Dr. Carl A Erdman

DOCTOR OF ENGINEERING INTERNSHIP EXPERIENCE AT BALL AEROSPACE SYSTEMS DIVISION, BOULDER, COLORADO

An Internship Report

by

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Submitted to the College of Engineering

of Texas A & M University

in partial fulfillment of the requirements for the degree of

DOCTOR OF ENGINEERING

April 1988

Major Subject: Electrical Engineering

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ABSTRACT

Doctor of Engineering Internship Experience at

Ball Aerospace Systems Division, Boulder, Colorado

Wiley J. Larson, Texas A & M University

Chairman of Advisory Committee: Dr. Jo W. Howze

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April 1988

This monograph describes the author's internship experience at Ball Aerospace Systems Division, Boulder, Colorado. A system-level spacecraft design procedure is presented. It describes a spacecraft design flow with emphasis on the interactions among subsystem design decisions. The preliminary subsystem trade studies are discussed.

Several financial and marketing issues of spacecraft contract proposal development are discussed.

The linear design and analysis of the attitude control system for the Air Force Starscan spacecraft is presented. The equations of motion are developed and several attitude control structures and saturation torque limiting schemes are analyzed.

To My Mother, Pauline M. Larson,

and In Memory of My Father, Jack L. Larson

ACKNOWLEDGMENTS

The author wishes to express his appreciation to those people that made significant contributions to the success of this internship.

Gratitude goes to my advisor, Dr. Jo W. Howze, and the members of my Advisory Committee for providing the opportunity to participate in the internship and for their help and support during the entire degree program.

Mr. Bill Follett, my internship advisor, has my sincere appreciation for sharing his unique insight and vision, and for providing invaluable guidance during the intern-ship. I respect Mr. Follett for the determination and courage he displayed battling cancer (and winning), while still meeting his obligations.

My thanks to the people at Ball Aerospace Systems Division for making the intern-ship not only a valuable experience, but a very enjoyable one. Special thanks to Gwen Scohy for her help and enthusiam in editing this paper.

My deepest appreciation and affection goes to my wife and family for supporting me in this undertaking. I could not have done it without them.

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1.0 Introduction

The Doctor of Engineering (DE) program, as administered by Texas A & M University, prepares individuals for professional engineering activities in the public and private sectors. It emphasizes engineering practice, not research, and aims to provide the basis for developing competent engineering leadership capabilities. The required courses provide an excellent blend of business, managerial and technical information that was extremely useful during the internship.

The one-year internship is a significant component of the DE program. The experience gained is the special ingredient that makes the program a unique and viable educational opportunity. This internship has been exciting and insightful. The business and political aspects were every bit as interesting as the technical challenges.

The objectives of the internship as described in the DE Program Manual are twofold: (1) to enable the student to demonstrate and enhance his or her abilities to apply both knowledge and technical training by making an identifiable contribution in an area of practical concern to the organization or industry in which the internship is served, and (2) to enable the student to function in a non-academic environment in a position in which he or she will become aware of the employer's approach to problems, in addition to those approaches of traditional engineering design or analysis.

The author participated in an internship with the Directorate for Spacecraft Design and Development of Ball Aerospace Systems Division (BASD), Boulder, Colorado, from May 1987 through May 1988. The internship advisor, Bill Follett, is the Deputy Director for Spacecraft Design and has been employed by BASD for over 25 years. He is BASD's ultimate authority in system-level spacecraft design.

Ball Aerospace Systems Division is a subsidiary of Ball Corporation, a Fortune 500 billion-dollar company with an operating budget well in excess of $100 million. Its primary customers are the National Aeronautics and Space Administration (NASA) and the Department of Defense (DoD). BASD has successfully designed, built and operated numerous scientific instruments and spacecraft. The BASD internship provides a unique opportunity to learn about and contribute to their spacecraft activities.

Figure 1-1 depicts the internship morphology. The Time axis is actually the life cycle of a spacecraft program and begins with the initial concept of a spacecraft mission, which occurs during the program planning phase, and proceeds through satellite mis-

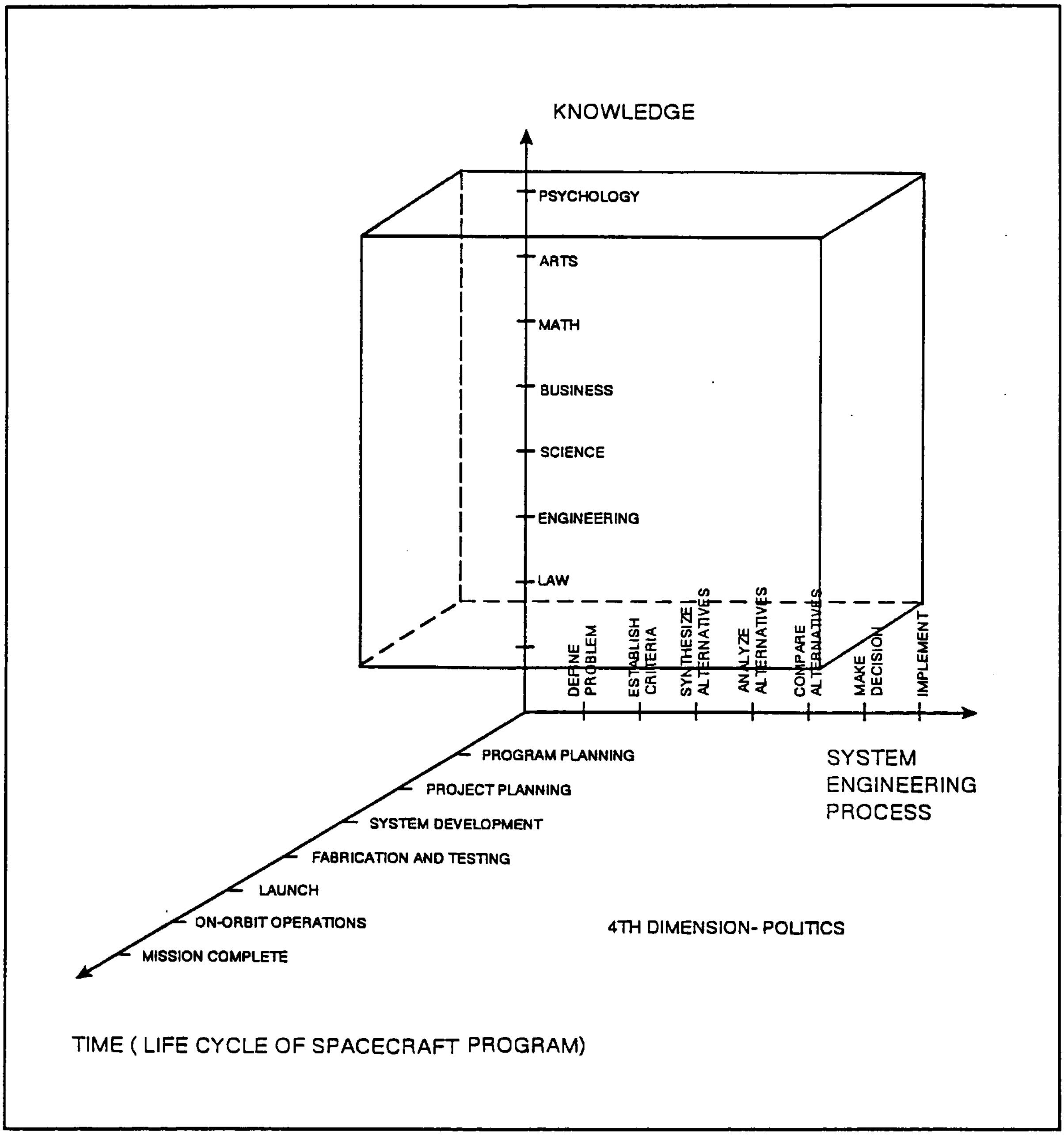

Figure 1-1 Internship Morphology

sion completion. This period of time typically ranges from 7 to 20 years, depending on the program. The Knowledge axis incorporates the various disciplines required for the endeavor. Note that many of the areas are addressed in the DE curriculum. The third dimension is the Systems Engineering Process, which begins with the definition of the problem and progresses through the indicated steps to the implementation of the proposed solution.

A fourth dimension is politics, which permeates the entire process. Adequate interpersonal skills are mandatory for successful working relationships within the company. Contractor/contractor, contractor/Government and Government/Government political relationships must be recognized and dealt with appropriately to keep the process running smoothly.

The author has extensive experience in spacecraft integrated testing, launch and early on-orbit operations. The box in Figure 1-1 depicts the overall content of the BASD internship activity: the knowledge acquired and applied, the period in the spacecraft life cycle and the general problem-solving process. Note that the intern- ship activities are concentrated in the project planning and system development phases of the life cycle, and expand the author's knowledge and experience base to encompass the complete life cycle of a spacecraft development program.

The specific objectives of the BASD internship include:

(1) Refine and publish a draft Spacecraft Systems Design Handbook (SSDH) in collaboration with the internship advisor to acquire and demonstrate breadth of knowledge the of systems-level spacecraft design activity.

(2) Develop a preliminary attitude control subsystem design for BASD's Starscan spacecraft, as the program schedule permits, thus demonstrating a depth of knowledge of this design process.

(3) Develop an understanding of selected managerial and political principles governing the spacecraft industry.

The material that satisfies objective 1 is documented in Chapter 3, efforts that satisfy objective 2 are discussed in Chapter 4, and the information in Chapters 2 and 3 satisfy objective 3.

2.0 Spacecraft Proposal Development

The author participated in several spacecraft proposal development activities at BASD to better understand the business development efforts of a spacecraft contractor. It was one of the most exhilarating and intellectually stimulating endeavors that the author has experienced. The process is straightforward, but it is steeped in urgency and intense interpersonal relations.

The Government (either NASA or DoD) determines its needs and develops spacecraft mission concepts. Spacecraft program requirements and specifications are documented and transmitted to potential developers, usually commercial contractors, via a request for proposal (RFP). Upon receipt of the RFP, the contractor develops a proposal for the Government that specifies, among other things, a detailed design coupled with cost and schedule estimates, and a management plan.

Typically, several contractors provide proposals; therefore, each contractor must provide the most cost-effective and technically correct design because the Government will select the proposal it considers best. There are several aspects of the proposal effort the author found particularly interesting.

First, the Government provides the contractor with a list of criteria that will be used to judge the adequacy of the proposal and to rank the competing proposals. The contractor must determine which criteria are most important to the Government and then develop an appropriate proposal strategy. Obviously, if the estimated relative importance of the criteria is not correct, the proposal will not be successful. Overall contractor competition could be improved if the Government were more specific about its priorities.

Second, the Government must advertise all RFPs valued at $25,000 or more so that potential contractors have an equal opportunity to compete for the contract. The Government advertises in publications such as the *Commerce Business Daily* (CBD). But it is common knowledge in the industry that if a spacecraft design house learns about a pending spacecraft RFP through the CBD, it is too late to develop a competitive proposal. Thus the key is to determine, as early as possible, what RFPs are being generated and by whom. This is the responsibility of the contractor's advance marketing organization.

Company management must decide whether it will compete for the contract. At BASD, this decision is based on the company experience base, the anticipated competition, the customer, knowledge of other pending RFPs and the customer's funding source and capability. If the customer is the Government, contractors may avoid acceptable RFPs because of the Government's erratic funding policies. It is strongly recommended that the contractor review both the relative priority of the program within the Government and the funding available. Occasionally, a program is funded in one year, and, if it has low priority, the funds are terminated the next year because of a budget cut.

The decision to develop a proposal must be carefully weighed because a company has a finite amount of bid and proposal funds. Relatively small spacecraft proposals can cost hundreds of thousands of dollars, so management must be confident that they can win the contract and that the Government will fund the effort through completion before committing to proposal development.

If the company decides to pursue the contract, an intensive pre-proposal effort is begun before RFP receipt. This includes ascertaining the payload or experiments and what concepts are most important to the customer. Then the pre-proposal team begins developing potential spacecraft systems. The author believes that most aerospace companies have similar practices, and the result is that the companies all start early and, therefore, remain competitive.

Just prior to receiving the RFP, the pre-proposal team is augmented so that every spacecraft subsystem is represented, and formal proposal efforts begin. Typically, the proposal team is a microcosm of the company, but if a major contract is at stake, the people selected are the company's best. This is understandable since the proposal team completes the initial design of the spacecraft and lays the foundation for a program that may exist for 10 to 12 years. If the proposal is inadequate for technical, cost or schedule reasons, the company will either forfeit the contract or be bound by its mistakes. Therefore, the contractor is motivated to develop the best proposal possible.

The most astounding aspect of the proposal effort is that even for a major program, it spans only three to six months. Yet, during this short period, all subsystems are designed to exacting standards, a detailed schedule is devised, detailed cost estimates are generated and a near-letter-perfect proposal is created. In a very limited amount of space the successful proposal demonstrates that the company is technically and fiscally competent in all aspects of the program.

The environment is rich in intellectual and interpersonal stimulation. BASD's proposal team members displayed uncommon dedication and enthusiasm.

Spacecraft Proposal Development - A Systems Perspective

The spacecraft proposal development process can be viewed from a systems engineering perspective as shown in Figure 2-1. The customer provides an RFP that specifies the technical performance and schedule requirements, and the contractor ascertains the associated customer funding capability. This model can be used at various levels of the design effort: systems (e.g., subsystem mix), subsystems (e.g., type of attitude control scheme) or component level (e.g., type of batteries for the power subsystem).

The process depicted in Figure 2-1 begins by determining the requirements and constraints stated in the RFP. This is accomplished by having every proposal team member read the RFP from cover to cover and independently develop lists of "stated" requirements and constraints, which are then integrated into a single list.

Additional derived and assumed requirements and constraints are injected into the process. If not managed properly, they may become the nemesis of the spacecraft design process and could ultimately flaw the proposal. Most derived requirements and constraints follow logically from those stated in the RFP. However, some may be improperly derived, and care must be taken to eliminate the improper ones. Assumed re-

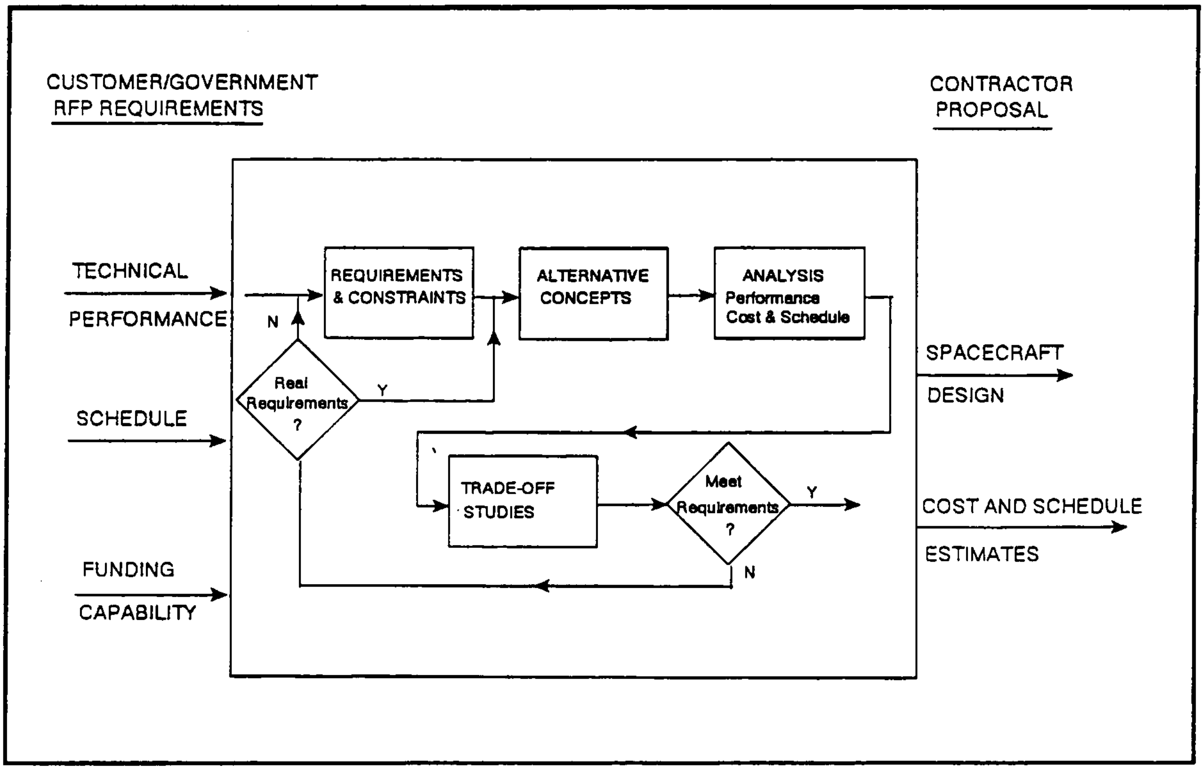

Figure 2-1 Spacecraft Proposal Development Process

quirements and constraints do not flow directly from those stated; rather, they are assumptions that must be made to continue with the process. One must remember that they are flexible requirements yet could unnecessarily limit potential alternatives. For example, suppose a stated requirement is that the spacecraft bus provide sufficient electrical power for the entire spacecraft. For a satellite in low-Earth orbit, LEO an acceptable derived requirement might be that solar arrays are used to generate the electricity. An unacceptable assumed requirement is that the standard solar cell used on the last mission is to be used. Solar cell efficiencies have improved by three to four percent in the last five years. Selecting an older, less efficient cell might unduly increase the size, weight and cost of the array.

This example also points out that alternative concepts should always be considered. Some engineers tend to use the same hardware/software in their designs because they have used them before successfully. There is a lot to be said for incorporating heritage into spacecraft designs. Often, previously designed and flown alternatives are desirable.

But often there is newer and, perhaps, better technology available. Therefore, the rule, at all levels, is to develop alternatives before selecting a design.

Once alternatives are developed, each is analyzed for technical performance, and cost and schedule implications. Then the alternatives are compared and the best one is selected. On one hand, good systems design practice demands that alternatives be considered. On the other hand, a caution is to not become overzealous with the process. If cost is considered, "better" is the enemy of "good enough." The results of a successful satellite development proposal are (1) an adequate technical design that is cost-competitive and available within the required time frame and (2) a signed contract.

Proposal Cost Estimating

Once an adequate spacecraft design has been developed, cost must be determined and an appropriate price must be selected. The pricing strategy is crucial to the success of the proposal effort: If the contractor designs the ultimate satellite but prices it improperly, (S)he has spent hundreds of thousands of dollars only to have the contract awarded to a competitor. To remain in business the contractor must make a profit, but (s)he must ensure that the price quoted to the customer is competitive. Occasionally, these competing requirements cannot be satisfied. Therefore, the first step in determining a competitive price for the system is to develop a cost estimate.

Three methods of cost estimation are typically used in the Government/contractor arena.[2] The most prevalent method is the bottom-up or grassroots cost estimate. The basis of this estimate is the work breakdown structure (WBS), like that shown in Figure 2-2.

The idea is to separate the effort into reasonably small work packages that contain identifiable tasks and components. A project schedule is provided and the number of hours, level of expertise (which determines pay scale) and resources required to accomplish each task are estimated for each work package. Ideally, each estimate is generated by the person responsible for the work package. The work package estimates are added

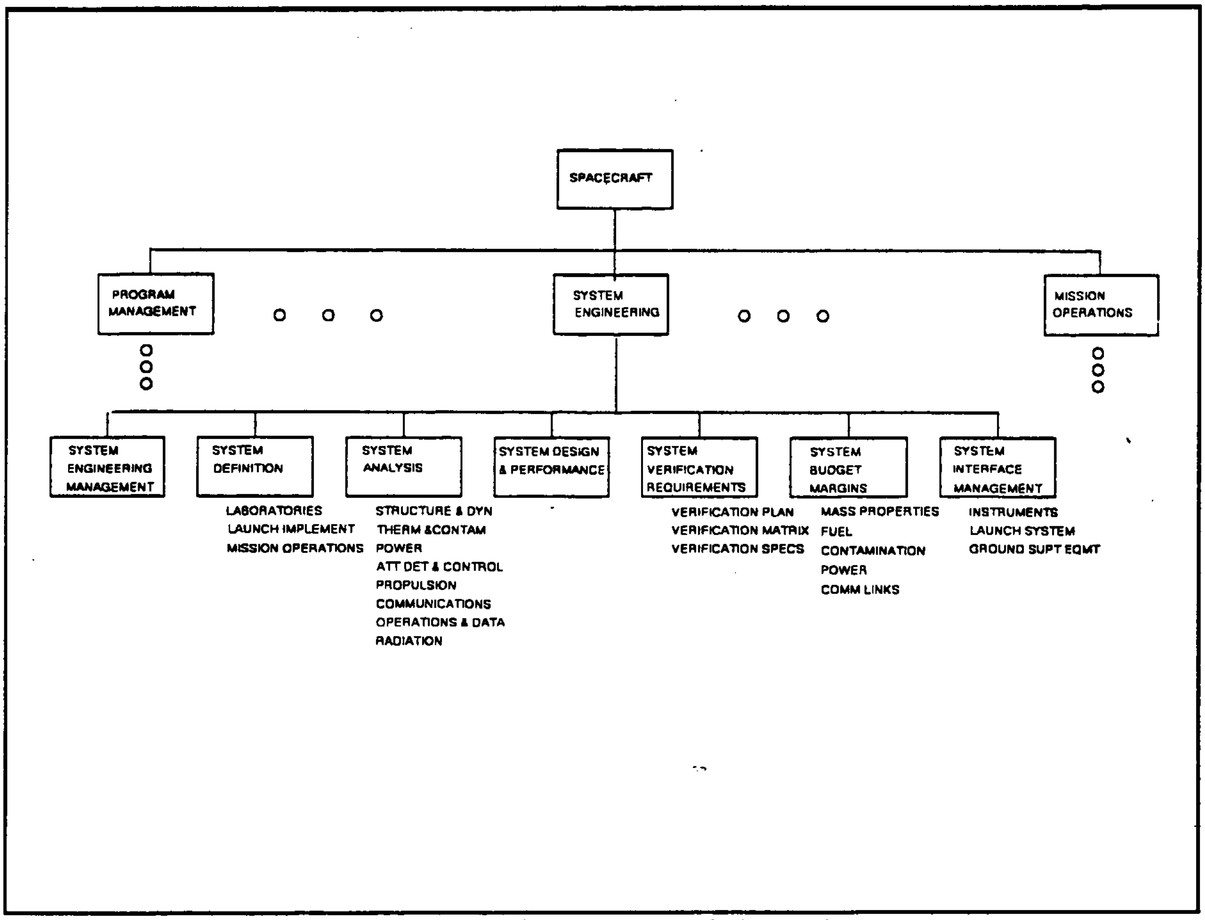

Figure 2-2 Example Work Breakdown Structure

from the bottom of the WBS to the top, hence the name. The result is the total number of man-hours required to complete the project. The total hours are multiplied by the appropriate labor rates and the appropriate overhead rate is applied. The result is a bottom-up estimate of the project's cost.

Many managers feel that a grassroots estimate is higher than that produced by other methods. The benefit of this type of estimate is that the people who are ultimately going to accomplish the work are required to plan the effort. This provides some degree of accountability and also provides management with more insight into the design. In addition, problem areas may thus be identified and resolved.

The Government typically uses a parametric cost estimate to help ascertain the appropriate cost of a satellite program. Many spacecraft have been built since the late 1950s, and studies have revealed that for similar spacecraft, the cost of the subsystems and overall spacecraft can be estimated fairly accurately using several basic parameters. [2] These include, but are not limited to, spacecraft mass, power, data rate, type of attitude stabilization and required launch vehicle. Usually contractors are not privy to the dollar values associated with these parameters. The result of this activity is an independent cost estimate for the program that is used to assess the acceptability of the contractor's price.

In addition to the grassroots estimate, the contractor typically uses a historical cost estimation technique based on the actual man-hours, computer CPU time, equipment and facilities required for a similar, previously developed spacecraft. The price ultimately quoted in the proposal may differ from all of the estimates previously mentioned.

One pricing strategy that is useful to consider is illustrated in Figure 2-3. The key to this strategy is to ascertain the Government funding profile. The sum of the funds allocated for all years constitutes the Government funding limit for the program. The funding profile can be determined in various ways, one of which is to review congressional, presidential and lower-level Government budget submissions for items relating to the proposed spacecraft program. This is particularly easy if the DOD considers it a major program (> $40 million), because it will be listed as a separate line item in the budget.

The Government funding profile changes from year to year based on the political and budgetary climate, which poses significant problems for both the Government and the contractor. Unfortunately, the funding profile is usually stretched over a longer term. This causes the contractor to reduce the level of effort and stretch out subcontracts, ultimately costing the Government more money. The point is that the funding profile is dynamic and must be constantly watched by both parties.

Once the Government funding limit is determined, a management reserve for both the Government and contractor is subtracted, resulting in the contractor's upper limit price. This price is then allocated to the term of the contract, resulting in the contractor pricing profile. The contractor attempts to match the pricing profile to the Govern-

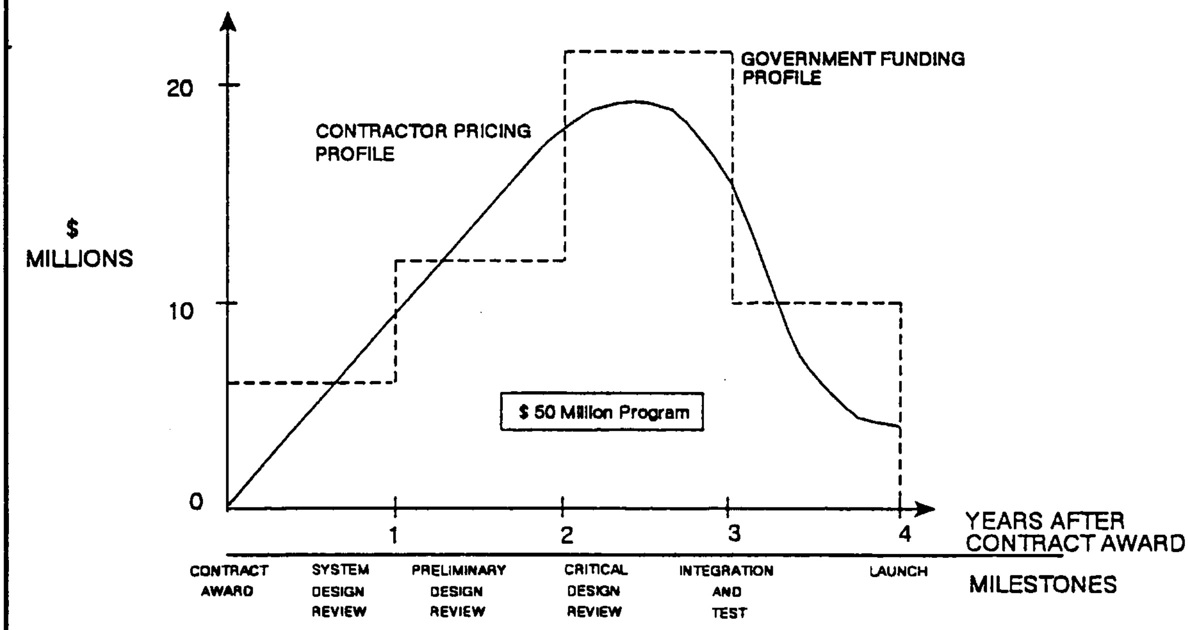

Figure 2-3 Proposal Pricing Strategy

ment funding profile to avoid the need for more Government funds or further contractor investment. Of course, the contractor must determine if (s)he expects to accomplish the proposed work for the price quoted.

The upper limit price quote becomes the "should" cost for the spacecraft program. The "should" cost is divided into total labor hours, computer time, facilities and equipment required, and is distributed among the various work packages shown in the WBS. The percentage of the total labor hours and computer time allocated to each work package is determined by management based on the historical data previously mentioned. The resulting figures become the "should" cost for each subsystem. The "should" cost and grassroots estimates for each subsystem are compared and any discrepancies are negotiated. If the process is working properly, the result is a price that is both competitive and acceptable to the work package managers.

Before discussing the details of systems-level spacecraft design, one important aspect of the spacecraft proposal development process must be considered: *People* perform the tasks within the process. Their personalities, backgrounds and goals must be understood and accommodated so the process can proceed successfully.

Key participants are the program manager, systems engineer (SE) and subsystems engineers. The primary goal of the program manager is to develop a technically adequate product on schedule for a competitive cost; (s)he is the primary advocate for cost and schedule. The subsystems engineers' primary concern is the technical performance of their subsystems. Their goals may vary from developing an exceptional state-of-the-art subsystem to maintaining the status quo with previously developed, proven designs and equipment. The SE is dedicated to ensuring that overall spacecraft system performance is acceptable. (S)he is the primary technical advocate for the program and is concerned with "making it work."

The following chapter discusses the spacecraft design process from the systems engineering perspective.

3.0 Spacecraft System Design

This chapter discusses a spacecraft system design process for satellites in LEO from the perspective of the spacecraft SE. This activity must be discussed carefully because spacecraft systems engineering is an art, and every artist has his/her own perspective and method of accomplishing similar tasks. The goals here are to identify the primary responsibilities of the SE and to provide a model to facilitate the discussion of the overall spacecraft configuration development. Fifteen design paramters are identified that can be used to control the design of the spacecraft. The intent is to maintain a system level perspective and to avoid, as much as possible, the details of subsystem level implementations.

There are many systems within a spacecraft program. This chapter deals primarily with the spacecraft, the launch vehicle and the ground support systems that are inextricably tied together to form the basis of a spacecraft design. The emphasis is on the spacecraft system, with reference to the others as necessary.

The spacecraft can be divided into two distinct elements, the payload and the spacecraft bus, as shown in Figure 3-1. The payload may be a communications package, meteorological equipment, reconnaissance equipment or a collection of scientific or military experiments. The spacecraft bus is divided into subsystems that support the requirements of the payload and the associated mission. A plethora of information is available about the design of individual subsystems, but the literature is lacking in information about the systems engineering effort required to integrate the subsystems into a viable and cost-effective spacecraft system.[3,4,5]

Systems Engineer's Perspective

Dr. J.R. Stuart, known in the industry as Captain Satellite, uses an analogy that accurately describes the function of the spacecraft SE. It says that many people know how to grow terrific carrots, tomatoes, lettuce and cucumbers, and there are many

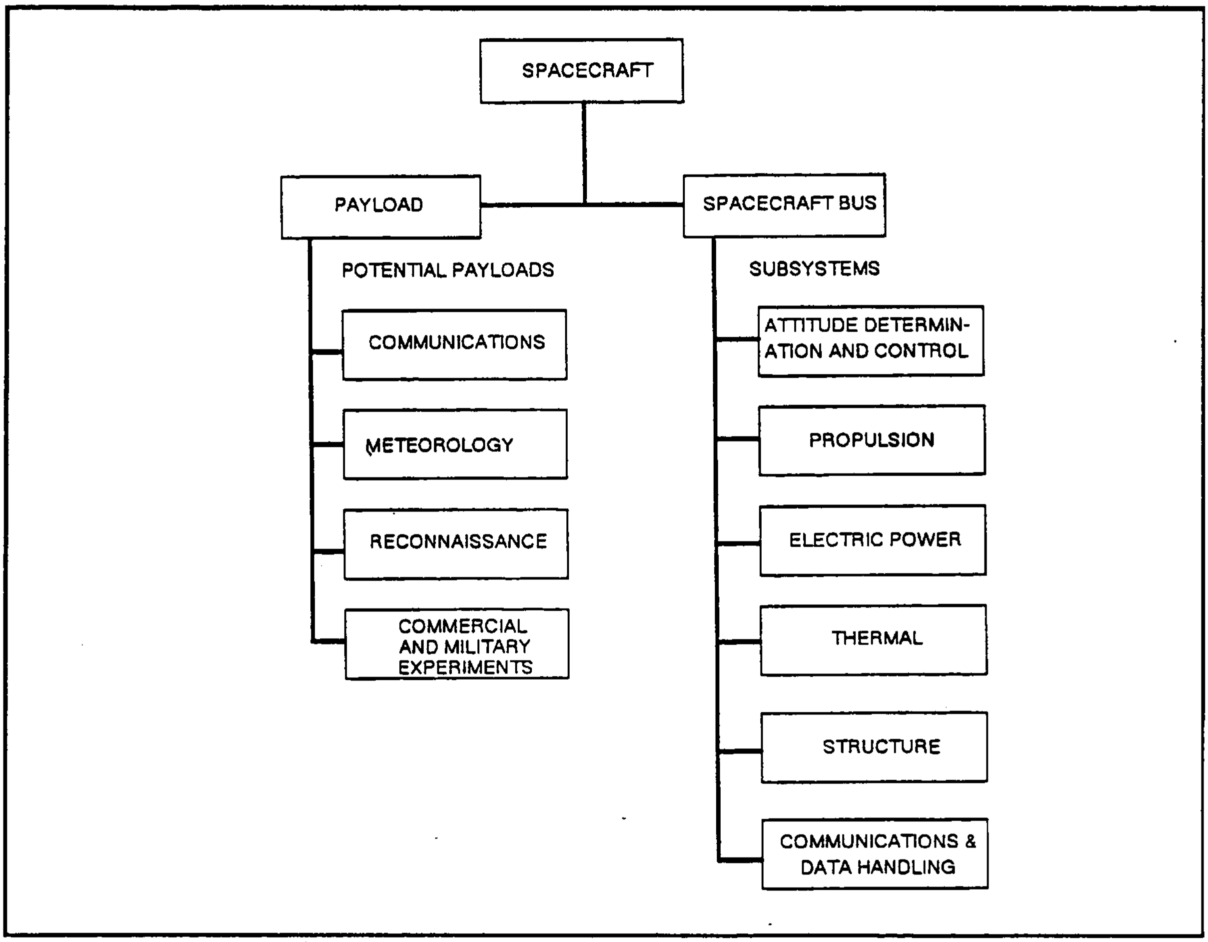

Figure 3-1 Typical Spacecraft Block Diagram

good salad dressings available, but there are few who can combine these ingredients into a truly exceptional salad. The analogy follows for a good SE.

The SE has three primary responsibilities in the design process. First, (s)he is charged with ensuring that the systems engineering process is executed effectively. This requires a strong leader who can perform the associated group maintenance functions required of any effective manager. The key to success is to properly define and control the requirements and constraints on the system and to communicate them to the appropriate subsystems engineers. The importance of this task cannot be overstated. If the requirements are improperly stated, too broad or too restrictive, the result will almost always be a system that does not meet the needs of the mission or is not cost- effective, or both.

Second, the SE must initiate, perform and control system-level or intrasubsystem trade studies. The approach used maintains a fairly equal level of effort among the subsystems, while ensuring that system requirements are met. The intent is to consider different implementations within each subsystem and to determine what the resulting impacts are on the remaining subsystems. In most cases, the implementation selected is the one that meets the requirements while minimally impacting the other subsystems.

Finally, the SE is ultimately responsible for the interfaces between subsystems within the spacecraft and the interfaces between the spacecraft system and other program elements, such as the launch vehicle system.

The spacecraft SE does not have the luxury of delving into the details of a particular subsystem design, except when it is required by a system trade analysis. The SE must constantly determine to what extent potential subsystem implementations will affect other systems and subsystems. The following section provides a systematic approach to spacecraft design.

A System-Level Preliminary Spacecraft Design Approach

This section discusses the major factors that affect spacecraft cost and configuration. Key determinants of the ultimate configuration are examined using a system-level spacecraft design approach. This section will help the reader to understand the spacecraft design process and to develop a cause-and-effect mentality for spacecraft subsystem interactions.

In general, the cost of a spacecraft is highly dependent on its total mass, total power, attitude pointing, data rate, reliability, testing, documentation and launch vehicle requirements. The requirement for high reliability translates, in part, to subsystem component redundancy and a resulting increase in mass and power requirements, and it is addressed as such. Severe part and component testing requirements improve overall system reliability but tremendously increase cost and schedule. Testing and documentation requirements are determined primarily by the rules and regulations governing procurement for the sponsoring Government agency. Since they are not directly controllable design parameters, they are not discussed further.

Primary cost parameters are augmented in the preliminary system design process, shown in Figure 3-2, and the resulting set of key design parameters is used to develop a baseline spacecraft configuration.

Preliminary System Design Process

The preliminary system design process can be divided into preliminary mission design and preliminary spacecraft design. Mission design integrates payload requirements with orbital options and constraints and determines, if necessary, which launch vehicle is most appropriate. The intent is to divide payload, mission and launch vehicle requirements into the key design parameters (KDPs) used to develop potential system and subsystem designs and to perform trade-offs among them.

Typically, some of the KDPs are provided directly by the payload sponsor's RFP. They become the "stated" requirements mentioned previously and, as such, must be satisfied. They are indicated in Figure 3-2 by the single-direction arrows on the key design parameters.

The double-direction arrows denote derived and assumed requirements that should be used in the initial design, but may be changed after proper analysis and negotiation. They are integral to proceeding with spacecraft design.

The preliminary design effort is represented at the bottom of the figure. The procedure is iterative, and decisions within one subsystem will almost always affect the other subsystems in varying degrees. Actual design and trade studies may begin with one of several subsystems. This approach begins with the attitude determination and control subsystem (ADACS) and proceeds to the thermal control system.

An individual or small group of people might execute the cycle once to initially size and configure a spacecraft. Typically, many trade studies are performed within each sub-

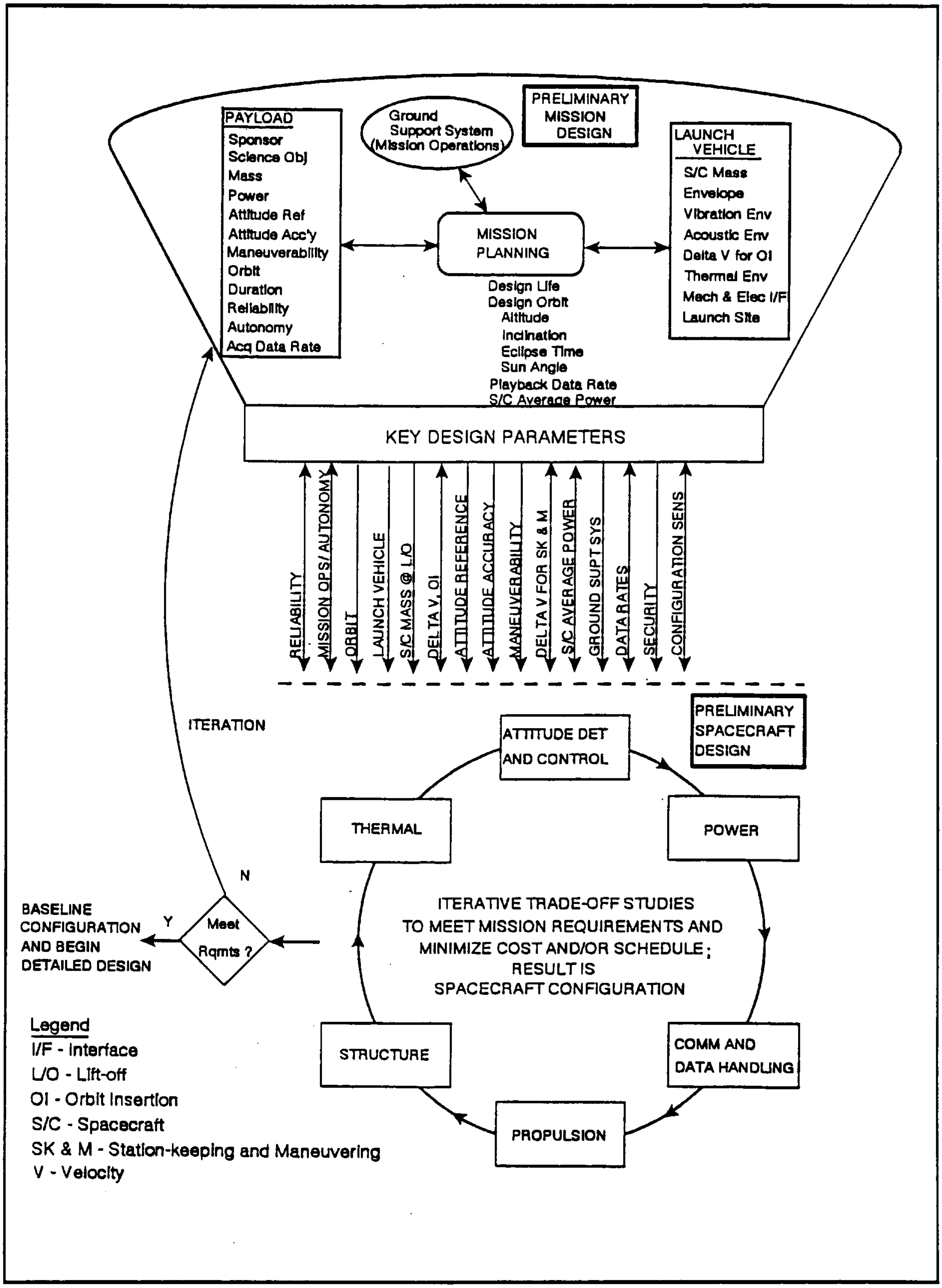

Figure 3-2 Preliminary System Design Process

system, impacts on other subsystems are considered, total performance is reviewed and the results are compared to the requirements. If all requirements are satisfied and cost and/or schedule are minimized, the design becomes the baseline configuration. If the requirements cannot be satisfied after exhausting system and subsystem options, the overall system design process is iterated, as shown in the figure. This process continues either until all requirements are satisfied and the cost and/or schedule constraints are met, or the sponsor's requirements change.

Detailed mission and spacecraft design activities are now discussed within the framework of the preliminary system design process. Specific numbers are avoided, but many approaches, assumptions and example calculations are provided in the *Ball Aerospace Spacecraft Systems Design Handbook*.[6] The purpose of the preliminary mission design (PMD) effort is to develop realistic estimates for the KDPs so that preliminary spacecraft design activity progresses smoothly.

Preliminary Mission Design

The primary elements considered in PMD are the payload and launch vehicle. The function of mission planning is to consider all aspects of the mission, determine the appropriate orbit, integrate the elements into a cohesive unit and provide KDPs for the remaining activity.

Payload

The most important element of mission design is the payload. Agencies or units within NASA and DoD are the sponsors of most payloads. Knowledge of the sponsor alone can yield insight into several KDPs. The sponsoring agency is required to procure space hardware and software in accordance with its stated policies and regulations, which specify (among others) testing, documentation and reliability requirements. These requirements are tailored for each program, and this information is usually provided in the RFP.

Specific elements of NASA and DoD maintain dedicated ground support systems, so in many cases the appropriate ground support system can be determined simply by knowing the sponsoring agency. From the user's perspective, ground support systems are not very dynamic, in that they maintain fairly constant capabilities. For example, antenna sizes, types and associated frequency bands remain constant, and the effective isotropic radiated power (EIRP) required to guarantee performance at the input to the antenna is published. This information coupled with the orbit parameters can be used to size the spacecraft transmit antenna(s) and identify the need for RF amplification. A similar procedure can be used for the receiving antennas.

The ground support system is composed of a finite number of ground stations at fixed locations. Given the groundtrack of the spacecraft, the specific ground stations involved in the mission are easily identified. Then the amount of time available to download data and the time between data transmissions can be estimated. This information, combined with the payload data acquisition rate, yields an estimate of the playback data transmission rate from the spacecraft to the receiving station, which is a KDP.

The sponsoring agency documents its scientific/mission objectives in the RFP. These objectives typically specify, among other things, the type of technology used in the payload, the type of experiments to be conducted and the motivation for the mission. These objectives then set the tone for the entire design activity.

As a result of in-house mission planning, the payload sponsor determines many of the essential requirements for the mission and documents them in the RFP. They include, but are not limited to, payload mass, power, orbit, mission duration, spacecraft attitude reference, attitude pointing accuracy, data acquisition rate, and maneuverability requirements. Sometimes even the launch vehicle is specified.

A significant responsibility of the mission design team is to check the stated requirements for accuracy and to augment them, as necessary, to provide a complete list of KDPs. The orbit and launch vehicle provide two insightful examples of augmentation.

Often, the sponsor partially specifies the orbit. For example, of the six classical orbital elements required to completely specify a satellite's position in an orbit, one or two may be specified. The attitude and inclination may be stated as a range of values, e.g., 300 to 500 km or 57 degrees to 90 degrees, respectively. The mission designer must select the appropriate value or range of values and augment the orbital elements, as required. Once the orbit is specified, several additional parameters are fixed. For a specified orbit, the altitude, inclination and eccentricity are defined. The orbit lifetime is established based on drag effects at altitude. The maximum sun-Earth-satellite eclipse time is established, and the range of the sun angle (i.e., angle between the sun vector and orbit normal) is identified. Orbit selection has far-reaching implications and, once determined, the launch vehicle can be considered.

Launch Vehicle

The launch vehicle is a critical component of a successful mission and must be selected carefully because it is also one of the most expensive. Consider two cases. The first case is when the sponsor specifies a launch vehicle, which tremendously simplifies the problem. Given the launch vehicle performance curves and the desired parking orbit, one can calculate maximum spacecraft mass at lift-off. The change in velocity and mass of propellant required to transfer to the desired orbit can also be estimated. The mass of propellant is subtracted from the total spacecraft mass at lift-off and the result is the allowable spacecraft mass. This mass can be allocated to the subsystems via mass-fractions developed from similar existing spacecraft. The result is an estimated allowable mass for each subsystem.

The more interesting problem occurs when the launch vehicle is not specified. Given a desired orbit and a payload mass, an estimated spacecraft mass can be calculated via mass-fractions. The performance charts for a finite number of available launch vehicles are analyzed, resulting in a list of acceptable launch vehicles. For a single spacecraft there are usually just one or two expendable launch vehicles (ELV) that are appropriate. Final launch vehicle selection is based on performance, availability and cost.

Once the launch vehicle is identified, several KDPs and constraints are established. As before, the upper limit for spacecraft weight at lift-off is fixed. The volume or envelope constraints on the spacecraft launch configuration are set. The launch phase vibration, acoustic and temperature levels are established. The launch vehicle/spacecraft interface is fairly well defined, and even the probable launch site becomes apparent. The preliminary mission design procedure may require several iterations before an acceptable set of KDPs is established.

Key Design Parameters

The key design parameters (KDPs) are distilled from the preliminary mission design process. They constitute the overall system-level configuration drivers for the spacecraft. A major change in a key parameter will have a significant impact on the total spacecraft configuration. Here, the term *parameter* has a broader interpretation than usual. It includes some non-quantitative items because many of them act as parameters in the truest sense for the design process.

This particular set of KDPs was determined by reviewing the information required by each subsystem designer to completely specify the subsystem configuration. The list of items required by all subsystems was molded into an integrated list of information required to adequately bound, but not overly constrain, the design activity. Much of the information was related and could be traced to a specific source. In many cases, the source became the key design parameter.

A word of caution is necessary. Since every spacecraft design problem is different, it is difficult, if not impossible, to provide for every contingency. An attempt is made to generalize the KDPs for the broadest possible application, but keep in mind that they are just typical parameters. The parameters and approaches presented should be tailored to the specific design situation.

KDPs are presented in Table 3-1. The overall contribution of each KDP to the ultimate spacecraft configuration is discussed below. In addition, the derived information and parameters that are available from a combination of KDPs is briefly examined.

Reliability

Reliability and design life requirements specified in the RFP significantly impact spacecraft configuration. Reliability is specified either as a figure-of-merit, 0.9999, or as a redundant part and component requirement, e.g., no single-point failures. The design life requirement is stated as the period of time that spacecraft operations must be reliable, such as one year with a goal of three years.

Table 3-1 Key Design Parameters

FIFTEEN KEY DESIGN PARAMETERS
RELIABILITY
MISSION OPERATIONS/ AUTONOMY
ORBIT
LAUNCH VEHICLE
SPACECRAFT MASS AT LIFT-OFF
DELTA V FOR ORBIT INSERTION
ATTITUDE REFERENCE
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MANEUVERABILITY
DELTA V FOR STATION KEEP & MAN
SPACECRAFT POWER
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Design life and reliability requirements are aimed at solving different problems, but the combined effect is to enhance the reliable operation of the spacecraft. A design life requirement is directed at curbing parts, component and surface degradation mechanisms. Some parts and components simply wear out after a certain period of time and various surfaces on the spacecraft lose their original qualities over time. The task of the designer is to ensure that the spacecraft is built to last for the specified design life.

Reliability addresses the failure of certain parts, components and surfaces. Redundant components, dual strings of components and cross-strapping of components is aimed at reducing the ill effects of a failed component, therefore allowing the mission to continue. Another aspect of reliability is the testing necessary to ensure that the parts and components are qualified to a certain level of confidence. There are three major impacts of reliability requirements on spacecraft procurement. In general, as the reliability and design life requirements increase, so do the mass, the time required to design and build and, ultimately, the cost of the spacecraft.

Mission Operations/Autonomy and Ground Support System

Spacecraft on-orbit operations are highly dependent on the particular ground support system used and the degree of spacecraft autonomy that is required. Most spacecraft programs use existing ground support stations to monitor and control the spacecraft. The configuration of most ground stations is fairly static so the sizes and types of antennas and frequency bands available are usually predetermined. The magnitude of the input power required to ensure station performance is documented in the ground station user's manual. The software and data manipulation capabilities of some ground stations are more dynamic and flexible, so more effort may be required to fully use their capability in these areas. Ground support stations are usually sponsored by an agency of NASA or DoD and prioritize their support of spacecraft missions accordingly. The amount and type of support available to a particular spacecraft mission depends on its priority within the system. The level of autonomous spacecraft operations required depends on the amount and type of support available from the ground stations.

In general, the trend is toward highly autonomous vehicles. Operating a ground station is expensive. It is often economically feasible to reduce the spacecraft to ground interface to a minimum. In addition, the spacecraft should be designed to accommodate occasional ground station failures.

Even if a spacecraft is autonomous it must transmit data to the ground at fairly periodic intervals (hours or days). These intervals are referred to as the revisit time. A spacecraft may have a revisit time of, say,100 hours. This information can be used to bracket the amount of data storage capability required on board.

Orbit

Using the classical orbital elements, an orbit is specified by its altitude, inclination and eccentricity. One of the toughest aspects of spacecraft analysis and design is to adequately understand the geometry of the situation. This includes the orbit, the relationship of the primary axis of the spacecraft to the sun and Earth and the relationship of sensors and actuators on the spacecraft to various objects. Specifying the orbit is the first step in understanding the geometrical requirements and constraints of the mission.

There is an infinite number of orbits available, but several are used repeatedly. A geostationery orbit is circular with an attitude and inclination of 36,000 km and 0 degrees, respectively. Its notable feature is that a spacecraft in this orbit remains stationary above a specific longitude on the equator. This monograph deals primarily with satellites in LEO at altitudes ranging from 250 to 1,000 km.

Several inclined orbits are used often. A 28.5 degree inclined orbit requires a minimum amount of propellant for a satellite launched from Cape Kennedy, Florida. The maximum inclined orbit directly attainable from the Cape is 57 degrees, based on range safety requirements. The lowest inclination orbit directly attainable from Vandenberg AFB, California, is 58.2 degrees. Polar orbits are inclined at 90 degrees and sun- synchronous orbit inclinations depend on altitude, but start at about 97 degrees.

Several parameters are fixed once the orbit is specified. For example, the maximum sun-Earth-satellite eclipse time is established, which in turn affects the size or capacity of the batteries required for the mission. The maximum and minimum values of the sun angle in spacecraft coordinates are fixed and these values affect the size and configuration of the solar arrays as well as spacecraft temperature variations. Finally, altitude directly affects the orbital lifetime of the spacecraft.

Given the orbit, a satellite ground track can be generated and used in conjunction with the ground support station locations to determine the available playback data transmission time and rate.

Data Rates

The data acquisition and playback rates are of immediate concern. The acquisition data rate is typically determined by the payload sponsor. It specifies how fast the data will be generated and provided to the spacecraft bus for transmittal. Given the spacecraft revisit time and the data acquisition rate, data storage requirements can be estimated.

The playback data rate is estimated by dividing the total amount of stored data by the time over the ground station. Using the input power required by the ground antennas, the frequency band, the orbit and the playback data rate, the size of the spacecraft transmit antenna aperture can be estimated. The need for RF amplification may also be identified. A similar approach is used to determine the appropriate spacecraft receiving antenna aperture size. The antenna gain, aperture size and RF amplification are also dependent on the spacecraft attitude reference and pointing capabilities. In general, if no RF power is required, high antenna gain (large size) and accurate pointing are necessary.

Launch Vehicle

The launch vehicle KDP, constrains the spacecraft structural configuration more than any other. The selection of a launch vehicle system dictates the maximum envelope and mass of the spacecraft for launch. The launch environment dictates the maximum values for and profiles of the vibration and acoustic levels to be experienced. The mechanical and electrical interface between the spacecraft and launch vehicle is constrained. Given the orbit, even the launch site and attendant interfaces are fairly well determined.

Delta V Required for Orbit Insertion

Given the launch vehicle, associated performance capability and mission orbit, the post-separation change in velocity required to place the spacecraft in the proper orbit from the parking orbit can be calculated. Using a typical propulsion technology, it is possible to estimate the mass of propellant required to place the spacecraft in the proper orbit.

Spacecraft Mass at Lift-off

Once the launch vehicle and mission orbit are selected, the spacecraft mass at lift-off is determined by reviewing the launch vehicle performance capability. The mass of the propellant required to provide the change in velocity necessary to attain the mission orbit is subtracted from the spacecraft mass at lift-off to arrive at an estimate of the "dry" spacecraft mass. This mass then can be allocated to the various subsystems via the mass-fractions associated with previously launched spacecraft.

Attitude Reference

This is a critical design parameter. The primary axis of the spacecraft or the payload line of sight may be directed at or away from the center of the Earth resulting in a nadir or zenith pointer, respectively. The spacecraft may be an inertial pointer, indicating that the primary axis is pointing to a star, the sun or to other targets in inertial space. The attitude reference coupled with the orbit parameters further defines the geometric conditions and constraints imposed on the design. The attitude reference provides insight into potential thermal concerns and potential solar array configurations. For example, a low-cost solar array configuration for a sun-pointed spacecraft is a fixed array, oriented perpendicular to the primary axis.

Attitude Pointing Accuracy

The attitude pointing accuracy requirements dictate the complexity of the ADACS. Usually, the intent is to point the primary axis of the payload in a particular direction with a certain accuracy. There are two components of attitude pointing accuracy: drift and jitter.

The drift requirement is intended to restrict the slower tendency of the primary axis to move away from the prescribed position. Typically it is expressed in ^{o}/hr. The jitter requirement provides a limit for the high rate motions of the primary axis about the intended direction. Jitter is typically expressed in ^{o}/sec.

An analogy may be helpful here. Assume you are driving by a house in a slow moving vehicle and you want to photograph the house. As the vehicle moves past the house, you try to keep the house centered in the viewing frame of the camera. This is analogous to correcting for drift. If the result is a picture that is out of focus the cause might have been rapid or nervous (jittery) movement of the camera during shutter operation. Using a vehicle-mounted tripod might correct for the jitter and enhance picture resolution.

The level of accuracy required will, in most cases, dictate the class of attitude stabilization required. If the accuracy requirement is on the order of 2 to 3 degrees, a passive, low-bandwidth (slow) method of stabilization is appropriate. On the other hand, if the system is required to point to within 0.1 degree, an active, high- bandwidth (fast) stabilization method is required.

In general, the current attitude must be determined more accurately than specified by the pointing requirement. If a stringent attitude determination requirement is levied, a tremendous amount of analysis is required to demonstrate that an attitude determination scheme will provide the required accuracy.

Maneuverability

There are two types of maneuvering to be considered: slewing and tracking. A slewing requirement specifies how the primary axis is moved from an initial position to the target. Minimum distance and minimum time solutions are examples of slewing requirements. A tracking requirement typically specifies the target that the primary axis must follow, such as a star or missile. The requirement to track a relatively slow moving star is easily satisfied, whereas the requirement to track a missile is more difficult.

Spacecraft Average Power

The average payload power or operating power and duty cycle requirements are usually provided in the RFP. The average power required by the spacecraft can be estimated in several ways. Given the payload average power requirement, power fractions may be used to *roughly* bracket the total spacecraft average power necessary. An alternate method is to assume a spacecraft configuration, identify specific components and electronic boxes, assume a duty cycle and then total the average power required. The intent is to determine a ballpark figure for total average power required so solar array and battery sizes can be determined. The power estimate is updated as more information becomes available.

Security

If the payload data is classified, encryption and decryption equipment is necessary. It is usually provided by the government but adds mass and power requirements to the system. In addition, ground support stations and mission operations may be thereby constrained.

Configuration Sensitivities

This parameter is a catch-all for the configuration requirements that do not fit within other catagories. Payload, instrument and attitude determination field of view (FOV) requirements can dictate certain instrument, sensor and actuator relative locations. Thermal considerations may further restrict equipment placement.

The primary purpose for developing KDP values is to provide sufficient information to the subsystems engineers to facilitate parallel subsystem trade studies. Some KDPs are stated in the RFP and others are derived or assume but are necessary to develop a preliminary spacecraft design.

Preliminary Spacecraft Design (PSD)

Given KDP values, trade studies within each subsystem are performed. One must remember that the spacecraft design process is highly iterative and decisions made within a particular subsystem will almost always have an impact on the other subsystems. This is symbolized by the circular arrangement shown in Figure 3-3.

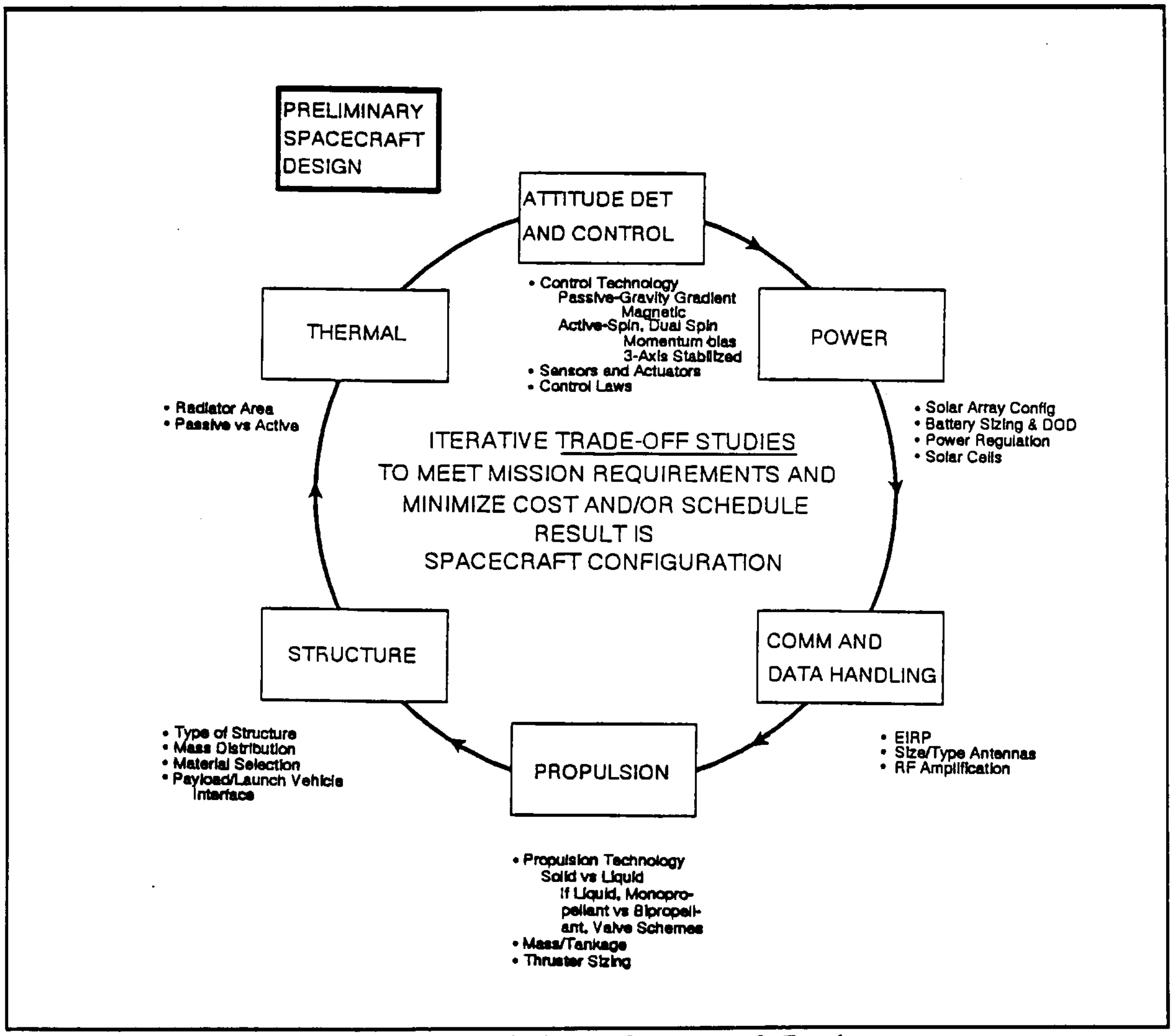

Figure 3-3 Preliminary Spacecraft Design

During subsystem trade analyses, it is important to maintain a system perspective and evaluate the impact of a subsystem option on all other subsystems and the overall system performance. The amount of time and effort devoted to subsystem trade analyses is a function of the complexity of the particular mission and the amount of time available to perform the tasks.

The discussion on the PSD process could begin with any subsystem. In this case attitude determination and control is treated first, followed by the power, communications and data handling, propulsion, structure and thermal subsystems.

KDPs that have a profound impact on the subsystem design are noted within each subsystem description. Reliability, autonomy and configuration requirements affect all subsystems and are mentioned only where there is a major impact. The most important trade-offs are discussed and the impact of each option on the other subsystems is reviewed.

Attitude Determination and Control Subsystem

The KDPs that significantly impact the ADACS design are the orbit, spacecraft mass (inertias) at lift-off, attitude reference, attitude pointing accuracy and maneuverability.

Attitude Control Technology Trade Study

The first trade-off considered is whether to use a passive or active control approach. Then several methods of attaining the control are considered. Once the methodology is selected, subsystem trade studies can proceed in parallel. Sensors and actuators are selected to complement the methodology and alternative nutation damping methods and control laws are examined.

For spacecraft in LEO, the passive vs active control trade-off is fairly simple. In general, if a spacecraft is required to be inertially pointed, highly accurate and/or maneuverable, passive methods, such as gravity gradient or pure magnetic control, are not appropriate. There are exceptions, and occasionally a passive approach can be augmented in a cost-effective manner to meet the requirements. This must be determined on an individual basis. On the other hand, if the primary axis of the spacecraft is pointed directly at or away from the center of the Earth with two-to- three degree accuracy and no maneuvering requirements, a passive system may be appropriate.

If the attitude control requirements can be satisfied with a passively controlled, nadir-pointing spacecraft, other factors must be considered. For example, highly directional communication antennas must be avoided and RF amplification will be required if the spacecraft is not precisely pointed.

If KDPs indicate that an active control approach is required, trade-offs among the active control methods must be considered. Basically, there are two types of active control methods: momentum-based and zero-momentum. Momentum-based methods include spin, dual-spin and momentum-bias stabilization, which are based on the gyroscopic stiffness provided by a spinning mass with angular momentum. The type of spinning mass differentiates the methods. If the entire spacecraft spins during operation, spin stabilization is being implemented. A dual-spin satellite is composed of a spinning section and a despun section. The spinning section provides the angular momentum and the despun section provides a platform for the payload. Usually, a smaller, constantly spinning mass inside a satellite, called a momentum wheel, typifies a momentum-bias satellite.

Each of the three momentum-based stabilization techniques is appropriate for specific purposes. A spacecraft is typically spin-stabilized during transfer from the parking orbit to the final mission orbit. The dual-spin and momentum-bias stabilized spacecraft can provide stable, nonspinning platforms for the payload. Both methods yield accurate (0.1-degree), all-attitude pointing and low rate (slow) maneuvering capabilities. The primary differences are the spacecraft structure and magnitude of the angular momentum vector. For spacecraft of similar mass, the momentum-bias spacecraft offers more nonrotating real estate than its dual-spin counterpart. The magnitude of the angular momentum vector for the spinner and momentum-bias spacecraft is typically measured in thousands and tens of N-m-s, respectively. These methods are capable of accurately positioning one of the three spacecraft axes. Often, the decision to use a particular method is based on the company's previous experience.

The terms *three-axis stabilized* and *zero-momentum* are used interchangeably. A three-axis stabilized spacecraft usually has a small rotatable wheel on each spacecraft axis for attitude control. These wheels are called reaction wheels and are distinguished from momentum wheels since they are normally static, are rotated in either the clockwise or counterclockwise directions and usually provide less torquing capability. Attitude corrections and maneuvers are performed by rotating the appropriate reaction wheel(s). The changing angular momentum introduces a torque and the spacecraft moves accordingly. Zero-momentum spacecraft are capable of providing highly accurate (0.01 degree), all-attitude pointing and dynamic maneuvers for all three spacecraft axes, if necessary.

Once a candidate control method is identified, the impact of the selection on the other subsystems is examined. In general, once the orbit, attitude reference and stabilization method are determined, the essential spacecraft geometry is established. The sun-Earth-satellite geometry can affect all subsystems in varying degrees. Potential solar array, communication antenna and thermal control configurations and constraints become apparent. Payload and attitude control sensor FOV options can be examined. Next, trade-offs among attitude determination and control sensors and actuators are performed.

Attitude Determination and Control Sensor and Actuator Trade Study

Attitude determination and control sensors and actuators are selected to be compatible with the chosen stabilization method and to satisfy the attitude reference, pointing accuracy and maneuvering requirements. If cost were not a factor, the most flexible and accurate sensors and actuators would be used in most designs. The challenge for the subsystem designer is to satisfy the requirements with the most inexpensive components.

The ADACS sensors determine the spacecraft attitude so that appropriate attitude corrections can be implemented. Many sensors, capable of providing a variety of accuracies, are available. Quality accelerometers and gyroscopes provide highly accurate translational and rotational rate information, whereas horizon, sun and star sensors provide position information. Horizon, sun and star sensors are implemented to provide vectors from the satellite to the Earth, sun and stars, respectively; sun and star sensors provide an inertial reference. Given the sensor locations on the spacecraft, it is possible to use these vectors to determine the attitude of the spacecraft.

Magnetometers provide less accurate (two-to-three-degree) information by sensing the magnitude and direction of the Earth's magnetic field. In general, magnetometers are less accurate than Earth sensors, Earth sensors are less accurate than sun sensors and sun sensors are less accurate than star sensors. Note, however, that a high-quality Earth sensor may be more accurate than a low-quality sun sensor, etc. The key is to select the quality and type of sensor that provides an acceptable level of accuracy for the particular application at the lowest cost.

Similar criteria exist for ADACS actuators. The actuators are selected to complement the attitude control methodology and provide the necessary control authority to meet the attitude pointing accuracy and maneuvering requirements. In general, reaction wheels provide more control resolution than thrusters or magnetic torque rods, and thrusters can be sized to provide more control resolution than torque rods. Thrusters and reaction wheels are used to meet high rate slewing and tracking requirements, and magnetic torque rods may be used to meet low rate maneuvering requirements.

A very expensive, highly accurate and maneuverable three-axis stabilized ADACS includes star sensors, gyroscopes and control moment gyroscopes (CMGs). At the opposite end of the active control spectrum is a reasonably inexpensive momentum- bias system using a momentum wheel for control, magnetometers and horizon sensors for determination and torque rods for momentum dumping. An even less expensive nadir pointing system might use a gravity gradient passive control approach with no sensors or actuators.

Sensors and actuators can affect other subsystems. Optical sensors, for example, require an unobstructed FOV of their target and may be implemented in pairs. This restricts the options for locating them on the spacecraft, but, more importantly, their FOV requirements restrict potential locations for other hardware, such as antennas, solar arrays and boom appendages.

Thrusters may be used for station-keeping or attitude control. They emit hot gases that may impinge on spacecraft surfaces, resulting in off-nominal thrust and heat. Thrusters require a certain amount of propellant to meet the design life criterion. If the attitude pointing requirements are stringent, an inordinate amount of propellant may be required, so thrusters may not be appropriate for certain applications.

The electromagnetic field created by torque rod operation may interfere with the magnetometer information and may adversely affect the operation of other on-board electronic equipment. This effect could be countered by properly placing the rods and electronic equipment, shielding the equipment or constraining of the torque rod operation to periods when other equipment is not operating. The important point is that the decisions made within a subsystem may adversely affect other subsystem and system components. These effects must be identified and examined as early as possible to avoid potentially devastating problems in the future.

Power Subsystem

The KDPs impacting the design of the power subsystem are the mission orbit, launch vehicle, attitude reference, payload, spacecraft average power required and design life (an element of reliability). The spacecraft average power requirement may be negotiable, since it is based on a particular equipment duty cycle. Often, the duty cycle can be adjusted to reduce the average power required during certain periods, such as eclipses. In addition, the selected attitude control methodology may further constrain the power subsystem design.

Three primary trade studies are required to adequately size the spacecraft power subsystem: power source/configuration, power storage/amount and power regulation.

There are several fairly exotic power sources available, but for LEO spacecraft with mission durations greater than one year, solar arrays are typically used.

Solar Array Configuration Trade Study

For a stated spacecraft average power requirement, the size and shape of the solar array is dependent on the geometry defined by the orbit, spacecraft attitude reference and the attitude control methodology. The maximum and minimum values of the angle between the sun vector and orbit normal are readily calculated. This angle is often referred to as the beta, or sun angle, but the definition of this angle varies. Given the attitude reference, the range of values for the angle between the sun vector and a particular spacecraft surface normal can be determined.

Maximum power is obtained from a flat plate solar array if the angle between the sun vector and the surface normal is zero. The power output of the array tapers off as the angle between the sun and surface normal vector increases or as the cosine of the angle decreases. Given the spacecraft average power requirement, the size of the flat plate solar array in normal sun conditions is easily calculated for the beginning of the mission. As the design life increases so does the size of the array, to compensate for solar cell degradation.

In most cases a single flat plate solar array sized to provide the total power required by a satellite is not feasible, since the array must fit inside the shroud of the launch vehicle and it is not always possible to point the array directly at the sun. Numerous configuration options exist, but an acceptable option provides a projected area equal to the size of a single flat plate array with normal sun.

Several configurations are considered to demonstrate the critical issues involved with system-level solar array design. There are two basic types of arrays, stationary and articulated. Normally, stationary arrays are subject to a wider variation in sun angle than articulated arrays, so more surface area is required to provide the same amount of power.

Stationary arrays may be fixed to the spacecraft body or deployed. Body-mounted arrays are attractive, since no mechanisms are required, reliability is enhanced and they easily fit within the launch vehicle shroud. They are particularly useful in dual spin applications, where the solar cells are mounted on the spinning portion of the satellite. Disadvantages are that they restrict valuable exterior spacecraft real estate and can pose thermal control problem because the surface of the array gets very hot. In deployed array situations the back of the array functions as a radiator to dissipate the heat, but a body-mounted array does not dissipate the heat as efficiently and transmits the heat inside the spacecraft. Occasionally this effect is desirable, but if not considered and dealt with properly, it could adversely affect spacecraft performance.

Deployed arrays typically operate at lower temperatures and solar cell efficiency is enhanced, but they must fit inside the shroud. Usually they are locked in position for launch, but once in orbit, they are moved to a predetermined position. The attendant deployment mechanisms provide an opportunity for single point failures that could significantly degrade spacecraft operation.

Articulated solar arrays are attractive since they can be maneuvered to reduce the sun angle, so the use of smaller arrays is possible. In some cases they maintain normal sun conditions except during eclipse. Normally, they are deployed and share the same concerns as deployed arrays. In addition, the articulation mechanism complicates the sun pointing algorithm and introduces another potential failure mode.

The resulting solar array configuration may incorporate aspects of several of the options listed above. Selection of a particular option is based on mass, cost and reliability considerations.

The selected configuration can impact all other subsystems. For example, if large deployed arrays are used, they could introduce flexible body motions into the attitude control system. The arrays could impact the location of various sensors and antennas due to FOV restrictions or vice versa. The heat generated by the arrays must be addressed and the structure subsystem must support them during launch and on-orbit operations.

Battery Size Trade Study

KDPs that affect the number of battery cells required are the orbit (eclipse time), design life (reliability) and spacecraft average power. The ultimate battery size is a trade-off between mass, reliability and cost.

Batteries are sized to provide the spacecraft average power requirements during the launch through orbit insertion and sun-Earth-satellite eclipse periods. Normally the eclipse period is the driver because it is usually longer, and only because essential spacecraft housekeeping functions are operated during the launch phase.

Once the orbit is specified, the maximum time of the eclipse period, if it exists, is established. The orbit has a tremendous effect on battery size requirements. For example, a minimal battery and solar array size is possible when the spacecraft is in a "daylight" sun-synchronous orbit. In this optimum case the orbit precesses about the Earth at the same rate as the apparent motion of the sun, avoiding an eclipse. But, mission requirements usually preclude the use of this type of orbit.

Given the eclipse period, design life and satellite average power required, battery size can be determined. This calculation is complicated by reliability considerations. Battery operation degrades with time, increased temperature and number of discharge cycles. The design life coupled with the number and duration of eclipses yields the number of times the batteries are discharged. Batteries are limited to a certain number of discharge cycles, called cycle life. Battery reliability requirements are typically stated as the maximum level of battery discharge allowable, called depth-of-discharge (DOD). For example, if a battery is required to have a minimum DOD of 25 percent, it should only be drained to 75 percent of its total capacity.

For a given cycle life, as the DOD requirement is decreased (increased reliability), the number of cells or capacity required to provide the necessary power is increased, and the resulting mass and cost of the batteries increases.

If the mass of the battery required is prohibitive, operational workarounds may be incorporated to reduce the average power required during eclipse. One way to accomplish this is to turn off nonessential equipment and alter the payload/equipment duty cycle so that less power is required during eclipse. Another option is to adjust the orbit parameters to reduce the eclipse time.

Control of battery temperature is critical. Typically, nickle-cadmium batteries operate best at about 10 degrees C, so the extreme temperature swings in space require that appropriate temperature control methods be used. An attempt is made to place batteries on the cool side of the spacecraft (if there is one) and to use heaters to maintain the prescribed temperature.

When initially sizing the batteries, it is assumed that standard cells will be used. If the need arises, more efficient, lighter and less temperature-sensitive batteries are available at an increased cost. Once the solar arrays and battery sizes have been determined, the power regulation method should be addressed.

Power Regulation Trade Study

The primary purpose of power regulation is to integrate the solar array and battery capabilities to provide adequate power to the load and to recharge the batteries. KDPs affecting power regulation trade-offs are spacecraft average power and reliability.

The simplest power regulation approach is to use the direct energy transfer (DET) method, where the solar arrays, batteries and spacecraft load are connected in parallel. Ideally, when the arrays are receiving sun, they provide power to the load and charge the batteries; during eclipse, the batteries provide power to the load. Unfortunately DET is insufficient, since load and battery charging requirements are not matched with the power provided by the arrays. So batteries can be overcharged and their performance degraded. Therefore power regulation is usually required, except in lower-powered and less sophisticated systems. For larger, more sophisticated systems, power matching between the load and batteries and the solar arrays is necessary.

Two basic approaches are used to manage solar array power in a DET system: power dissipation and power avoidance. Solar array power dissipation is accomplished by passing the undesired current through large resistors via a shunt regulator, which dissipates excessive power by generating heat. One consideration is where to locate the resistors. Some may be placed within the spacecraft and others on the deployed solar arrays, depending on the spacecraft's thermal configuration. If the interior spacecraft temperature is hot, power dissipating resistors may be placed either on the exterior of the spacecraft or on the arrays. If placed on the arrays, the solar cell temperatures are raised and their efficiency is decreased.

Power avoidance is accomplished by solar array switching and suboptimal solar array pointing. Switches can be used to turn off various strings of solar cells, thereby eliminating excess power. Articulated arrays increase the sun angle and reduce the amount of power produced to an acceptable level. This method requires a more complicated solar array pointing algorithm, but it is useful in certain situations. The final power regulation scheme may be a combination of the methods mentioned above. The thermal and attitude control subsystems are most affected by the power regulation method selected.

If increased power subsystem efficiency is required, the simplicity of the DET method is foregone and a power matching system is inserted between the solar array and batteries. A form of series regulation is used to match the impedences of the array and battery/load combination so that maximum power is transferred to the load.

Mission success depends on the adequate performance of the power subsystem. Numerous issues arise when discusssng power subsystem configuration. Several of the more pressing issues are: the load or equipment duty cycle, single vs multiple bus, bus voltage, regulated vs unregulated bus and total power subsystem mass. A measure of performance typically used is watts/kg.

Communications and Data Handling Subsystem

KDPs that influence the design of the communications and data handling (C&DH) subsystem are mission operations/autonomy, orbit, attitude reference, attitude pointing accuracy, ground support system, data rates and security. System-level analyses required to adequately size the C&DH subsystem include determining the EIRP of the spacecraft transmit/receive antenna, size of the antenna aperture, RF amplification and the amount of data storage required.

Amount of Data Storage

The amount of data that must be transmitted to the ground via the spacecraft communications link is a function of the payload data acquisition rate, orbit, ground support system and the degree of spacecraft autonomy. Given the orbital parameters and the location of the ground support stations, the number of stations overflown in a day is estimated. A certain percentage of these station passes is used to transmit data to the ground. This percentage is an expression of spacecraft autonomy. If only a few passes per day are allowed, the vehicle is more autonomous than if many passes per day are acceptable. The amount of time per day that a spacecraft is acquiring data can be estimated and multiplied times the acquisition data rate to assess the amount of data that must be stored. The playback data rate can then be approximated by dividing the amount of stored data by the time available to transmit to the ground.

Effective Isotropic Radiated Power for the Transmit Antenna Trade Study

Several options for spacecraft transmit antenna EIRP can be developed. The user's manuals for most ground support systems specify the ground receiver antenna gain, noise temperature, frequency band(s) and available modulation coding techniques. The orbital parameters coupled with the location of the ground stations yields the slant range from the spacecraft to the ground station. A simple calculation provides the free-space transmission loss. This information is combined to determine required spacecraft transmitter EIRP. Trade-offs are conducted among various modulation coding schemes.

Antenna Gain vs RF Amplification Trade Study

The spacecraft transmit antenna EIRP is the product of the antenna gain and transmitter power. The trade-off here is between a high gain, directional antenna and increased RF amplification. If the attitude reference and pointing accuracy permit, a highly directional antenna may be selected in lieu of increased amplification; otherwise, an omnidirectional higher power system may be required. Antenna gain is a function of antenna size and channel frequency. The result of this trade study is the antenna aperture size and an estimate of the RF amplification required to provide the desired link margin.

Propulsion Subsystem

KDPs that impact the propulsion subsystem design are the orbit, launch vehicle, spacecraft mass at lift-off and the change in velocity required for orbit insertion, station-keeping and maneuvering. The attitude control technology selected directly affects the propulsion delta V options. Trade studies are conducted to determine the best propulsion technology for the various phases of the mission, propellant mass and tank/motor sizes and the appropriate thruster sizes, if required.

Propulsion Technology Trade Studies

The primary trade-off is between solid or liquid propulsion technology. The measure of propellant performance is specific impulse, I_{sp}. Performance is improved and less mass of propellant is required with increased I_{sp}. Liquid propellants generally have higher I_{sp}s, but solid propellant systems are less complex and, perhaps, more reliable, because they do not require the plumbing and valve schemes associated with liquid systems. Liquid systems can be throttled on and off, whereas solid systems are ignited and operate until burnout. Solids do not have a restart capability. Liquid engines can be gimbaled to provide direction thrust control, but most existing solid motors do not have thrust vector control.

A solid propellant motor is a highly reliable choice for orbit insertion but is not appropriate for on/off station-keeping and maneuvering operations. Liquid propellant systems function equally well for orbit insertion and on-orbit operations.

In many missions the propulsion technology is mixed. A solid propellant motor is selected for orbit insertion and a liquid propellant or momentum-based system is chosen for on-orbit operations. Other missions may rely on a liquid system to meet all delta V requirements.

If a liquid propellant is selected for on-orbit operations, a choice must be made between a monopropellant and bipropellant system. Monopropellants require a catalyst to ignite and produce the thrust, whereas bipropellants ignite upon mixing. A bipropellant system requires more plumbing but no ignition system. Both schemes have a limited life based on the amount of propellant carried into orbit.

Momentum-based maneuvering systems enjoy an unlimited life and can provide finer impulse resolution than propellant-based systems. The ultimate configuration is based on design life and cost considerations.

Maintaining orbit parameters such as attitude and inclination is called station-keeping. Station-keeping requires a translation of the spacecraft, so momentum systems are not appropriate. Thrusters usually provide the orbit maintenance capability.

Propellant Mass/Tank Size Trade Studies

These trade studies attempt to determine the appropriate size of motor or tank required and the amount of propellant necessary to best meet mission requirements. Since most solid motors burn until propellant depletion, the correct mass of propellant for orbit insertion must be determined.

The appropriate mass of liquid propellant must be calculated to meet mission requirements and provide a margin of safety. In addition, the thrusters must be sized to provide the proper resolution of control authority. The final configuration is based on propellant mass, reliability and cost considerations. Once the propulsion technology and motor/tank sizes are determined, the structure subsystems engineer can develop preliminary designs.

Configuration and Structure

KDPs that have the most influence on the configuration and structure of the spacecraft are the launch vehicle, spacecraft mass at lift-off and configuration sensitivities. The spacecraft configuration/structure must also accommodate requirements and constraints dictated by all other subsystems. The three trade studies that are typically performed are configuration, type of structure and structural materials.

Configuration Trade Study

Trade studies are performed to identify the spacecraft configuration that best satisfies the requirements and constraints on the system. The maximum volume of the satellite is dictated by the size of the expendable launch vehicle shroud or the space shuttle cargo bay. Configuration sensitivities include FOV and structural stiffness requirements for mounting the payload, instruments and sensors. Spin balance, spacecraft inertia ratio, equipment accessibility and thermal control requirements further restrict configuration options. Various configurations are developed in conjunction with structural concepts. A successful spacecraft configuration/structure accommodates the conflicting requirements at the lowest cost.

Structure Trade Study

Structural concepts are developed to support specified spacecraft configurations. The options provide the stiffness required to accommodate the launch phase vibration and acoustic environments as well as the real estate necessary to mount the payload and subsystem equipment. Thermal conductivity and equipment access requirements are also addressed.

Several basic types of structures are used in spacecraft construction. A monocoque structure is usually a straight conical section constructed of sheet metal. It provides support for heavy loads because it provides uniform load paths. It is easily manufactured but tends to be heavy compared to other alternatives. Reinforced monocoque structures provide additional load carrying capability and are typically used as an interface adapter between the launch vehicle and the spacecraft.

Aluminum honeycomb structures are constructed by placing a light aluminum honeycomb material between two flat aluminum sheets. Numerous structural configurations are possible but fabrication is more costly than with the monocoque method. The result is a moderately light, stiff structure with excellent thermal conductivity characteristics and flexible mounting opportunities. This type of construction is typically used to build equipment bays and deployed solar array structures.

Tubular truss structures are easily constructed and are fairly light. They provide excellent load paths for extended structures but reduced thermal conductivity and limited mounting opportunities restrict their usage. They can be augmented with various plates and panels to improve their usefulness.

A typical spacecraft structure incorporates several of these methods. A honeycomb structure may be used to house equipment, and honeycomb, monocoque or semi-monocoque and tubular truss construction is used for load bearing elements.

Structural Material Trade Study

Conventional, lower-cost materials, such as sheet metal made of aluminum, are used for the structure of most spacecraft. When potential weight and/or thermal problems are identified, beryllium and graphite/epoxy composite materials may be used to augment the structural design. They may provide the required thermal characteristics with an accompanying weight savings. Graphite/epoxy composite materials are about 35 percent lighter than aluminum for the same volume of material and their modulus of elasticity is two to three times that of aluminum. They provide a necessary option, but are usually avoided in spacecraft design because the material and attendant tooling capability are expensive. These composite materials can cost two to ten times more than aluminum, depending on the complexity of the structure.

Once the configuration and structure of the spacecraft have been determined, a meaningful preliminary multinode stress and thermal analysis can be performed.

Thermal Control Subsystem

KDPs that affect the design of the thermal control subsystem are the orbit, spacecraft average power, attitude reference and design life. Many thermal control design considerations are driven by decisions made in other subsystems. For example, the thermal control problem is eased by selecting a spinning vs a fixed-body attitude control scheme because the sun's radiation is distributed more evenly throughout the spacecraft body. The spacecraft duty cycle also has an impact on the severity of the thermal control problem.

Batteries require fairly cool operating temperatures, whereas propellants usually require a warm environment. The thermal control subsystem must maintain all equipment within the temperature range dictated by the temperature specifications of the spacecraft equipment. Given the orbit, spacecraft average power and radiation intensities from the sun, Earth radiation and albedo, the size of the deep space radiator required to maintain a specified temperature for a single-node spacecraft can be calculated. This is similar to determining the required size of a flat plate solar array oriented normal to the sun's rays. It brackets the problem, but this alone cannot be relied on to identify potential problems.

The thermal control scheme selected depends almost entirely upon the spacecraft configuration, orientation and equipment duty cycles. Ideally, the thermal environment can be maintained using passive thermal control techniques, but in most practical situations a combination of passive and active techniques is employed. A key to an efficient thermal control subsystem is to inject passive thermal control philosophies into the spacecraft configuration at the beginning of the design process.

Passive Vs Active Thermal Control Trade-off

This trade-off decision is not usually made at the beginning of the design process. Unless there is an overriding consideration that requires active thermal control methods, passive techniques are assumed. A payload with a mix of extremely high ($> 40^{o}$ C) or extremely low ($< 0^{o}$ C) temperature requirements or extremely tight temperature tolerances (3-5^{o} C) is a candidate for active thermal control. Obviously the decision is program-dependent, but if a payload has extreme temperature requirements, the cost of the performance analysis for a passive scheme may be greater than the cost of the active control components.

Passive thermal control techniques include, but are not limited to, surface coatings (paint and multilayer insulation) with various emittance and absorptivity characteristics, extended radiators, phase-change materials and component placement. Many successful passive thermal control schemes have been documented using a combination of these techniques. Passive methods are attractive because they tend to be simpler, lighter, less power-consuming and, therefore, less expensive than active methods.

Active thermal control methods include moveable surfaces (louvers), heaters, variable conductance heat pipes, controllable surface finishes (e.g., LCDs) and component duty cycling. These methods are used to satisfy spacecraft temperature requirements where passive methods are not feasible. Active thermal control methods typically require additional mass and power and tend to be more complex and expensive than passive methods.

This concludes the discussion on subsystem trade studies, but keep in mind that the preliminary baseline spacecraft configuration is defined when all stated requirements are satisfied and the cost and/or time required to complete the project is minimized. Many iterations are necessary. The results of certain trade studies may indicate that one or several key design parameters must be changed, or the impact to the performance or cost of the system would be unacceptable. In this case derived or assumed requirements may be changed, or the customer's stated requirements may be challenged and corrected, if necessary.

Conclusions - Preliminary System Design Process

The preliminary system design process, depicted in Figure 3-2, describes the elements of the preliminary mission and spacecraft design activities. Insights pertaining to what should be accomplished and when are provided, but the details of how to perform the stated tasks are provided elsewhere. The *BASD Spacecraft Systems Design Handbook* presents many approximations and approaches that may be useful in the process.

The purpose of preliminary mission design is to determine and/or verify the appropriate values of the KDPs by integrating the payload, launch vehicle and mission planning requirements.

Fifteen key design parameters drive the configuration and cost of the spacecraft, because they are used as a basis to perform system and subsystem trade studies. Many design parameters are specified in the RFP (stated requirements), but others (derived and assumed requirements) are estimated to provide the spacecraft design team with enough data to perform the preliminary spacecraft design activity. The derived and assumed parameters must be carefully scrutinized to ensure that they do not unnecessarily constrain the design, and they must be continuously updated to reflect the results of the system and subsystem trade studies.

The preliminary spacecraft design process is discussed, using Figure 3-3 as a guide. If the process is performed by one or two people for a quick response to a study request, the flow indicated is appropriate. If the process is executed by a proposal design team, the key design parameters should be identified so the team can perform the trade studies in parallel.

The key design parameters that affect each subsystem are identified and the most important system-level trade studies are discussed. The emphasis is on the interaction between subsystems with reference to the implementation within subsystems, as necessary.

In the following chapter, the details of the design and implementation of the attitude determination and control subsystem of a specific spacecraft are discussed.

4.0 Starscan Attitude Control Subsystem Development

This chapter documents the analysis and design of the Starscan attitude determination and control subsystem (ADACS). Following a discussion of the Starscan mission and requirements, the primary ADACS elements are described.

The linearized equations of motion for the momentum-bias satellite are developed and a nominal state-space description is provided. Two linear control structures and associated design procedures are examined using both classical and modern control methodologies.

Several nonlinear saturation torque limiting schemes are evaluated in conjunction with the proposed control structures, and overall closed-loop system performance is examined. The advantages and disadvantages of each design are discussed.

The Starscan program was terminated several months after the contract was awarded because of budget cuts in the Government program office. Had the contract continued, additional work would have included developing a state estimator (constant gain Kalman filter) and analyzing the entire closed-loop system including the nominal plant, controller and state estimator. This chapter describes the process used to complete general control system design and analysis.

Starscan Mission Overview

Starscan is a scientific satellite program funded by the Department of Defense. The purpose of the one-to-three-year mission is to provide a data base sufficient for the final design of operational space systems for a variety of possible defense applications requiring complex gamma ray and neutron detection capabilities.

The payload to be supported by the spacecraft bus is comprised of two experiments and a source deployment subsystem (SDS). The experiments, the advanced nuclear gamma ray analysis system (ANGAS) and the advanced neutron detection and analysis system (ANDAS), are sponsored by the Defense Advanced Research Projects Agency and are built by the Lockheed Missiles and Space Company.

The ANGAS is designed to demonstrate and evaluate the performance of a number of new approaches and methods of gamma ray detection in space. The instrument will explore advances in high-sensitivity fine-energy resolution gamma ray spectroscopy, gamma ray imaging and associated on-board data analysis. Similarly, the ANDAS will investigate neutron detection in space, exploring advances in increased resolution neutron spectroscopy, neutron radiography, neutron imaging and smart post-processing analysis.

The SDS is a retractable radioactive source package that is deployed by a coilable longeron boom mechanism. The source package provides calibration signals to the ANGAS and ANDAS instruments. The boom mechanism is designed to position the source package two to five meters above the ANGAS and ANDAS. The lightweight source package, located on the tip of the boom, is designed to have a relative rotation with respect to ANGAS of up to 5 rpm about an axis that is coaligned with the ANGAS look direction and is centered on the ANGAS detection array.

A diagram depicting the major components of the spacecraft bus, experiments and body coordinate frame is shown in Figure 4-1. The body coordinate frame, shown as the 1, 2 and 3 axes, will also be referred to as the roll, pitch and yaw axes, respectively.

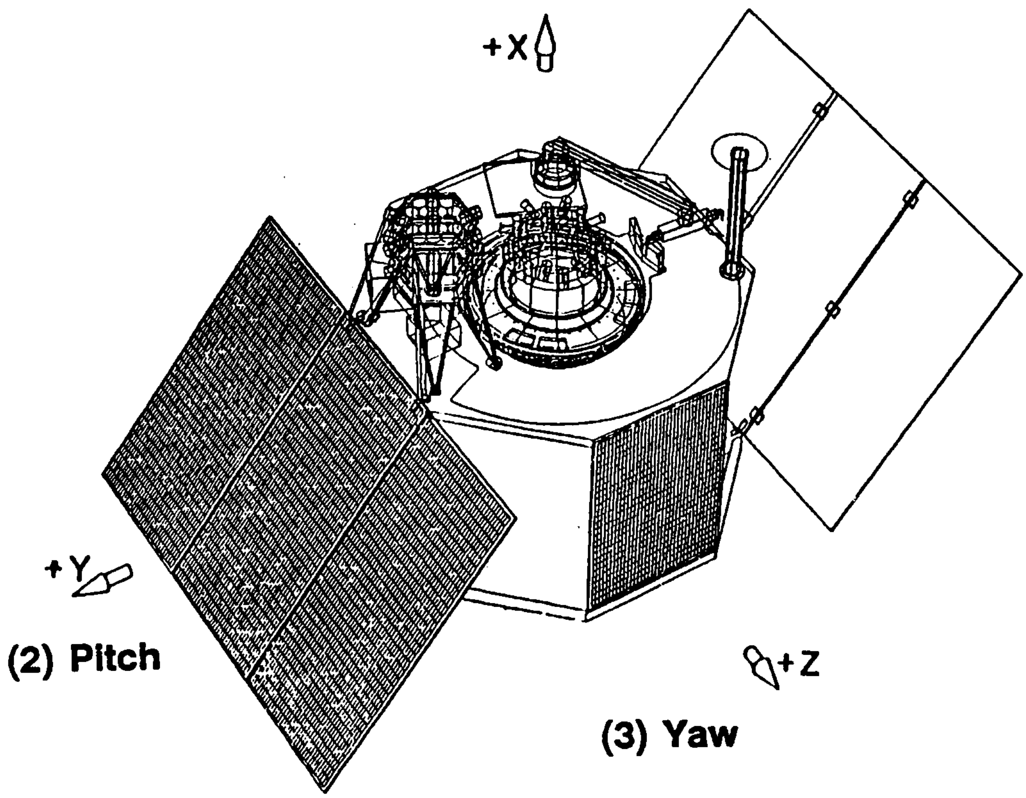

Figure 4-1 Starscan Spacecraft

The orbit selected for the mission is a 546-km (295-nmi) sun-synchronous circular orbit inclined at 97.5 degrees to the equatorial plane. The orbital period and frequency are 95.6 minutes and 0.001 rad/sec, respectively.

The final BASD Attitude Determination and Control Subsystem Analysis Report will address five primary modes of operation: attitude acquisition, inertial spacecraft pointing, large angle maneuvers, slow spin about the roll axis and extended boom operations. This report is limited primarily to the linear analysis of the attitude control system for the acquisition and inertial pointing modes, and introduces certain nonlinearities associated with large angle maneuvers.

Attitude Determination and Control Subsystem

Top-level ADACS Requirements

The principal contractual requirements for the attitude control system (ACS) are listed in Table 4-1. The Starscan spacecraft is an all-attitude vehicle; therefore, the ACS should be capable of pointing the roll axis at any point in inertial space.

Table 4-1 ADACS Performance Requirements

Requirement / Mode	Inertial Pointing Mode	Spin Mode	Extended Boom Operations
Accuracy (deg)	<1	<1	<20
Jitter (deg)	<0.25	<0.25	N/A
Rvft (deg/1hr)	<1	<1	<10
Knowledge (deg)	<0.25	<0.25	<10
Roll Rate (RPM)	<30	0.4-30	<5
Roll Rate Accuracy	N/A	±1%	N/A
Duration	Hrs/Days	Days	24-48 hrs

Derived Requirements

Several of the derived requirements are qualitative rather than quantitative and reflect rather general control philosophies. The linear design must be stable and robust to parameter variations. The large angle slew maneuver executed between pointing locations must be as direct as possible (providing quality of motion). The maximum disturbance torque is 5×10^{-4} N-m at a frequency equal to the orbital rate. Tracking of objects is not required.

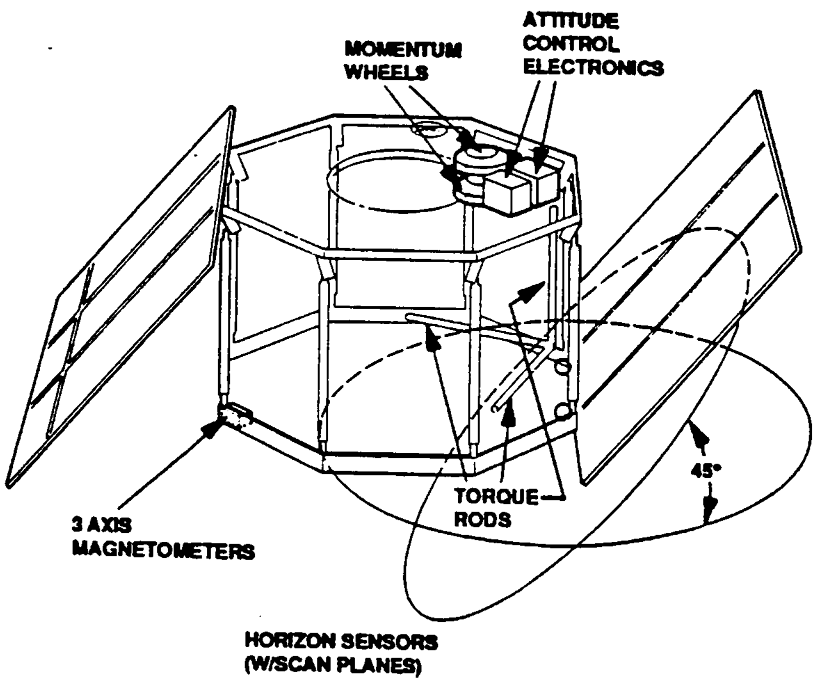

Figure 4-2 ADACS Components

ADACS Description

The Starscan spacecraft, as depicted in Figure 4-2, is a momentum-bias vehicle capable of all-attitude pointing and slow spinning about the ANGAS experiment line of sight (roll axis). The vehicle uses two scanning horizon sensors and vector magnetometers

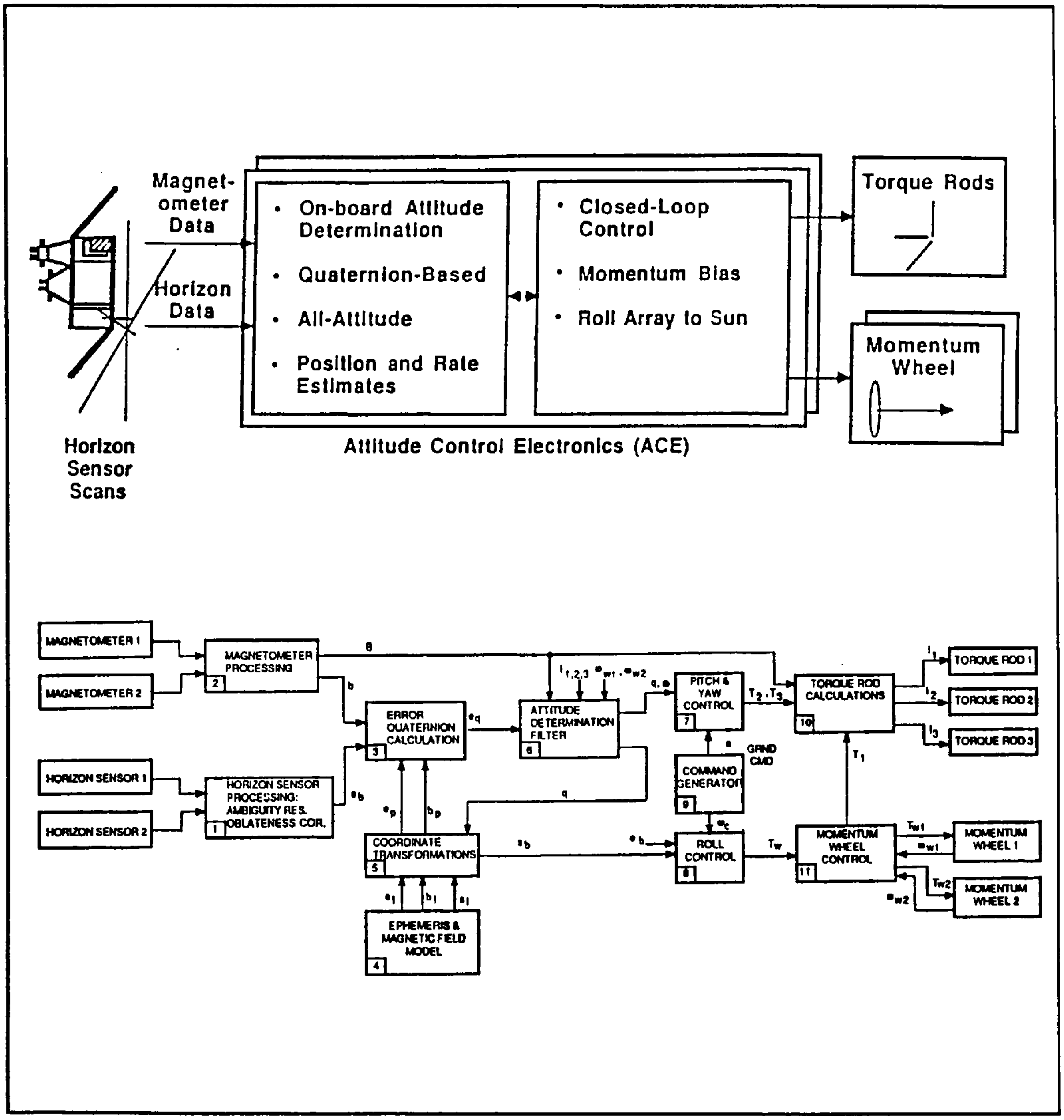

Figure 4-3 ADACS Block Diagram

for attitude determination, and two momentum wheels and three electromagnets (torque rods) for control.

The ADACS is unusual in that it is a three-axis maneuverable momentum-bias system, and the angular rate information is provided by a state estimator rather than by sensors (e.g., gyroscopes). The overall block diagram is shown in Figure 4-3. This design and analysis effort will deal primarily with the highlighted blocks.

Description of the Design and Analysis Activity

The ADACS performs the functions of attitude determination and control in an integral fashion. Although this report will not address the overall nonlinear attitude determination problem, the linear behavior of the determination filter and the closed-loop system will be analyzed.

The design and analysis of the ACS (for the inertial pointing mode) may be accomplished in six basic steps:

1. The nonlinear equations of motion for the spacecraft were developed and linearized, resulting in a nominal state-space plant model.

2. Several potential control structures for the nominal plant were developed assuming full-state information was available.

3. The closed-loop performance of the proposed plant/controller combinations was investigated using both frequency domain and time domain analysis techniques.

4. Several nonlinear torque limiting schemes were included in the closed-loop system proposed in step 3, and the resulting systems were simulated.

5. Based on the results of steps 2 to 4, two potential controllers were selected and a full-state estimator (constant gain Kalman filter) was developed. The closed-loop performance of the entire closed-loop system was investigated, as described above, and the stochastic performance was examined.

6. The best controller/estimator combination was selected, and satisfactory performance was verified via a six-degree-of-freedom Starscan nonlinear simulation.

Steps 1 to 4 were performed and are documented in this chapter. The Starscan contract was terminated by the Government, and steps 5 and 6 were, therefore, not accomplished.

Equations of Motion

In this section the nonlinear Euler equations of motion for the Starscan momentum-bias spacecraft are developed using spacecraft body coordinates. Then the equations are linearized and expressed in state-space form. Nominal parameter values are substituted into the resulting equations and the frequency response of the plant is investigated.

The coordinate frames of interest are the Earth-centered inertial (ECI) and spacecraft body coordinate frames shown in Figure 4-4. The ECI frame is described by X, Y and Z, where the X-axis points from the center of the Earth to the vernal equinox (first point of Aires), the Z-axis points from the center of the Earth through the North Pole and the Y-axis completes the orthogonal set.

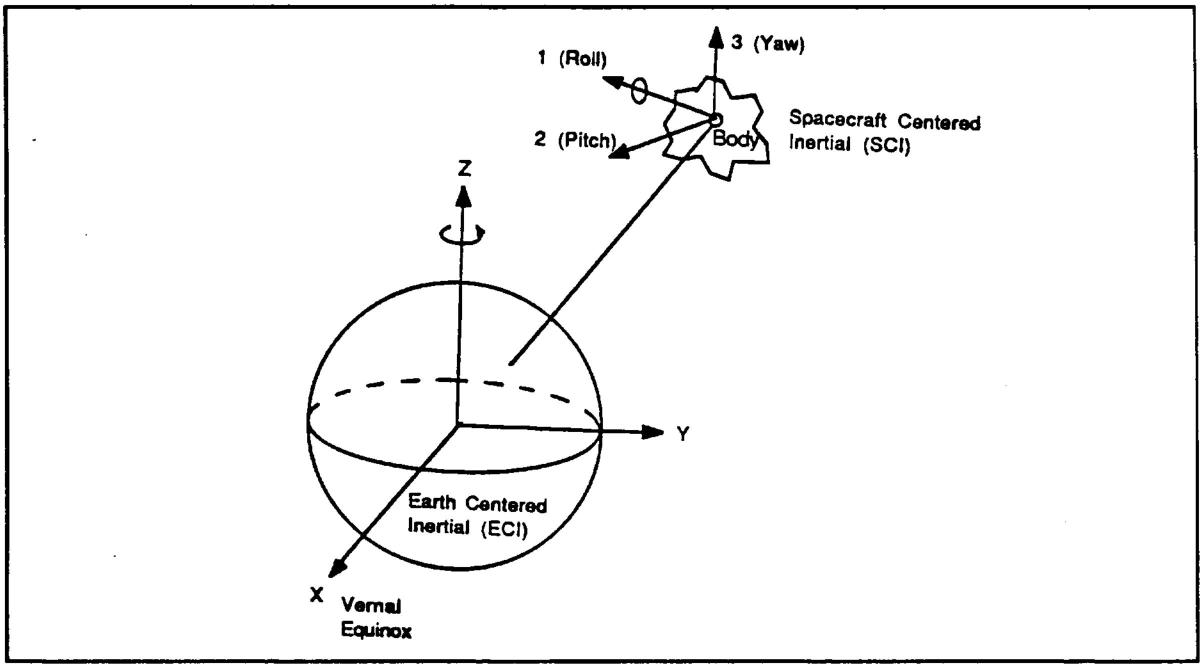

Figure 4-4 Inertial and Body Coordinate Frames of Ref

The Euler dynamic equations of motion in the spacecraft body frame for a momentum-bias spacecraft are derived from various sources [7,8]. The equations are shown below in vector form.

$$I\dot{\underline{\omega}} = T_D - \dot{\underline{h}} - \underline{\omega} \times (I\underline{\omega} + \underline{h}) \tag{4-1}$$

where: h = Angular momentum of the wheel in the 1, 2 and 3 axes

T_D = Disturbance torques about the 1, 2 and 3 axes

$\underline{\theta}$ = Angular displacement about the 1, 2 and 3 axes

$\underline{\omega}$ = $\dot{\underline{\theta}}$, Angular velocity about the 1, 2 and 3 axes

$\dot{\underline{\omega}}$ = Angular acceleration about the 1, 2 and 3 axes

I = Inertia matrix, 3 x 3 diagonal matrix

Ignoring wheel misalignment, there are no components of wheel angular momentum along the 2 and 3 axes. Expanding Equation (4-1) into component form yields the non-linear equations for inertial angular acceleration about each axis in body coordinates.

$$I_1\dot{\omega}_1 = T_1 - \dot{h}_1 - (I_3 - I_2)\omega_2\,\omega_3 \qquad \text{Roll Axis} \tag{4-2}$$

$$I_2\dot{\omega}_2 = T_2 - (-I_3\,\omega_1\,\omega_3 + I_1\,\omega_1\,\omega_3 + h_1\,\omega_3) \qquad \text{Pitch Axis}$$

$$I_3\dot{\omega}_3 = T_3 - (I_2\,\omega_1\,\omega_2 - I_1\,\omega_1\,\omega_2 - h_1\,\omega_2) \qquad \text{Yaw Axis}$$

If the angular velocities are assumed to be very small, as would be the case for an inertially pointed spacecraft, the product of two angular velocities is negligible and the linearized version of Equation (4-2) results.

$$I_1\ddot{\theta}_1 = T_1' - \dot{h}_1 = T_1 \qquad \text{Roll Axis} \quad (4\text{-}3)$$

$$I_2\ddot{\theta}_2 = T_2 - h_1\dot{\theta}_3 \qquad \text{Pitch Axis}$$

$$I_3\ddot{\theta}_3 = T_3 + h_1\dot{\theta}_2 \qquad \text{Yaw Axis}$$

A very convenient result of the linearization is that the roll axis is decoupled from the pitch and yaw (transverse) axes. This will simplify later analysis.

Note that Equation (4-3) is expressed in body coordinates. Typically it is useful to express the satellite motion in inertial coordinates. The transformation from the spacecraft body coordinates ($\underline{r}_b$) to inertial coordinates ($\underline{R}_i$) may be represented by

$$\underline{R}_I = S_{321}\ \underline{r}_b \qquad (4\text{-}4)$$

where S_{321} is the direction cosine matrix for a 3-2-1 Euler angle rotation. For

Ψ = Angle rotation about the Z-axis

θ = Angle rotation about the Y-axis and

ϕ = Angle rotation about the X-axis

the resulting direction cosine matrix is

$$\begin{bmatrix} c\theta c\psi & c\theta s\psi & -s\theta \\ (-c\phi s\psi + s\theta s\phi c\psi) & (c\phi c\psi + s\phi s\theta s\psi) & s\phi c\theta \\ (s\phi s\psi + c\phi s\theta c\psi) & (-s\phi c\psi + c\phi s\theta s\psi) & c\phi c\theta \end{bmatrix} \qquad (4\text{-}5)$$

where, C(angle) and S(angle) represent the cosine and sine of the angle, respectively.

If the motion is restricted to small angles, the following substitutions can be made in Equation (4-5):

$$\cos(\text{angle}) = 1$$

$$\sin(\text{angle}) = \text{angle}$$

$$\text{product of two angles} = 0$$

and the linearized direction cosine matrix emerges as:

$$S_{L321} = \begin{bmatrix} 1 & \psi & -\theta \\ -\psi & 1 & \phi \\ \theta & -\phi & 1 \end{bmatrix} \tag{4-6}$$

For small angles, the product of S_{L321} and r_b is approximately equal to r_b. This result indicates that for the inertial pointing mode, it is possible to adequately analyze the spacecraft motion in body coordinates.

Now the linearized spacecraft model can be developed from the equations in (4-3). Angular velocity is simply the derivative of angular position and the rate of change of momentum is a torque, so the equations become:

$$\ddot{\theta}_1 = T_1/I_1 \qquad \text{Roll Axis} \quad (4\text{-}7)$$

$$\ddot{\theta}_2 = T_2/I_2 - (h_1/I_2)\dot{\theta}_3 \qquad \text{Pitch Axis}$$

$$\ddot{\theta}_3 = T_3/I_3 + (h_1/I_3)\dot{\theta}_2 \qquad \text{Yaw Axis}$$

The linear second-order differential equations in (4-7) can be represented in state-space form by making the following state assignments:

$$X_1 = \theta_1 \qquad X_3 = \theta_2 \qquad X_5 = \theta_3$$

$$X_2 = \dot{X}_1 = \dot{\theta}_1 = \omega_1 \qquad X_4 = \dot{X}_3 = \dot{\theta}_2 = \omega_2 \qquad X_6 = \dot{X}_5 = \dot{\theta}_3 = \omega_3$$

$$\dot{X}_2 = \ddot{\theta}_1 = \dot{\omega}_1 \qquad \dot{X}_4 = \ddot{\theta}_2 = \dot{\omega}_2 \qquad \dot{X}_6 = \ddot{\theta}_3 = \dot{\omega}_3$$

These assignments are particularly useful since they have physical meaning. That is, the x_1, x_3 and x_5 states represent angular displacement about the roll, pitch and yaw axes, respectively and the x_2, x_4 and x_6 states are angular rates about the same axes.

The resulting state-space description of the momentum-bias spacecraft, hereafter referred to as the plant, are shown below in the form $\dot{\underline{x}} = A\underline{x} + B\underline{u}$, $\underline{y} = C\underline{x} + D\underline{u}$.

$$\begin{bmatrix} \dot{X}_1 \\ \dot{X}_2 \\ \dot{X}_3 \\ \dot{X}_4 \\ \dot{X}_5 \\ \dot{X}_6 \end{bmatrix} = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & -h_1/I_2 \\ 0 & 0 & 0 & 0 & 0 & 1 \\ 0 & 0 & 0 & h_1/I_3 & 0 & 0 \end{bmatrix} \begin{bmatrix} X_1 \\ X_2 \\ X_3 \\ X_4 \\ X_5 \\ X_6 \end{bmatrix} + \begin{bmatrix} 0 & 0 & 0 \\ 1/I_1 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 1/I_2 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 1/I_3 \end{bmatrix} \begin{bmatrix} T_1 \\ T_2 \\ T_3 \end{bmatrix} \qquad (4\text{-}8)$$

$$\underline{Y} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} X_1 \\ X_2 \\ X_3 \\ X_4 \\ X_5 \\ X_6 \end{bmatrix} + \underline{0} \begin{bmatrix} T_1 \\ T_2 \\ T_3 \end{bmatrix}$$

where:

h_1	= Angular momentum of the momentum wheel
I_1, I_2, I_3	= Moment of inertia for the 1, 2 and 3 axes, respectively
T_1, T_2, T_3	= Control torques for the 1, 2 and 3 axes, respectively

Figure 4-5 is the plant block diagram.

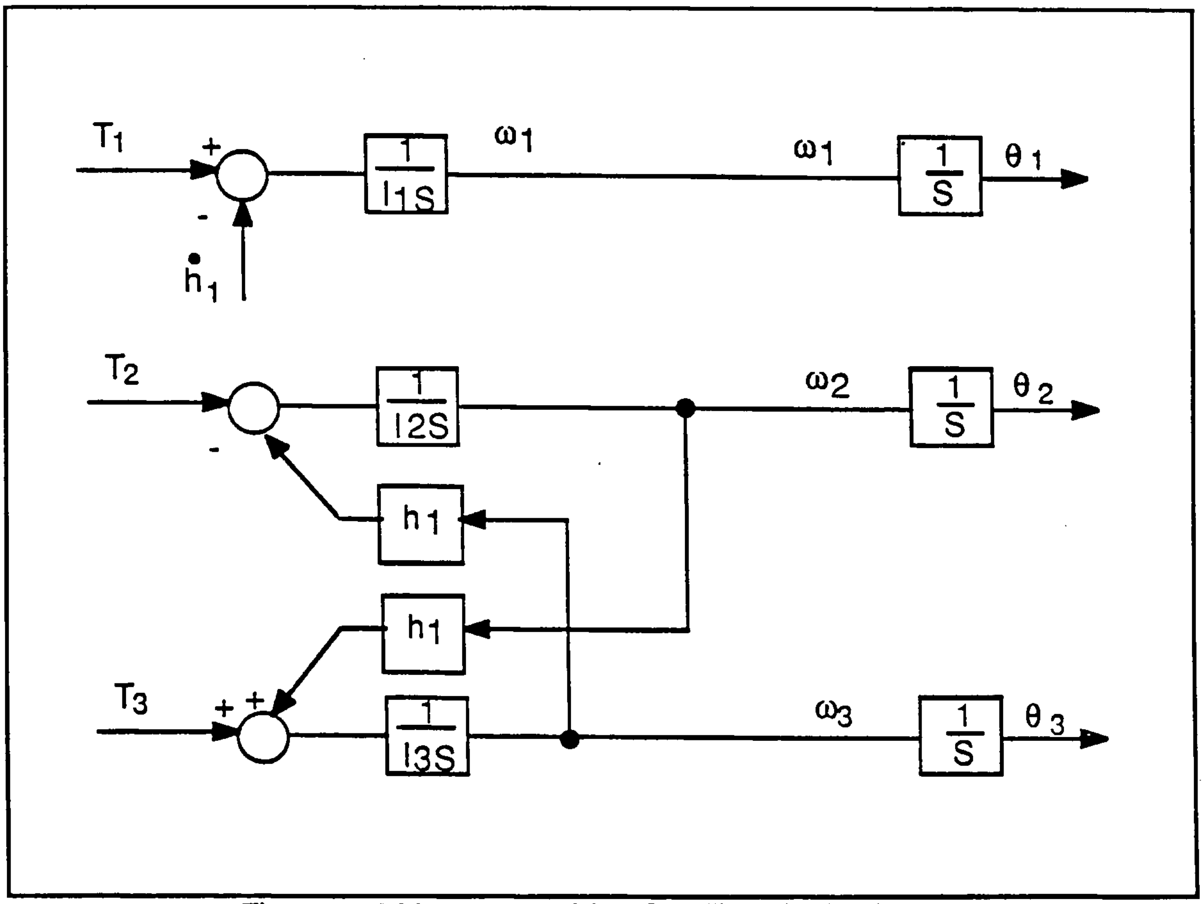

Figure 4-5 Momentum-bias Satellite Block Diagram

The plant parameters are the inertias for the roll, pitch and yaw axes and the angular momentum of the momentum wheel. The nominal values for the plant parameters are:

$$h_1 = 50 \text{ N-m-s}$$
$$I_1 = 1{,}000 \text{ kg m}^2$$
$$I_2 = 900 \text{ kg m}^2$$
$$I_3 = 1{,}100 \text{ kg m}^2$$

The resulting state-space description of the nominal plant is

$$\dot{\underline{x}} = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & -.0555 \\ 0 & 0 & 0 & 0 & 0 & 1 \\ 0 & 0 & 0 & .0454 & 0 & 0 \end{bmatrix} \underline{x} + \begin{bmatrix} 0 & 0 & 0 \\ .0010 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & .0011 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & .0009 \end{bmatrix} \underline{T} \qquad (4\text{-}9)$$

$$\underline{Y} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 \end{bmatrix} \underline{x}$$

Note that the plant is both controllable and observable. This may be verified by checking that the rank of both the controllability matrix [A,B] and observability matrix [C,A] is six.

The characteristic equation for the uncontrolled plant is:

$$s^4 (s^2 + h^2/I_2I_3) = 0 \qquad (4\text{-}10)$$

and the root locus is shown in Figure 4-6.

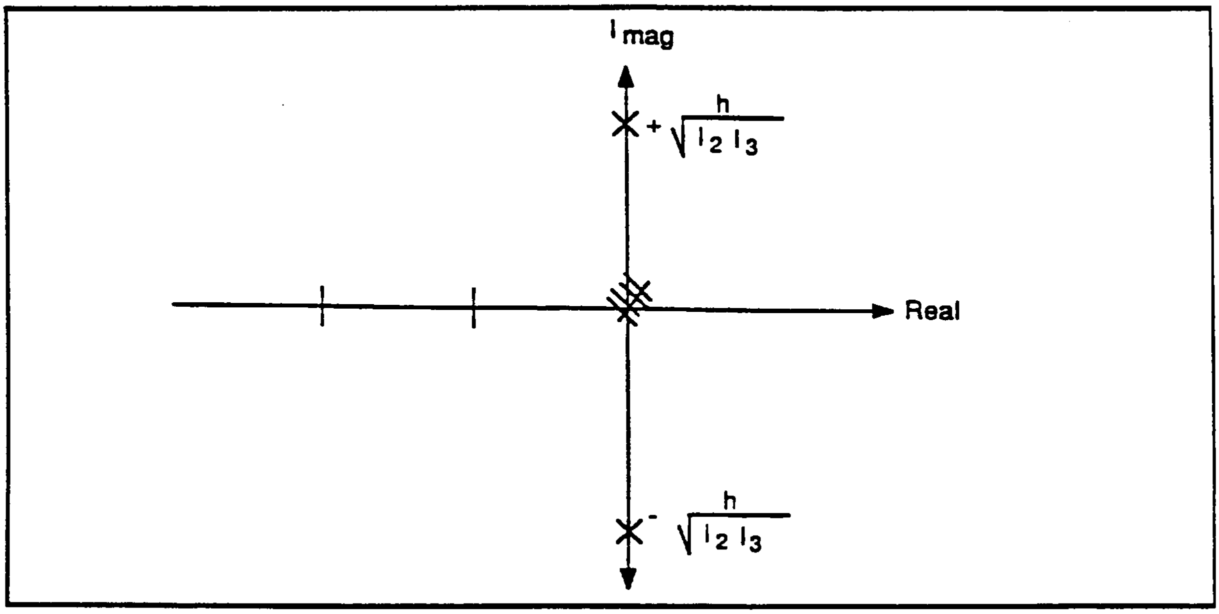

Figure 4-6 Nominal Plant Root Locus

The $s^2 + h^2/I_2I_3$ term in the characteristic equation yields the plant's inherent nutation frequency. For the nominal plant, the nutation frequency is at 0.05 rad/sec as evidenced by the Bode plot of the θ_2/T_2 and θ_2/T_3 transfer function shown in Figure 4-7.

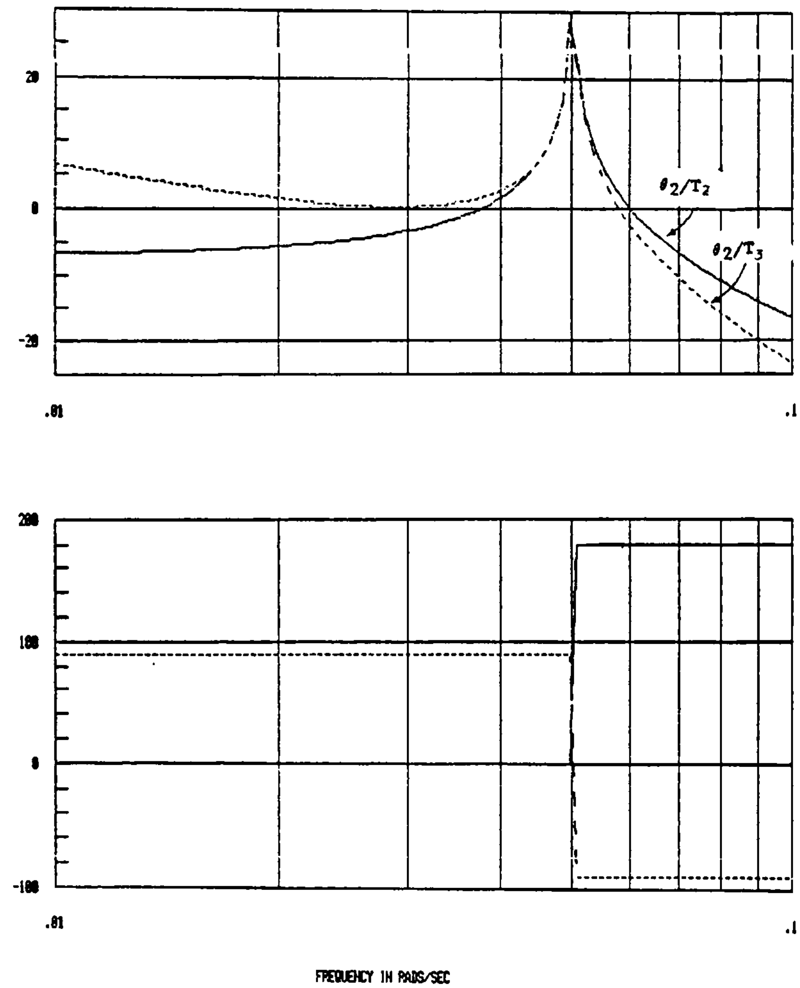

Figure 4-7 Bode Plot of the Nominal Plant

Next, the appropriate controller/control law must be selected for this system.

Potential Controller Structures

In the following effort several control laws are examined with respect to their transfer functions, torque disturbance rejection capability and time response to several different commanded inputs and disturbance torques.

Recall that angular rates are not directly observed. For control design, however, we rely on a state estimator to provide full-state feedback information so the output matrix, C, is assumed to be a 6 x 6 identity matrix. Also note that the roll axis is decoupled from the transverse axes. Therefore the roll axis is designed and analyzed separately from the pitch and yaw axes.

The model used for the analysis and design of the transverse axes is shown in Figure 4-8. Note that $\underline{T}$ is the control torque, $\underline{T}_d$ is the disturbance torque, $\underline{T}_A$ is the total applied torque, $\underline{x}$ is the actual state vector and $\underline{x}_r$ is the reference state vector. The matrix K is

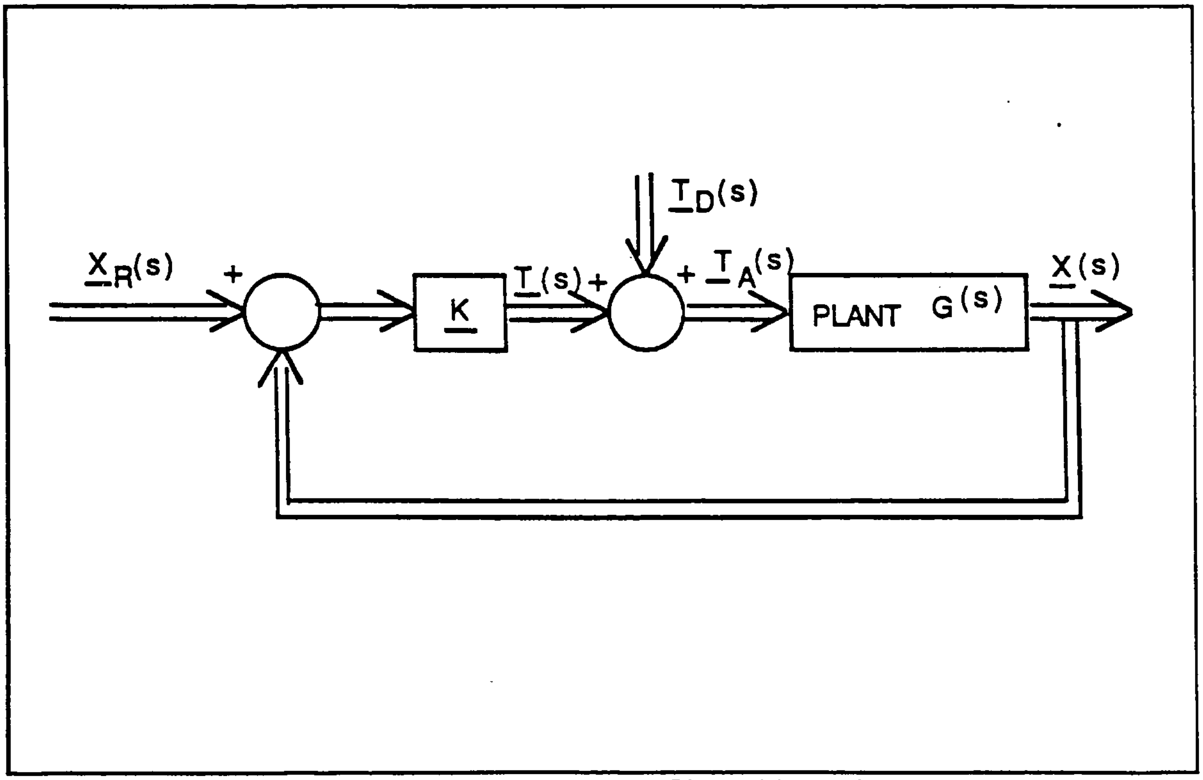

Figure 4-8 Transverse Axes Closed-loop System

a 2 x 4 gain matrix that multiplies the error vector yielding the control law. It has the following form:

$$K = \begin{bmatrix} k_{21} & k_{22} & k_{23} & k_{24} \\ k_{31} & k_{32} & k_{33} & k_{34} \end{bmatrix} \tag{4-10}$$

The resulting control law has the following form:

$$\begin{aligned} T_2 &= k_{21}\,\theta_{2e} + k_{22}\,\omega_{2e} + k_{23}\,\theta_{3e} + k_{24}\,\omega_{3e} \\ T_3 &= k_{31}\,\theta_{2e} + k_{32}\,\omega_{2e} + k_{33}\,\theta_{3e} + k_{34}\,\omega_{3e} \end{aligned} \tag{4-11}$$

where θ_{ie} and ω_{ie} are the angular displacement and angular rate error signals, respectively.

Several well-known control structures for spacecraft can be derived from Equation (4-11). If k_{23}, k_{24}, k_{31} and k_{32} are set equal to zero, a zero-momentum controller results. In this control scheme the control law for a particular axis depends only on its angle and rate errors. If k_{21}, k_{24}, k_{32} and k_{33} are set equal to zero, a pure precession control law results. In this approach the control law for a particular axis, say, T_2, depends on its rate error and the cross-axis angle error. If none of the k's are forced to zero, a hybrid of these two classic approaches results. The precession and hybrid controllers are examined in this report.

Both classical and modern control analysis and design techniques are used for this problem. Classical techniques are used to determine the gains for the pure precession controller, and the linear quadratic regulator approach is used to select gains for the hybrid controller.

Using the transverse axes block diagram in Figure 4-9, a signal flow graph was developed as shown in Figure 4-10. Mason's gain rule was used in conjunction with the signal flow graph to determine the overall characteristic equation for a full gain matrix, K.

$$\begin{aligned} & s^4 + \left(\frac{K_{22}I_3 + k_{34}I_2}{I_2I_3}\right)s^3 \\ & + \left(\frac{k_{21}I_3 + k_{33}I_2 + k_{22}K_{34} + h^2 - k_{32}h + k_{24}h - k_{32}k_{24}}{I_2I_3}\right)s^2 \\ & + \left(\frac{k_{21}k_{34} + k_{22}k_{33} + k_{23}h - k_{31}h - k_{32}k_{23} - k_{24}k_{31}}{I_2I_3}\right)s + \left(\frac{k_{21}k_{33} - k_{31}k_{23}}{I_2I_3}\right) = 0 \end{aligned} \tag{4-12}$$

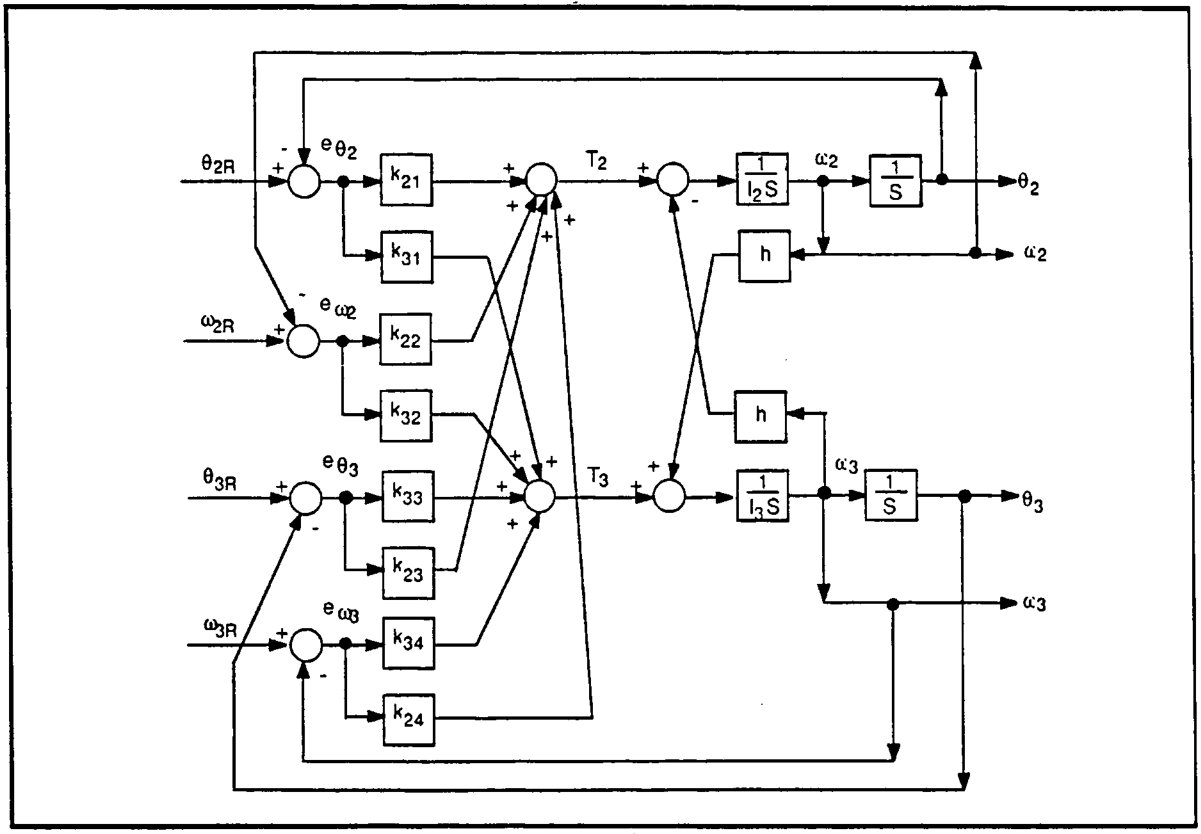

Figure 4-9 Transverse Axes Block Diagram

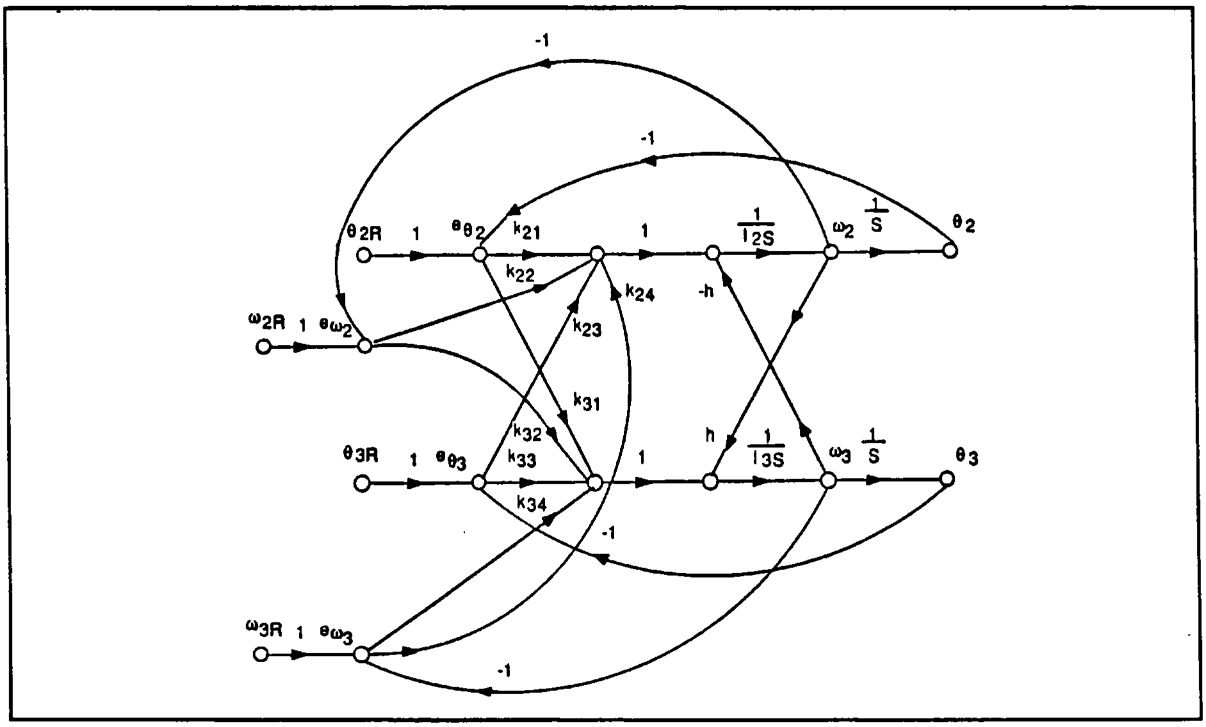

Figure 4-10 Transverse Axes Signal Flow Graph

Precession Controller

The closed-loop transfer functions and characteristic equation (Δ_P) in variable form for the precession controller are shown in Equation (4-13).

(4-13)

$$\Delta_P = s^4 + \frac{K_{22}I_3 + K_{34}I_2}{I_2I_3}s^3 + \frac{K_{22}K_{34} + h^2}{I_2I_3}s^2 + \frac{(K_{23}-K_{31})h}{I_2I_3}s + \frac{(-K_{31}K_{23})}{I_2I_3}$$

$$\theta_2/\theta_{2R} = \frac{\frac{K_{23}h}{I_2I_3}s \; \frac{-K_{31}K_{23}}{I_2I_3}}{\Delta P} \qquad \theta_3/\theta_{2R} = \frac{-s\left(\frac{K_{23}}{I_3}s + \frac{K_{22}K_{23}}{I_2}\right)}{\Delta P}$$

$$\theta_2/T_{d2} = \frac{s\left(\frac{1}{I_2}s + \frac{K_{34}}{I_2I_3}\right)}{\Delta P} \qquad \theta_3/T_{d2} = \frac{\frac{h}{I_2I_3}s + \frac{K_{23}}{I_2I_3}}{\Delta P}$$

The gain matrix for the precession controller is:

$$K = \begin{bmatrix} 0 & k_{22} & k_{23} & 0 \\ k_{31} & 0 & 0 & k_{34} \end{bmatrix} \tag{4-14}$$

where, for stability of the closed-loop system, all coefficients of s must be positive and the following constraints on values of K result:

(4-15)

$K_{22}I_3 + K_{34}I_2 > 0$

$K_{22}K_{34} + h^2 > 0$

$(K_{23}-K_{31})h > 0$

$-K_{31}K_{23} > 0 \longrightarrow$ either $K_{31} < 0,\ K_{23} > 0$ or $K_{31} > 0,\ K_{23} < 0$

The gain selection procedure for the precession controller is fairly straightforward. In general, the desired pole locations are identified and the resulting desired characteristic equation is developed. Then the coefficients of like powers of s of the desired characteristic equation and the variable form characteristic equation are equated, and the appropriate values of k are determined.

The desired fourth order characteristic equation can be divided into two second-order equations, the biquadratic form, each having a desired damping ratio (ζ) and frequency (ω). This results in a desired characteristic equation of the form:

$$s^4 + 2(\zeta_1 \omega_1 + \zeta_2 \omega_2)s^3 + (\omega_1^2 + \omega_2^2 + 4\zeta_1 \zeta_2 \omega_1 \omega_2) s^2$$
$$+ 2(\zeta_1 \omega_1 \omega_2^2 + \zeta_2 \omega_1^2 \omega_2)s + \omega_1^2 \omega_2^2 \qquad (4\text{-}16)$$

Equating coefficients of like powers of s in Equations (4-13) and (4-16) and substituting $\alpha = \omega_1\omega_2$, $\beta = \omega_1+\omega_2$, $\omega_n^2 = h^2/I_2I_3$ and $\zeta_1 = \zeta_2 = \sqrt{2}/2$, results in:

$$\frac{k_{22}}{I_2} + \frac{k_{34}}{I_3} = \sqrt{2}\,\beta \qquad (a)$$
$$\left(\frac{k_{22}}{I_2}\right)\left(\frac{k_{34}}{I_3}\right) + \omega_n^2 = \beta^2 \qquad (b)$$
$$\frac{(k_{23}-k_{31})h}{I_2I_3} = \sqrt{2}\,\alpha\beta \qquad (c)$$
$$\frac{-k_{31}k_{23}}{I_2I_3} = \alpha^2 \qquad (d)$$
$$(4\text{-}17)$$

Squaring Equation (17a) and subtracting twice Equation (17b) results in:

$$\left(\frac{k_{22}}{I_2}\right)^2 + \left(\frac{k_{34}}{I_3}\right)^2 = 2\omega_n^2 \qquad (4\text{-}18)$$

The angular rate gains (k_{22} and k_{34}) are independent of ω_1 and ω_2. They depend only on the plant parameters, h_1, I_2 and I_3.

A slightly different approach yields a more interesting result. Solving Equation (17b) for (k_{34}/I_3), substituting into Equation (4-17a) and solving for k_{22}/I_2:

$$\frac{k_{22}}{I_2} = \frac{\sqrt{2}}{2}(\omega_1 + \omega_2) \pm \sqrt{\omega_n^2 - \frac{(\omega_1 + \omega_2)^2}{2}} \tag{4-19}$$

$$\frac{K_{34}}{I_3} = \frac{\sqrt{2}}{2}(\omega_1 + \omega_2) \mp \sqrt{w_n^2 - \frac{(\omega_1 + \omega_2)^2}{2}}$$

Note there is no acceptable solution when

$$\frac{(\omega_1 + \omega_2)^2}{2} > \omega_n^2 \tag{4-20}$$

A similar result is achieved by solving Equations (4-17c) and (4-17d) for:

$$K_{23} = \frac{\sqrt{2}}{2h}\omega_1\omega_2(\omega_1 + \omega_2)I_2I_3 \pm \frac{\omega_1\omega_2 I_2 I_3}{h}\sqrt{\frac{(\omega_1 + \omega_2)^2}{2} - \omega_n^2} \tag{4-21}$$

$$K_{31} = \frac{\sqrt{2}}{2h}\omega_1\omega_2(\omega_1 + \omega_2)I_2I_3 \mp \frac{\omega_1\omega_2 I_2 I_3}{h}\sqrt{\frac{(\omega_1 + \omega_2)^2}{2} - \omega_n^2}$$

In this case, there is no acceptable solution if

$$\frac{(\omega_1 + \omega_2)^2}{2} < \omega_n^2 \tag{4-22}$$

Comparison of Equations (4-20) and (4-22) implies that for the case where ζ_1 and $\zeta_2 = \sqrt{2}/2$

$$\omega_n^2 = \frac{(\omega_1 + \omega_2)^2}{2} \tag{4-23}$$

Precession Controller Design Procedure

The design procedure for the case where $\zeta_1 = \zeta_2 = \sqrt{2}/2$ is as shown below for

$$\omega_n^2 = h^2/I_2I_3$$

(1) Select a desired ω_1 which is related to settling time, $\sigma_1 = \zeta_1\omega_1 = \sqrt{2}/2\omega_1$

(2) Compute $\omega_2 = \sqrt{2}\omega_n - \omega_1$ (4-24)
where, $0 < \omega_1 < \sqrt{2}\omega_n$ and $0 < \sigma_1 < \omega_n$

(3) Calculate the resulting gains

(4-25)
(4-26)

In this specific case the potential pole locations are shown in Figure 4-11. The possible pole locations are restricted to the 45° line and must be within the bounds of the dashed box. Note that the box limit is driven by plant parameters. If I_2 and/or I_3 increase significantly (as may occur with a deployed payload or boom extension), ω_n would decrease and the potential operating range would decrease significantly from that shown in the figure. Precession control would then become less attractive.

An alternate approach to gain selection is to remove the constraint that $\zeta_1 = \zeta_2 = \sqrt{2}/2$ and allow $\zeta_1 \neq \zeta_2$. Assuming symmetric gains, define the parameters a and b as:

$$a = k_{22}/I_2 = k_{34}/I_3 \tag{4-27}$$
$$b = k_{23} = -k_{31}$$

Then,

$$a = \zeta_1\omega_1 + \zeta_2\omega_2 \tag{4-28}$$
$$a^2 + \omega_n^2 = \omega_1^2 + \omega_2^2 + 4\zeta_1\zeta_2\omega_1\omega_2$$
$$b\,\omega_n^2/h = (\zeta_1\omega_2 + \zeta_2\omega_1)\omega_1\omega_2$$
$$b/I_2I_3 = \omega_1\omega_2$$

Combining the equations in (4-28) in the appropriate fashion yields

$$\zeta_1^2 + \zeta_2^2 = 1 \tag{4-29}$$

The alternate design procedure for this approach is as follows:

(1) Select a desired settling time for the first mode, $\zeta_1 \omega_1$, then

(2) $\zeta_2 = (1 - \zeta_1^2)^{1/2}$

(3) $\omega_2 = (\omega_n - \zeta_2 \omega_1)/\zeta_1$

(4) $k_{22}/I_2 = k_{34}/I_3 = \zeta_1 \omega_1 + \zeta_2 \omega_2$

(5) $k_{23} = -k_{31} = I_2 I_3 \omega_1 \omega_2$

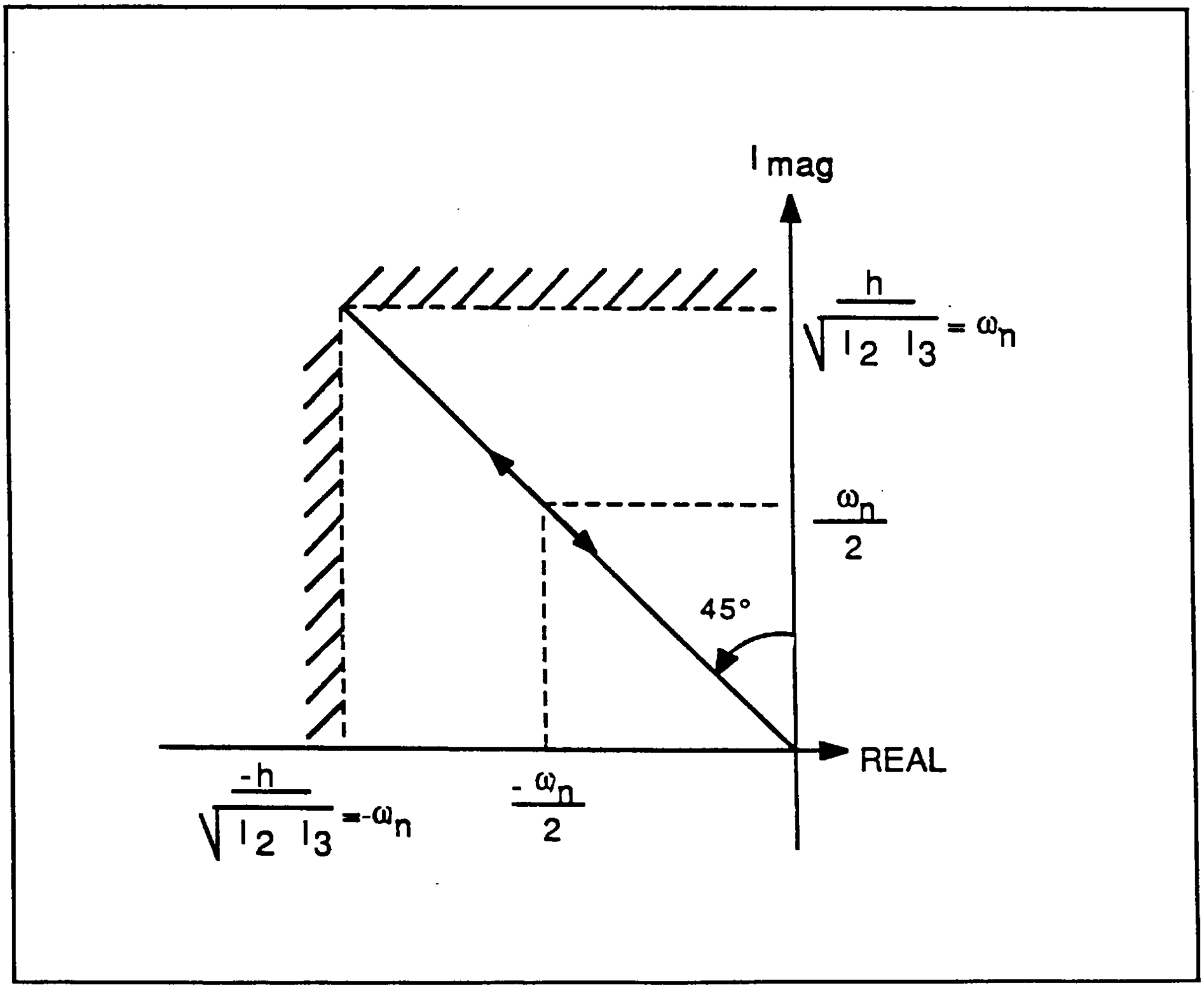

Figure 4-11 Critically Damped Pole Locations

This design procedure results in pole locations as shown in Figure 4-12. The angle, α, is defined as

$$\alpha = \mathrm{acos}(\zeta_1)$$

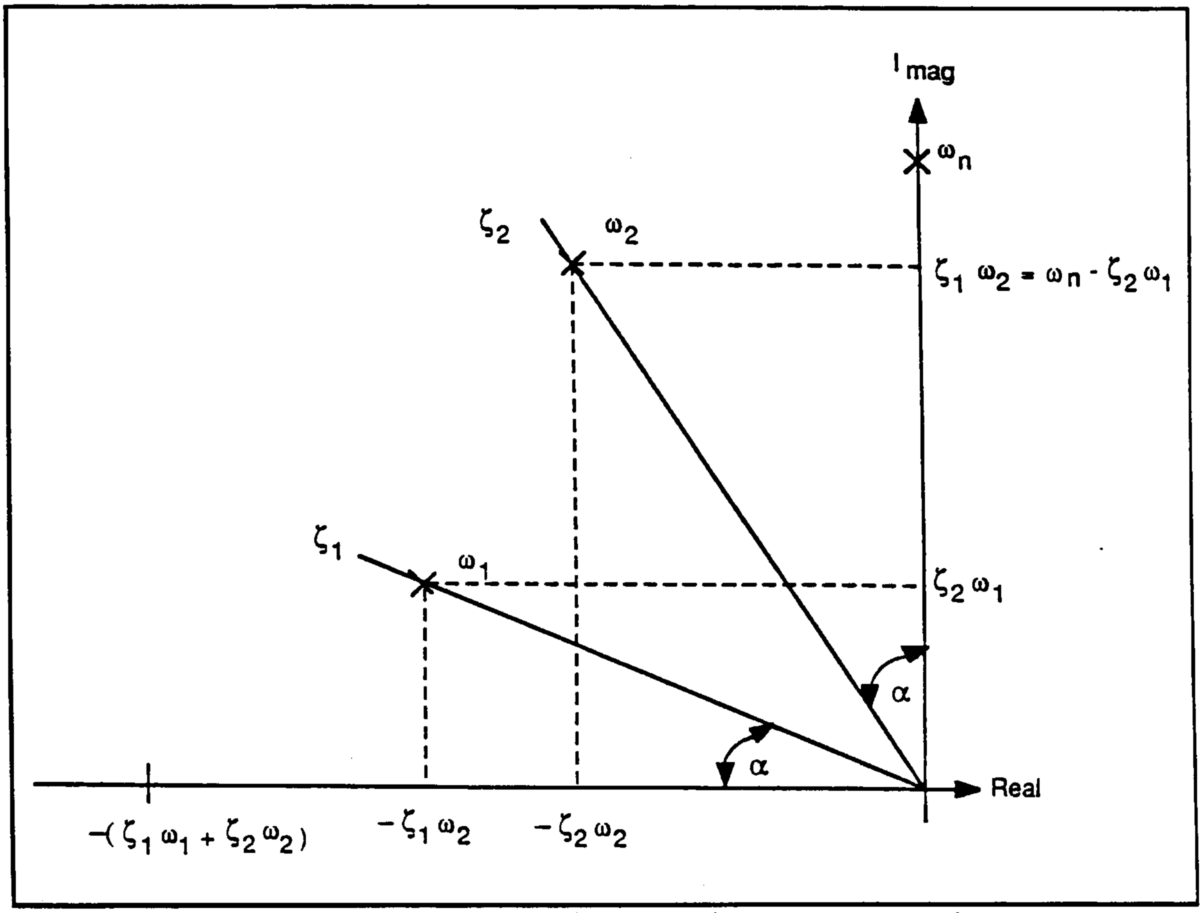

Figure 4-12 Pole Locations for Alternate Approach

This procedure is more general than the first since the damping ratios are not restricted to be 0.707 for both transverse modes, although this occurs when α is 45 degrees. A drawback to this method is that if the damping ratio is increased in one mode it is decreased in the other.

Using the first approach ($\zeta_1 = \zeta_2 = \sqrt{2}/2$) with desired pole locations of

$$s = -0.005 + j\,0.005 \tag{4-28}$$

$\omega_1 = 0.00706$ and $\omega_2 = 0.06365$. This results in other poles being located at

$s = 0.045 + j\,0.045$ and

$$K = \begin{bmatrix} 0 & 45 & .45 & 0 \\ .45 & 0 & 0 & 55 \end{bmatrix}$$

The closed-loop pole locations are shown in Figure 4-13.

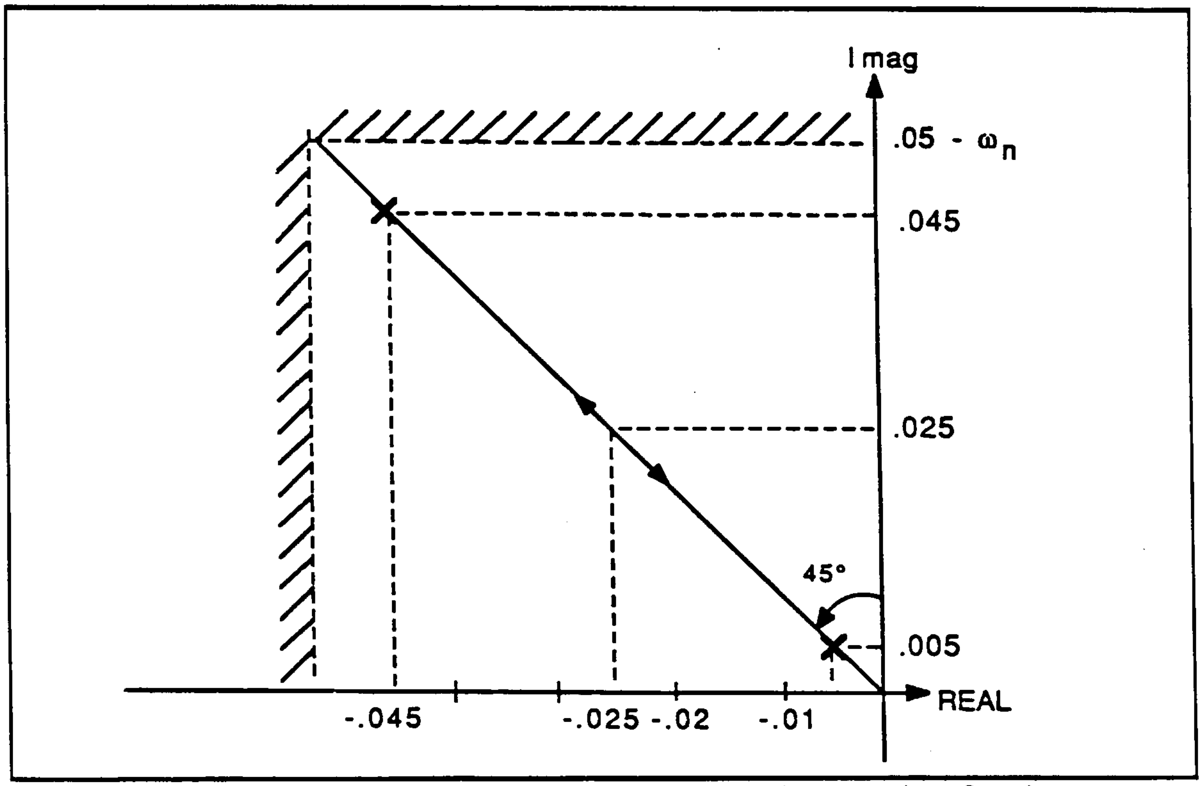

Figure 4-13 Closed-Loop Pole Locations (Precession Cont)

Closed Loop Precession Controller Performance

The performance of the closed-loop precession controller/plant combination is depicted in the following diagrams.

Figures 4-14 and 4-15 are the Bode plots (overall transfer functions) of θ_2/θ_{2r}, θ_2/θ_{3r} and θ_3/θ_{3r}, θ_3/θ_{2r}, respectively. Note the similarity between the pitch and yaw channels, and the bandwidth of θ_2/θ_{2r} is about 0.03 rad/sec.

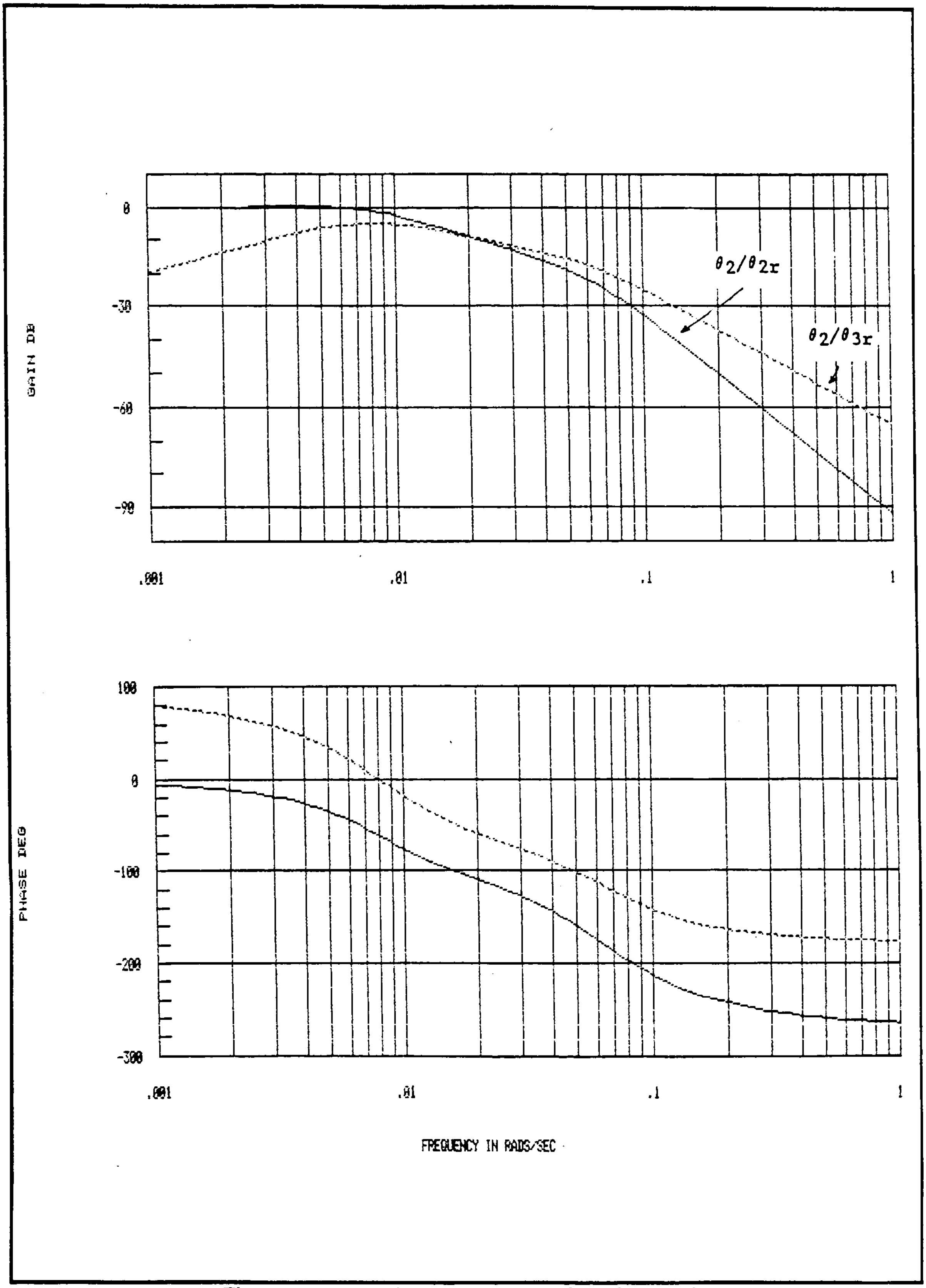

Figure 4-14 Bode Plot, Precession Controller

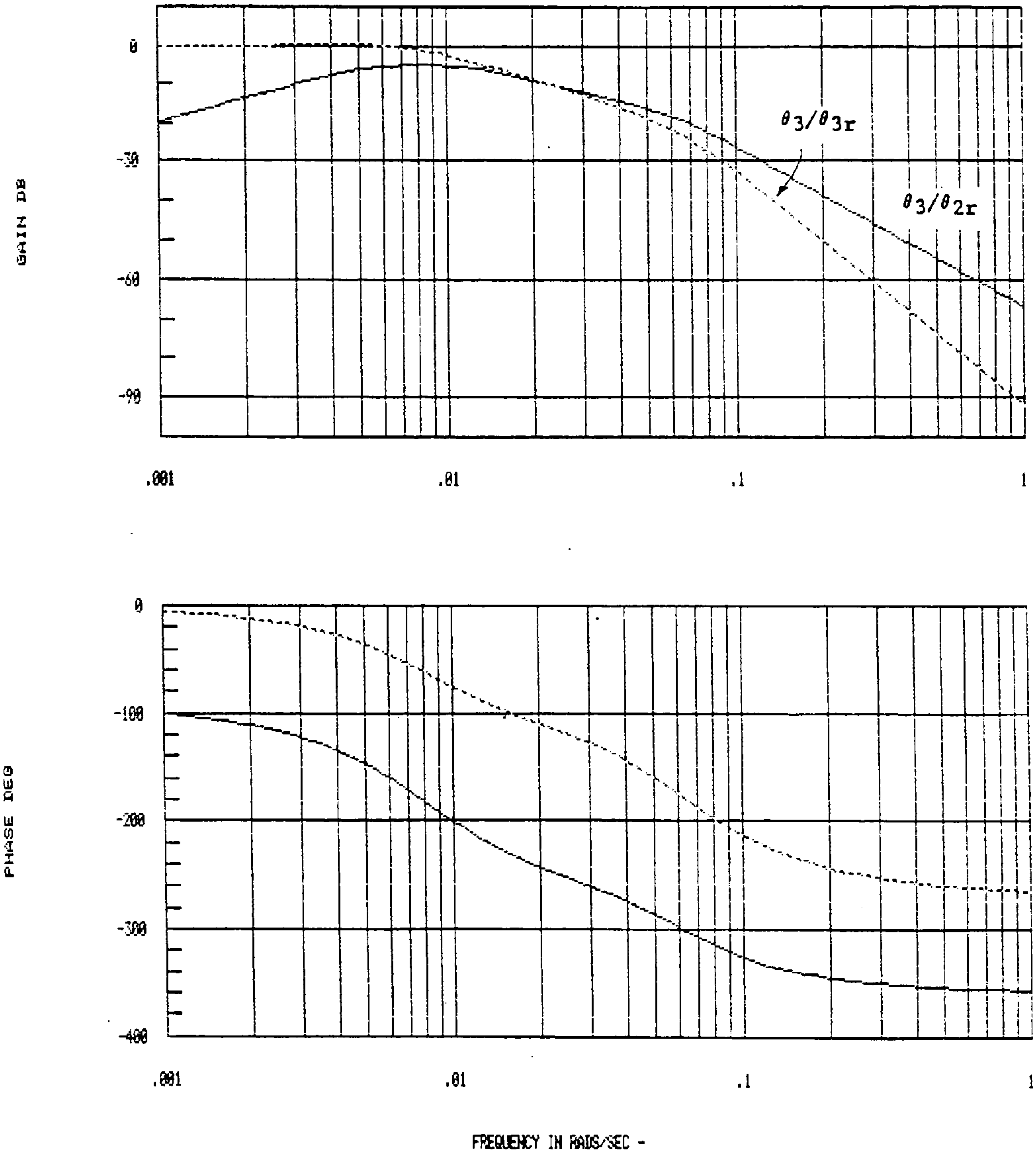

Figure 4-15 Bode Plot, Precession Controller

Figures 4-16 and 4-17 are the disturbance transfer functions for θ_2/T_{d2}, θ_2/T_{d3} and θ_3/T_{d2}, θ_3/T_{d3}, respectively. The spacecraft will experience gravity gradient and magnetic field disturbance torques with a maximum estimated value of 4×10^{-4} N-m at the orbital rate and twice the orbital rate, respectively. Using the maximum position error allowed, 0.1°, the derived limit for the disturbance transfer function is calculated as follows:

$$\theta/T_d = [0.1^{\circ} \times \pi \text{ rad}/180^{\circ}]/[4 \times 10^{-4} \text{ N-m}] = 4.36 \text{ rad/N-m} = 12.8 \text{ dB}$$

Note that at the orbital rate, θ_2/T_{d2} is -11 dB and θ_3/T_{d2} is 8 dB.

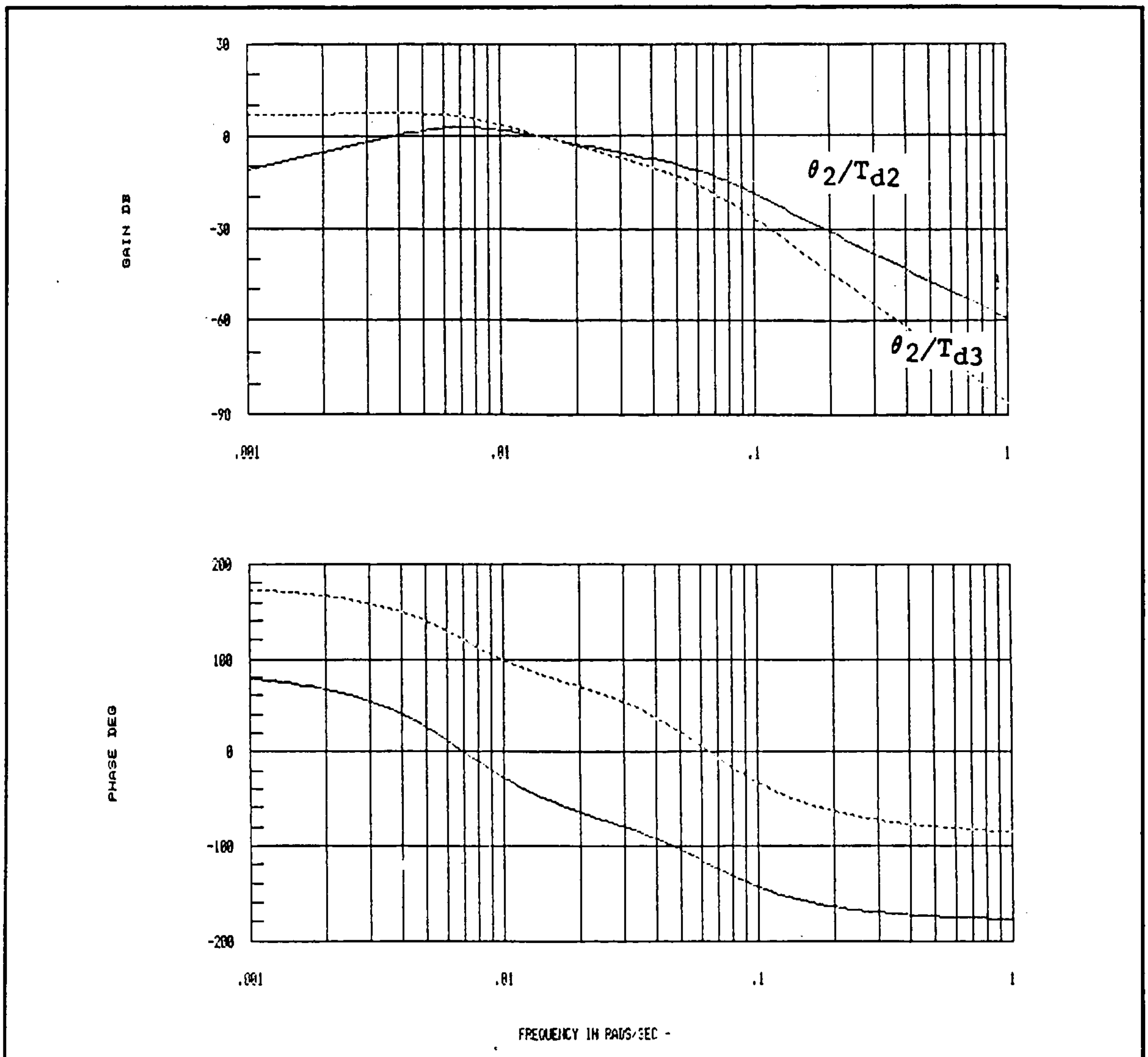

Figure 4-16 Disturbance Transfer Functions, Precession Con

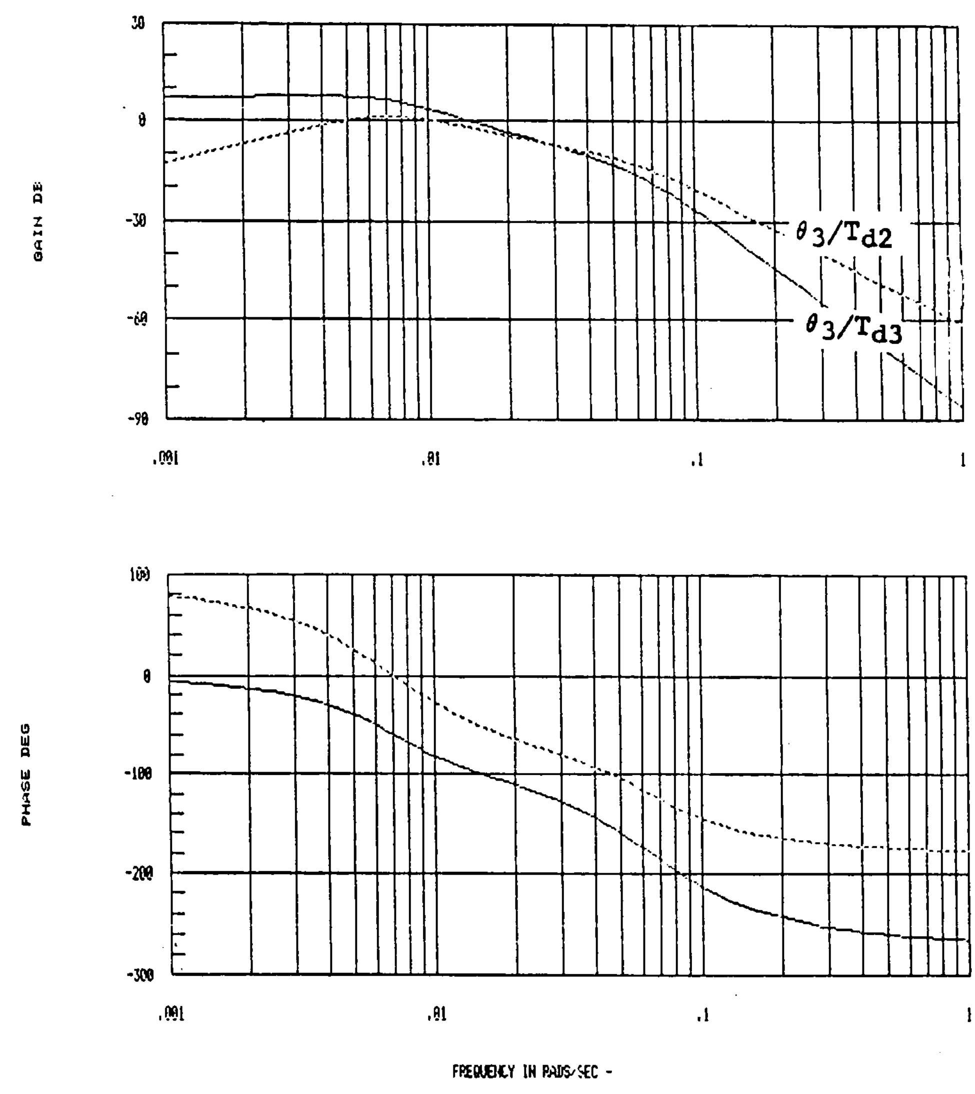

Figure 4-17 Disturbance Transfer Functions, Precession Con

The time response for the linear closed-loop precession system with an applied step input in pitch and yaw is shown in Figure 4-18. The settling time for both is less than 800 seconds.

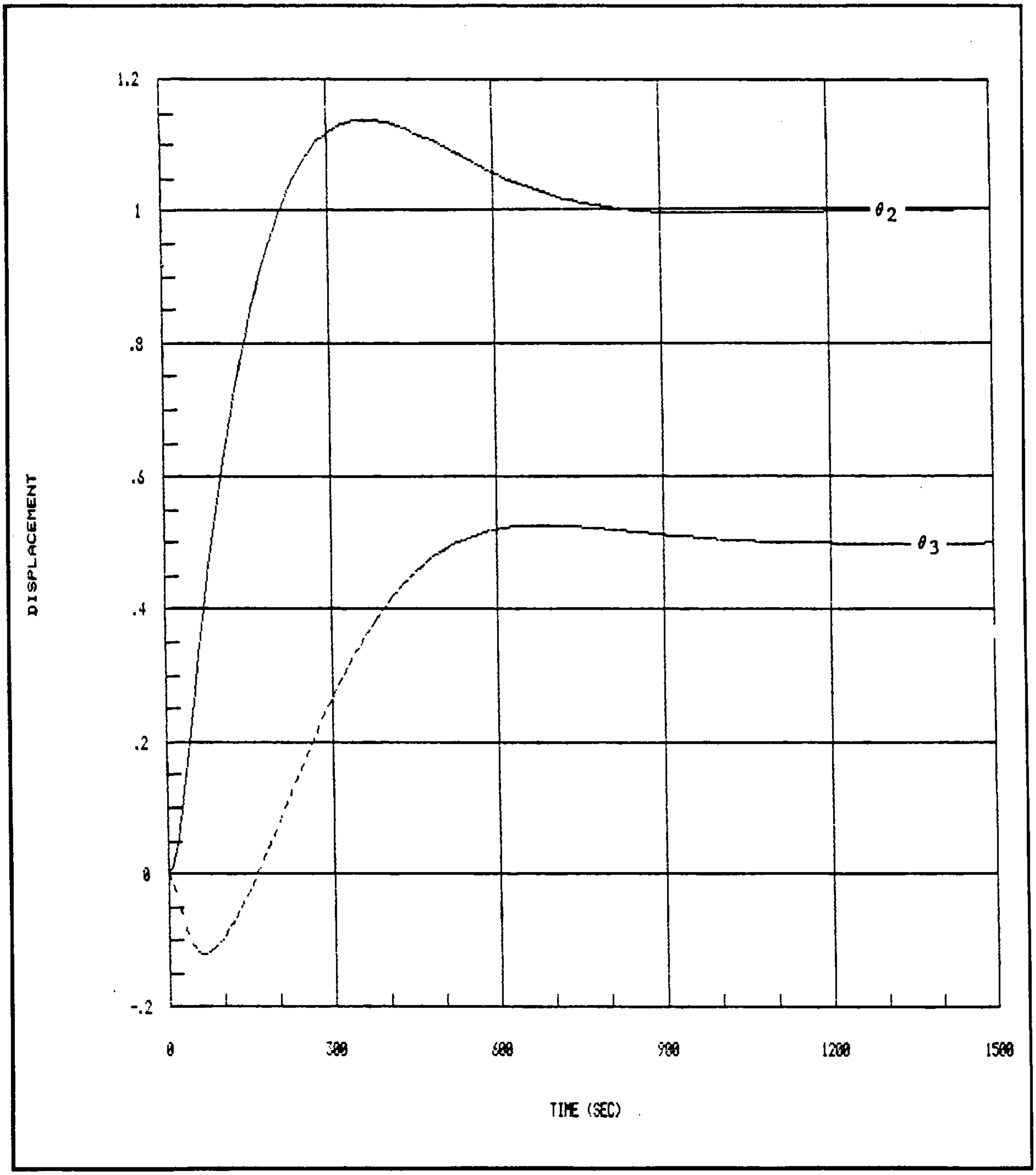

Figure 4-18 Pitch/Yaw Time Response, Step Input, Prec Cont

The relative motion of the roll axis is shown in Figure 4-19.

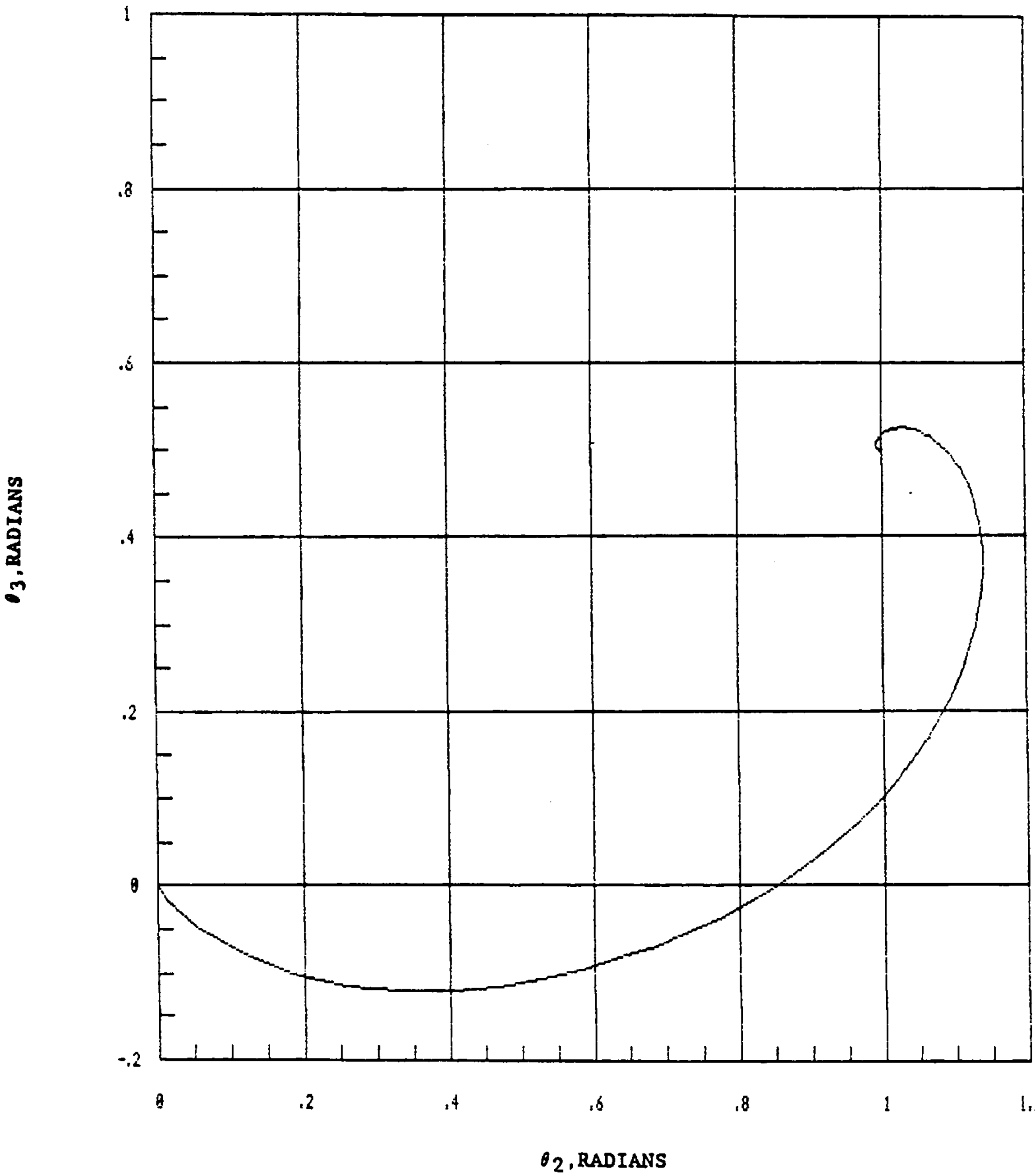

Figure 4-19 Pitch/Yaw Relative Motion, Precession Cont

Figures 4-20 and 4-21 demonstrate that the linear system is robust to variations in the gain parameters K_{23} and K_{22}, respectively. Both gains depend on the inertias I_2 and I_3 which are known to within 10 percent of their actual value. The gains were varied by ± 20% and resulting changes in pole locations show that the closed-loop system remains stable. Similar results occur for the k_{31} and k_{34} gains.

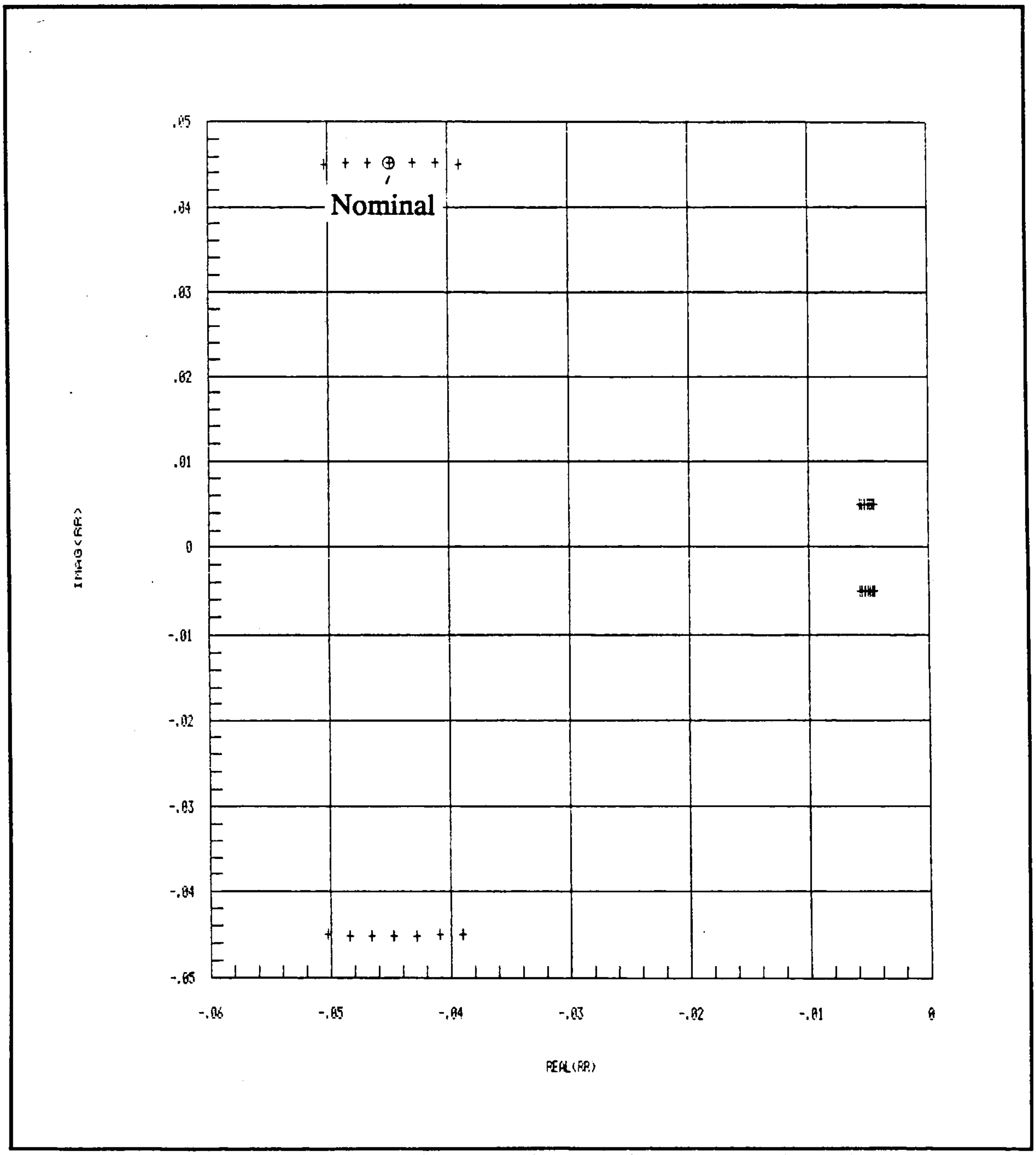

Figure 4-20 Migration of Pole Locations, k23 +/- 20%

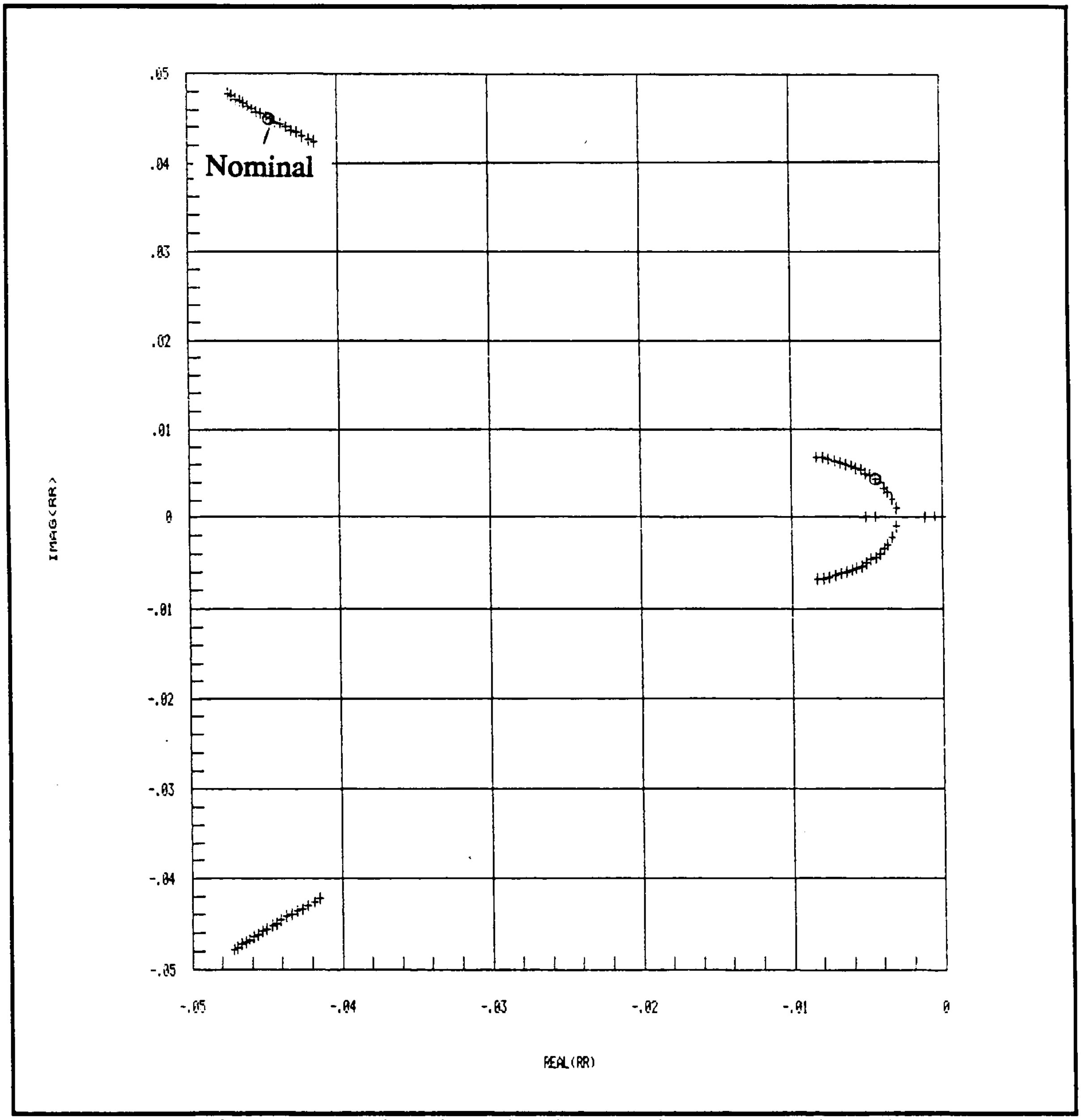

Figure 4-21 Migration of Pole Locations, k22 +/- 20%

The angular momentum of the momentum wheel is nominally 50 N-m-s; however, it may vary from 30 to 70 N-m-s under certain circumstances. Figure 4-22 shows that the transverse axes remain stable to this plant parameter variation.

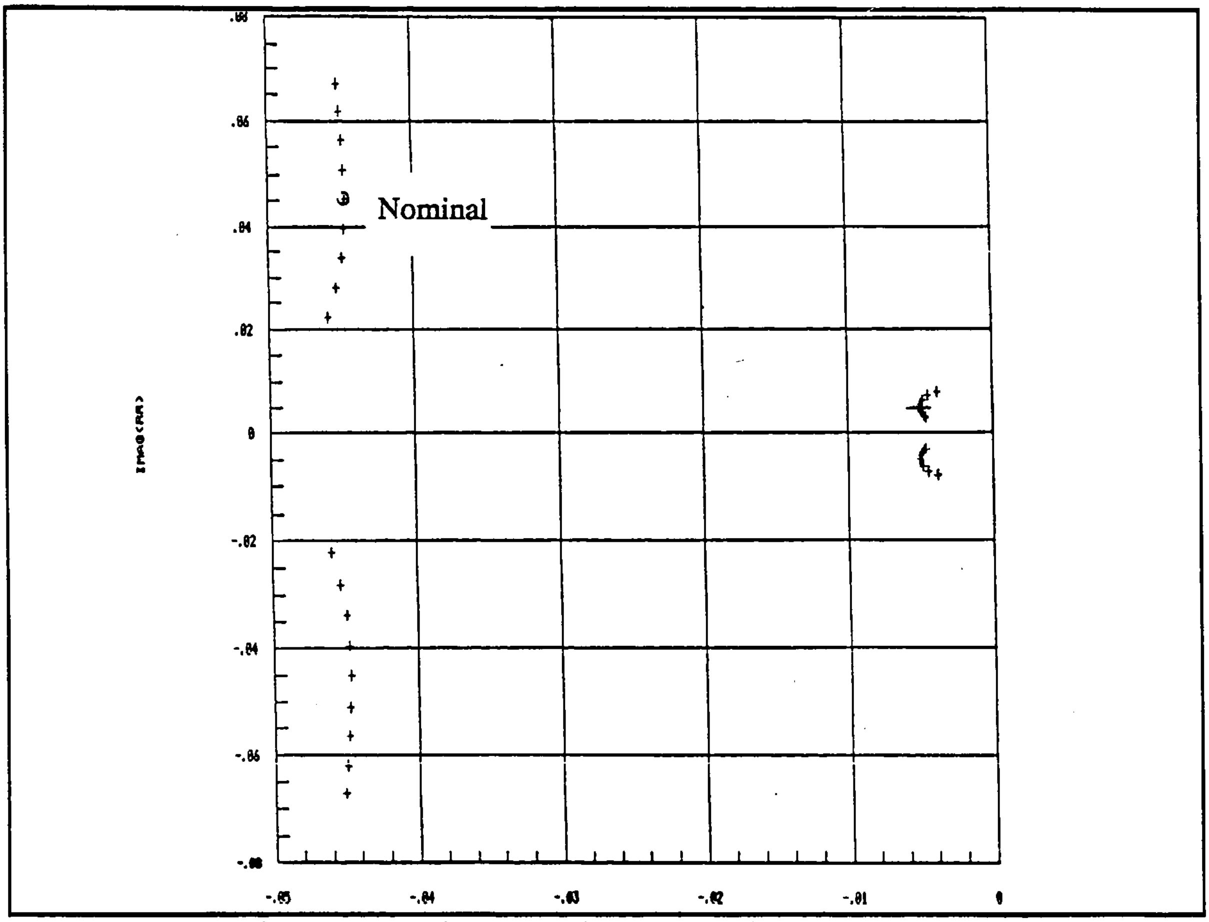

Figure 4-22 Migration of Pole Locations, h +/- 40%

Conclusions - Nominal Precession Controller

The precession controller is an acceptable controller for the nominal momentum-bias satellite transverse axes. The overall transfer functions are acceptable with a bandwidth of 0.03 rad/sec. The disturbance rejection capability for step disturbances satisfies the requirements, and the system has a settling time less than 800 seconds. The attitude maneuver shown in Figure 4-19 is not spacially direct, but this behavior is addressed in the nonlinear torque discussion.

If the inertias, I_2 and I_3, were increased significantly, the size of the box containing the potential pole locations would decrease dramatically (see Figure 4-11). The usefulness of the precession controller becomes questionable as the inertias become large due to the extention of a long boom with a tip mass, e.g., 40,000 N-m^2 for a 20m boom vs 1,000 kg m^2 as is presently envisioned.

Linear Quadratic Regulator

In this section an alternate control law for the Starscan ACS will be developed using the deterministic linear quadratic regulator (LQR) approach. [9]

Prior to presenting the LQR equations and designing the controller, it is helpful to review the block diagram and transfer functions for a standard model of the multi-input multi-output system.

In general, the ACS can be characterized as shown in Figure 4-23.

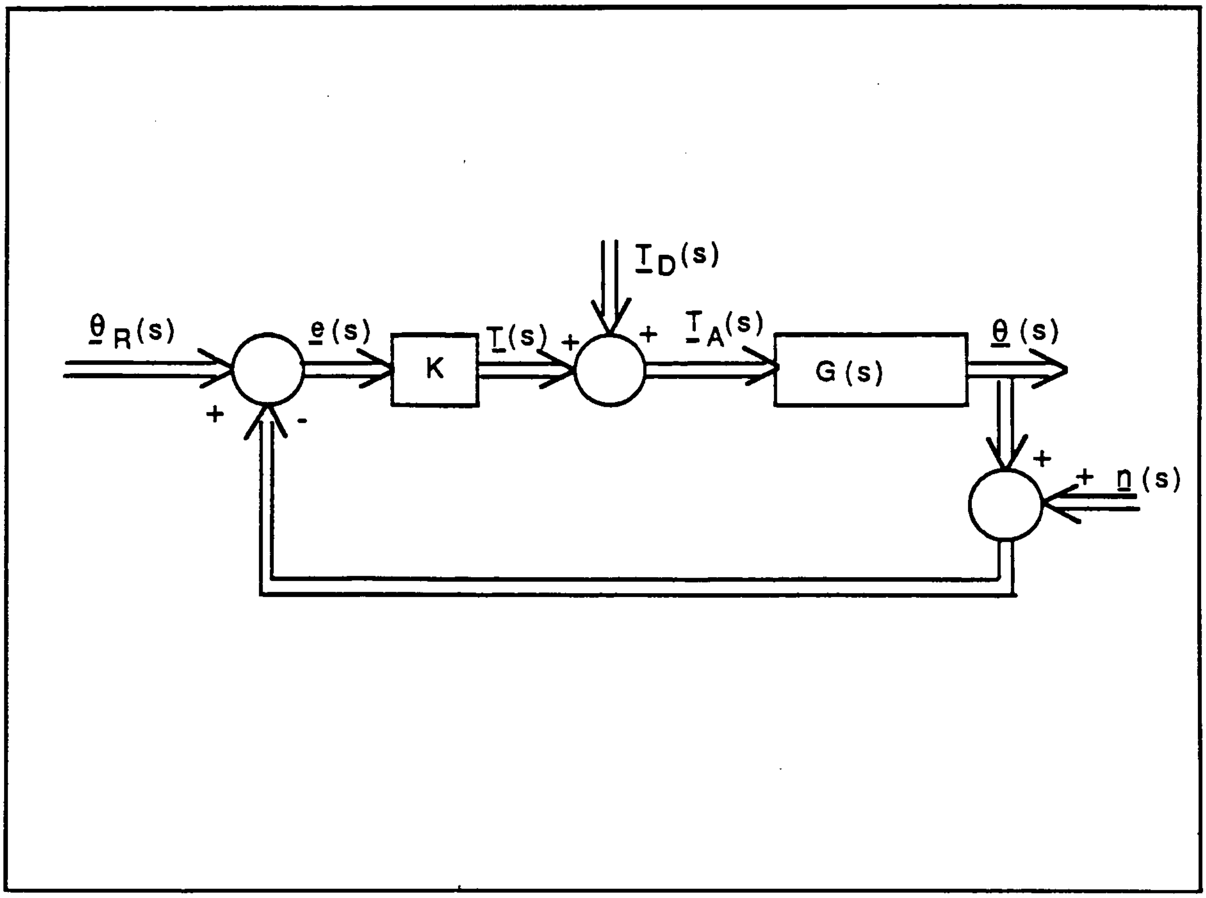

Figure 4-23 Starscan ACS Multi-input Multi-output Model

The input to the system, $\underline{\theta}_r(s)$, is the commanded angular position of the satellite. The resulting commanded torque, $\underline{T}(s)$, is the output of the gain block, K, and the disturbance torque is represented by $\underline{T}_d(s)$. The output of the nominal plant, G(s), is the actual attitude of the spacecraft, $\underline{\theta}(s)$. Sensor noise inherent in the system is represented by $\underline{n}(s)$. The error signal, $\underline{e}(s)$, is the difference between $\underline{\theta}_r(s)$ and $(\underline{\theta}(s) + \underline{n}(s))$.

The error and output transfer functions are shown in Equations (4-29) and (4-30).

$$e(s) = (I + G(s)K)^{-1}(\underline{\theta}_r(s) - \underline{n}(s) - G(s)\underline{T}_d(s)) \tag{4-29}$$

$$\underline{y}(s) = (I + G(s)K)^{-1}G(s)K\,(\underline{\theta}_r(s) - \underline{n}(s)) + (I + G(s))^{-1}G(s)\underline{T}_d(s) \tag{4-30}$$

Several definitions are used extensively in the literature and are presented here for later use. The product G(s)K is called the loop transfer function matrix. The quantity $(I + G(s)K)^{-1}G(s)K$ is defined as the closed-loop transfer function matrix and $(I + G(s)K)$ is the return difference transfer function matrix.

There are several somewhat conflicting requirements imposed on the system. To maintain the commanded attitude, $\underline{\theta}_r(s)$, the quantity $(I + G(s)K)^{-1}G(s)K \cong I$, the identity matrix. On the other hand, to reduce the effect of signal noise and torque disturbances on the system, $(I + G(s)K)^{-1}G(s)K$ must be small at the appropriate frequencies. The LQR design must satisfy these competing requirements.

The linear quadratic (LQ) optimal control problem, discussed in many references [e.g., 9, 10], is to minimize the quadratic cost function

$$J = \int_0^{\infty} [\underline{x}^T(t)\,Q\,\underline{x}(t) + \underline{u}^T(t)\,R\,\underline{u}(t)]\,dt \tag{4-31}$$

subject to the constraints

$$\dot{\underline{x}}(t) = A\underline{x}(t) + B\underline{u}(t)$$

and

$$\underline{x}(0) = \underline{x}_0.$$

The vectors $\underline{x}(t)$ and $\underline{u}(t)$ represent the states of the system and the control law, respectively. The Q and R matrices are the state and control cost matrices, respectively. Assume that $Q = Q^T \geq 0$ and $R = R^T > 0$.

The classic solution to the LQ problem was obtained by Kalman in 1960 under the further assumption that [A,B] is controllable and [A, $\sqrt{Q}$] is observable. Kalman showed that the optimal control expressed in feedback form is

$$\underline{u}^*(t) = -B^T R^{-1} K \underline{x}(t) \qquad (4\text{-}32)$$

and the minimum value of J is

$$J^* = \underline{x}_0^T K \underline{x}_0 \qquad (4\text{-}33)$$

where K satisfies the steady-state algebraic matrix Riccatti equation

$$A^T K + KA + Q - KBR^{-1}B^T K = 0 \qquad (4\text{-}34)$$

Among the various solutions to Equation (4-34) is the unique solution that is positive definite. Moreover, the closed-loop matrix (A - BG(s)) is asymptotically stable.

LQR Design Procedure

In general, the solution to this problem is to select the Q and R matrices, solve the algebraic matrix Riccatti equation, Equation (4-34), for the gain matrix, K, and calculate the control law via Equation (4-32). The key issue is how to select the appropriate state and control cost matrices.

The structure of the Q and R matrices for the transverse axes is:

$$Q = \mathrm{diag}[q_{\theta 2}\ q_{\omega 1}\ q_{\theta 2}\ q_{\omega 3}] \tag{4-35}$$

$$R = \mathrm{diag}[r_{t2}\ r_{t3}] \tag{4-36}$$

where the $q_{\theta}i$'s are penalties associated with transverse axis angular position errors, the $q_{\omega}i$'s are the penalties for angular rate errors and the $r_t i$'s are the penalties for commanded torques.

In the recent literature it has been demonstrated that the individual values of Q and R are not as important as the relationship of Q to R. [10] This can be readily verified in a scaler case where the ratio, q/r, uniquely defines the optimization problem.

The strategy is to fix either Q or R and vary the other to understand what the effects are on the closed-loop system. Intuitively it seems appropriate to let Q = I and penalize the control torque magnitude by assigning large values to the matrix R. This makes sense because magnetic torque rods are used to provide attitude corrections and the resulting control torques are small, on the average about 0.014 N-m per axis. Two parameters, r_{t1} and r_{t2}, are available to place the poles of the system. The transverse axes are somewhat symmetric given the nominal inertias. We desire the pitch and yaw axes' performance to be similar so the control torque penalties are set equal to one another. The result is that the designer has only one parameter to vary for pole placement.

On the other hand, if we let R = I and a similar rationale is used for the state cost matrix, Q, the result is

$$q_{\theta 2} = q_{\theta 3} = q_{\theta} \tag{4-37}$$

$$q_{\omega 2} = q_{\omega 3} = q_{\omega}$$

and two parameters are available for pole placement. The structure of the state and control cost matrices becomes

$$\begin{aligned} Q &= \mathrm{diag}[q_{\theta} \quad q_{\omega} \quad q_{\theta} \quad q_{\omega}] \\ R &= \mathrm{diag}[1 \quad 1] \end{aligned} \tag{4-38}$$

To gain further insight into the system it is beneficial to let $q_{\omega} = 1$, let q_{θ} vary from 0.1 to 100, solve the LQR problem and plot the resulting closed-loop eigenvalues. The results are shown in Figure 4-24.

Several characteristics are noteworthy. As the position penalty is increased from 0.1 to 100 the poles move to the left, the response time of the system is decreased and a corresponding increase in bandwidth occurs. In addition, this LQR approach yields modes with nearly equal settling times.

Using a similar strategy but setting $q_{\theta} = 1$ and varying q_{ω} results in the pole locations shown in Figure 4-25.

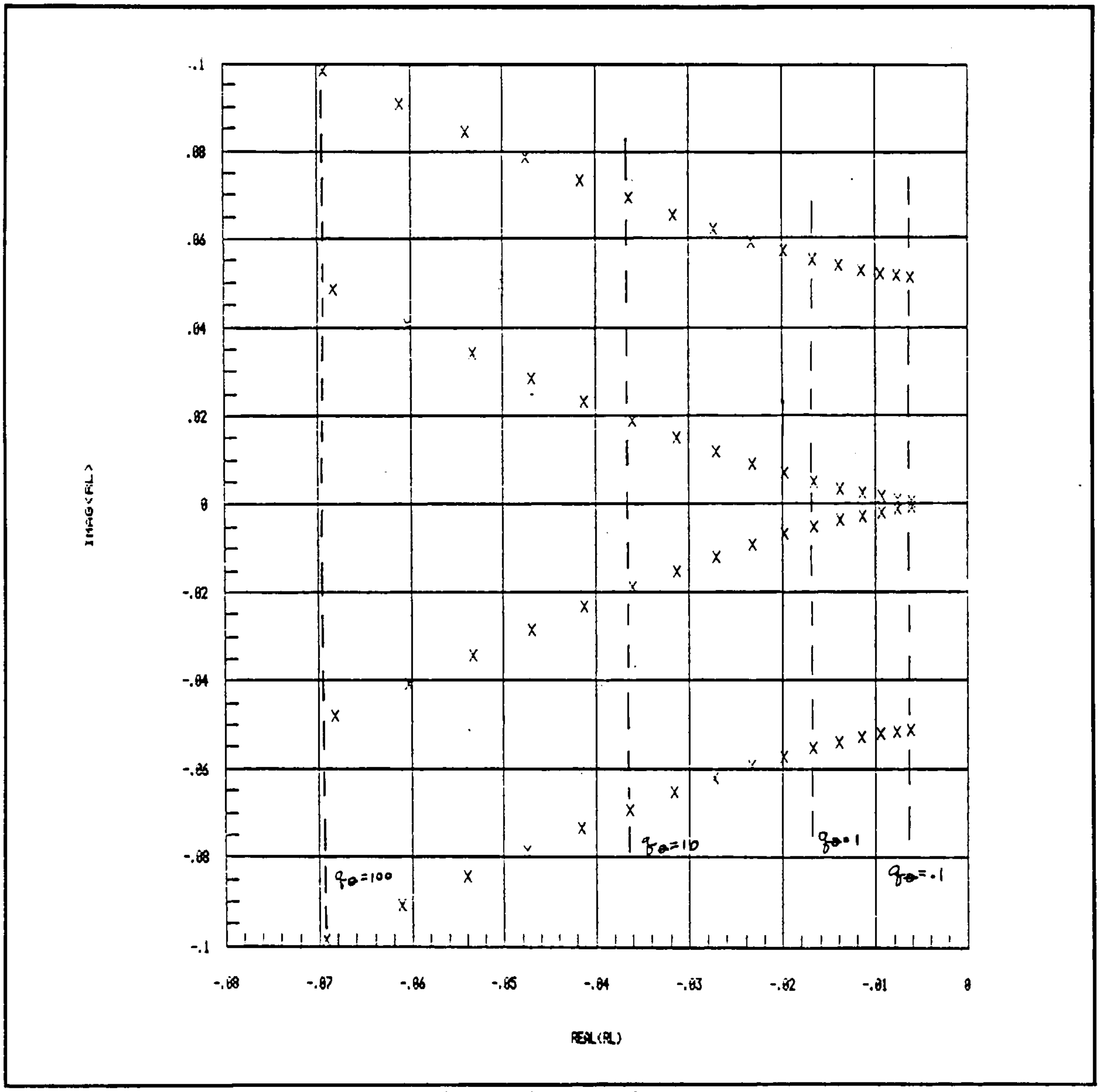

Figure 4-24 Resulting Pole Locations, 0.1 < q < 100

Note that the closed-loop system is relatively insensitive to the variations in the penalty for angular rate errors. Even if $q_{\omega} = 0$, the closed-loop system is stable. This makes sense, since by controlling angular position we have inherent control of the angular rate.

The interesting result is that once again there is only one useful parameter for pole placement, namely the position cost parameter, q_{θ}.

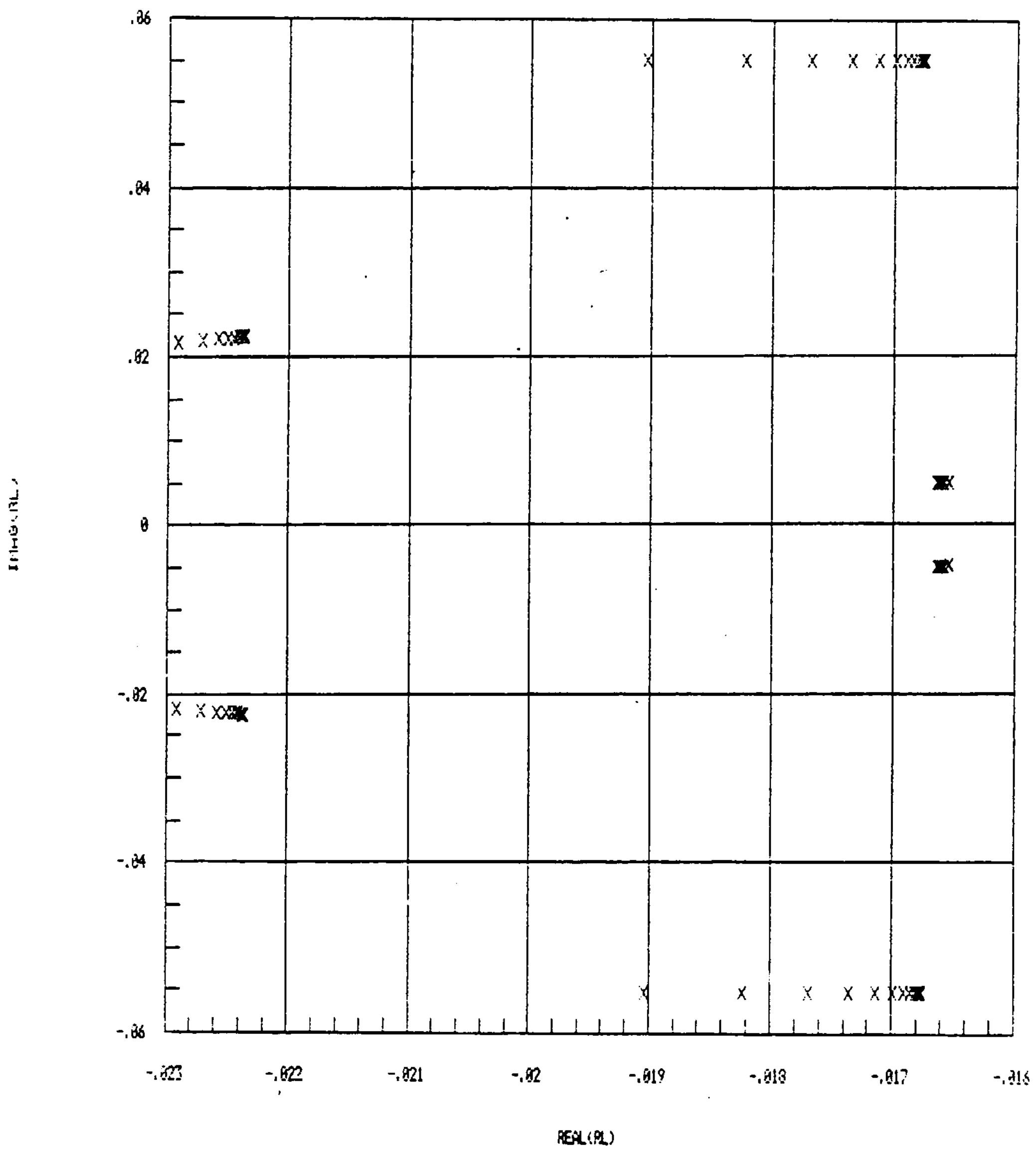

Figure 4-25 Resulting Pole Locations, 0.1 < q < 100

To compare the nominal precession controller and the LQR design, a similar bandwidth LQR controller (θ_2/θ_{2r}, using asymtotic approximation) was developed for the transverse axes. The selected state and control cost matrices are

$$Q = \mathrm{diag}[2 \quad 1 \quad 2 \quad 1]$$

$$R = \mathrm{diag}[1 \quad 1]$$

The gains via the algebraic matrix Ricatti equation are

$$K = \begin{bmatrix} .92 & 40.71 & 1.07 & 0 \\ -1.07 & 0 & .92 & 45.01 \end{bmatrix}$$

Note that the additional gains k_{21} and k_{32} are equal. The closed-loop pole locations are shown in Figure 4-26.

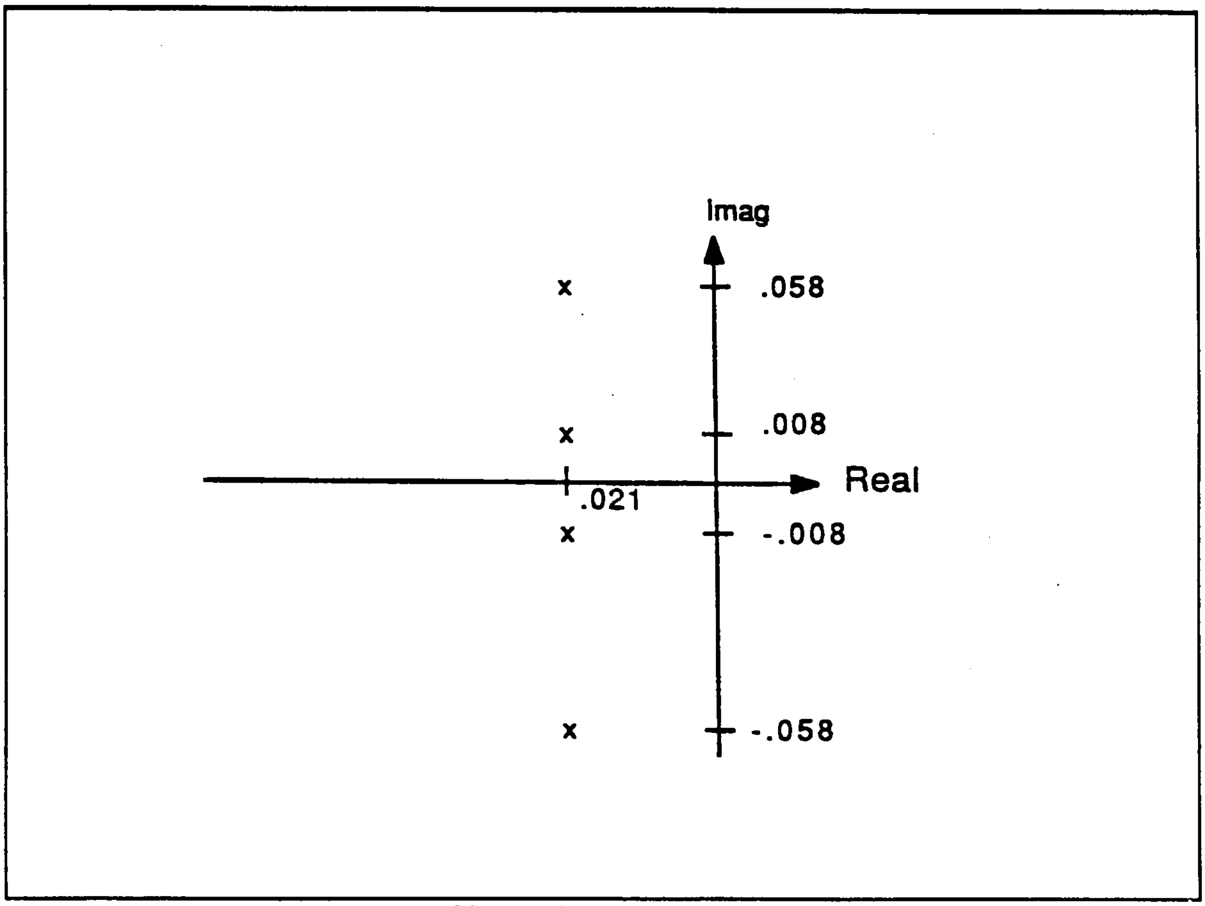

Figure 4-26 Closed-loop Pole Locations (LQR)

Closed-loop LQR Performance

The performance of the closed-loop LQR controller/plant combination is depicted in the following figures.

Figures 4-27 and 4-28 are the Bode plots (overall transfer functions) of $\theta_2/\theta_{2r}, \theta_2/\theta_{3r}$ and $\theta_3/\theta_{3r}, \theta_3/\theta_{2r}$, respectively. Note the similarity between the pitch and yaw channels and the fact that their bandwidth is about 0.03 rad/sec (similar to the bandwidth of the precession controller).

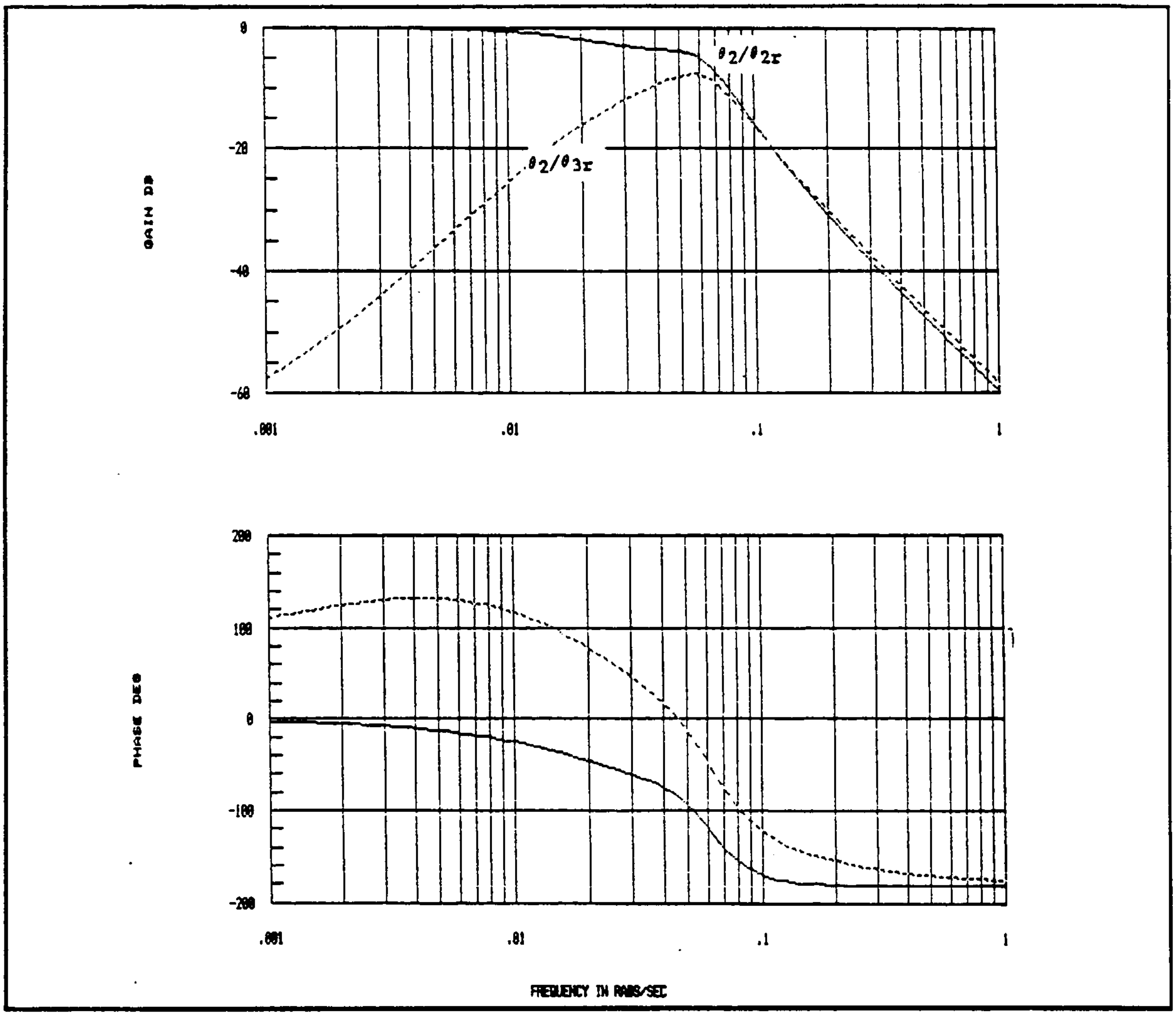

Figure 4-27 Bode Plot, LQR Controller

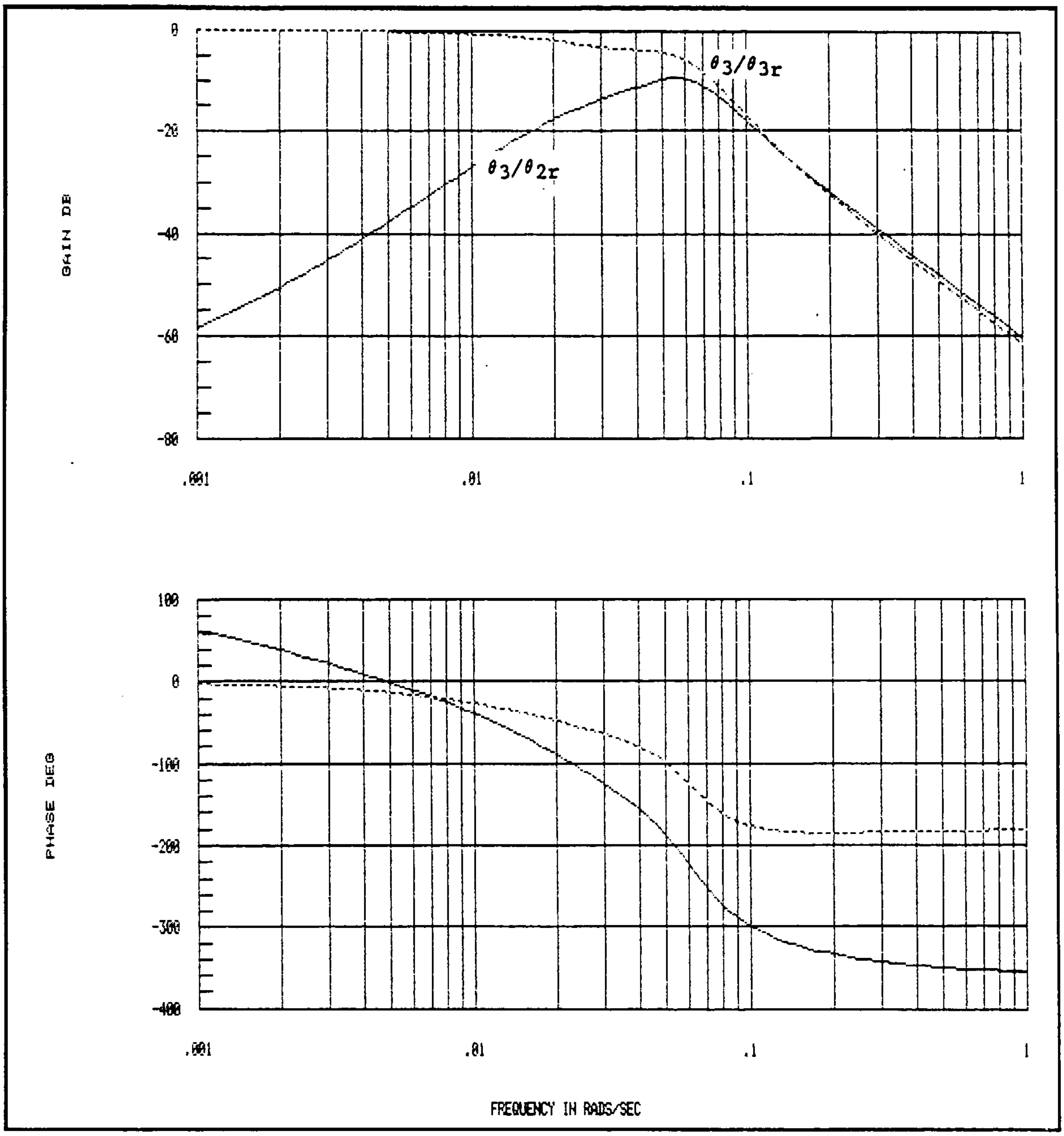

Figure 4-28 Bode Plot, LQR Controller

Figures 4-29 and 4-30 are the disturbance transfer functions for θ_2/T_{d2}, θ_2/T_{d3} and θ_3/T_{d2}, θ_3/T_{d3}, respectively. At the orbital rate θ_2/T_{d2} is -7 dB and θ_3/T_{d2} is -5 dB.

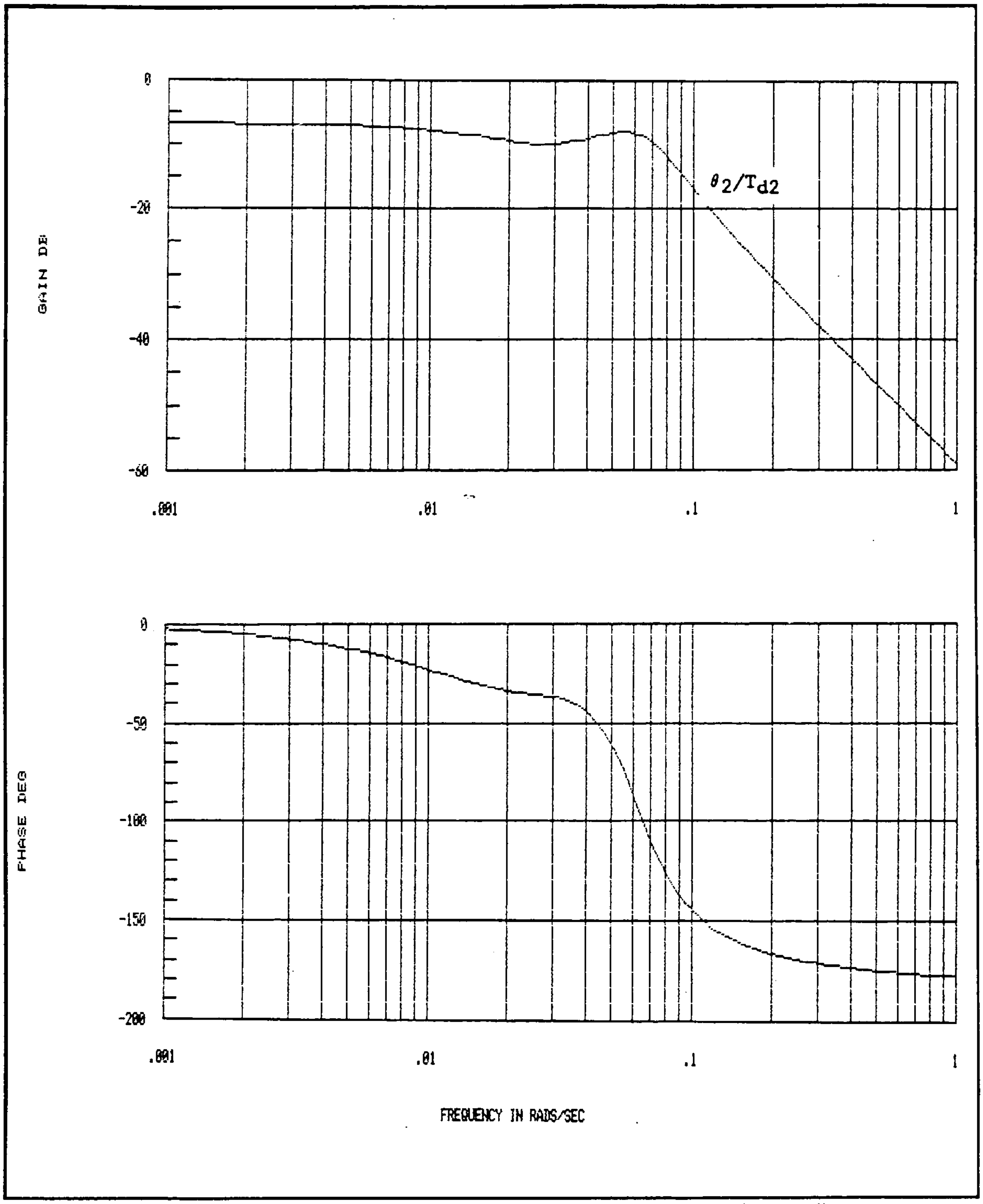

Figure 4-29 Disturbance Transfer Functions, LQR Controller

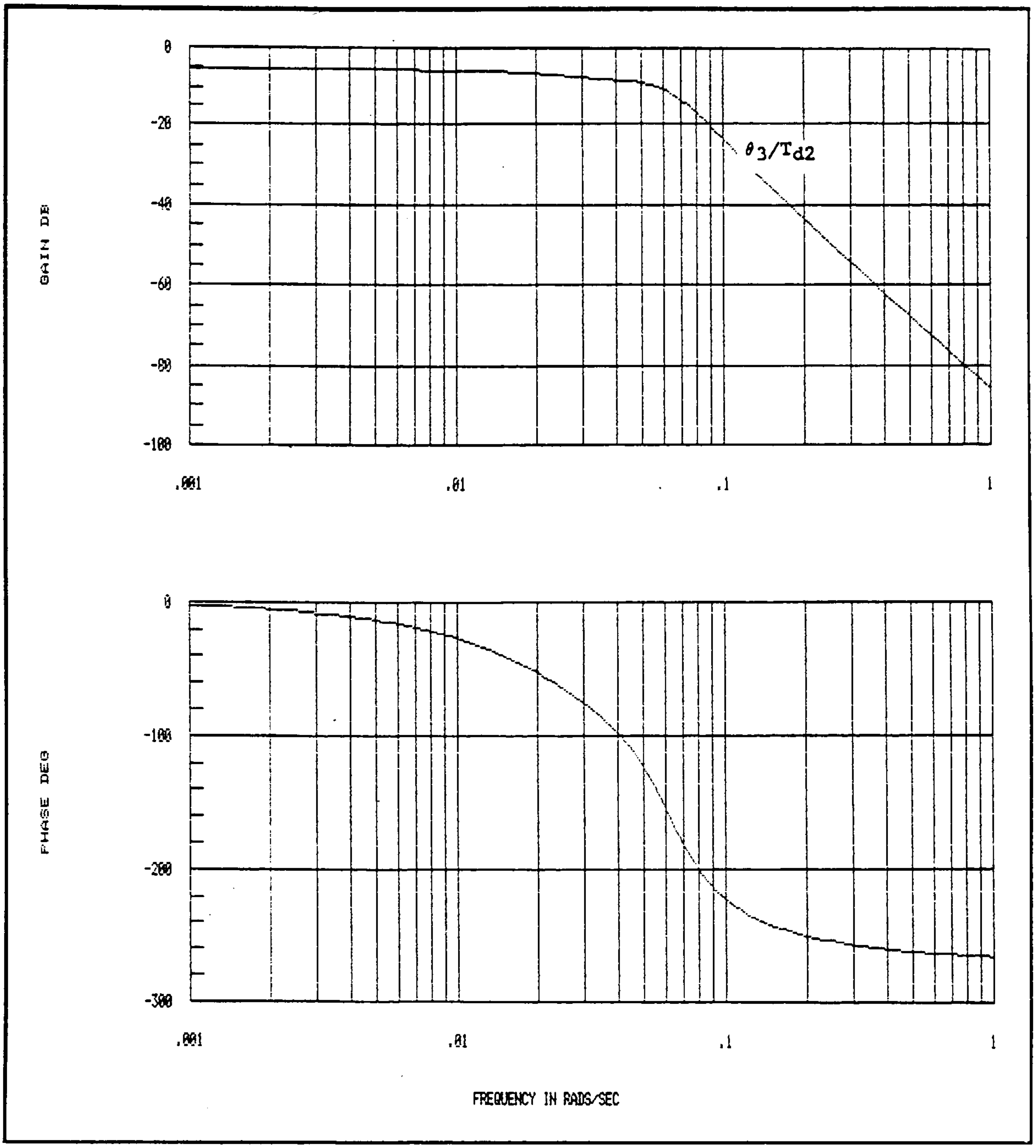

Figure 4-30 Disturbance Transfer Functions, LQR Controller

The time response for the linear closed-loop LQR system with an applied step input in pitch and yaw is shown in Figure 4-31.

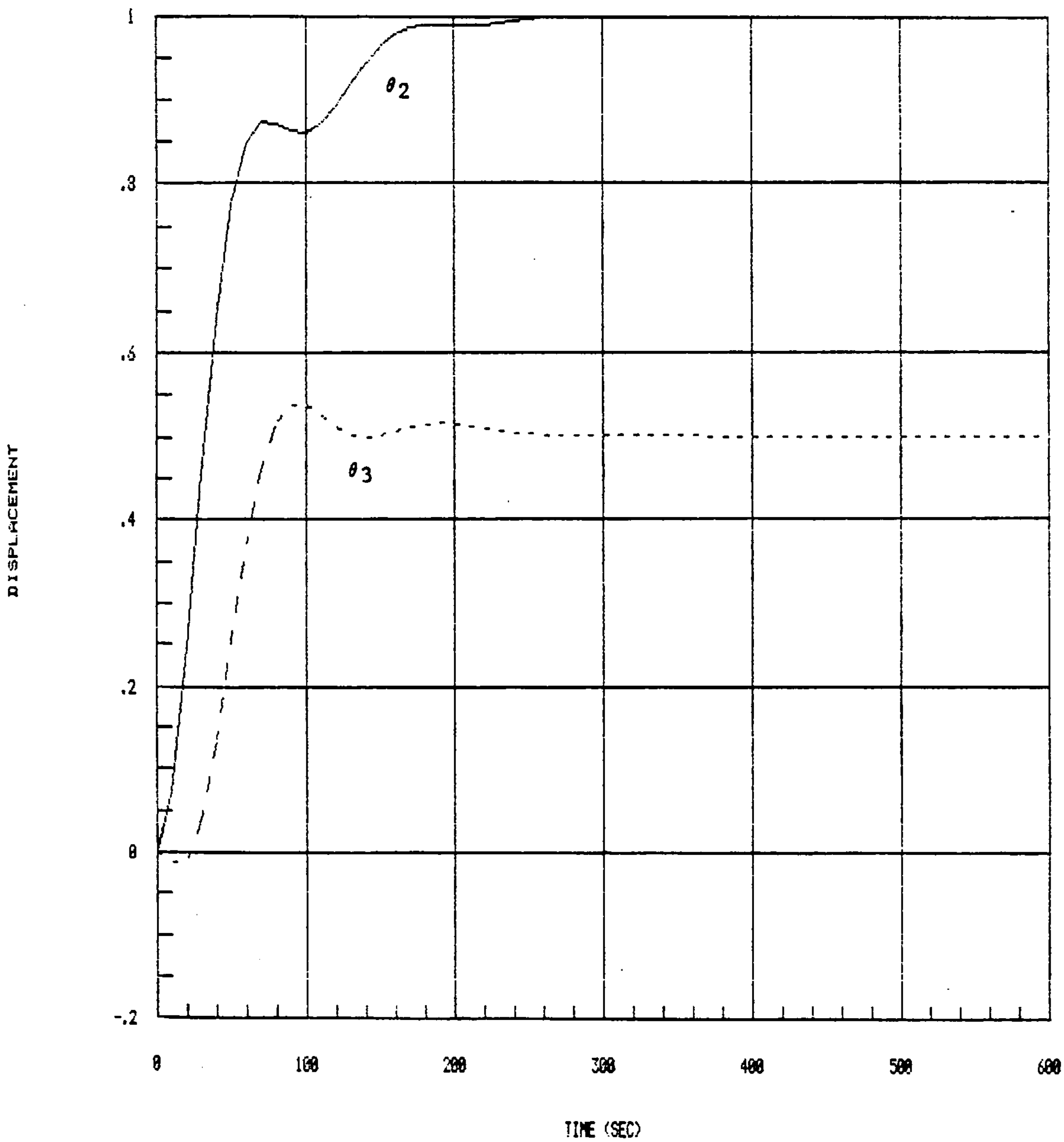

Figure 4-31 Pitch/Yaw Time Response, Step Input, LQR Cont

The settling time for both is less than 400 seconds. The relative motion of the roll axis is shown in Figure 4-32. The motion appears to be indirect when compared to the motion of the precession controller, and the maximum excursions are slightly less than those predicted for the precession controller.

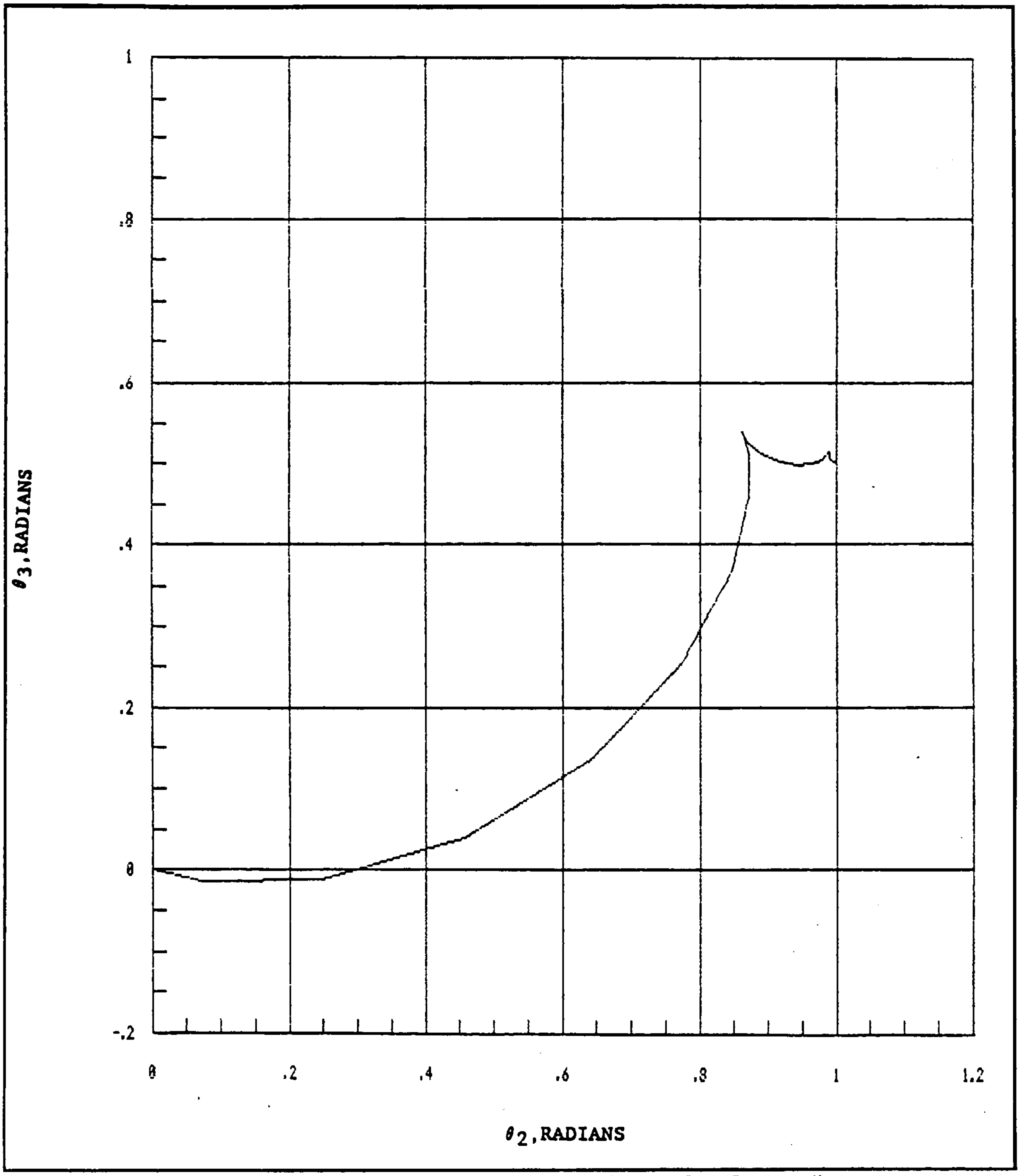

Figure 4-32 Pitch/Yaw Relative Motion, LQR Controller

Figure 4-33 shows the resulting pole locations for variations in the k_{21} gain parameter by $\pm$ 20%. A similar result occurs for k_{33} variations. The closed-loop system is robust to these variations. In fact, the gains k_{21}, k_{33} can be set equal to zero with very little effect on the pole locations.

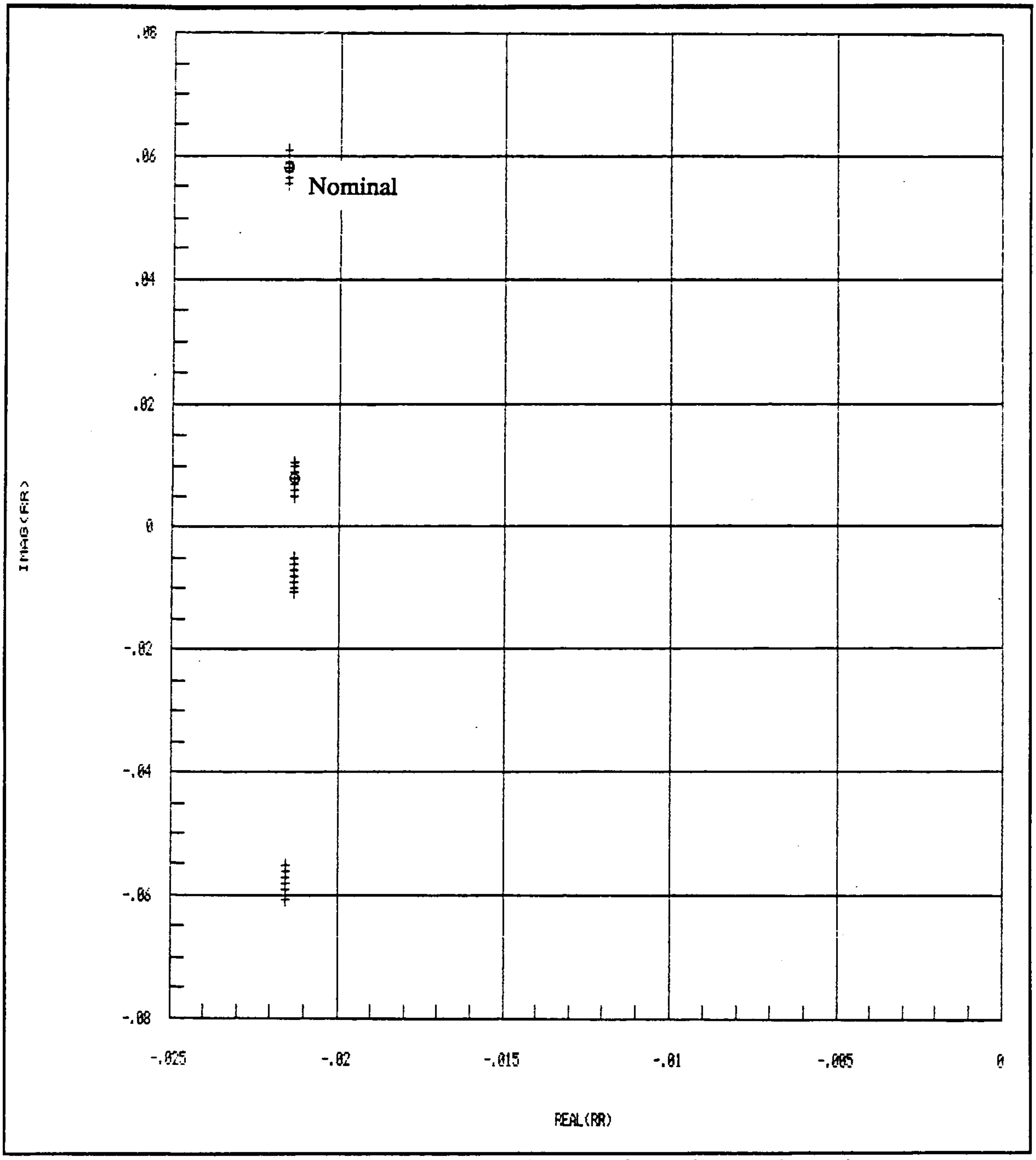

Figure 4-33 Migration of Pole Locations, k21 +/- 20%

Figures 4-34 and 4-35 demonstrate that the linear system is robust to the gain parameters k_{23} and k_{22}, respectively. Both gains were varied by $\pm$ 20% and the resulting pole locations indicate that the closed-loop system remains stable. Similar results occur for k_{31} and k_{34}. It was also determined that the linear system remained stable when each of the gains was set to zero individually.

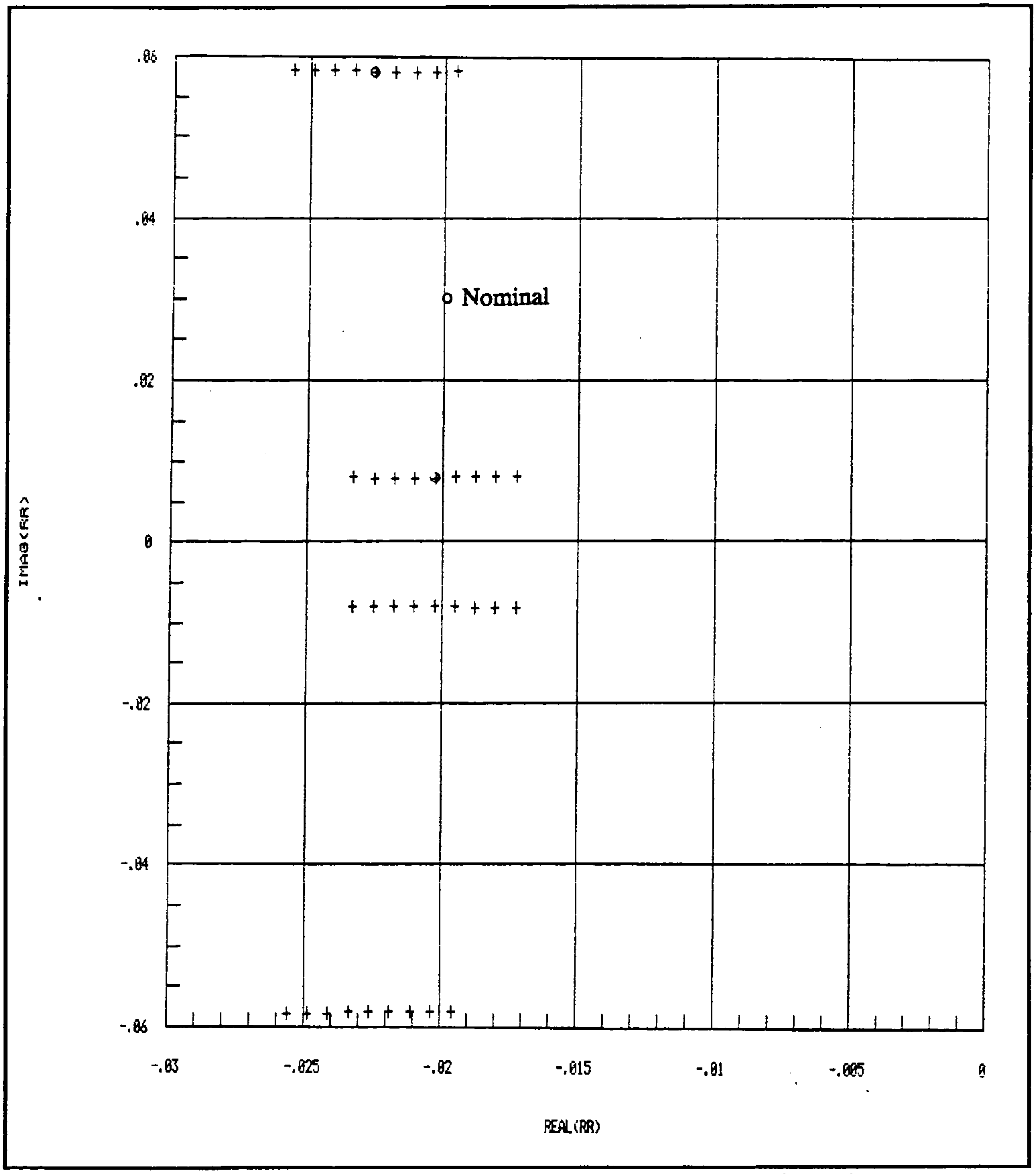

Figure 4-34 Migration of Pole Locations, K23 +/- 20%

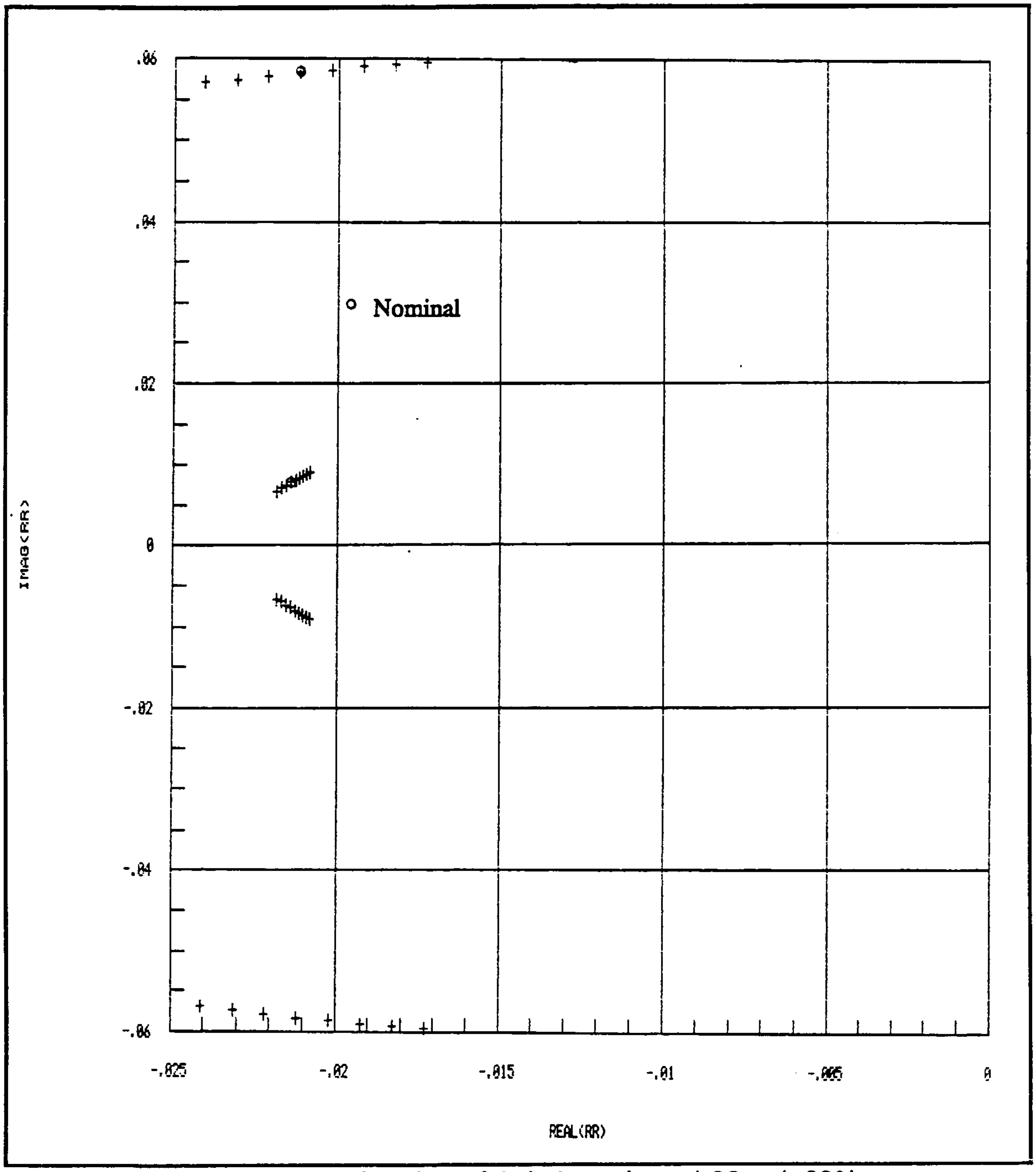

Figure 4-35 Migration of Pole Locations, k22 +/- 20%

The angular momentum of the momentum wheel is nominally 50 N-m-s; however, it may vary from 30 to 70 N-m-s under certain circumstances. Figure 4-36 shows that the transverse axes remain stable to this plant parameter variation. In fact, the LQR system remains stable when h angular momentum is zero.

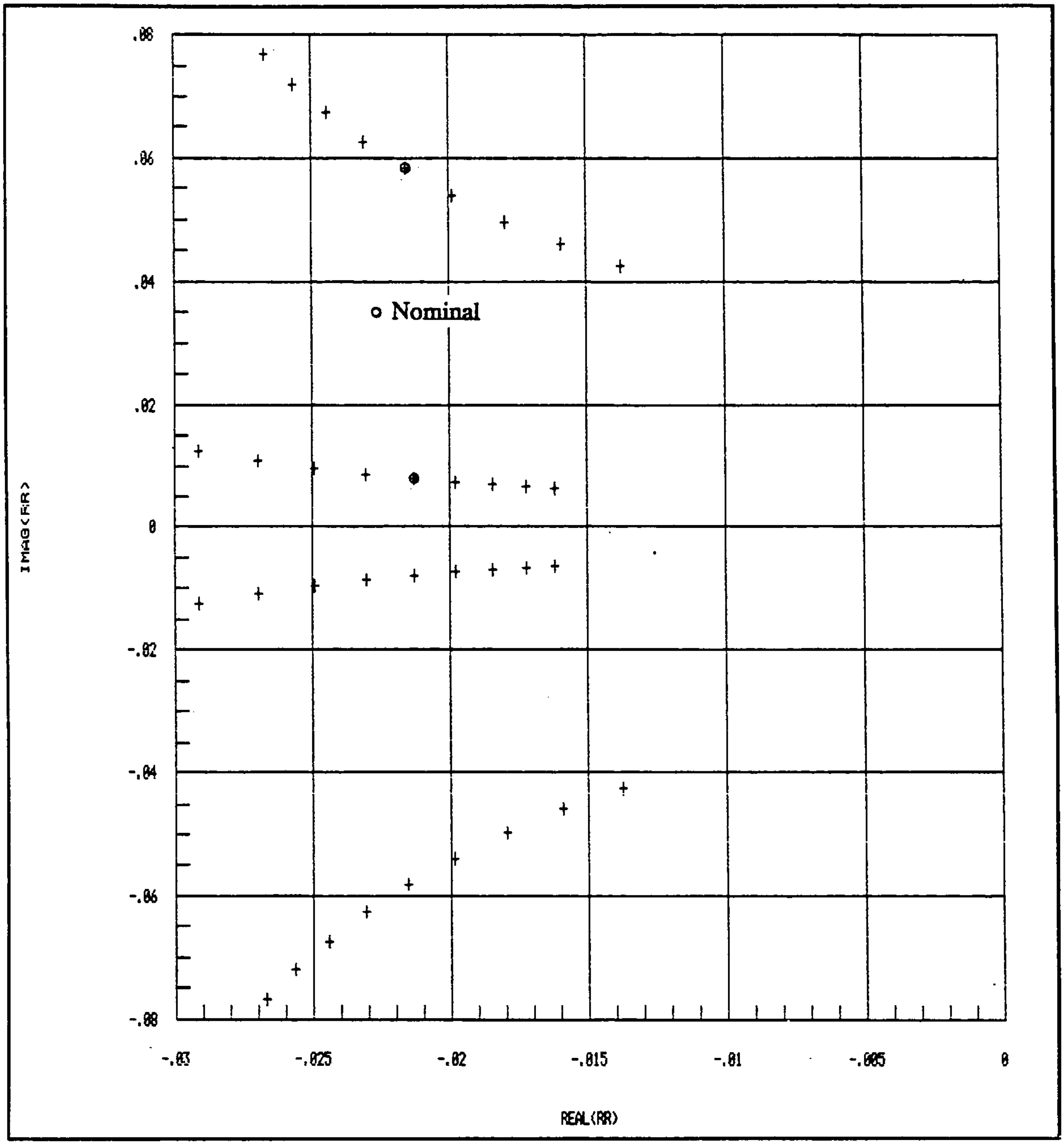

Figure 4-36 Migration of Pole Locations, h +/- 40%

Results of Linear Design and Analysis - Transverse Axes

The results of the linear design and analysis activity are summarized in Table 4-2. The bandwidth of both the nominal precession controller and the LQR is 0.03 rad/sec, based on asymtotic approximation.

Table 4-2 Comparison of Results - Transverse Axes

	Nominal Precession	LQR Q-diag (2 1 2 1)
θ_2/θ_2 Bandwidth	.03 r/s	.03 r/s
θ_2/θ_2 @.001 r/s	0 dB	0 dB
θ_3/θ_2 @.001 r/s	-20 dB	-57 dB
θ_2/Td_2 @.001 r/s	-11 dB	-7 dB
θ_3/T_{d2} @.001 r/s	+7 dB	-5 dB
Settling time (linear) (step input)	≃800 sec	≃350 sec
Large Angle Motion	Smooth	Indirect
Insensitivity to Parameter variations		
Inertias	Acceptable	Acceptable
Angular Momentum	Acceptable	Acceptable

The disturbance rejection capability for both designs is acceptable since the magnitudes of the disturbance transfer functions are less than the maximum allowed, 12.8 dB. The LQR design seems to have a more "balanced" response to disturbances.

The large angle motion associated with the LQR design, Figure 4-32, was indirect when compared with that of the precession design, Figure 4-19. The maximum excursion associated with the LQR motion was slightly less than that for the pure precession controlled motion.

Both designs are robust to gain and plant parameter variations. The angular momentum of the momentum wheel is not expected to deviate from the nominal value of 50 N-m-s by more than +/- 10 percent; but it is interesting to note that when the angular momentum is set equal to zero, the precession controlled system becomes conditionally stable, whereas the LQR design remains stable.

Next, a realistic estimate of the control system's torque producing capability will be developed. Then several nonlinear saturation torque limiting schemes will be investigated.

Nonlinear Saturation Torque Limiting

There are two reasons for introducing a nonlinear torque limit into the control design problem at this point. First, and most important, is the fact that the combination of the magnetic torque rods on the spacecraft and Earth's magnetic field at orbital altitude severely limit the amount of torque that can actually be produced by the control system. So, torque limiting more accurately simulates the spacecraft environment. Second, it will be demonstrated that a certain type of torque limiting will refine the quality of motion for large angle slew maneuvers.

The magnitude of the torque authority available to control spacecraft attitude is limited by the magnetic ACS. Also the magnetic torque rods, shown in Figure 4-2, are aligned parallel to each of the spacecraft body axes. Each torque rod is capable of developing a magnetic moment, B, of 350 amp-m^2 or 350 N-m/Tesla.

The torque available is calculated by the cross-product of the magnetic field strength, M, in tesla, and the magnetic moment, $T = M \times B$ N-m. The orbital average torque available about a single axis was obtained via simulation to be 0.014 N-m. The effect of this limited torque authority on the time response of the system is dramatic.

The model used to evaluate the potential torque limiting schemes is shown in Figure 4-37.

Two saturation limiting approaches were investigated: independent torque limiting (ITL) and directional torque limiting (DTL). The ITL technique is fairly standard and uses independent saturation limiters on each axis as shown in Figure 4-38. Note that the saturation limits have been set to 0.01 N-m.

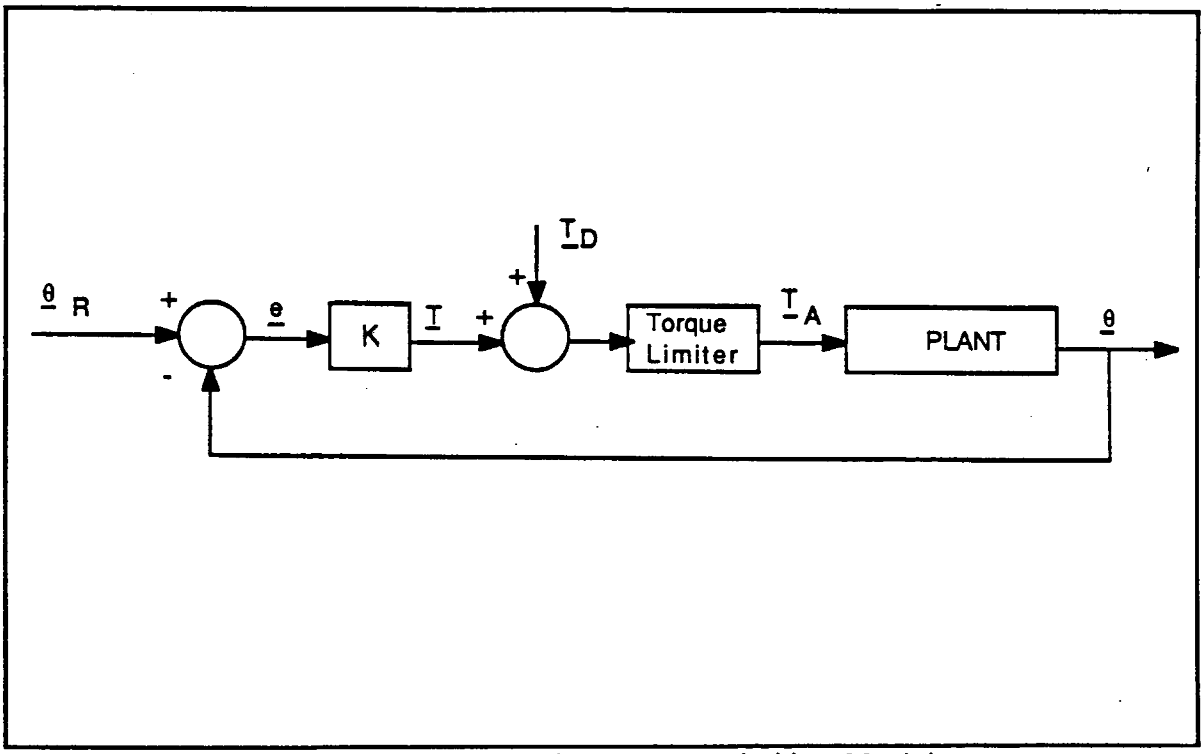

Figure 4-37 Saturation Torque Limiting Model

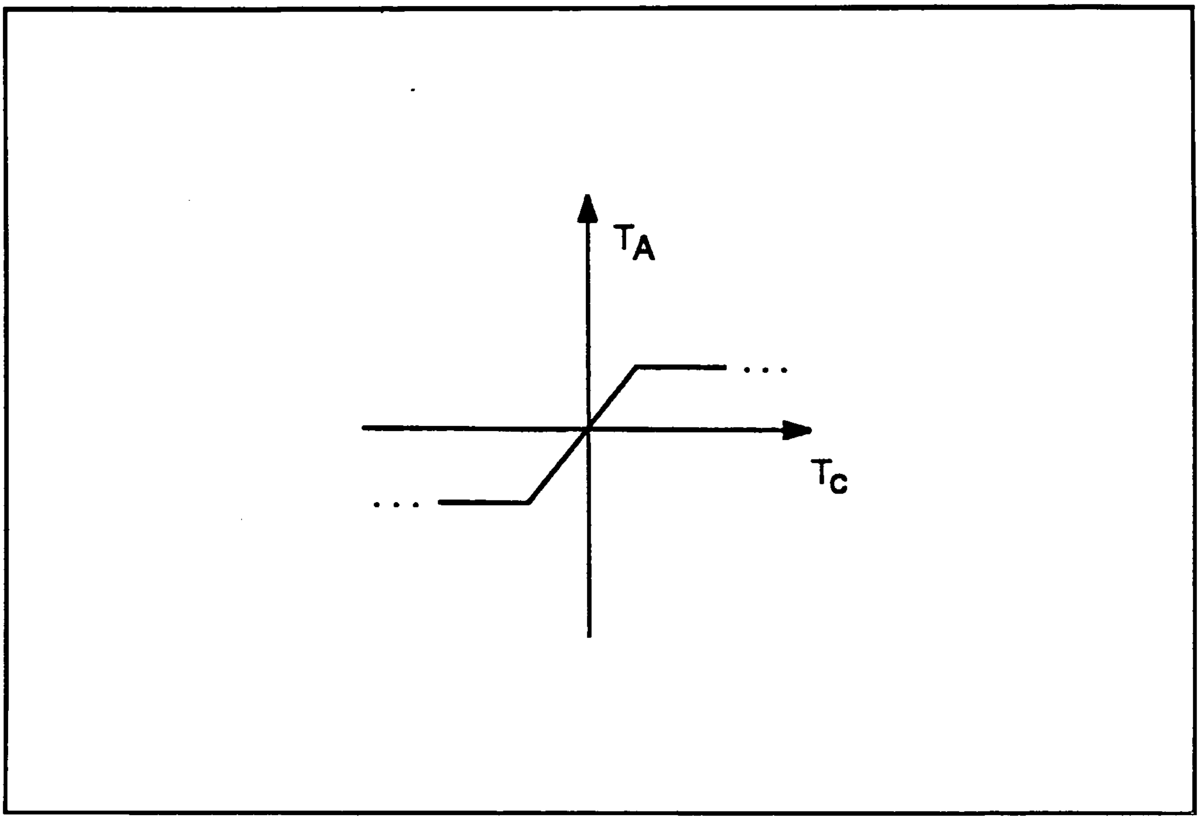

Figure 4-38 ITL, Saturation Limiter

In effect, placing these limiters on each transverse axis restricts the torque authority as shown in Figure 4-39. If the commanded torques, T_2 and T_3, are within the saturation limits, the angle ϕ_c is maintained. However, if one of the commanded torques exceeds the saturation limit, the result, shown in Figure 4-40, is that the angle between the applied torques, ϕ_d, is distorted. If both of the commanded torques exceed the saturation limit, the result is $\phi_d = 45$ degrees, independent of the relative magnitudes of the original commands.

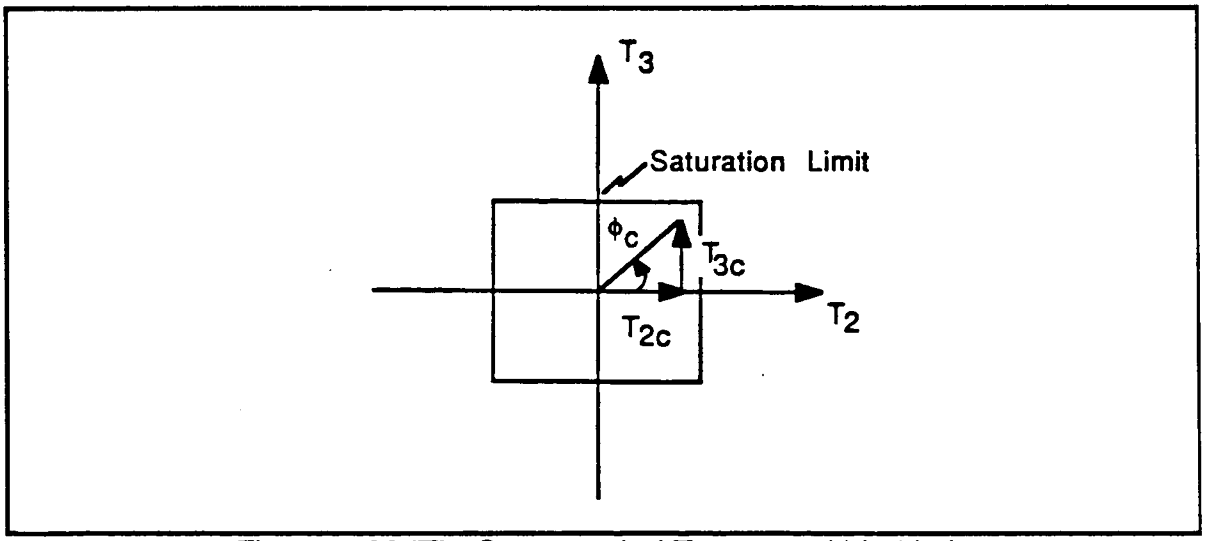

Figure 4-39 ITL, Commanded Torques within Limits

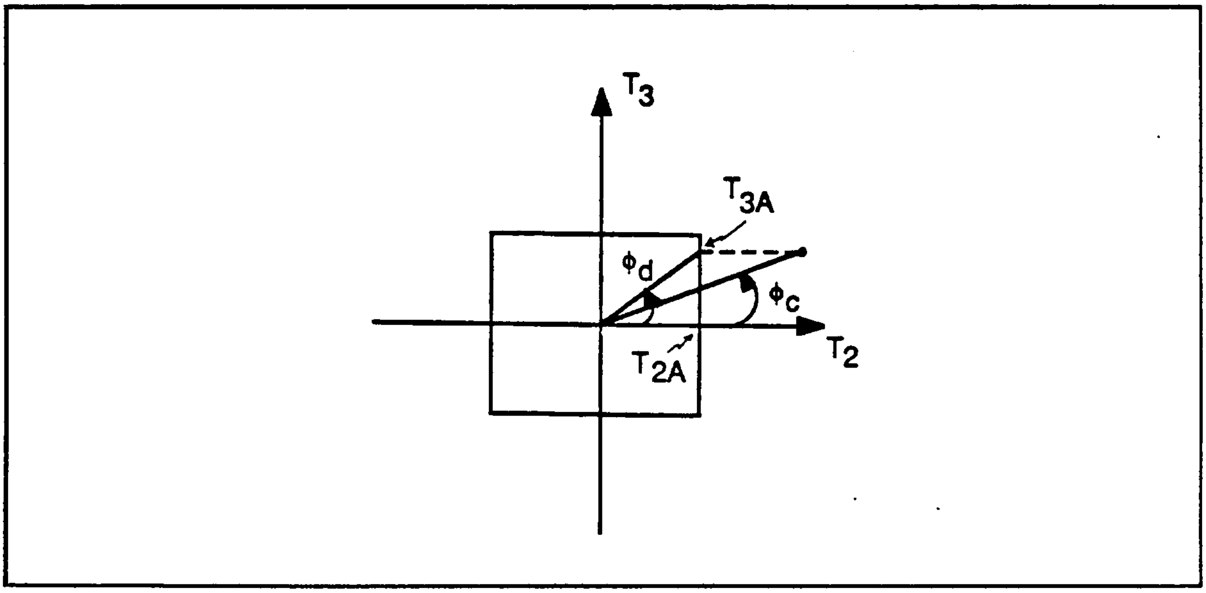

Figure 4-40 ITL, Commanded Torques out of Limits

In the DTL scheme the angle between the commanded torques, ϕ_c, is maintained in all cases. The angle, ϕ_c, is calculated initially; then, if one of the commanded torques exceeds the saturation limits, it is reset to the limit and the corresponding torque for the other axis is computed using ϕ_c. This is illustrated in Figure 4-41.

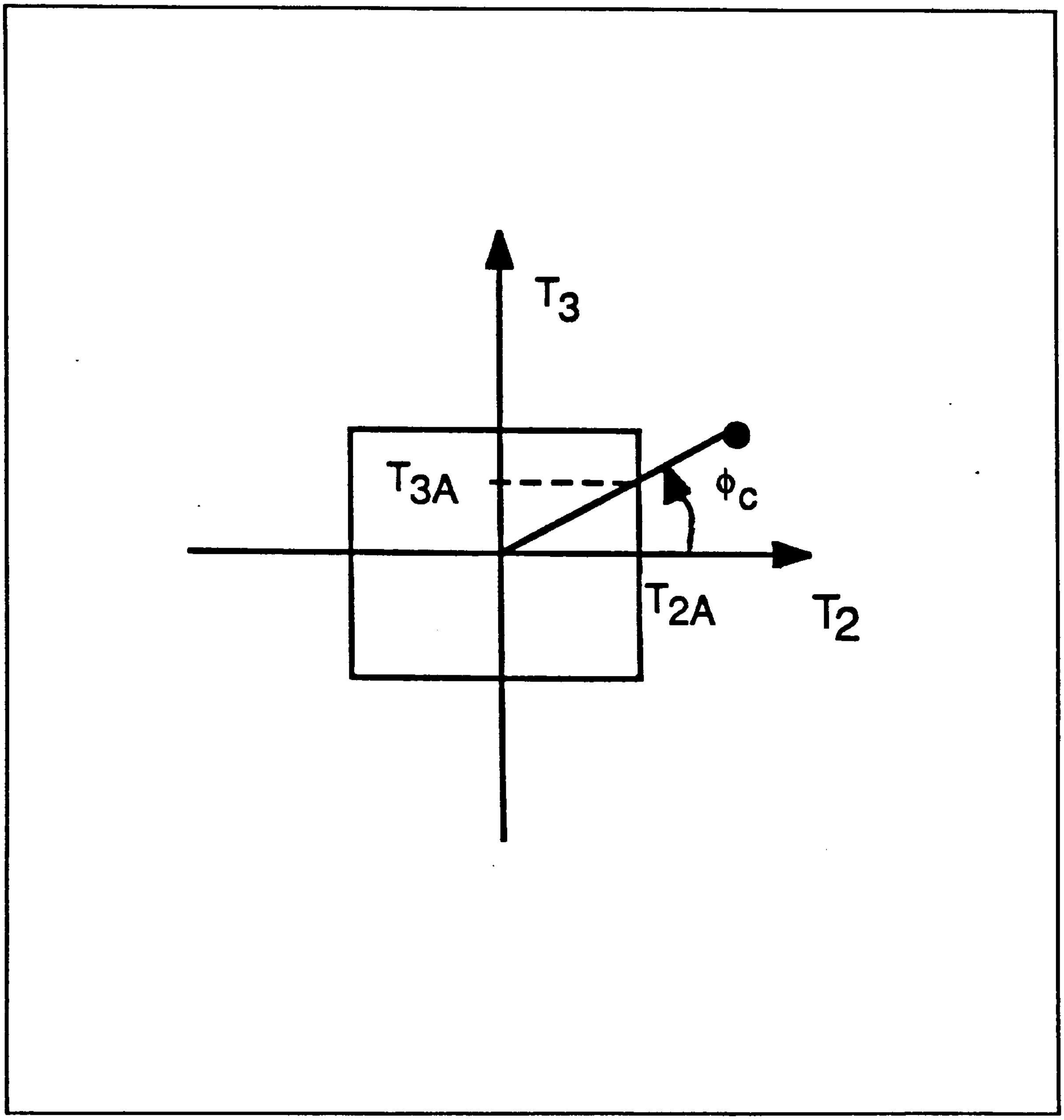

Figure 4-41 DTL, Commanded Torques out of Limits

The ITL and DTL techniques were simulated on a digital computer using the model shown in Figure 4-37. The results of the simulation for 57° and 28° reference inputs for θ_{2r} and θ_{3r}, respectively, are shown in Figure 4-42 for the precession controller and Figure 4-43 for the LQR design.

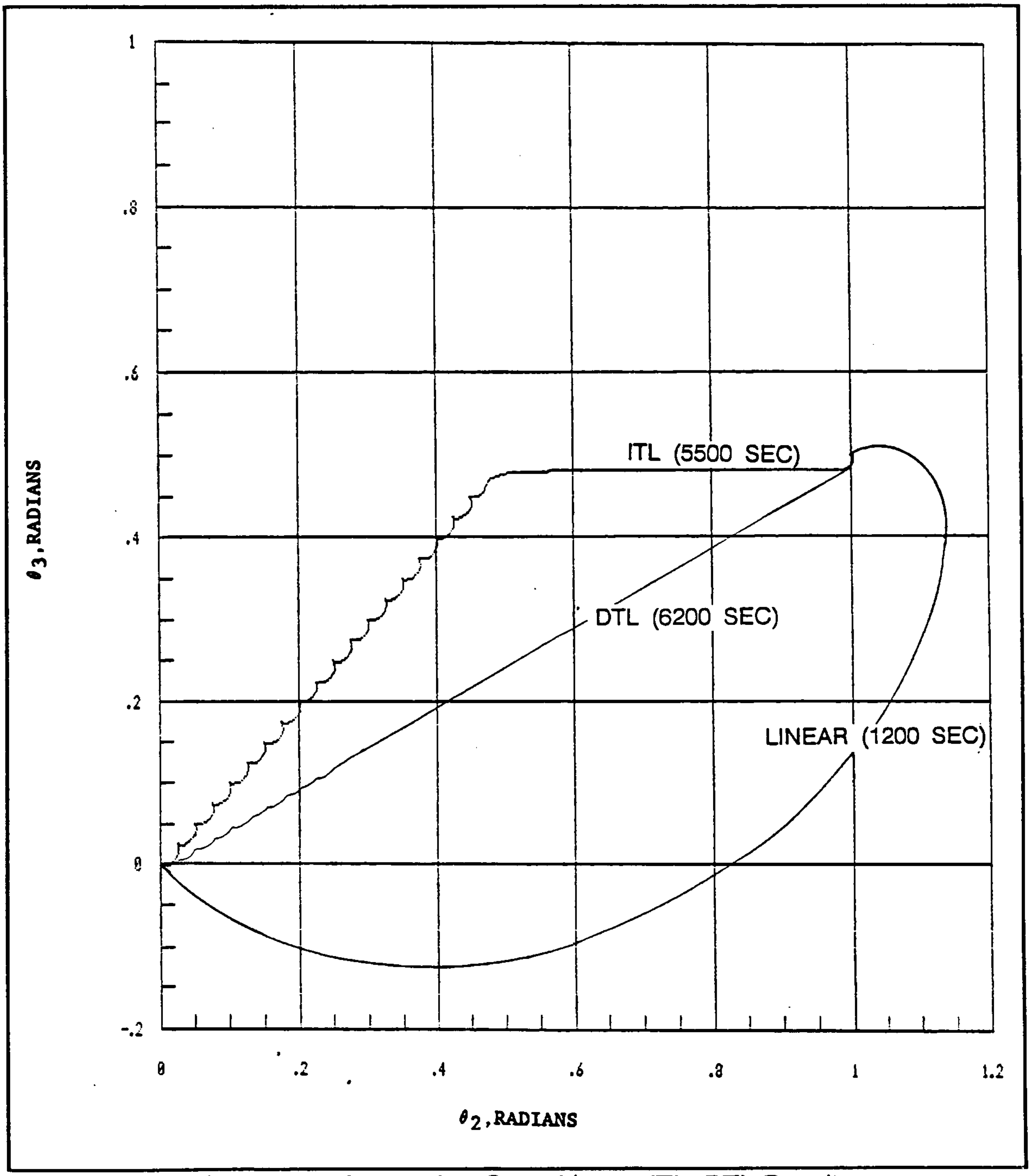

Figure 4-42 Precession Cont, Linear, ITL, DTL Results

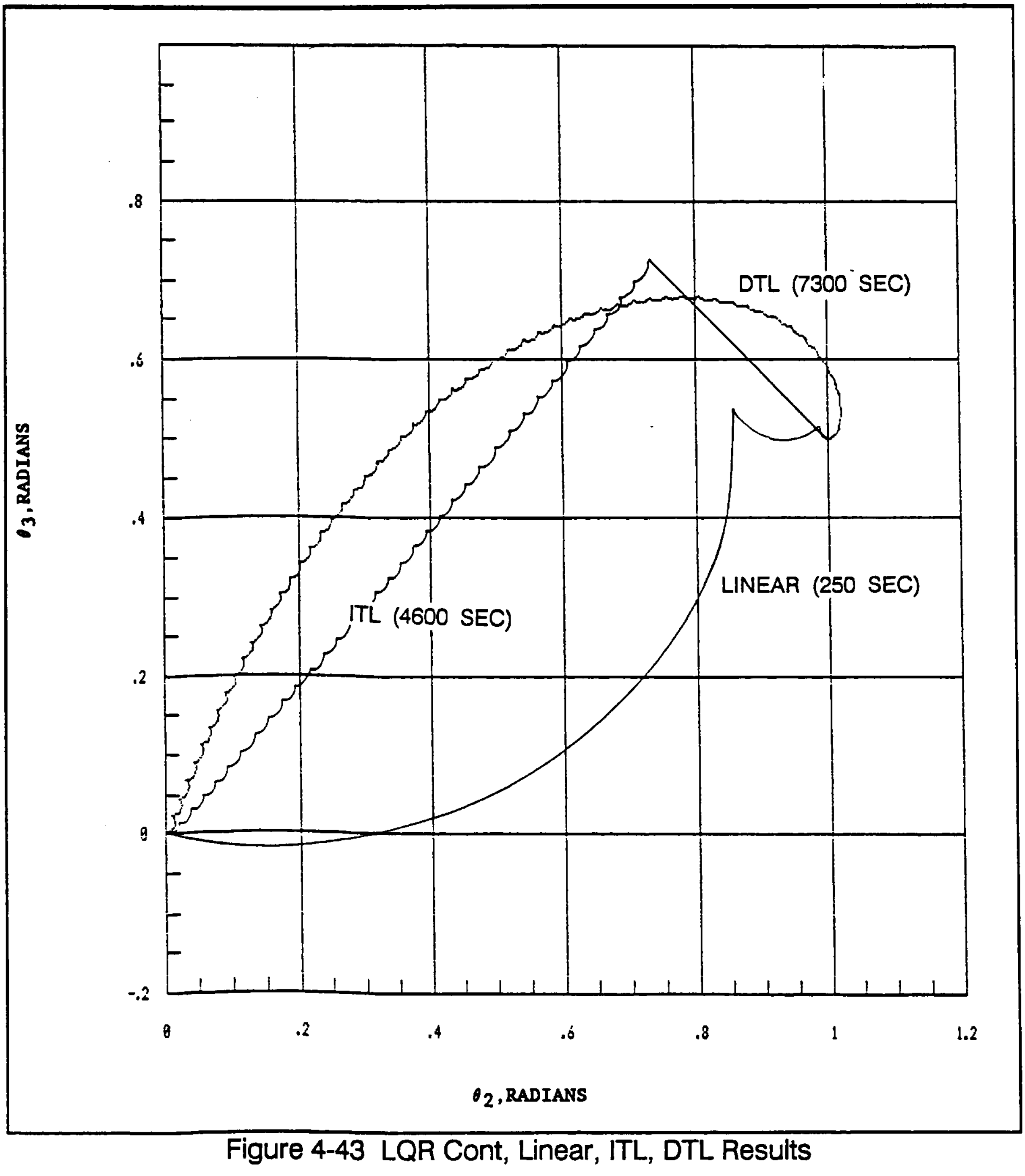

Figure 4-43 LQR Cont, Linear, ITL, DTL Results

Results of Saturation Torque Limiting

Table 4-3 summarizes the response to 57^{o} and 28^{o} step inputs to the pitch and yaw channels, respectively, for the independent and directional torque limiting schemes.

Table 4-3 Comparison of Torque Limiting Schemes

SCHEME	LINEAR	ITL	DTL
DESIGN			
Precession			
Settling Time	800 sec	5500 sec	6200 sec
Motion	circuitous	dogleg	direct
LQR			
Settling Time	250 sec	4600 sec	7300 sec
Motion	erratic	dogleg	circuitous

Naturally occurring independent torque limiting will significantly increase the settling times predicted in the linear analysis as shown in Table 4-3. The artificially induced DTL scheme will cause yet another increase in settling time for the large angle slew maneuver.

For the given plant, if the desire is to provide a direct slewing capability with little regard for the time required, the DTL approach may be appropriate. Several complexities are introduced with the addition of DTL.

The DTL scheme is implemented via software, so the algorithm must be generated and validated, resulting in additional cost and complexity. The code will require knowledge of the time varying saturation limit so that the angle between the applied torques can be maintained. This may require more effort.

An operational question must be addressed: How often will the mission require large attitude changes? If the answer is on the order of several maneuvers per day or week, perhaps a faster, less direct method is appropriate. On the other hand if the answer is seldom, on the order of once every couple of weeks or months, then the direct method may be acceptable.

Future Work

In the analysis thus far there appears to be a strong correlation between the speed of the closed-loop system and the "indirectness" of the nonlinear slew maneuvers. This effect should investigated further.

The two controller designs satisfy the overall requirements of the ADACS. Next a state estimator should be developed to predict the angle and angular rate information. But recall that angular rate information is not available. Once a satisfactory estimator has been designed, the entire closed-loop performance should be investigated.

5.0 Summary of Activities and Conclusions

The Doctor of Engineering internship at Ball Aerospace Systems Division was an overwhelming success. Objectives were satisfied and even surpassed in several cases due, in great part, to the excellent environment provided by Ball Aerospace and Mr. Bill Follett. Numerous opportunities were made available for the author to learn about and contribute to the spacecraft development business at BASD.

Internship accomplishments include:

- Developed a low-cost tactical satellite concept
- Created a systematic spacecraft design process
- Developed a draft *Spacecraft Systems Design Handbook*
- Performed Starscan satellite attitude control system design and analysis
- Produced various products for inclusion in BASD contract proposals

The author participated in an ongoing United States Space Command low-cost satellite panel discussion that culminated in the tactical satellite concept. The purpose was to develop concepts and methods to be used to provide low-cost orbital resources for various DoD elements. This concept was embraced by Space Command and was provided to the Defense Advanced Research Projects Agency for inclusion in the LIGHTSAT initiative. This activity also supported an advance marketing effort at BASD.

A systematic spacecraft design process was created to provide valuable insights into the necessary flow of activity, as well as a list of key design parameters and a description of the potential impact of various decisions on the overall spacecraft configuration. Major system-level trade studies were identified.

Elements of this design process were used in the largest multimillion-dollar contract proposal that BASD has developed to date. This document also will be used to train aspiring systems engineers in the spacecraft design process and as a basis for the description of the design process at BASD in future contract proposals.

The draft *Spacecraft Systems Design Handbook* was developed to complement the spacecraft design process. It provides numerous methods and approaches to initially size various spacecraft elements and components. It also will be used as a reference for spacecraft system studies at BASD.

The author performed and documented significant portions of the attitude control system linear design and analysis for the Starscan spacecraft, sponsored by the Air Force. The results of the activity were documented as a BASD systems engineering report and were provided to the Air Force Program Office.

Several tasks resulted in products that were or will be included in BASD contract proposals. A bottom-up cost estimate for the development of the attitude determination and control subsystem for a pair of NASA spacecraft was created in support of the overall proposal costing/pricing effort. The author also completed a low-cost, quick-turnaround component procurement study that was used to justify the feasibility of a low-cost, compressed schedule satellite procurement for DoD.

The Doctor of Engineering coursework provided a wealth of information that proved to be invaluable during the internship. Electrical and aeronautical engineering courses provided valuable insights that benefited the attitude control system design activity. The ethics course helped prepare the author to deal more effectively with several Government/contractor ethical dilemmas, and the financial and accounting coursework provided the necessary background for the cost estimating and pricing endeavors. System design courses proved to be the backbone of the spacecraft systems engineering activity. Other internship activities were based on advance marketing principles that were foreign to the author, and BASD's George Sayre provided the necessary information. Perhaps a marketing course should be added to the curriculum.

The author deeply appreciates the efforts of Dr. Howze, the Doctor of Engineering Committee at Texas A & M University, and Mr. Bill Follett and others at BASD. This exceptional opportunity would not have been possible, nor would it have been as successful, without their dedicated efforts, help and guidance.

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VITA

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Major Larson has fourteen years of documented successes in technical and leadership roles from radio equipment repairman as an Airman, to aircraft flight tester, to teacher, to Government program manager.

He has over five years of spacecraft testing and launch experience as spacecraft launch program manager and senior technical advisor for launch operations at Vandenberg AFB, California.

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