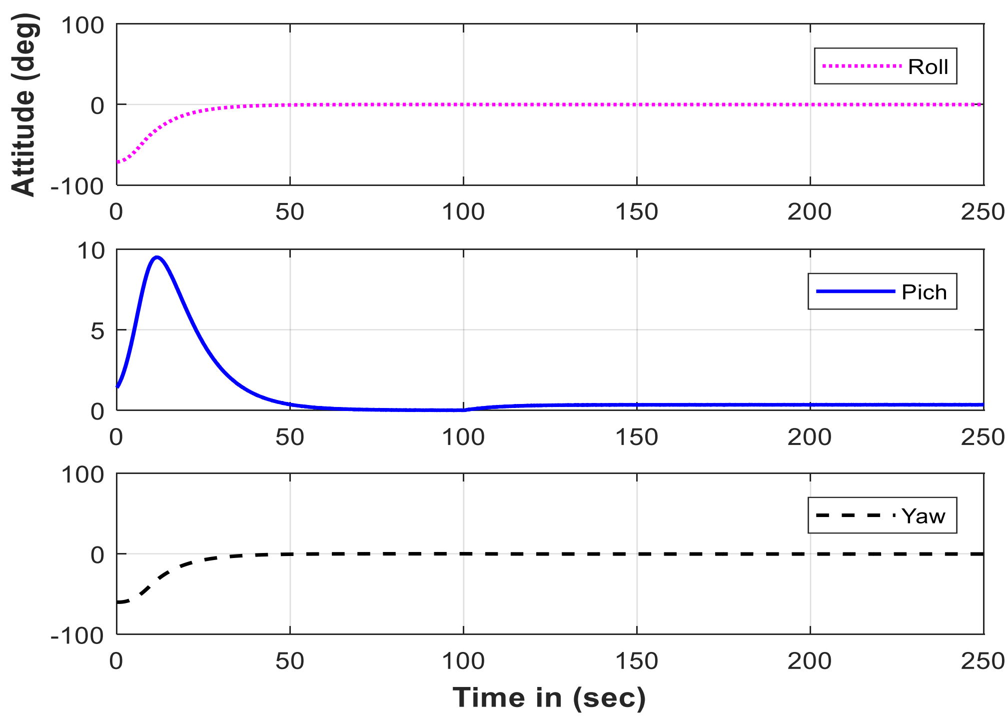

Figure 5. 7 Euler angle time response of integral BC based attitude control systems with linear virtual control inputs [6],when actuator and sensor fault occurs.

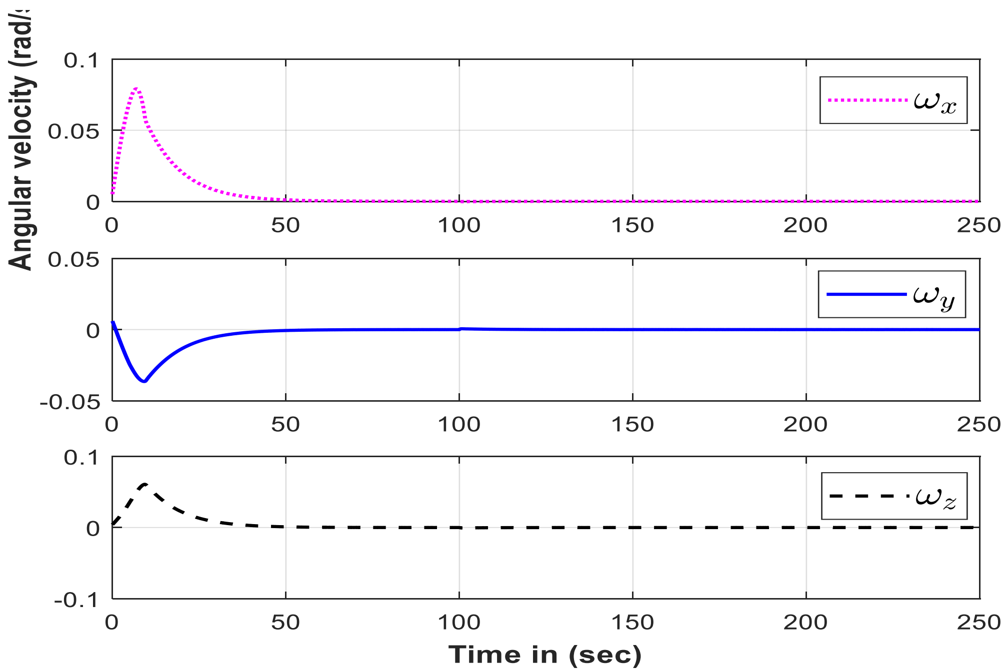

Figure 5. 8 *Angular velocity Time response of integral back stepping controller-based attitude control systems with linear virtual control inputs [6], when a fault occurs.*

Table 5.1 Time response comparison of proposed AFTC and integral BC with linear virtual control inputs in cause of Euler angle

No	Controller	Euler angle (Roll)		
		Rise time(sec)	Settling time(se)	Overshootc(%)
1	AFTC	11.333	24.144	1.299
2	Backstepping with linear virtual input	21.566	40.47	7.57
		Euler angle (Pitch)		
1	AFTC	7.56	1.005	4.652
2	Backstepping with linear virtual input	10.755	1.064	2.724
		Euler angle (Yaw)		
1	AFTC	12.491	20.709	3.633
2	Backstepping with linear virtual input	21.333	40.578	1.956

From Table 5.1, In cause of actuator and sensor faults occurred in the system with addition of model uncertainty and external disturbance transient responses for both controllers, including rise time, overshoot, and settling time, are compared to allow for an analysis of the results. The proposed active fault tolerant controller performed better than the integral back-stepping controller in terms of overshoot, rise time and settling time with respect to roll angle. With the proposed active fault tolerant controller the system parameters are given by settling time was 24.144 second, rise time was 11.33 seconds, overshoot was 1,299% compared to 21.566 second,24.144 second and 7.57% for the BC with linear virtual input. In terms of rise time and settling time the proposed controller performed better than the integral back-stepping controller with respect to pitch and yaw angles, while in terms of overshot the integral back-stepping controller performed better than the proposed controller, when the overshooting for proposed AFTC with respect to pitch and yaw were 4.652%,3.633% and integral back-stepping were 2.724% and 1.956% respectively. Comparing the backstepping controller with the suggested fault-tolerant controller reduces the settling times of the attitude roll, pitch and yaw by 67.6%, 5.9% and 95.9 %, respectively and rise time by 90.3%, 42.3% and 70.8% respectively. So from the result it is conclude that the performance of the system and effectiveness greatly improved by the proposed FTCS.

Table 5.2 Time response comparison of proposed AFTC and integral BC with linear virtual control inputs in cause of angular velocity

No	Controller	Angular velocity(x)		
		Rise time(sec)	Settling time(se)	Overshootc(%)
1	AFTC	16.401	28.727	8.258
2	Backstepping with linear virtual input	21.319	45.567	4.965
		Angular velocity(y)		
1	AFTC	0.728	33.522	1.420
2	Backstepping with linear virtual input	0.736	48.619	9.053
		Angular velocity(z)		
1	AFTC	0.526	23.598	7.222
2	Backstepping with linear virtual input	20.927	48.202	6.594

From Table 5.2, In cause of actuator and sensor faults occurred in the system with addition of model uncertainty and external disturbance transient responses for both controllers, including rise time, overshoot, and settling time, are compared to allow for an analysis of the results. The proposed active fault tolerant controller performed better than the integral BC in terms of overshoot, rise time and settling time with respect to x-axis of angular velocity. With the proposed active FTC the system parameters are given by settling time was 0.728 second, rise time was 33.522 seconds, overshoot was 1.42% compared to 0.736 second, 48.619 second and 9.053% for the BC with linear virtual input. In terms of rise time and settling time the proposed controller performed better than the integral BC with respect to x-axis and z-axis of angular velocity, while in terms of overshot the integral BC performed better than the proposed controller, when the overshooting for proposed AFTC with respect to x-axis and z-axis of angular velocity were 4.652%,3.633% and integral BC were 2.724% and 1.956% respectively. Comparing the backstepping controller with the suggested fault-tolerant controller reduces the settling times of the angular velocity x, y and z-axis by 58.6%, 45% and 104.3 %, respectively. So from the result it is conclude that the performance of the system and effectiveness greatly improved by the proposed FTCS.

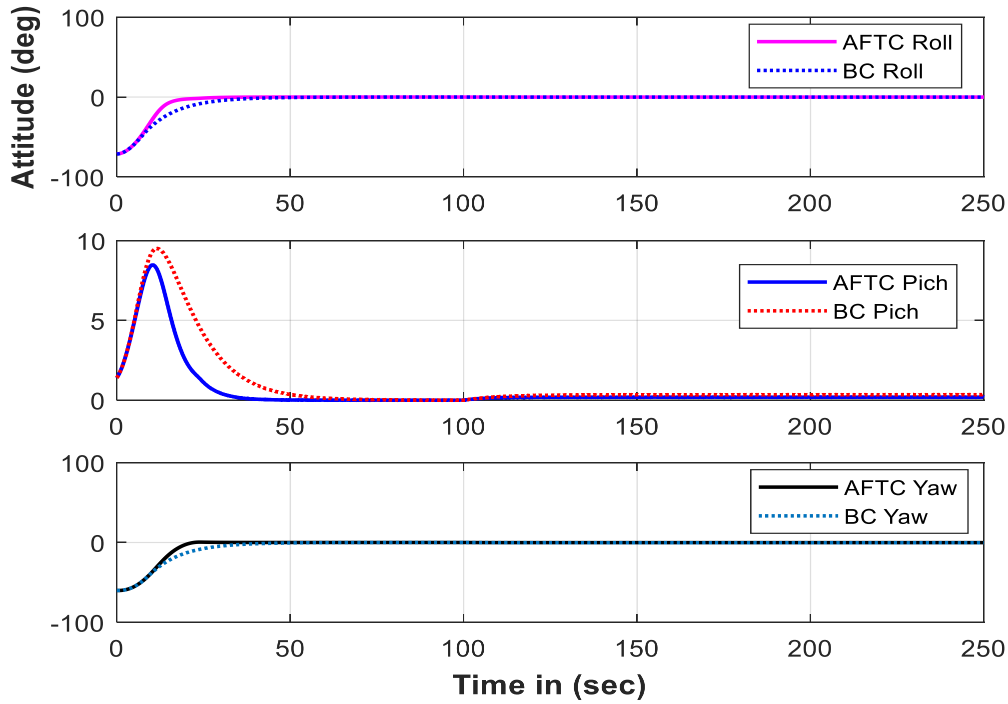

Figure 5. 9 Euler angle time response of both AFTC and BC with linear virtual input under consideration when a fault occurs

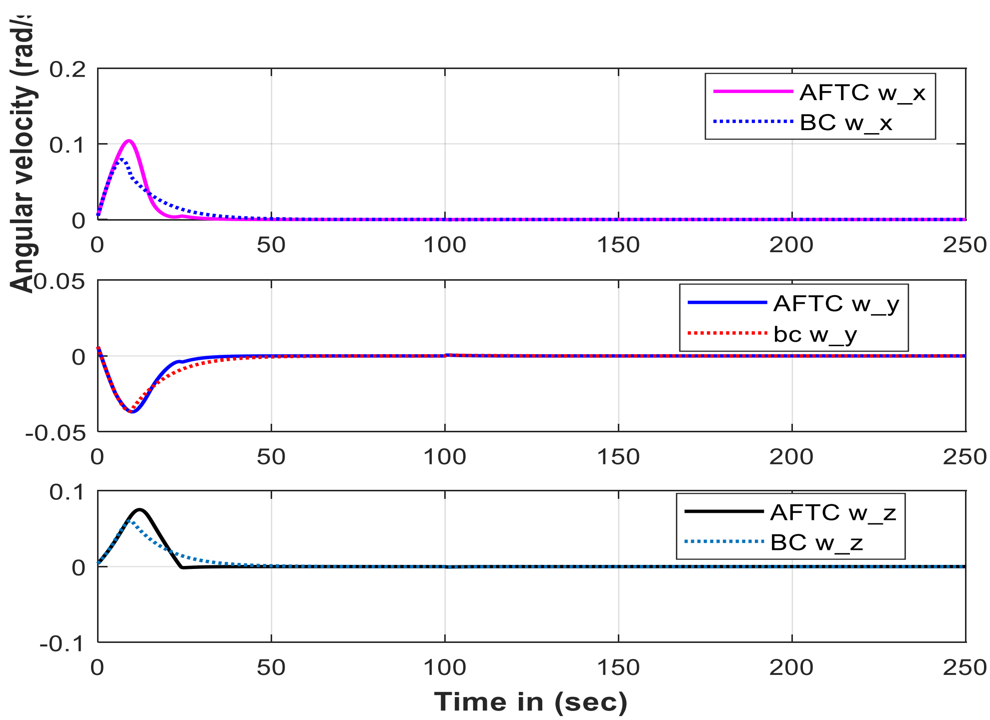

Figure 5. 10 Angular velocity time response of both AFTC and BC with linear virtual input under consideration when a fault occurs

CHAPTER SIX

CONCLUSION AND RECOMMENDATION

6.1 CONCLUSION

For spacecraft attitude control systems with faulty actuators and sensors, model uncertainties, external disturbances and actuator saturations, an AFTCS design is Angular velocity and this design is based on integral BC with a nonlinear virtual input in this research. First, the dynamic and kinematic equations are provided to describe the attitude systems of a rigid spacecraft. The system is sensitive to actuator and sensor problems at certain times, and a fault detection strategy is proposed to identify these times and prevent false alarms from model uncertainties and external disturbances. The overall fault effects are estimated instead of each individual fault in order to provide a straightforward structure for fault identification. Exponentially estimating the overall fault effects to the necessary accuracy level is possible using the established indirect fault identification technique. After that, a FTC law is created to account for actuator faults, sensor faults and saturation constraints based on the estimated information regarding actuator and sensor faults. The Lyapunov technique is used to analyze the stability of closed-loop attitude control systems for rigid spacecraft.

Comparing the integral BC with the suggested fault-tolerant controller in terms of the settling times and rise time of the attitude and angular velocity the suggested fault-tolerant controller have smaller settling times and rise time. So, it is confirmed that the performance of the system and effectiveness greatly improved by the proposed FTCS.

6.2 RECOMMENDATION

A constant fault detection threshold was presented in this thesis, however time-varying fault detection threshold will be the future work to increase the sensitivity of fault detection. Future studies may minimize or avoid the undesirable transient when moving from a normal controller to a reconfigurable one after the problem has been correctly identified. Utilizing the desired and the degraded reference model is one method for resolving the switching transient.

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