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A SURVEY OF CASE HISTORIES INVOLVING SPACECRAFT DYNAMIC INTERACTION

Contract NAS 12-110

30 March 1968

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Electronics Research Center
Cambridge, Massachusetts

TRW SYSTEMS GROUP

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SPACECRAFT DYNAMIC INTERACTION

DESIGN SURVEY

NASA/ERC DESIGN CRITERIA PROGRAM

GUIDANCE AND CONTROL

Prepared for

National Aeronautics and Space Administration
Electronics Research Center
Cambridge, Massachusetts

Contract NAS12-110

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1. INTRODUCTION

Recent experience has indicated that often the attitude control performance of a spacecraft is considerably poorer than that expected based upon mathematical analysis. This discrepancy is often attributed to an inadequate model representing the dynamic behavior of the spacecraft. It is evident that such discrepancies become even more apparent under the more stringent performance criteria and increased size, weight, and complexity of the spacecraft involved in recent programs. In order to focus greater attention upon potential problem areas, the Electronics Research Center has authorized TRW, as part of their design criteria contract, to perform a survey of case histories of spacecraft that have exhibited, in flight, an unexpected dynamic behavior. In addition, cases in which dynamic modeling effects of lesser importance were considered in detail during the vehicle's design, were also to be included in this survey, since such a design practice often identified possible problem areas and motivated corrections that prevented the spacecraft's flight performance from being impaired.

This survey was limited to obtaining the most readily accessible information and results in the least time penalty. Thus, for the most part, spacecraft involved in military missions are not considered in this report. Since civilian space programs are all administered by the various NASA agencies, primary emphasis was placed upon contacting these agencies to uncover examples for the survey. Generally this involved either telephone calls or personal visits by the author to NASA engineers who have come in closest contact to such spacecraft dynamic problems. In a couple of instances, talks were conducted with people associated either directly or indirectly with universities. No industrial contacts were made (other than TRW), nor was an extensive literature survey performed, as this was outside the scope of the investigation. Unfortunately, much of the detailed information concerning past dynamic problems is buried in company documents that are not easily accessible. Some of these documents have been accumulated by NASA agencies, however.

The approach adopted in discussing survey examples with the engineers visited consisted essentially of the following questions to be answered for each case.

1) Name of Satellite
2) Physical Description (size, weight, shape, etc.)
3) Launch Date and Orbit Characteristics
4) Mission Objectives
5) Attitude Control Requirements
6) Type of Attitude Control
7) Nature of Secondary Dynamic Motion Encountered or Anticipated
8) Cause of Secondary Dynamic Motion
9) Nature of Preflight and Postflight Analyses Performed

10) Actual or Expected Remedies

11) Influence Upon Future Design Practice

12) Responsible Government Agency

13) Responsible Civilian Agency

14) Pertinent Available Published Reports

15) Source of Information

Appendix includes each of the survey spacecraft in the above format, although in most cases some of the above questions either could not be answered or were not pertinent.

This report now continues with a discussion (Section 2) of the approach conventionally adopted in the design of attitude control systems. It consists of two phases: the first involves a fairly simple primary model of the spacecraft dynamics; and the second investigates the influence of secondary modeling effects. This section also considers the performance criteria generally encountered in control system design, and categorizes conventional control systems according to the tools available for their analysis.

Section 3 adapts the general concepts of the second section to the survey considered in this report; and Section 4 summarizes the case histories of the Appendix, demonstrating major trouble areas of the past as well as present day design trends.

2. CONTROL SYSTEM DESIGN AND ANALYSIS - THE GENERAL CASE

2.1 THE CONVENTIONAL DESIGN AND ANALYSIS APPROACH

The great majority of spacecraft designed have possessed some form of attitude control system. These have ranged from rather simple systems for controlling the spin axis of a spinning vehicle, to highly sophisticated control systems employing a variety of sensors to provide extremely accurate three-axis control. One of the most important elements of control system design is an accurate analytic representation of the plant to be controlled, in this case, the spacecraft. Despite the rather wide range of spacecraft that have come under scrutiny, the design of an attitude control system normally follows a fairly conventional approach consisting of two phases.

1) A reasonably simple mathematical model of the spacecraft's dynamic behavior is adopted, and an attitude control system designed in conjunction with this model in order to meet the mission performance criteria. This model generally includes only the primary dynamic effects in order to avoid undue complexity. After roughly sizing the control system hardware required, the control system is designed and analyzed for the spacecraft in its normal mode of operation, in which the attitude errors are normally very small relative to some ideal reference. This design utilizes all the techniques of conventional control theory along with analog and/or digital simulation aids, and generally results in a design that is relaible, and provides attitude stability and reasonable insensitivity to external disturbance and system parameter changes. Utilizing the same model, the attitude performance of the spacecraft in acquisition (in which case the attitude angles are normally large) is then determined. The control system is subsequently modified, if necessary, to meet the performance criteria in this mode of operation.

   The mathematical model assumed for the spacecraft in this phase will be termed the <u>primary dynamic model</u>, and the resulting dynamic behavior will be called the <u>primary dynamic motion</u>. It is assumed that the control system designed in this phase using the primary model results in a primary dynamic motion that meets the performance criteria.

2) Having arrived at a tentative control system design, it is now necessary to determine whether the design remains adequate when a more sophisticated mathematical model of the spacecraft is employed. This may require use of an accurate computer simulation which may be impractical for synthesis but holds extreme merit in design verification. If a general simulation fitting the analyst's needs does not exist, and if time and/or budget make it impossible to generate a new simulation, it is often impossible to adequately fulfill the objectives of Phase 2.

   Any mathematical modeling effect considered in Phase 2 but not Phase 1 will be called a <u>secondary modeling effect</u>; and the degree to which the dynamic behavior differs from the primary dynamic motion will be termed the <u>secondary dynamic motion</u>.

Since it has been assumed that the control system performs adequately with the primary model, it is normally desirable that the secondary dynamic motion have little effect upon the adopted performance criteria.

Dynamic models of spacecraft utilized in the analysis of their performance can be categorized by several characteristics used in their description. The left hand column of Table 2-I indicates the pertinent characteristics; the other columns give specific identifications normally encountered for each. Thus, for example, the primary model for a dual spin satellite in normal mode operation involves a constant parameter, two body, four degree-of-freedom system undergoing small attitude deviations relative to its ideal motion.

As has been previously noted, small attitude deviations can normally be assumed when the spacecraft is in its normal mode of operation. However, the other entries of Table 2-I depend strictly upon the nature of the spacecraft itself. Those characteristics judged to be of prime importance become part of the primary model. The remaining entries of Table 2-I are either taken as secondary modeling effects for study under Phase 2, or are ignored entirely.

Aside from the third characteristic of Table 2-I, the conventional primary models that have been utilized in past control system designs are:

1) One, rigid, constant parameter, body with one (or sometines three) rotational degrees-of-freedom. This model covers, by far, most of the spacecraft designed in the past.

2) Two or more, rigid, constant parameter, coupled bodies with three rotational degrees-of-freedom. Most gravity gradient and dual-spin spacecraft fit in this category, as well as vehicles whose movable apendages have significant inertia properties (e.g. large solar array).

3) One flexible body with one (or three) degrees-of-freedom and viscous (velocity) structural damping. Some gravity gradient spacecraft and the larger present day vehicles fit into this category.

2.2 SPACECRAFT PERFORMANCE CRITERIA

The conditions under which a spacecraft operates can usually be characterized as either mission-operational or

Table 2-I. Characteristics of Spacecraft Dynamic Models

1) Number of bodies	1		2 or more	
2) Rotational degrees-of-freedom	1		more than 1 per body	
3) Attitude deviations	small		large	
4) Property of bodies	rigid, constant parameter	rigid, changing parameter	flexible	
5) Flexibility deviations			small	not small
6) Flexible energy dissipation			none	structural damping

mission-nonoperational. In the former category, the spacecraft is normally performing its mission objective and thus, usually experiences only small attitude errors. The performance criteria, in this case, usually involve the attitude error and/or the error rates since the spacecraft pointing inaccuracy often influences the basic mission objective. The accuracy to which a satellite can be expected to hold its reference is influenced by several factors.

1) External disturbance forces and torques acting upon the vehicle.

2) Significant dynamic modeling effects not accounted for in design.

3) The degree to which the spacecraft parameters (and upon which the control system design is based) are unknown.

4) The degree to which the control system receives unreliable information concerning the attitude state of the spacecraft.

Typical examples of the mission-nonoperational mode are the initial spacecraft acquisition and any subsequent reorientations. In this case, the attitude errors often become quite large making it impossible to validly linearize the corresponding dynamic and kinematic equations. Attitude accuracy is generally not a severe requirement in this mode, being replaced, first by stability requirements, and then, usually, by the desire to make the spacecraft mission-operational within a relatively short time. A reasonable criterion, in this case, is to design the control system such that the spacecraft's nonoperational time is minimized.

Several criteria are important in both modes of operation, particularly, the required control effort (fuel or power) needed to do the job. The expected control effort is evidently influenced by the same factors affecting the attitude accuracy noted above.

The desire to increase the control system (and sensor) reliability and yet reduce its weight and cost is also applicable in most practical situations. Finally, a design should be selected which makes it possible to determine, with reasonable ease and confidence, the expected dynamic behavior of the satellite. A simple design that is impossible to analyse has questionable merit, unless a simple flight test program exists for its evaluation.

The most pertinent criteria commonly encountered in spacecraft applications have just been reviewed. The importance of one relative to the others is obviously a function of the satellite's mission. Often one of these criteria will be selected for minimization on the condition that the remainder satisfy certain constraints. Thus, for example, a typical mission-operational criterion is the minimization of average control effort while restricting the attitude error to less than some maximum bound. The roles of these two criteria can clearly be reversed by changing the nature of the mission.

2.3 GENERAL TYPES OF SPACECRAFT ATTITUDE CONTROLLERS

A spacecraft attitude control system generally consists of a number of disjoint controllers, which are defined as identifiable separate torquing devices. Most practical controllers can exert torque only about a single spacecraft axis, and thus can be said to have an output dimension of one. A three-axis attitude control system normally employs three such controllers, but less than three can sometimes suffice by taking advantage of dynamic coupling in the system. Some controllers can provide independent torques along two "noncolinear" spacecraft axes and therefore are said to have an output dimension of two. A two-degree-of-freedom control moment gyro typifies this situation. Evidently controllers of output dimension three can also be devised.

Conventional spacecraft controllers can be categorized in many ways. To the control system analyst, a convenient breakdown is based upon the impact that the various controllers have upon the methods that can be employed in design and analysis. The following represents this kind of categorization.

1) Controller quiescent for its input (normally some attitude variable) within some deadzone about its ideal value.

2) Controller active for "non-ideal" inputs.

   a) Controller output continuous with its input
      i) Linear output-input relationship
      ii) Nonlinear output-input relationship

   b) Pulse control

Category 1 represents a large number of past controller designs. Bang-bang control of an attitude variable using either gas jets or reaction wheels is the most popular example in this group. The conventional tools of analysis for this category are use of the phase plane, the rate diagram, state transition equations, and, less frequently, describing function methods. Often computer simulation yields the most information concerning the system dynamic behavior.

Category 2a provides the most advantageous tools for analysis of normal mode control. In this case, the system equations are linear and the frequency domain methods of Bode, s-plane methods, and root locus analysis all come within the analyst's grasp. A firm understanding of the system behavior can be obtained without resorting to time response simulation. Category 2a-(i) often possesses an advantage over 2a-(ii) if a single-axis study is being conducted, because the dynamic and kinematic system equations are usually linear even for large attitude deviations. In this case, the above mentioned methods still apply. Typical of continuous, linear controllers are systems employing reaction wheels or proportional gas jets in conjunction with linear electronic circuitry. An example of a continuous, nonlinear controller is a control moment gyro whose input torque depends linearly upon its error signal.

Category 2b is typified by fixed torque gas jets whose on-time frequency and/or pulsewidth depend in some way upon the controller error signal. Such systems are generally difficult to analyse in normal mode operation and must be treated by methods similar to those of Category 1. However, when the controller error signal is large, the pulse repetition rate is generally beyond the bandpass of the system dynamics, the latter then being sensitive only to the average control impulse imparted. Thus, for large errors, the pulse controller can be replaced by an equivalent linear controller, and the analysis is considerably simplified particularly in single-axis studies.

The above breakdown applies even if the controller input information is sampled rather than continuous. Conventional z-transform techniques apply to Category 2a in normal mode, and the analytic methods for the other categories remain essentially the same.

3. CONVENTIONS ADOPTED IN THIS STUDY

The majority of case histories to be documented in this report concern spacecraft which, in the initial control system design phase, can be represented dynamically as a single, constant parameter, rigid body. Effects judged to be of lesser importance were usually considered at a later date (and then often only briefly) or ignored entirely. This lack of attention to these latter effects is responsible for the inclusion of many of the examples to be subsequently cited.

In view of the design approach adopted for most of these spacecraft, and in order to present a unified definition when referring to the various case histories, it will be assumed that the primary model for each of these examples consists of a single, constant parameter, rigid body. Effects falling outside this definition are then the secondary modeling effects considered in Phase 2 of the conventional design approach, or possibly in postflight analysis. Secondary effects arise primarily from three causes.

1) Flexibility of the spacecraft main body, either dynamically or thermally induced.

2) Movable controlled or uncontrolled elements attached to the spacecraft main body. Controlled antennae or solar arrays fit in the former category; gyros and dampers belong to the latter.

3) Changing main body mass or moments of inertia. Examples of this are spacecraft employing mass expulsion for velocity correction or for control; or the motion of crew members in a manned satellite.

In each documented example one or more of these effects was either present, or was at least considered at some point in the attitude control system design. Many of these examples involved secondary modeling effects that resulted in excessive secondary dynamic motion in actual flight.

4. RESULTS AND CONCLUSIONS OF THE SURVEY

The detailed results of the survey are contained in the Appendix which documents each of the spacecraft on which a secondary modeling effect was either considered in design or encountered in flight. The results based on this appendix are summarized chronologically in Table 4-I where the symbol "A" indicates an effect actually encountered, and the symbol "D" applies to a design consideration.

The secondary modeling effect involving spacecraft flexibility is broken into two types: that involving flexibility of the main body itself, and that concerned with structural deformations of appendages or booms. The latter are, likewise, considered to be two types: deformations that are dependent solely upon the structural properties of the booms, and those that depend not only upon these properties but upon temperature gradients within the member.

The second main category involves nonrigid auxiliary bodies on the spacecraft. As can be seen by Table 4-I, solar arrays furnish most of the examples. Solar arrays are treated as auxiliary bodies, rather than a flexible part of the main body, since the paddles themselves are normally fairly rigid with most of the deformation taking place in a short complient shaft which attaches it to the rest of the vehicle. Fuel slosh is also treated as an auxiliary body due to its equivalent representation as a damped spring-mass system. Other examples of considered auxiliary bodies are the scan platform of Mariner IV, the sun shade of OAO 1 and the ergometer utilized by Apollo astronauts for exercise.

The third major category is that involving changing parameters in what would otherwise be a perfectly rigid spacecraft. For the most part the concern here is with changes in the location of the vehicle center-of-mass causing an upsetting torque when the spacecraft is in a powered flight mode of operation.

The following conclusions are immediate consequences of the survey.

1) Flight problems of the past have predominently involved flexibility of the spacecraft, and in most cases the secondary dynamic motion was attributed to deformation of one or more appendages, or booms.

2) Past examples of boom deformation have usually involved not only its structural characteristics but, in addition, have been severely dependent upon boom temperature gradients. This has been particularly true of gravity-gradient stabilized satellites. Mathematical modeling of such effects are very complicated. The greatest hope appears to be in the area of boom technology – the construction of extendible members that are tortionally much stiffer, reflect most of the incident radiation, and conduct heat with sufficient ease to cause a fairly uniform temperature throughout the appendage.

3) With the exception of two of America's earliest satellites, which were spun up about an axis of minimum moment of inertia, main-body flexibility has created no flight problems thus far. In fact, main-body flexibility was

Table 4-I. Summary of Secondary Modeling Effect Case Histories

Spacecraft	Launch Date	Type of Attitude Control	Nature of Secondary Modeling Effect: Flexibility of Spacecraft: Booms: Thermal-Structural	Flexibility of Spacecraft: Booms: Structural	Flexibility of Spacecraft: Main Body	Effect of Auxiliary Bodies: Solar Array	Effect of Auxiliary Bodies: Slosh	Effect of Auxiliary Bodies: Other	Changing Spacecraft Parameters
Explorer I	1 Feb 1958	Spin stabilized			A				
Pioneer I	11 Oct 1958	Spin stabilized			A				
Mariner II	22 July 1962	Gas jets for pitch and roll control				D			D
Alouette I	28 Sept 1962	Spin stabilized	A						
1963-22A	15 June 1963	2-axis gravity gradient	A						
1963-49B	6 Dec 1963	2-axis gravity gradient	A						
Ranger(s)	28 July 1964- 21 March 1965	Gas jets for pitch and roll control				D			D
Explorer XX	25 Aug 1964	Spin stabilized	A						
Mariner IV	28 Nov 1964	12 gas jets for full control				D			D
1964-83D	12 Dec 1964	2-axis gravity gradient	A						
OAO 1	8 Apr 1966	3 magnetic torquers; 18 jets; 6 wheels						D	
Surveyor(s)	30 May 1966- 8 Sept 1967	6 gas jets; 3 vernier engines			D				
GGTS	16 June 1966	2-axis gravity gradient							A
OGO III	7 June 1966	3-axis gas jets and reaction wheels		A					
Lunar Orbiter(s)	10 Aug 1966- 1 Aug 1967	Gas jets—3-axis				D			
OV1-10	11 Dec 1966	3-axis gravity gradient	A	A					
Dodge		3-axis gravity gradient	A						
OGO IV	28 July 1967	3-axis gas jets and reaction wheels	A						
RAE	mid 1968	3-axis gravity gradient	D						
SERT II	(1969)*	Gravity gradient with Roll-Vee gyros			D	D			
Apollo	(1970)	Total of 44 gas jets			D		D	D	D
Voyager	(1975)	3-axis gas jets			D		D		
Galactic Probe	(1973)	Spin stabilized			D				
AOSO	Canceled	Gas jets, reaction wheels, magnetics			D	D			
Space Probe	-	-	D	D	D				

*( ) - predicted launch date,D - Problem considered in design, A - problem experienced in flight.

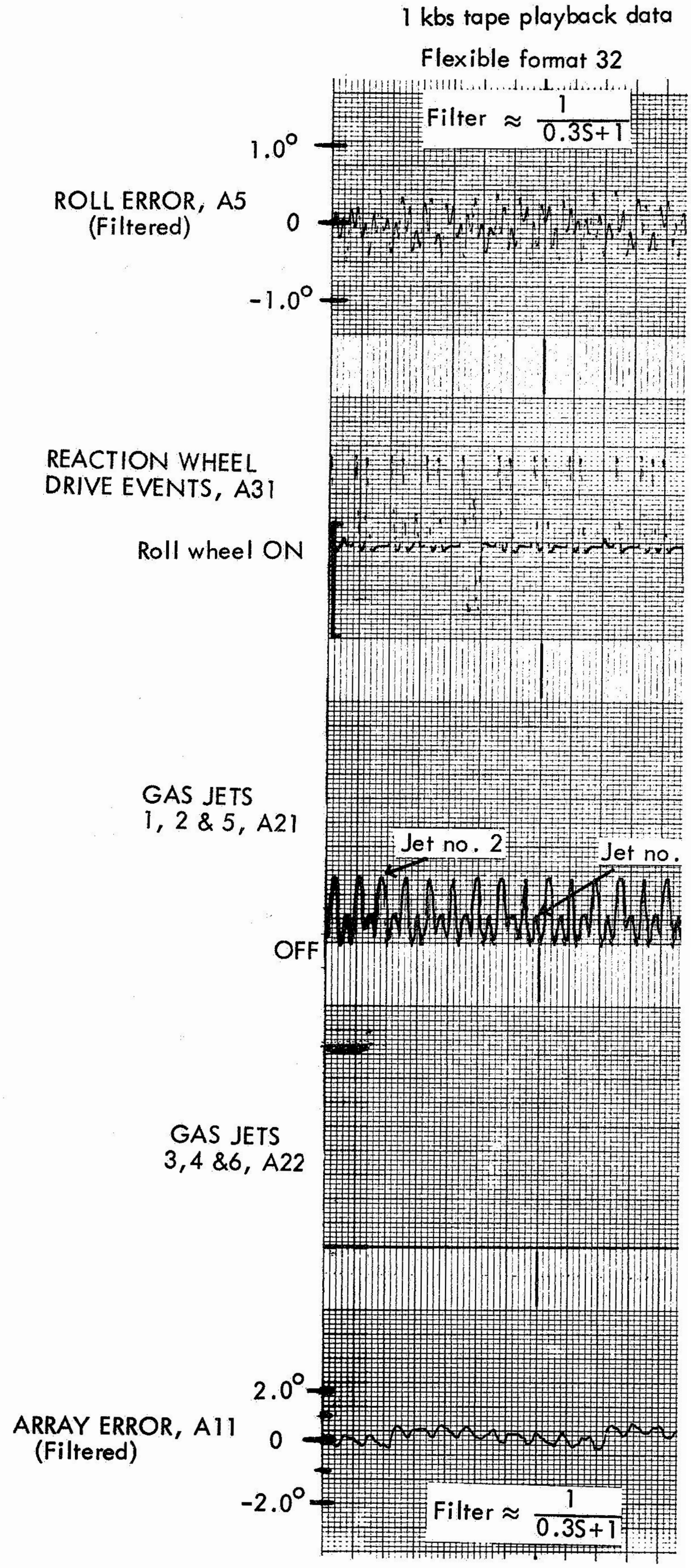
hardly even considered in most of the earlier spacecraft programs. But the trend is changing due to increasingly severe attitude accuracy requirements, and due to the need of present programs of larger, and thus more flexible, spacecraft.

4) Considerable design attention has been devoted to the influence of auxiliary bodies. Rarely have they resulted in significant secondary dynamic motion. In particular, attention has focussed right from the beginning on the effects of a nonrigid solar array, but no examples were found where the array motion had an adverse effect upon the spacecraft.

5) The effect of changing spacecraft parameters, particularly the location of its center-of-mass, has created no flight problems. It is likely that many potential problems of this nature were identified in the design phase and appropriate modifications made.

A vivid example of an adverse interaction between the spacecraft and its attitude control system occurred on OGO III which employs bang-bang reaction wheels with a ±0.4-deg deadzone as the primary roll control means; and a gas jet backup system to remove the momentum buildup caused by noncyclical disturbance torques. It was already noticed in OGO III's first orbit that the roll wheel was undergoing much larger excursions than anticipated and was apparently experiencing great difficulty in controlling the vehicle's roll angle. Eventually the solar array (which is controlled in roll) began to oscillate also and the roll wheel was driven to its saturation limit in its futile attempt to control the attitude. Eventually the gas jet deadzone was exceeded, and the spacecraft settled into a fuel-consuming gas jet limit cycle. (Fortunately, it was possible to alleviate this situation by ground command.) Figure 4-1 indicates the data telemetered from OGO III just as the gas jets joined the cyclic symphony. Analog simulation eventually attributed this problem to the flexible behavior of two experiment appendages deployed essentially along the vehicle pitch axis. The damping characteristic inherent in these booms was much less than anticipated, causing the flight behavior noted.

On some of the more sophisticated programs of the present, it is evident from the start that spacecraft flexibility must be viewed with great concern. This is particularly true for the massive Apollo for which the prime consideration centers upon the safety of its crew. In order to estimate its structural modes, the Apollo has been modeled in three dimensions as a system composed of a very large number of mass elements interconnected by flexible, massless beams of appropriate stiffness. The complexity surrounding such a simulation is evident from Figure 4-2 which illustrates the mass elements and the corresponding vectors denoting their deformation for one of the low-frequency structural modes. Despite the care involved in such an undertaking, uncertainties in assigning model parameter values, and the effects of nonlinearities such as backlash in joints, result in about a 20-percent uncertainty associated with the model frequencies obtained, and an even lower confidence in the mode shapes. Thus, the control system designer is faced with the challenge of designing a system that will perform adequately for quite a range of possible plants, even in cases where potential problem areas have been identified. Uncertainties in the system dynamic model are, in fact, the major obstacle facing the designer. There appears to be fairly general agreement that once the spacecraft plant has been identified tangibly, the problem of designing an adequate control system is usually much less severe.


1 kbs tape playback data
Flexible format 32
Filter ≈ $\frac{1}{0.3S+1}$
1.0°
ROLL ERROR, A5
(Filtered)
0
-1.0°
REACTION WHEEL
DRIVE EVENTS, A31
Roll wheel ON
GAS JETS
1, 2 & 5, A21
Jet no. 2
Jet no.
OFF
GAS JETS
3,4 &6, A22
2.0°
ARRAY ERROR, A11
(Filtered)
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Filter ≈ $\frac{1}{0.3S+1}$

11 A

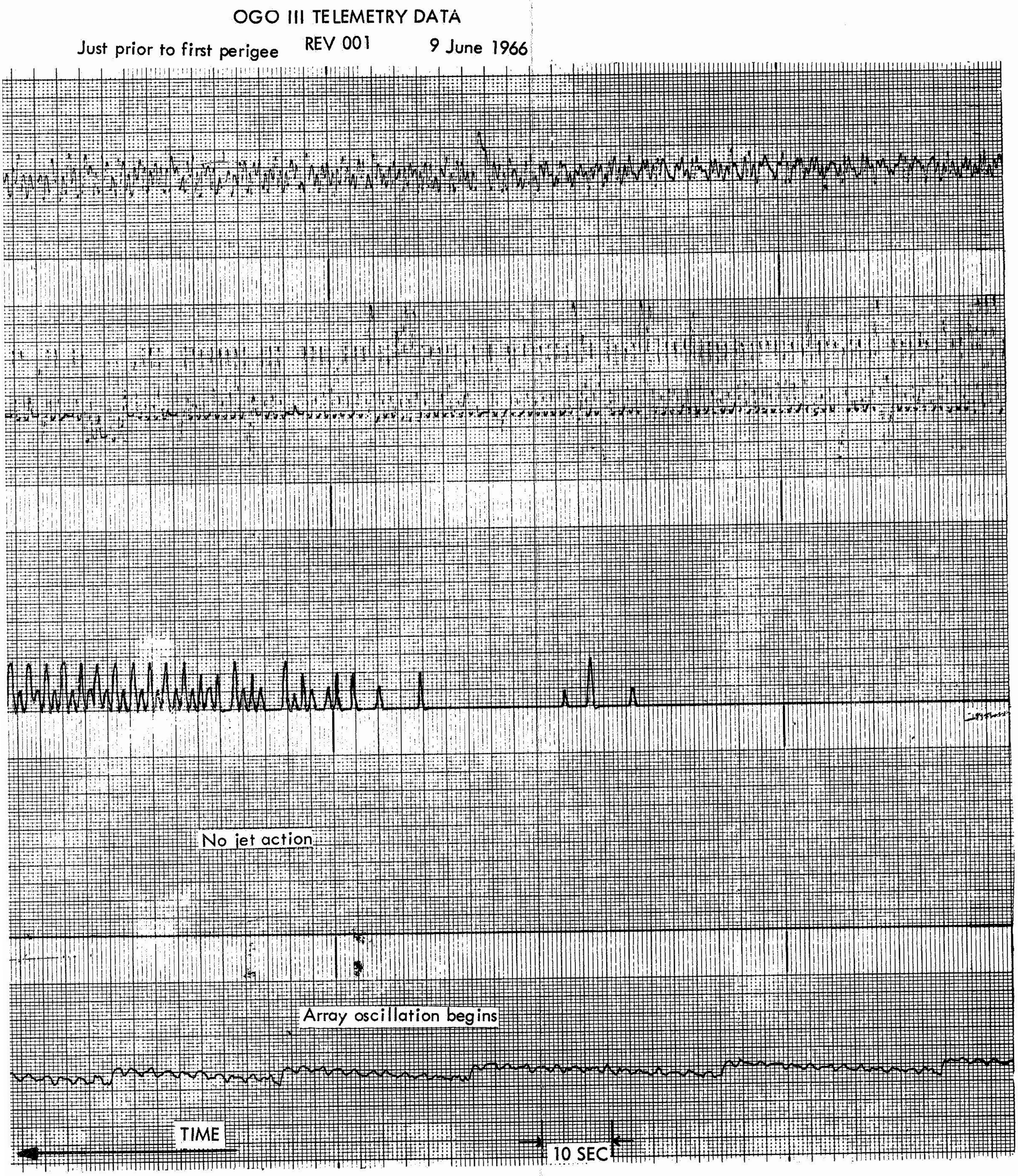
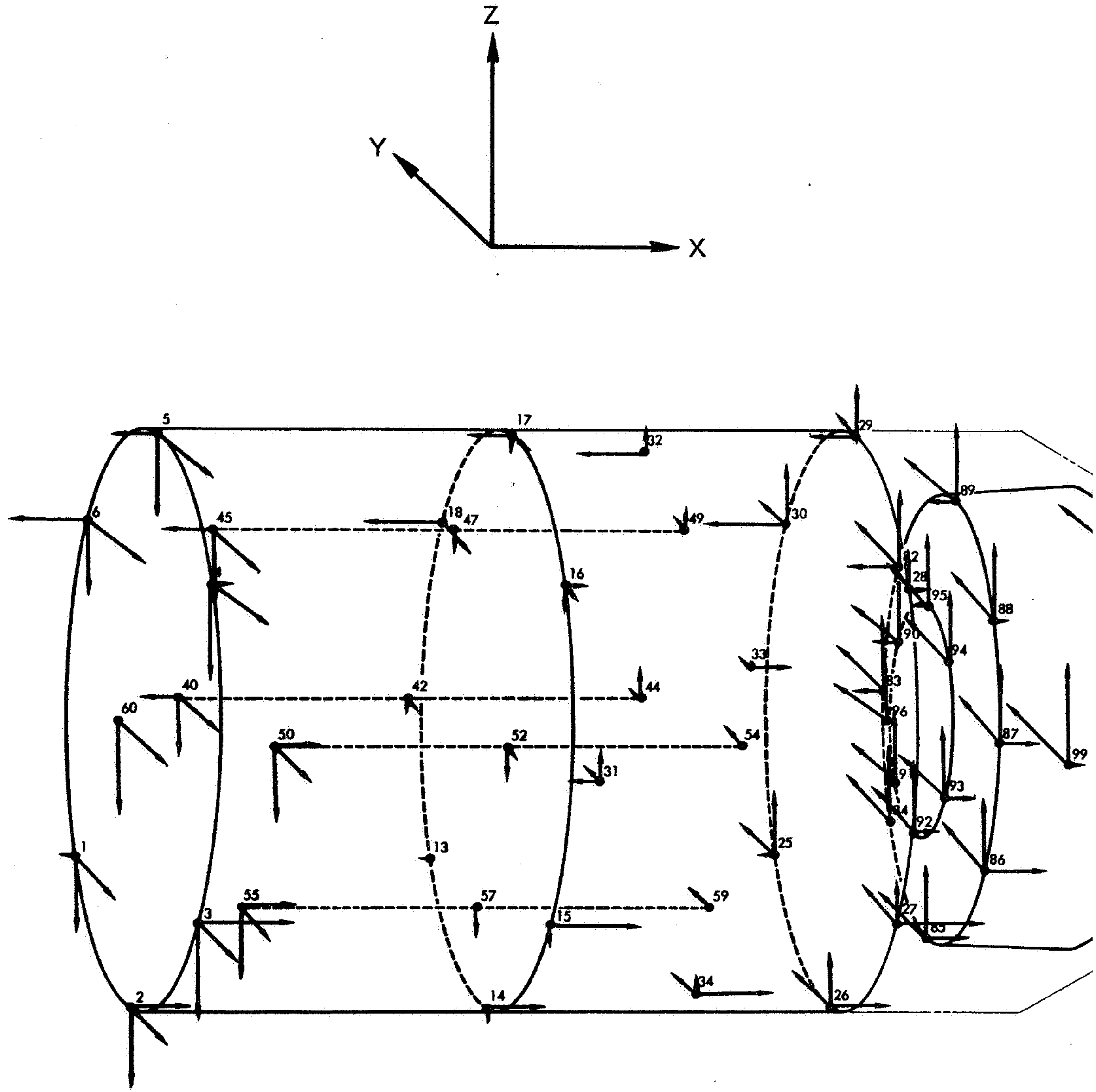


Figure 4-1. OGO III Telemetry Rev 001 Expanded Scale Start of Gas Limit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12A

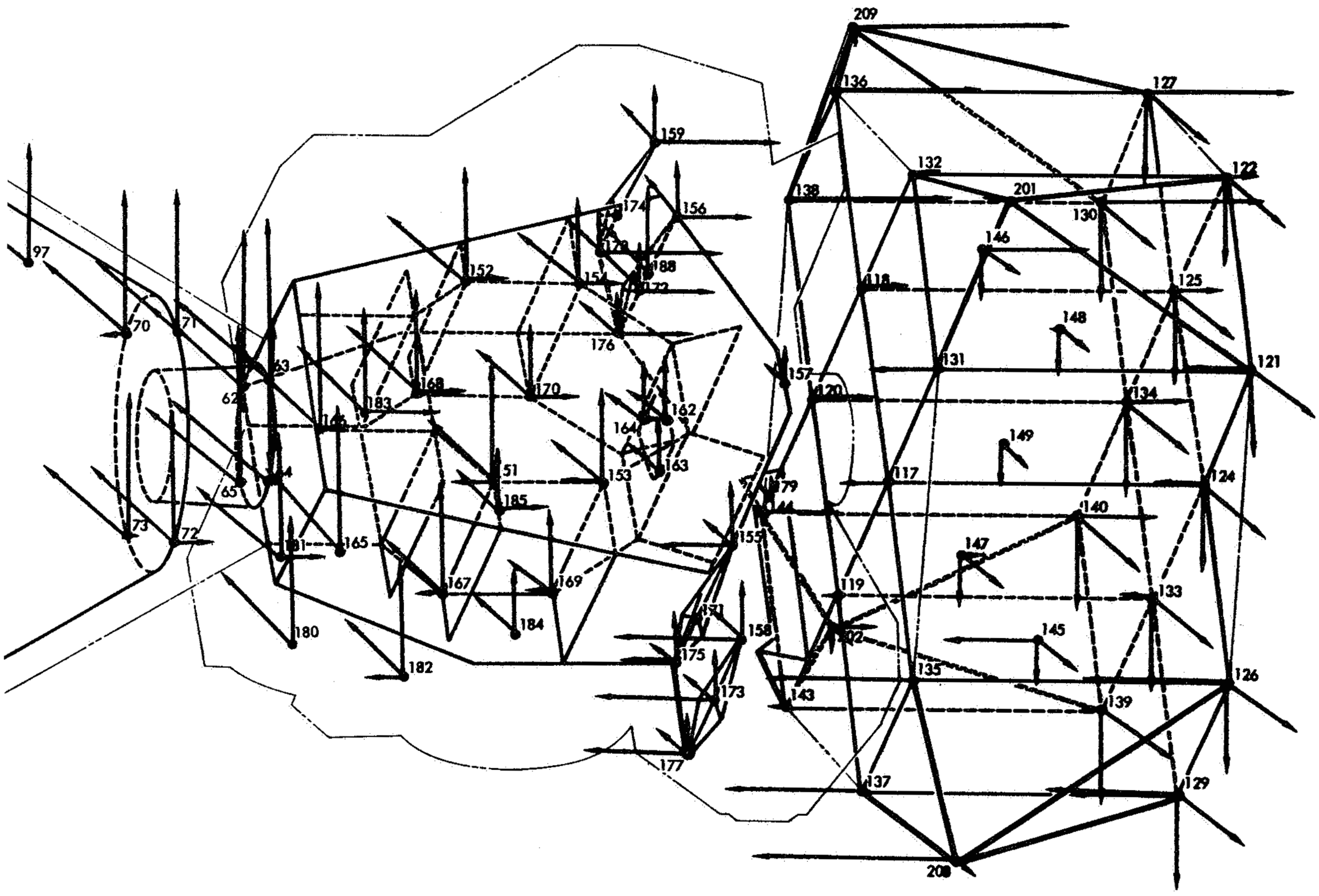

Figure 4-2. CSM/LM Mode 1
Frequency = 2.20 Hz

APPENDIX
CASE HISTORIES OF SECONDARY DYNAMIC MOTION

NAME OF SPACECRAFT: SPACE PROBE

Physical Description: Small center body from which extend two very large thin-film solar array panels. Length from tip to tip ranging from 45 to several hundred feet in later versions. Weight - 600 lb minimum.

Mission Objectives: Designed for interplanetary data gathering missions and utilizing an ion engine for thrust control.

Attitude Control Requirements: The ion engine thrust vector must be pointed with reasonable accuracy. Control about the thrust vector is of lesser importance.

Cause of Secondary Dynamic Motion: The thin-film solar arrays, which are extended like a window-shade by deployable booms, results in a highly flexible structure which, in addition, is sensitive to temperature gradients. These characteristics can have severe effects upon the controllability of the spacecraft.

Responsible Government Agency: NASA/JPL

NAME OF SPACECRAFT: MARINER IV

Physical Description: Octagonal main body 50 in. across. Four solar panels extend from one end of this body which also holds an antenna dish. Spacecraft is 9.5 ft high and 18-22 ft across. Weight - 565 lb.

Launch Date: 28 November 1964

Mission Objectives: Perform planetary experiments during Mars flyby.

Type of Attitude Control: Twelve gas jets utilizing information from sun sensors, 3 gyros, and a Canopus star tracker.

Nature of Secondary Dynamic Motion: (1) The same problems considered to be of possible significance for Ranger were considered also for Mariner IV (See the documentation of Ranger I-IX) (2) On Mariner IV, a scan platform tracking the planet was felt to possibly cause sufficient coupling with the spacecraft and control system to result in excessive attitude error. (3) It was felt that the engine start transient might saturate the control system causing large pointing errors.

Preflight Analysis: (1) Same as for Ranger I-IX. (2) The motion of the scan platform was found to have little influence upon the controllability of the spacecraft. (3) The spacecraft was mounted in a test chamber and the engine fired. The attitude control system adequately handled the ensuing transient.

Responsible Government Agency: NASA/JPL

NAME OF SPACECRAFT: OV1-10

Physical Description: Cylinder with hemisphere at each end, 27 in. in diameter by 55 in. long. Weight - 287 lb.

Launch Date: 11 December 1966 into a 93-deg inclined orbit of apogee 479 mi. and perigee 403 mi.

Mission Objectives: Measure earth, background, and space radiation.

Source: Bob Fischell of JHU/APL

Type of Attitude Control: 3-axis gravity gradient control utilizing 6 extendible booms.

Nature of Secondary Dynamic Motion: The satellite motion is erratic and for some reason turns upside down every 28 days.

Responsible Government Agency: AF Office of Aerospace Research

Responsible Civilian Agency: General Dynamics/Convair

NAME OF SPACECRAFT: GALACTIC JUPITER PROBE

Physical Description: A 9-ft antenna dish from which extend 4 booms of either 7 1/2 or 19 ft, weight - 550 lb.

Launch Date: About 1973

Mission Objectives: To provide scientific information by exploring deep space in the vicinity of Jupiter.

Attitude Control Requirements: Point spin axis to within 1 deg of earth.

Source: Bill Reddish of NASA/GSFC

Type of Attitude Control: Spin stabilized with probably a fluid damper for removing wobble.

Nature of Secondary Dynamic Motion: Boom flexibility could conceivably limit the pointing accuracy.

Preflight Analysis: Preliminary analysis has indicated that this problem is not significant.

Responsible Government Agency: NASA (Study Phase)

NAME OF SATELLITE: 1964 - 83D

Physical Description: Wedge-shaped 6 in. by 10 in. by 12 in. Weight-10 lb.

Launch Date: 12 December 1964 into a 90-deg inclined orbit whose apogee was 582 mi, and perigee 558 mi.

Mission Objectives: Investigate ham radio phenomena.

Attitude Control Requirements: Less than ±20-deg pointing error.

Type of Attitude Control: Gravity gradient using a silver-plated 100-ft long boom and magnetic rods.

Nature of Secondary Dynamic Motion: The satellite settled into a 12-deg amplitude attitude pointing oscillation.

Cause of Secondary Dynamic Motion: Thermal gradients in the boom as in satellites 1963 — 22A and 1963 — 49B. The greater oscillation level is attributed to removal of the "lossy" spring for this vehicle.

Responsible Civilian Agency: American Radio Relay League

Source: Bob Fischell of JHU/APL.

NAME OF SPACECRAFT: ALOUETTE I

Physical Description: Oblate spheroid 42 in. in diameter and 34 in. high. A pair of antennas extended from the spacecraft equator, one pair 75 ft long, the other, 150 ft long; weight- about 300 lb.

Launch Date: 28 September 1962, 620-mi circular orbit inclined 80.5 deg.

Mission Objective: Measure variations in ionosphere electron density distribution; measure galactic noise; observe cosmic ray particles.

Type of Attitude Control: Spin stabilized.

Source: Bill Reddish of NASA/GSFC

Nature of Secondary Dynamic Motion: The spacecraft experienced a slowly decreasing spin rate.

Cause of Secondary Dynamic Motion: Solar heating deformed the antennae causing a nonstationary spacecraft center-of-mass/center-of-pressure relationship. The resulting torque due to solar pressure caused the lowering spin rate.

Remedy: Small paddles were affixed to the tips of the antennae to compensate for the above effect on the following vehicle, Alouette II.

Responsible Agency: Canadian Defense Research Telecommunications Establishment.

NAME OF SPACECRAFT: SURVEYORS I - V

Physical Description: Irregular spacecraft with solar panels and a high-gain antenna on one end and a tripod landing gear on the other end. Height 10 ft and 14 ft across its extended legs. Weight at launch — about 2200 lb; at lunar touch down — about 630 lb.

Launch Dates:

- Surveyor I — 30 May 1966
- Surveyor II — 20 September 1966
- Surveyor III — 17 April 1967
- Surveyor IV — 14 July 1967
- Surveyor V — 8 September 1967

Mission Objectives: Provide engineering and scientific data in support of the Apollo program.

Type of Attitude Control: 6 gas jets, 3 vernier engines, a sun acquistion sensor, sun and Canopus sensors, inertial reference unit, altitude marking radar, altimeter and doppler velocity sensors.

Cause of Secondary Dynamic Motion: In the lunar descent phase, a vernier engine mounted on each of the three legs is activated utilizing the inertial reference unit to set the thrust levels. The interaction of this control system with the flexible properties of the spacecraft, and particularly the legs, was felt to be a possible control problem.

Analysis: The Surveyor S-9 was subjected to vibration tests and found to possess 14 modes between 7 and 30 cps. The descent phase control system design was such that these modes were not excited.

Responsible Government Agency: NASA/JPL.

Responsible Civilian Agency: Hughes.

Source: Bill Gamin and Bill Carley of NASA/JPL.

NAME OF SPACECRAFT: PIONEER I

Physical Description: A double toroid; total height of the two cone frustrums and thin cylinder mid-section was 30 in.; mid-section diameter was 29 in. Spacecraft weight after firing injection rocket was 51.1 lb.

Launch Date: 11 October 1958 to an altitude of 70,700 mi.

Mission Objective: Measure radiation, micrometeorites, magnetic fields, study moon's surface.

Type of Attitude Control: Spin stabilized about an axis of minimum moment of inertia.

Nature of Secondary Dynamic Motion: The spacecraft experienced increasing wobble amplitudes.

Cause of Secondary Dynamic Motion: Only a rigid body has a stable spin equilibrium about the minimum moment of inertia axis. This spacecraft dissipated energy through structural deformation resulting in the observed wobble motion.

Responsible Government Agency: AFBMD

Responsible Civilian Agency: Space Technology Laboratories, Inc.

Source: TRW Systems Group

NAME OF SPACECRAFT: EXPLORER I

Physical Description: Cylinder 80 in long by 6 in. in diameter; weight - 30.8 lb.

Launch Date: 1 February 1958 at an orbit inclination of 33.34 deg to the equator; apogee 1185 mi; perigee 217 mi.

Mission Objective: Measure cosmic rays, micrometeor impacts.

Type of Attitude Control: Spin stabilized about the minimum moment of inertia axis.

Nature of Secondary Dynamic Motion: The spacecraft experienced increasing wobble amplitudes.

Cause of Secondary Dynamic Motion: Only a rigid body has a stable spin equilibrium about the minimum moment of inertia axis. This spacecraft dissipated energy through structural deformation resulting in the observed wobble motion.

Postflight Analysis: The problem was studied at JPL. For a report on the subject- "Effects of Energy Dissipation on the Free Body Motions of Spacecraft" by Peter W. Likins, JPL TR32-860.

Responsible Government Agency: AFBMD

Responsible Civilian Agency: JPL

Source: Technical Report noted above.

NAME OF SATELLITE: 1963 - 22A

Physical Description: Octagonal prism 18 in. across by 10 in. high; 4 solar blades 64 in. by 10 in., weight - 122 lb.

Launch Date: 15 June 1963, orbit inclined 90 deg with apogee of 418 mi. and perigee of 392 mi.

Mission Objectives: Provide navigation means for USN ships; increase knowledge of the earth's shape and gravitational field.

Attitude Control Requirements: Less than ±20-deg pointing error.

Type of Attitude Control: Gravity gradient with a "lossy" spring attached to its end. Magnetic rods were also utilized.

Source: Bob Fischell of JHU/APL

Nature of Secondary Dynamic Motion: The satellite eventually settled into a pointing error oscillation of 5 deg peak amplitude. An unexpected high-frequency component was also present in the response.

Cause of Secondary Dynamic Motion: Temperature gradients in the extendible boom caused it to deform, shifting the spacecraft principal axes as a function of time. The high-frequency effect was attributed to a thermal excitation when the satellite passes from eclipse into sunlight.

Remedies: Silver plating the BeCu boom causes most of the sun's incident energy to be reflected thus reducing temperature gradients in the boom.

Responsible Civilian Agency: Applied Physics Laboratory, JHU

NAME OF SATELLITE: 1963 - 43B, SNAP

Physical Description: Octogonal prism 18 in. across by 12 in. high. SNAP-9A, a cylinder 12 in. in diameter by 8 in. high with four side radiating fins; 4 aft skirt fins 18 x 12 in.; weight 166 lb.

Launch Date: 6 December 1963, into an 89.9 deg inclined orbit with a 598-mi. apogee and a 582-mi. perigee.

Mission Objectives: Provide means for navigation of U.S. Navy ships; demonstrate operation of the SNAP-9A power supply; increase knowledge of earth's shape and gravitational field.

Attitude Control Requirements: Less than ± 20 deg pointing error.

Type of Attitude Control: Gravity gradient with "lossy" spring and 100-ft silver-plated BeCu boom. Electromagnetics were also utilized.

Nature of Secondary Dynamic Motion: Static pointing errors of about 2 deg were noted with a negligible dynamic component.

Cause of Secondary Dynamic Motion: Thermal gradients in the boom shifted the spacecraft principal axes. However, use of silver plating on the booms resulted in a considerable performance improvement over Satellite 1963-22A.

Responsible Civilian Agency: Applied Physics Laboratory, JHU.

Source: Bob Fischell of JHU/APL

NAME OF SPACECRAFT: VOYAGER

Physical Description: Cylindrical with a dish-like structure on one end and a cone-like structure on the other end (TRW version). Weight-21,500 lb.

Launch Date: Program not active at present.

Mission Objectives: Unmanned exploration of the planets in search for information related to the origin of the solar system and nature of life.

Cause of Dynamic Interaction: (1) Structural flexibility of the vehicle and, more important, (2) Slosh of the large amounts of fuel required for this mission.

Type of Attitude Control: Gas jets in 3 axes using the sun and Canopus as inertial sensors.

Preflight Analysis: These problems have been analysed by TRW personnel and a feasible design recommended. In particular, a significant effort was devoted to developing mathematical models for low-g slosh of fluids.

Responsible Government Agency: NASA/JPL

Preliminary Design Contractors: Boeing, General Electric, TRW Systems.

Source: Terry Miwa of TRW Systems

NAME OF SPACECRAFT: RANGER I - IX, MARINER II

Physical Description: A 10-to 13-ft high cone formed by aluminum struts atop a 5-ft diameter hexagonal base with 2 large solar panels extending from the base measuring 15-17 ft tip to tip. Weight: Ranger I, II-675 lb; Ranger III, IV - 730 lb; Ranger V - 755 lb, Mariner II - 447 lb; Ranger VI to IX - 805 to 810 lb.

Launch Dates:

Ranger I	- 23 August 1961
Ranger II	- 19 November 1961
Ranger III	- 26 January 1962
Ranger IV	- 23 April 1962
Ranger V	- 18 October 1962
Ranger VI	- 30 January 1964
Ranger VII	- 28 July 1964
Ranger VIII	- 17 February 1965
Ranger IX	- 21 March 1965
Mariner II	- 22 July 1962

Mission Objectives: Ranger I, II - flight test of spacecraft and hardware; Ranger III, IV, V - rough land an instrumented capsule on the moon, obtain TV pictures of the moon's surface, study radiation and radar reflectivity of the lunar surface; Ranger VI-IX - provide scientific and topographic data concerning the moon's surface. Mariner II - perform Venus experiments.

Type of Attitude Control: Gas jets for pitch and roll control using sun and earth sensors and three gyros.

Nature of Secondary Dynamic Motion: (1) It was anticipated that, particularly during powered flight, flexibility of the antenna dish and two solar paddles could detrimentally interact with the spacecraft autopilot. (2) Effects of a changing center-of-mass due to fuel expenditure were expected to possibly degrade the control system performance.

Preflight Analysis: (1) Analysis was performed treating the spacecraft as a four-body system in order to evolve an acceptable design. (2) Shifting center-of-mass effects were considered and shown to be of insufficient importance.

Responsible Government Agency: NASA/JPL

Responsible Civilian Agency: None

Source: John Nicklas of TRW (formally of JPL); Steve Szirmay of NASA/JPL.

NAME OF SPACECRAFT: ORBITING GEOPHYSICAL OBSERVATORY (OGO III, IV)

Physical Description: Rectangular body 68 in. long by 33 in. square with 12 appendages (2 solar panels, two 22-ft booms, four 4-ft booms, 1 antenna boom, 2 gas jet booms, and 1 orbit plane experiment package (OPEP) boom). After deployment, an additional 30- or 60-ft Haddock antenna was extended from one of the solar panels.

Weight: OGO III — 1135 lb.

OGO IV — 1240 lb.

Launch Dates: OGO III—7 June 1966 into a 31-deg inclined orbit of 170 to 75,768 mi. OGO IV—28 July 1967 into an 86-deg inclined orbit of 256 to 564 mi.

Mission Objectives: Perform more than 20 experiments on geophysical and solar phenomena.

Attitude Control Requirements: ±0.4 deg in pitch and roll.

Type of Attitude Control: 3-axis gas jets and bang-bang reaction wheels utilizing information from 4 infrared horizon scanners and 2 sun sensors.

Nature of Secondary Dynamic Motion: OGO III — The vehicle settled into a roll oscillation exciting the roll reaction wheels and eventually the roll gas jets (until the latter were disabled). The solar array also oscillated in harmony with these effects. OGO IV — For the first two days after launch the spacecraft performed as expected. Then upon deployment of a 60-ft boom from one of the solar arrays, the vehicle began to experience a pitch oscillation exciting the pitch wheel and eventually the pitch controls jets (until they were disabled). Upon entering eclipse this characteristic successively dissappeared, reappearing in sunlit regions.

Cause of Secondary Dynamic Motion: OGO III — The problem was found to be caused by structural deformation of the OGO booms along the pitch axis. OGO IV — It was evident from the above sequence of events that deformation of the boom extending from the solar array was responsible for the problem.

Postflight Analysis: OGO III — The reaction torques resulting from the pitch boom deformations coupled dynamically into the roll control system which, in an effort to correct matters, then further excited these booms as well as the solar array. Analog simulation of the dynamic equations has indicated that this problem is extremely sensitive to the level of damping assumed for the booms. In fact, the 2-percent damping level originally assumed results in a well performing system. Evidently the damping factor of the boom is much less. OGO IV — In this case, the roll boom is evidently being thermally excited, with the control system playing little part in the action. Detailed analysis is still proceeding and no firm conclusions have been drawn at this point.

Remedies: OGO III — Time delay logic was incorporated into the roll reaction wheel system to prevent it from reversing its torque direction for 5 sec. Thus, the wheel cannot sustain a limit cycle of the type noted in OGO III. The solar array deadzone was also increased from 0.5 to 1.0 deg.

Influence on Future Design Practice: Long extendible booms attached to spacecraft will be analyzed in detail concerning their effect. Sometimes provision will be made on the actual vehicle for either retracting the boom or disposing of it, if the need arises.

Responsible Government Agency: NASA/GSFC.

Responsible Civilian Agency: TRW Systems Group.

Reference: "The OGO Attitude Control Subsystem Redesign as a Result of OGO III Experience — Vol. I." Report No. 67.7231.7-116 dated 30 June 1967.

Source: Keith McKenna of TRW and the above reference.

NAME OF SPACECRAFT: SPACE ELECTRIC ROCKET TEST 2 (SERT II)

Physical Description: Uses the Agena with two long wing-like solar arrays at one end.

Launch Date: To be launched late in 1969 into a 575-mi. polar orbit.

Mission Objectives: Test ion engines for at least a 6-month period.

Attitude Control Requirements: About ±1 deg in all three axes.

Type of Attitude Control: Gravity gradient system using Roll-Vee damping gyros with a gas jet backup. OGO horizon scanners are utilized for sensing attitude.

Nature of Secondary Dynamic Motion: Flexibility of the solar arrays and the Agena main body were considered to cause possible excessive attitude deviations.

Preflight Analysis: Conventional flexible body analysis techniques are being employed to study this problem.

Remedies: The problem is not considered to be of sufficient severity to have a great deal of impact upon the control system design.

Responsible Government Agency: Lewis Research Center

Responsible Civilian Agency: Lockheed

Source: Bill Reddish of NASA/GSFC

NAME OF SPACECRAFT: APOLLO

Physical Description: Basically a 12.8-ft diameter cylinder, 24.9 ft long, tapered to meet the Lunar Module which is 19 ft high and 30 ft across at its extended landing gear. Weight-about 94,500 lb.

Launch Date: 1968-1969 (manned)

Mission Objectives: Land two astronauts on the moon and safely return them.

Type of Attitude Control: 16 on-off gas jets on both the Service Module and Lunar Module, 12 gas jets on the Command Module. A great variety of sensors are used.

Cause of Secondary Dynamic Motion: (1) Flexibility of the docked configuration yields modal frequencies sufficiently low to impair the control system design. (2) The variable thrust LM engine has a complient support and thus, the engine deflection depends somewhat upon the thrust level. (3) The LM has two symmetrically placed fuel tanks. Should they be at different temperatures, transfer of fuel from one to the other could shift the vehicle center of mass, which, coupled with the engine thrust, possibly upset the controllability. (4) Venting of steam and purging of the fuel cell have the effect of external disturbance torques. (5) Crew motion or use of the ergometer (exercizer) can create internal reaction torques to affect the vehicle attitude.

Preflight Analysis: All of the above potential problems are being analyzed. To determine the flexible modes, use is being made of a computer program which characterizes the vehicle as a large number of lumped masses connected together by flexible, massless members.

Responsible Government Agency: NASA/MSC

Responsible Civilian Agencies: North American Aviation; Grumman

Source: Bob Goad and Art Goldberg of TRW Systems; Ken Cox of NASA/MSC

NAME OF SPACECRAFT: EXPLORER XX

Physical Description: Cylinder with cone at each end, 26 in. in diameter and 46.5 in. high. Three antennas extended from mid-section, two of 62-ft length and the other 122 ft long. Weight - 97 lb.

Launch Date: 25 August 1964 in an 80-deg inclined orbit of apogee 625 mi and perigee 540 mi.

Mission Objectives: To make radio soundings of the upper ionosphere at six fixed frequencies.

Type of Attitude Control: Spin stabilized at 2.2 rpm.

Nature of Secondary Dynamic Motion: The spacecraft experienced a slowly decreasing spin rate.

Cause of Secondary Dynamic Motion: Solar heating deformed the antennae changing the spacecraft center-of-mass/center-of-pressure relationship. The resulting torque due to solar pressure caused the lowering spin rate.

Responsible Government Agency: NASA/GSFC

Responsible Civilian Agency: Airborne Instruments Laboratory

Source: Holt Ashley Committee Meeting

NAME OF SPACECRAFT: RADIO ASTRONOMY EXPLORER (RAE)

Physical Description: a 36-in. cylinder, 30 in. high with 4 solar paddles affixed around the circumference and measuring 57 in. tip to tip. Four 750-ft booms are to be used as the antenna. Weight - 380 lb.

Launch Date: Mid-1968. Circular orbit of 3700-mi. altitude.

Mission Objectives: Study low-frequency radio emission from galactic regions.

Attitude Control Requirements: ±5-deg pointing accuracy

Type of Attitude Control: Gravity gradient, four 750-ft booms form an X while two additional 315-ft booms are used to damp librational motion.

Nature of Secondary Dynamic Motion: Structural flexibility of the 6 extendible, long booms and undergoing deflection due to thermal stresses caused by temperature gradients were considered to possibly cause excessive attitude errors.

Preflight Analyses: A large digital computer program developed by AVCO is being utilized to study these problems.

Remedies: Noting that the booms absorb only 8 percent of the incident solar radiation, 8 percent of the boom's surface consists of small punched holes. By painting the insides of booms black, 8 percent of the incident energy is absorbed by the rear inside of the boom which then results in a minimal effective temperature gradient.

Responsible Government Agency: NASA/GSFC

Responsible Civilian Agency: Fairchild-Hiller provides the booms and structure.

Source: Earl Angelo of NASA/GSFC

NAME OF SPACECRAFT: ADVANCED ORBITING SOLAR OBSERVATORY (AOSO)

Physical Description: Cylinder 119 in. long by 55 in. diameter. 8 solar panels, each of 20-ft span, extend from one end of the cylinder. Weight-1250 lb.

Launch Date: This project was canceled. The spacecraft was intended to go into a 380-mi altitude retrograde polar orbit.

Mission Objectives: Provide a platform for solar physics experiments.

Attitude Control Requirements: Pointing accuracy of ±5 arc sec.

Type of Attitude Control: Reaction wheels and gas jets with a backup magnetic unloading system. Sensors include gas bearing gyros, acquisition and fine sun sensors, and a Canopus tracker.

Nature of Secondary Dynamic Motion: (1) In scanning the sun in a raster scan, it was anticipated that flexibility in the solar paddle shafts could cause dynamic interaction with the main body control system. (2) The main body itself was considered sufficiently flexible to possibly cause excessive attitude pointing error.

Remedies: (1) Use of a notch filter tuned to the appropriate frequency made the control system insensitive to paddle motion. (2) Insufficient gain margin caused by the structural modes implied need for a design change. Whether such a change was made is academic as the project was canceled.

Responsible Government Agency: NASA/GSFC

Responsible Civilian Agency: Fairchild-Hiller

Source: Bill Reddish of NASA/GSFC

NAME OF SPACECRAFT: ORBITING ASTRONOMICAL OBSERVATORY (OAO1)

Physical Description: Octagonal cylinder 10 ft long and 7 ft wide. Two solar arrays of 3 panels each attached to opposite ends of the spacecraft giving it a wingspread of 21 ft. Two 9.5-ft balance weight booms were also included. Weight — 3900 lb.

Launch Date: 8 April 1966 — 500-mi circular orbit inclined 35 deg.

Mission Objective: Study over an extended time period the UV, X-ray and gamma ray regions.

Attitude Control Requirements: ±15 arc sec pointing accuracy for 50 min at a time.

Type of Attitude Control: 3 magnetic torques, 12 primary and 6 secondary gas jets; 3 coarse and 3 fine reaction wheels.

Nature of Secondary Dynamic Motion: A shade is utilized to protect the experiments from high-incident radiation levels. The question as to whether the shade motion could detrimentally affect the spacecraft pointing accuracy was considered.

Preflight Analysis: The effect of shade motion upon the spacecraft was evaluated and judged to be negligible.

Responsible Government Agency: NASE/GSFC.

Responsible Civilian Agency: Grumman.

Source: Bill Reddish of NASA/GSFC.

NAME OF SPACECRAFT: LUNAR ORBITER I - V

Physical Description: Truncated cone structure with 4 solar panels and a 3-ft parabolic antenna. Dimensions, 5.5 ft high and 12.2 ft across the solar panels. Weight - 850-860 lb.

Launch Dates:

L01 - 10 August 1966
L02 - 6 November 1966
L03 - 4 February 1967
L04 - 4 May 1967
L05 - 1 August 1967

All in lunar orbit

Mission Objectives: Determine possible Apollo landing sites; provide lunar photographs for scientific community.

Attitude Control Requirements: ±0.2 deg pointing accuracy

Type of Attitude Control: 3-axis stabilized by 8 gas jets. Five sun sensors are used for pitch and yaw reference; a Canopus startracker is used for roll reference, and an inertial reference is used when Canopus is not visible.

Nature of Secondary Dynamic Motion: (1) Attitude control thrusters mounted at the ends of the flexible solar panels were felt to result in possible control problems. (2) The solar array was sufficiently flexible to cause its lower modal frequencies to be within the control system bandpass, thus creating possible stability problems.

Remedies: (1) The attitude control jets were moved to the main body. (2) The solar array shafts were stiffened raising the vibrational frequencies to a safe point.

Responsible Government Agency: NASA/Langley

Responsible Civilian Agency: Boeing

Source: Clarence Gillis of NASA/Langley

NAME OF SATELLITE: DODGE

Physical Description: Truncated octogan 48 in. across flats by 32.6 in. high connected to a stack 62 in. high by about 14 in. in diameter. Weight-430 lb.

Launch Date: 1 July 1967 into a 7-deg inclined orbit of 18,161 mi. apogee and 17,959 mi. perigee.

Mission Objectives: To test gravity gradient principle at near synchronous attitudes.

Attitude Control Requirements: Pointing accuracy ±2 deg; yaw accuracy ±4 deg.

Type of Attitude Control: Gravity gradient (3-axis) with 10 motorized boom assemblies to achieve different configurations.

Nature of Secondary Dynamic Motion: Somewhat larger errors than those noted above were experienced. Also there has occurred a dynamic boom flutter of a 30-sec period and lasting 2 hr.

Cause of Secondary Dynamic Motion: (1) Temperature gradients causing boom deformation (2) The continuing experimentation program involving the gravity gradient booms also tends to excite the spacecraft.

Responsible Civilian Agency: Applied Physics Laboratory JHU

Source: Bob Fischell of JHU/APL

NAME OF SPACECRAFT: GRAVITY GRADIENT TEST SATELLITE (GGTS1)

Physical Description: Symmetrical polyhedron with 24 faces, 32 in. high and 36 in. in diameter. Weight-104 lb.

Launch Date: 16 June 1966 into a 0.25-deg inclined essentially synchronous orbit.

Mission Objectives: Test gravity gradient principle at high altitude.

Attitude Control Requirements: ±8 deg in pitch and roll.

Type of Attitude Control: Gravity gradient utilizing two 52-ft booms with magnetic ball dampers at each tip.

Nature of Secondary Dynamic Motion: The spacecraft seemed to steady out at an oscillation of about 15 deg. Subsequently, the vehicle damped out to a much smaller pointing inaccuracy.

Cause of Secondary Dynamic Motion: The cause for the original oscillation level was attributed to magnetic particles clinging to the ball dampers and thus, considerably reducing their effectiveness. This is essentially a dynamic interaction with the spacecraft environment.

Remedy: A magnetic storm evidently dislodged the particles rendering the ball dampers much more effective.

Responsible Government Agency: AF/SSD.

Responsible Civilian Agency: General Electric.

Source: Bill Reddish of NASA/GSFC.