NASA TECHNICAL MEMORANDUM

NASA TM X-73365

SHUTTLE/TETHERED SATELLITE SYSTEM CONCEPTUAL DESIGN STUDY

(NASA-TM-X-73365) SHUTTLE/TETHERED SATELLITE SYSTEM CONCEPTUAL DESIGN STUDY (NASA) 190 p HC A09/MF A01 CSCL 22A

N77-18199

Unclas
G3/16 17190

By Preliminary Design Office

December 1976

NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

TECHNICAL REPORT STANDARD TITLE PAGE

1. REPORT NO.	2. GOVERNMENT ACCESSION NO.	3. RECIPIENT'S CATALOG NO.
NASA TM X- 73365		
4. TITLE AND SUBTITLE Shuttle/Tethered Satellite System Conceptual Design Study		5. REPORT DATE December 1976 6. PERFORMING ORGANIZATION CODE
7. AUTHOR(S)		8. PERFORMING ORGANIZATION REPORT #
9. PERFORMING ORGANIZATION NAME AND ADDRESS George C. Marshall Space Flight Center Marshall Space Flight Center, Alabama 35812		10. WORK UNIT NO. 11. CONTRACT OR GRANT NO. 13. TYPE OF REPORT & PERIOD COVERED
12. SPONSORING AGENCY NAME AND ADDRESS National Aeronautics and Space Administration Washington, D.C. 20546		Technical Memorandum 14. SPONSORING AGENCY CODE

15. SUPPLEMENTARY NOTES

Prepared by Preliminary Design Office, Program Development

16. ABSTRACT

Tethers have been proposed for many space applications such as retrieving stranded astronauts, stationkeeping two orbiting vehicles, and suspending scientific payloads into the upper reaches of the atmosphere from a main orbiting vehicle. Subsequent to these proposals, a new ingredient has been added. Namely, a closed-loop control system has been added to the tether reel which improves control over the tethered satellite. In addition to increasing the stability of the tethered satellite along local vertical, this control system can be used for deployment and retrieval of tethered satellites. This conceptual design study describes a tether system for suspending a science payload at an altitude of 120 km from a Space Shuttle Orbiter flying at an altitude of 200 km. In addition to the hardware conceptual designs, various aspects concerning Orbiter accommodations are discussed.

17. KEY WORDS	18. DISTRIBUTION STATEMENT

19. SECURITY CLASSIF. (of this report)	20. SECURITY CLASSIF. (of this page)	21. NO. OF PAGES	22. PRICE
Unclassified	Unclassified	189	NTIS

MSFC - Form 3292 (Rev December 1972)

For sale by National Technical Information Service, Springfield, Virginia 22151

ACKNOWLEDGMENTS

The following persons outside the Preliminary Design Office have supplied information to include in this report: Mr. Z. Galaboff, Systems Dynamics Laboratory, and Mr. R. Kissel, Electronics and Control Laboratory, submitted trajectory information; Mr. K. Clifton, Space Sciences Laboratory, contributed to the Micrometeorite Study; and Mr. Dendy, Teledyne Brown Engineering, contributed to the hardware design and configuration. The following people with the Preliminary Design Office have participated in this study:

Technical Coordination Applications, Stabilization and Control, and Safety . . . . . Mr. Rupp

Configuration and Layout . . . . Messrs. Colley and Love

Interface and Technology . . . . Mr. Mundie

Structure and Mass Properties . . . . Mr. Tovar

Thermal Control System . . . . Mr. Fritz

Communication and Data Handling . . . . Mr. Reed

Control and Display . . . . Messrs. Reed and Haley

Power . . . . Mr. Vann

Mission Requirements . . . . Mr. Little

Crew Support . . . . Messrs. Dickerson and Burton

Ground Operations . . . . Mr. Dickerson

Timeline and Trajectories . . . . Messrs. Burton, Rupp and Mrs. Reisz

Tracking . . . . Messrs. Little and Hall

Micrometeorite Study . . . . Mr. Mundie and Miss Shields

Orbiter Pointing Accuracy . . . . Mr. Brandon

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Concluded)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Concluded)

LIST OF TABLES

TECHNICAL MEMORANDUM X- 73365

SHUTTLE/TETHERED SATELLITE SYSTEM CONCEPTUAL DESIGN STUDY

1.0 INTRODUCTION

1.1 Chronology

Tethered subsatellites have been proposed for a variety of space applications. One body of work was directed toward the retrieval of stranded astronauts by "throwing a buoy on a tether" from the rescue vehicle to an astronaut and then reeling the tether back. Eades and Wolf [1] described the equations of motion involved and a method whereby the proper initial velocity and constant tether tension can be determined which will cause the tethered buoy (generally called a subsatellite) to rendezvous with the stranded astronaut. Several problems were unsolved in this method; e.g. slight variations in the initial velocity of the subsatellite or errors in the tether tension cause large errors in the final state of the deployment or retrieval. The retrieval problem was dramatically illustrated in a study by the Marquardt Corporation [2] wherein the motion of the subsatellite was shown to rotate at ever-increasing angular rates about the main satellite as the tether was reeled in. All of these studies used an open-loop control law which applied precalculated tension to the tether for deployment or retrieval. None of these schemes were developed for actual rescue missions because of these problems involved with the schemes and because more suitable rescue techniques were subsequently developed.

A second body of study involved stationkeeping between two orbiting space vehicles called the Orbital Workshop (OWS) and the Lunar Module — Apollo Telescope Mount (ATM). As shown by this study, the ATM could be placed in a somewhat egg-shaped trajectory relative to the OWS by periodically tugging on a tether which connected the two vehicles [3]. This method of stationkeeping was found unattractive because of the difficulties in precisely determining and controlling the tugging forces to be applied and in providing the constant manned supervision which would be required to insure flight safety.

A third body of study, which culminated in two successful orbital space flights, involved experiments in which the manned Gemini space vehicle was tethered to an unmanned Agena vehicle [4]. The Gemini XI flight demonstrated a rotating configuration in which the Gemini was tethered to the Agena by 30.48 m (100 ft) of polyester webbing. The configuration was rotated using the Gemini thruster reaction control system (RCS) at a rate of approximately 0.9°/s. Centrifugal force maintained tension in the tether. The Gemini XII flight demonstrated a gravity gradient stabilized configuration. In this experiment, the Gemini vehicle oriented the tethered vehicles along local vertical using the Gemini RCS. The RCS was then turned off and gravity gradient successfully captured the configuration. Gemini flights XI and XII used the Gemini RCS to deploy and orient the tether system, and the tether, being of fixed length, was not used to actively control the motion of the space vehicles.

In a proposal to the Atmospheric, Magnetospheric, and Plasmas in Space (AMPS) science working group, Columbo et al. [5] described the use of a long tether system, called Skyhook, for low altitude research. In Reference 5, a tethered subsatellite containing experiments is deployed down to an altitude of approximately 117 km from a main satellite in orbit at 200 km. To deploy the tethered subsatellite, it was suggested that a large balloon be attached to the subsatellite. It was expected that atmospheric drag would pull the balloon behind the main satellite and downward, thereby deploying the tether to some distance. The balloon could then be burst and tether deployment could be continued by slowly unwinding the tether from a reel. The dynamics of the trajectories of the subsatellite using the balloon for deployment were not analyzed in Reference 5, however. Subsequent to Reference 5, studies, initiated by the Preliminary Design Office at the Marshall Space Flight Center (MSFC), showed the feasibility of deploying, stabilizing, and retrieving long tethered subsatellites with an active tether control system utilizing gravity gradient forces instead of atmospheric drag. The mathematical formulation of the control laws which enable the tether control system to function and a brief statement of system hardware requirements, as described by Rupp [6], resulted from this study.

Following the works of Columbo and Rupp, the Science and Engineering Directorate at MSFC instituted an investigation of several aspects concerning the feasibility of tethered satellites. This study addressed dynamics, control, thermal, aerodynamic, and communication aspects with the general conclusion that there was no evidence to indicate that the concept was infeasible. This study [7] was not intended to be a Phase A study and, as a result, important areas such as system requirements, Orbiter accommodation, and mechanical design and layout were not covered.

The most recent study in the control and dynamics discipline was accomplished by Kulla [8]. This study is in general agreement with the investigation of earlier references. The simulation results showed the shape of the tether during deployment and also investigated deployment and retrieval to a distance of 1 km.

The Phase A study documented herein was begun in May 1976. Concepts were defined early to support cost estimates which could be used in budget requests. The studies performed used previous references where possible and concentrated in those areas of requirements, interfaces, design, and layout which were not previously considered.

1.2 Applications

Tethered satellite systems have been proposed for use in many applications in addition to those previously described. Some of these are depicted in Figures 1-1 and 1-2. These applications can be classified as science applications or facility applications. The basic technology is similar for all applications; however, the system configuration might be vastly different from one application to another.

Typical science applications involve tether-deployed science payloads or using the tether itself as part of the science instrument. Low altitude science applications include gravity and magnetic field mapping, aeronomy, reentry research, Earth surveillance, and plasma physics.

A tether can also be used to deploy science payloads away from the Orbiter for the purpose of avoiding Orbiter induced environmental disturbances. Deployment can be along local vertical either to the Earth or away from the Earth. The distance which packages can be deployed is limited by practical considerations, e.g. volume of tether to be launched aboard the Orbiter.

A conducting tether can be used as an antenna. A free flying radio astronomy satellite is shown (Fig. 1-1) in which the tether acts as a dipole antenna. Low frequency emissions received by the long dipole antenna are converted to a microwave link and beamed to Earth. Other antenna configurations, such as a large loop antenna, are being investigated.

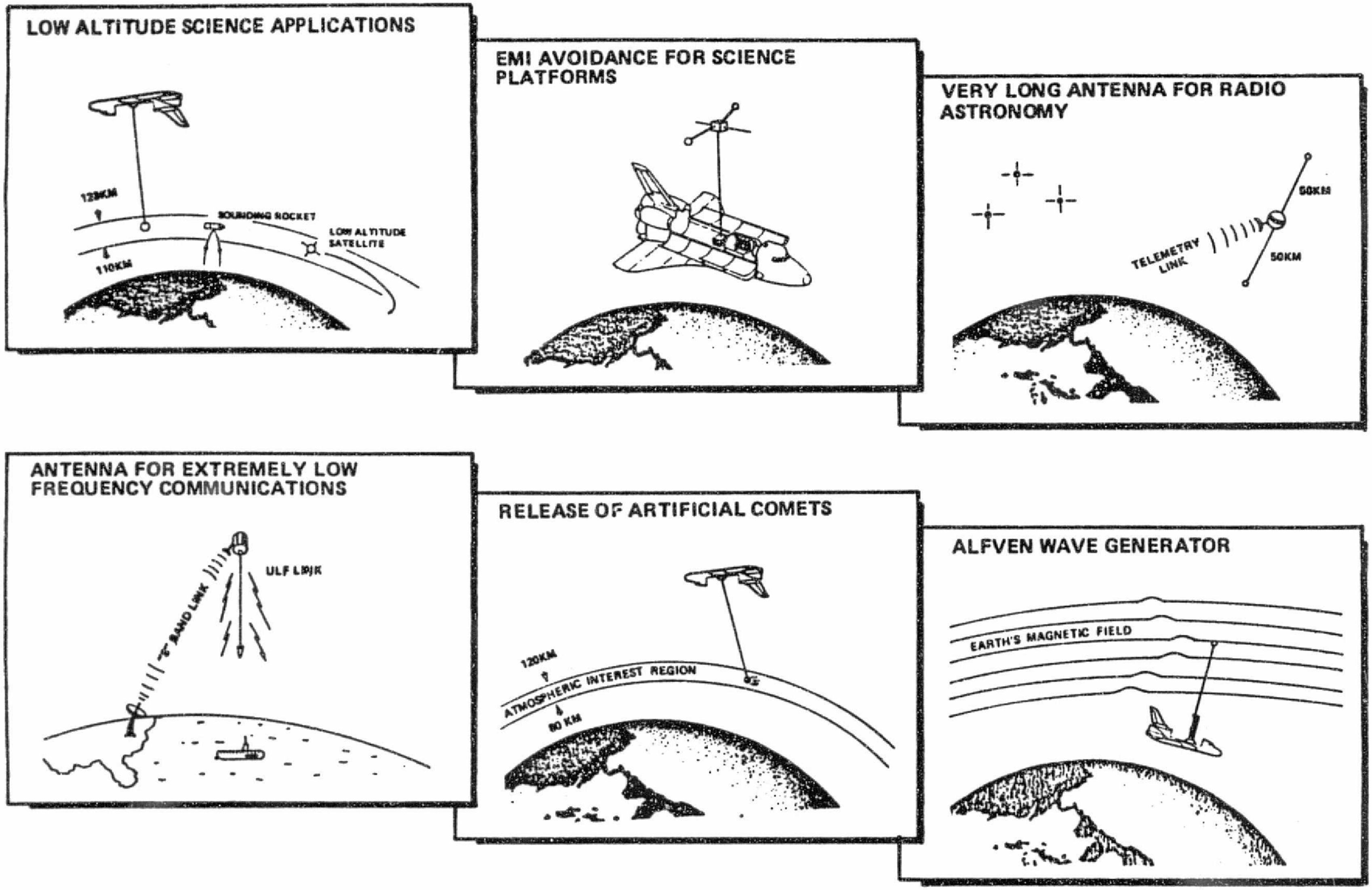


Figure 1-1. Tether system science applications.

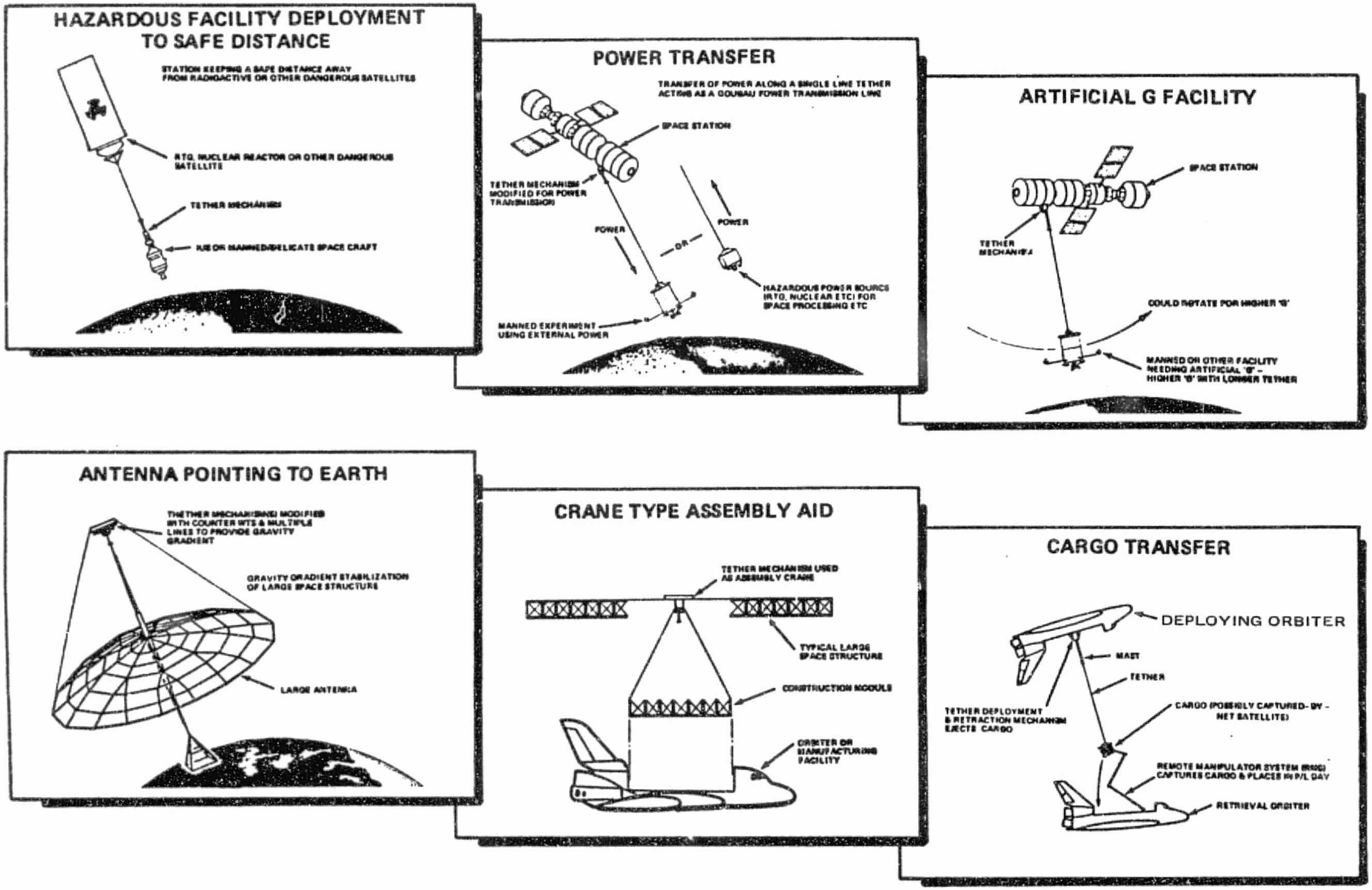


Figure 1-2. Tether system facility applications.

Another antenna configuration called a long wire antenna is shown in Figure 1-1. This antenna is utilized for communications with submarines. Communications are beamed via microwaves to the relay satellite, converted to low radio frequencies, and radiated to the Earth. The low frequencies can penetrate the surface of the ocean.

The tether system can be used to deploy packages for the release of various chemicals to stimulate various phenomena; e.g. barium can be released for ionization studies. Deploying a package into the upper atmosphere would allow chemicals to be released by ablation over a long path to study effects of comets.

The Earth's magnetic field can be studied by forcing current to flow in the tether. Propagation of the resulting field perturbation, called Alfven waves, can lead to a more thorough understanding of the dynamics of the Earth's magnetic field.

Facility applications are those which use the tether as a structure or crane (Fig. 1-2); e.g. a spacecraft can be tethered a safe distance from a hazardous object. The object could be towed by the spacecraft or just maintained a safe distance away for stationkeeping. This concept would allow a Tug without specially protected subsystems to tow a radioactive satellite.

In addition to providing mechanical support, a tether can also be used to transmit large amounts of power to a remote site by microwaves. A single conductor, surrounded by a dielectric, forms an efficient transmission line. This line, called a Goubau line, has the potential for transferring power more efficiently than conventional direct current or alternating current transmission lines.

Occasionally, the need for artificial gravity might arise in a manufacturing process. For example, a satellite at the end of a tether is subjected to an amount of gravity depending on the separation distance, and this gravity could be increased by swinging or rotating the facility about the main vehicle.

A large space structure, such as an antenna, can be gravity gradient stabilized using a tether to suspend a counterweight. In this way temporary stabilization of pieces of structure during assembly can be accomplished, thus simplifying the tasks of teleoperators.

A tether can be used as a crane to maneuver structural elements into place. The tether "crane" can be on either the supply vehicle or the receiving platform.

Cargo may also be transferred by using the tether either as a crane or as a high line.

Development of the present system concept is part of the NASA Office of Space Flight Five-Year Plan under the Orbiter Operations Capability Development line item. Development of a multiuse/multiapplication tether system facility is the goal of the plan.

The application driving the system studies described in this report is in support of the Magsat and Geosat programs within the NASA Office of Applications. This program utilizes satellites in Earth orbit to map the gravity and magnetic fields of the Earth. The orbit of these satellites is relatively high, on the order of 200 km (108 n. mi.) to maximize the lifetime of the satellites [9]; however, the high altitude causes the resolution of the data to be limited. Figure 1-3 shows the sensitivity required as a function of altitude for gravity gradient sensors. As shown in Figure 1-3, 100 harmonics of the gravity field could be sensed by a given instrument with a sensitivity of 0.01 Etvos Units at an altitude of 200 km. If the altitude is dropped to 120 km, the same instrument could sense slightly beyond the 200th harmonic. Spatial resolution also suffers with increased altitude because the object to be sensed must be nearly as large as the altitude height [10].

The tether system has been proposed to support satellites to make high resolution gravity and magnetic field maps of areas of interest detected by the lower resolution, higher altitude satellites. The preliminary estimate of the altitude which the tethered satellite should fly is 120 km.[1] This altitude is sufficiently low to greatly improve the map resolution but is still sufficiently high to minimize thermal and stability problems. Other applications compatible with the system, as it has been defined, will be investigated in subsequent study activities currently planned at MSFC.

1.3 Study Philosophy

This study was originally planned to address the design of a tether system which would accommodate a wide range of science requirements. The requirements were to be specified by an Ad-Hoc Science Advisory Group. The

1. Meeting at NASA Headquarters.

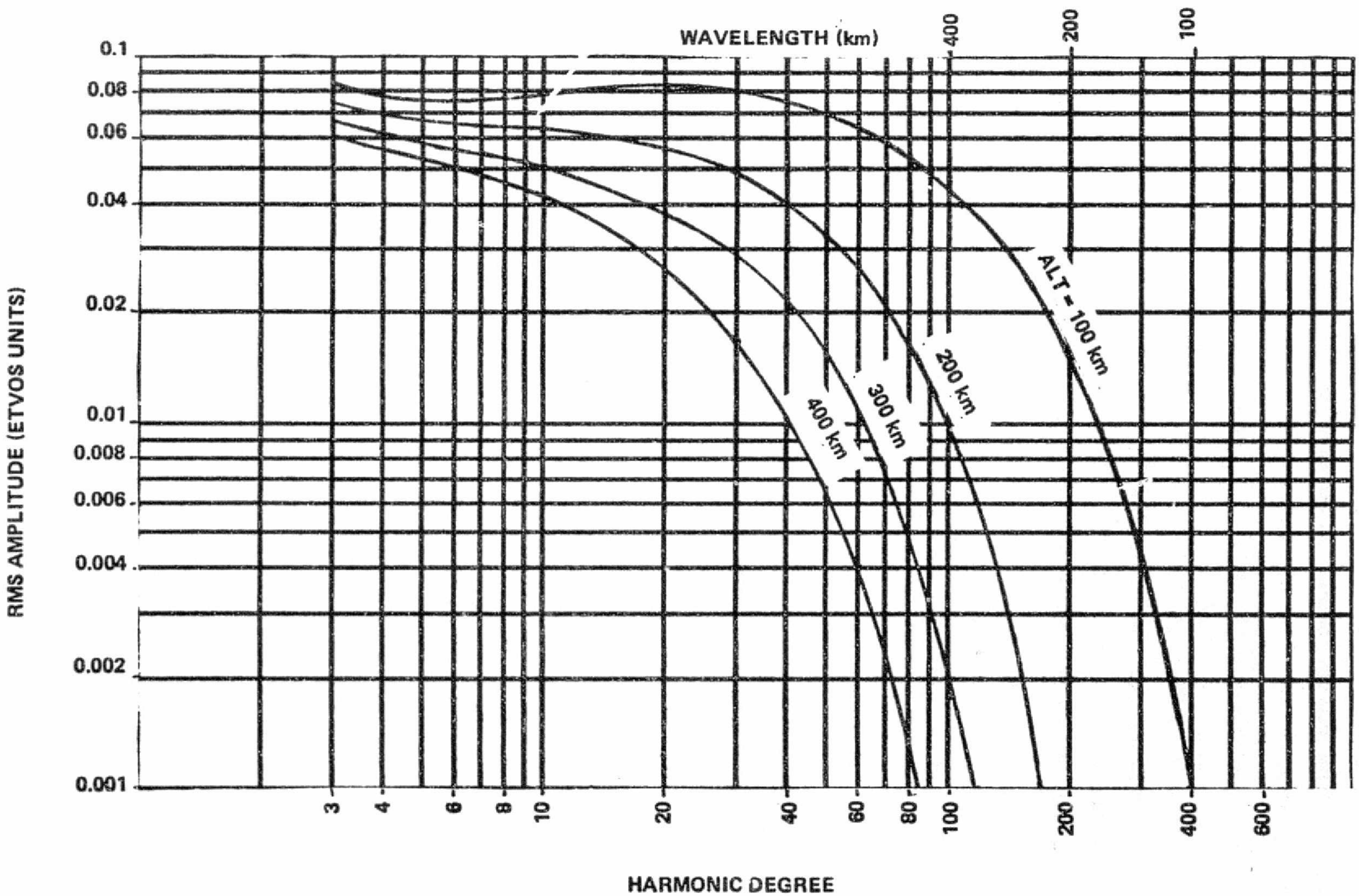


Figure 1-3. Radial second derivative sensitivity to altitude. (1 Etvos Unit = 10^{-9} gravities/m).

group was to be chaired and managed by Goddard Space Flight Center (GSFC) and co-chaired by MSFC. Membership was to be drawn from government agencies representative of the potential user community; however, for a variety of reasons the advisory group was not formed and, instead, preliminary requirements were generated by GSFC for gravity gradiometry and magnetic field mapping instrumentation.

At the same time these preliminary requirements were being generated, MSFC was asked to provide tether system cost estimates for demonstration flights (cost of science was not to be included in the estimate). In support of this request, preliminary system configurations were presented in sufficient detail to allow the cost to be estimated. A very minimum amount of engineering instrumentation was included in these designs to minimize the cost. Satellites were designed to house additional engineering instrumentation (such as cameras), and this cost was included in the total cost estimates. Measurements which would insure that the system meets the science requirements supplied by GSFC were not included because this cost should be borne by the user. Incorporation of additional instrumentation is not to be precluded by the tether system designs.

The study planning was flexible enough to address three system configurations. The choice between these systems is a programmatic problem; therefore, this documentation will include all three systems for future reference.

1.4 Design Philosophy

As shown in Reference 7, an open-loop scheme can be used to deploy a tethered satellite system. Such a system would simplify the system design to some extent but retrieval of the system will require some form of closed loop control. A reaction control system, with associated navigation and control systems, could be part of the tethered satellite. However, if retrieval can be accomplished by proper control of the reel mechanism, major portions of the usual satellite weight and expense can be eliminated.

Not all applications will justify the cost, in terms of system complexity or dedicated Orbiter time, involved to retrieve the satellite. Many applications can utilize expendable payloads as long as the required data are transmitted to the Orbiter. However, to accommodate the widest range of applications possible, the design of the tether system will include the capability to retrieve. Appropriate trade studies can then be made on a flight-by-flight basis to determine if a particular payload should be retrieved.

2.0 GUIDELINE SUMMARY

Earlier reports referenced in the previous section addressed several areas of concern regarding the feasibility of tethered satellite systems. The general guideline of this study is to make use of prior work to the maximum extent possible and to fill the gaps which remain in the system definition. Preliminary guidelines were published at the beginning of the definition study and these have been revised several times to respond to additional areas of concern and to better define requirements. The latest guidelines, presented in Appendix A, should agree closely with the studies which were actually performed.

2.1 Objectives

This study is for the development of a Shuttle/tethered satellite system to be operational in 1981 which will increase the potential for Shuttle utilization. The future operational concept envisions a tether system with closed-loop control, capable of supporting a payload module suspended from the Shuttle payload bay either toward or away from the Earth at distances up to 100 km from the Shuttle. The system would be capable of performing multiple round trip missions and would accommodate a wide variety of scientific payloads for purposes such as global mapping of the Earth's magnetic field and upper atmospheric exploration. The tether system also offers techniques for various facilities applications such as transferring cargo and erecting large space structures.

To support these objectives, the design of a tethered satellite system is undertaken which accommodates the magnetic and gravity mapping missions described in Section 1.2. As an intermediate step, definition is provided for a simplified and less costly system which can be flown on the orbital flight test (OFT) series to prove the system concept.

2.2 Plan

This study was accomplished mainly within Program Development with limited supporting studies furnished by Science and Engineering as the need arose and personnel were available. Documentation of the technical areas was accomplished within the Preliminary Design Office of Program Development. Cost, scheduling, and other programmatic support was furnished by the Payload Studies Office and the Program Planning Office. Documentation of the programmatic aspects of the study is separate from the technical documentation.

2.3 Schedule

Figure 2-1 presents a schedule of the activities performed during this definition and analysis study. This schedule was arranged to fit within the framework of preliminary project plans. Periodic status reports were presented to various NASA Headquarters offices during the course of the study.

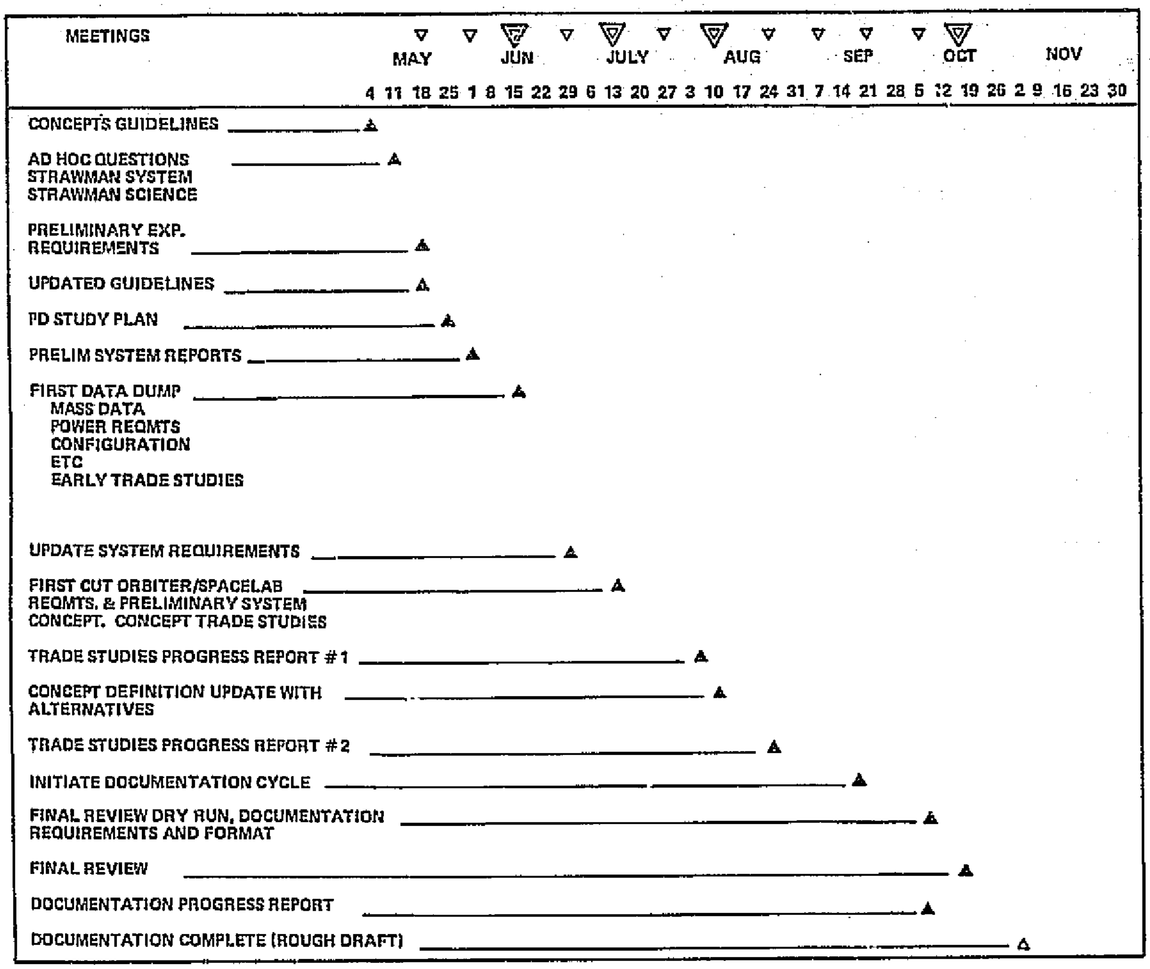


Figure 2-1. Analysis study schedule.

3.0 SYSTEM REQUIREMENTS

3.1 Operational System Application and Science Requirements

3.1.1 Magnetic Field Mapping

The first applications which have been identified in on-going programs are Earth magnetic and gravity field mapping. Preliminary data and a discussion of the requirements which the magnetic mapping mission imposes are contained in a letter from GSFC.[2] This discussion might be useful for future reference and is reproduced in Appendix B in its entirety.

3.1.2 Gravity Field Mapping

Gravity gradient sensors have not been selected or analyzed to determine their suitability for flying on the end of the tether. Sensors which would work well in free flying satellites might not be the best candidates for the tethered satellite missions. The study of sensors for tethered applications is proposed in RTOP 681-01-01 from GSFC. However, present plans do not call for this work to proceed in FY77. The only information available is from the Space Shuttle Payload Definition document under OP-02-A [9].

These data are presented in Appendix C.

3.1.3 Strawman Science Requirements

The science instrumentation requirements previously presented did not play a strong role in setting system requirements. The reasons for this are threefold. First, the science requirements are very preliminary and there is insufficient knowledge regarding the gravity gradient sensor requirements when operating in a tethered satellite. Second, the responsibility for defining gravity gradiometry and magnetometry instruments for tether application primarily rests with GSFC, which at the time of this study had no plans to begin this study through FY77. Third, there is a requirement from Headquarters to develop a tether system for an early demonstration flight. All of these reasons have served to cause the engineering aspects of the system to be developed using strawman science requirements.

---

2. Letter from R. A. Langel, Geophysics Branch, GSFC, to W. T. Roberts, dated May 10, 1976.

The strawman requirements have been incorporated into the study guidelines presented in Appendix A and will not be listed in detail here. A test flight of the operational system is planned for late 1981 and will demonstrate the capability to deploy and retrieve a 175 kg satellite to an altitude of 120 km from an Orbiter at 200 km. This flight as presently planned will be capable of testing candidate science payloads if available.

As part of the operational tether system, a satellite is designed incorporating limited engineering instrumentation. Earth viewing cameras are included on the satellite and on the pallet in the Orbiter payload bay. Photographs taken during a flight can be analyzed to provide knowledge of satellite position and stability. The design of the tether system will not preclude adding science instrumentation as required. The operational system with this satellite will require approximately 30 h to demonstrate the capabilities of the system to an altitude of 120 km.

3.2 System Verification and Demonstration Flight Requirements

The critical questions concerning the tether concept center on the system dynamics and control. The dynamics and control designs get difficult at two extermes — very long tethers when the satellite extends down into the atmosphere and very short tethers when the satellite is close to the Orbiter.

When tethered satellites are deployed down into the atmosphere, aerodynamic drag causes the satellite to trail the Orbiter. The drag can get large enough that it exceeds the gravity force and the satellite cannot be lowered further. In such cases, control of the swinging motion of the satellite may be lost and the satellite may rise and seek the stable equilibrium above the Orbiter. However, in actual practice the satellite cannot be lowered to an altitude low enough to cause loss of control without first exceeding the temperature limits of tether materials. Therefore, limiting the lowest operational altitude of the system to solve the temperature problem will also eliminate a major concern of control of low altitude tether systems. Another area of concern is the introduction of unwanted vibrations into the tether by high frequency disturbance forces such as variations in the atmospheric density. This problem will be analyzed using modal analysis and finite element simulations in future studies.

The finite element simulation will calculate the dynamic shape of the tether as a function of time given a set of initial condition and disturbance forces. To guarantee that the system will be stable for all conditions, all possible combinations of disturbance forces must be simulated with all possible system configurations which is an impossibly long task.

The modal analysis approach determines the system resonant frequencies, as well as the frequencies involved in the disturbance forces, and studies the coupling of the disturbances with the system. This analysis will describe the system stability over the entire frequency spectrum. However, the dynamic shape of the tether is not determined as a function of time. Both approaches taken together will be extremely powerful and complementary tools to describe system behavior.

No analysis will completely replace a flight test of the system. Questions will always arise over the completeness of the disturbance force model, the accuracy of the system models, the accuracy of the simulation, etc. This concern justifies the flight test of the system to the desired operational altitude of 120 km prior to an actual application mission.

There is concern for the control and dynamics of the tether system when the tethered satellite is close to the Orbiter. The gravity gradient forces are very small and cause the tension in the tether to diminish and therefore control authority is diminished. However, during such times, the satellite moves very slowly with respect to the Orbiter. Tether tensions measured in hundredths of a Newton ($0.01 \text{ N} = 0.036 \text{ oz}$) and relative velocities of a few centimeters per second are common for the early stages of tether system deployment and latter stages of retrieval. An early demonstration flight of a short (approximately 1 km) tether system would allow attention to be focused on this aspect of control and dynamics. Since the focus is on tether system dynamics, the satellite can be completely passive, thus reducing flight complexity and cost.

Two simplified versions, the limited operational system and the concept definition system, of the tether system are presented later in this report to meet the goals of an early demonstration flight. As will be seen, the concept definition system eliminates some tether system hardware at the expense of increased crew participation and additional concerns regarding Orbiter systems capability and availability.

4.0 DESIGN CONFIGURATION

The preceding section described requirements for the test flight of an operational tether system in late 1981. This tether system, which was designed to meet those requirements, is called the operational system. Two other tether system configurations, the limited operational system and the concept demonstration system, were designed to meet requirements for an early test flight to demonstrate the tether concept. The limited operational system is a simplified version of the operational system which can be upgraded to complete operational status after the successful test flight. The concept demonstration system is an even further simplified system which uses the Shuttle Remote Manipulator System (RMS) for satellite deployment and retrieval instead of a specially designed boom. This latter approach is an attempt to postpone some of the hardware development costs by flying a minimum of hardware on the first demonstration flight.

All three configurations are basically similar in that a tethered spherical satellite is clamped to a derrick-like support structure surrounding a reel mechanism on which the tether is wound. This integrated structure is mounted to a base which in turn mounts to the standard Spacelab pallet.

4.1 Operational Tether System Configuration and Layout

Two reel mechanisms were sized during the course of this study. The preferred, larger reel accommodates a 1 mm (0.039 in.) diameter tether of up to 100 km (62.14 miles) length. A smaller reel accommodates a 0.366 mm (0.014 in.) diameter tether, also of up to 100 km length. The larger diameter tether is preferred because of the increased factor of safety and decreased susceptibility to catastrophic (severing) micrometeorite damage. This trade study is described in Section 5.0. Both sizes are described; the larger diameter here in Section 4.1, and the smaller under alternate designs in Section 4.5.

4.2 Operational Tether System

Figure 4-1 shows the operational tether system mounted in the Orbiter payload bay. The 50 m (164 ft) extensible and retractable boom for satellite deployment and retrieval is shown extended and the satellite is shown deployed a short distance by the tether. The tether has a diameter of 1 mm (0.039 in.) and a length of 100 km (62.14 miles). The satellite has a diameter of 1.44 m (56.75 in.) and contains instrumentation for monitoring the performance of the

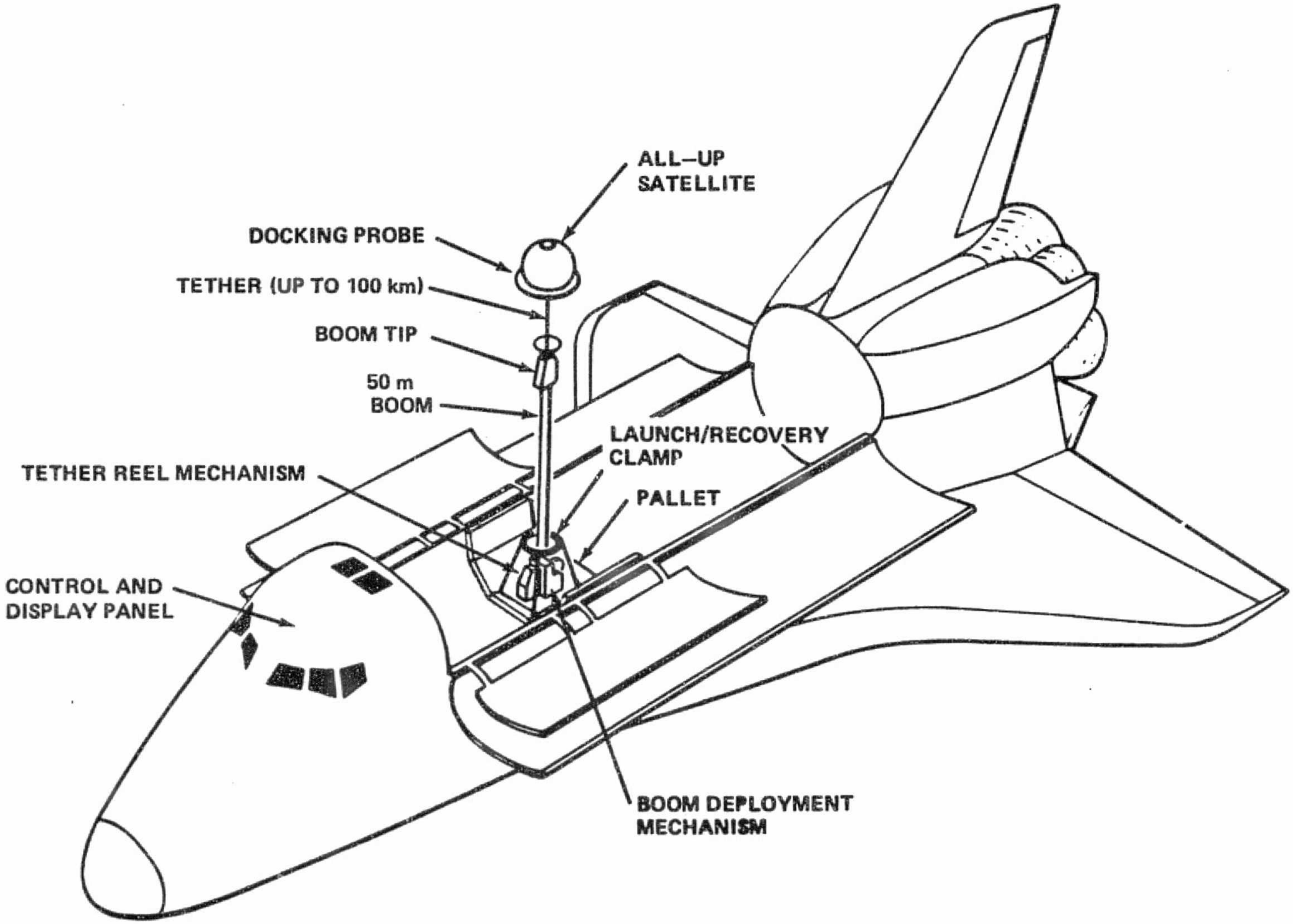


Figure 4-1. Operational system.

tether system as well as collecting atmospheric data. A docking probe is fitted to the satellite which mates to a drogue capture mechanism located on the tip of the boom. The launch/recovery clamp holds the V-ring of the satellite firmly for Orbiter launch and reentry. The 50 m (164 ft) boom deployment mechanism is mounted beside the reel mechanism on a base platform. The operational sequence is shown in Figure 4-2.

Details of the reel and boom mechanism are shown in Figure 4-3. In addition to a spool to wind the wire on, the reel mechanism includes a level wind mechanism to assure that the tether is wound uniformly on the spool. The boom is a configuration similar to that flown on the radio astronomy explorer. The boom material is a flat aluminum tape 0.3 mm (0.012 in.) thick which is formed into a 76 mm (3 in.) diameter tubular element by the boom deployment mechanism. Another spool, on which a flat conductor electrical cable running to the boom tip is wound, is driven by a chain drive from the same motor that drives the boom deployment mechanism. The tip of the boom contains a tensiometer to measure the tether tension; the satellite drogue capture mechanism; and a torque motor to maintain tension on the tether through the reel mechanism. Pyrotechnic guillotines are provided at the boom tip to release the tether and satellite, and at the base of the boom to release the tether, boom, and boom tip electrical cable. These guillotines are also designed to clamp the remaining tether to prevent snarling the loose line. Use of the Orbiter RCS may be required to assure that there is no fouling by the tether, satellite, or any other debris after the guillotines have been activated.

Further details of the reel and boom mechanisms, the boom tip mechanisms, and the satellite clamp are shown in Figures 4-4, 4-5, and 4-6, respectively. A more detailed explanation of the operation of the mechanisms is provided in Section 5.

4.3 Limited Operational System Configuration and Layout

The limited operational system is geometrically similar to the operational system, but with limited instrumentation of the 1.44 m (56.75 in.) diameter satellite and with the 1 mm (0.039 in.) diameter tether limited to 10 km (6.21 miles) or less. Otherwise, the two configurations are mechanically and geometrically similar. The 50 m (164 ft) boom and large reel mechanism are still used (Fig. 4-7). The operational sequence is shown in Figure 4-8.

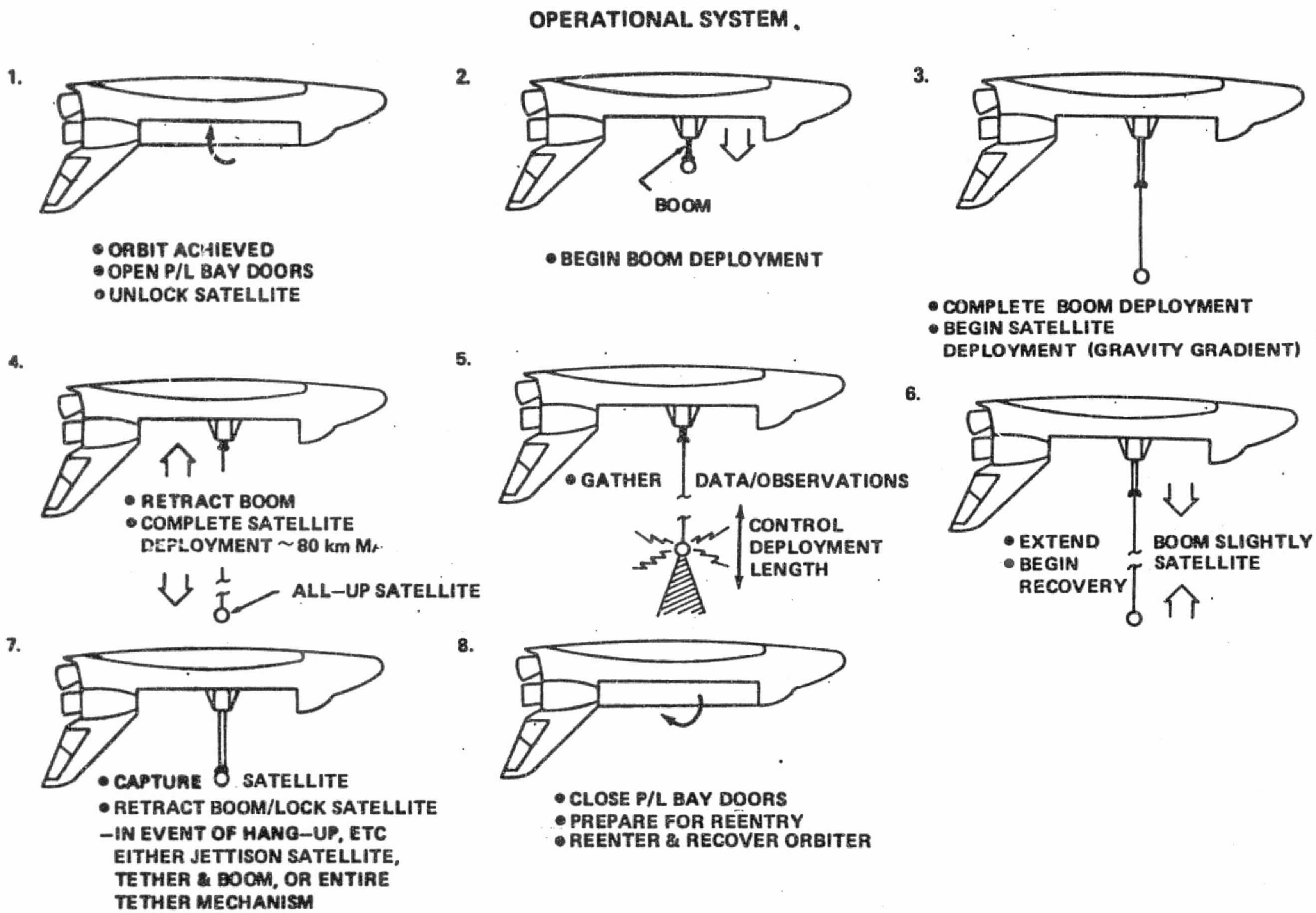


Figure 4-2. Operational sequence for the operational system.

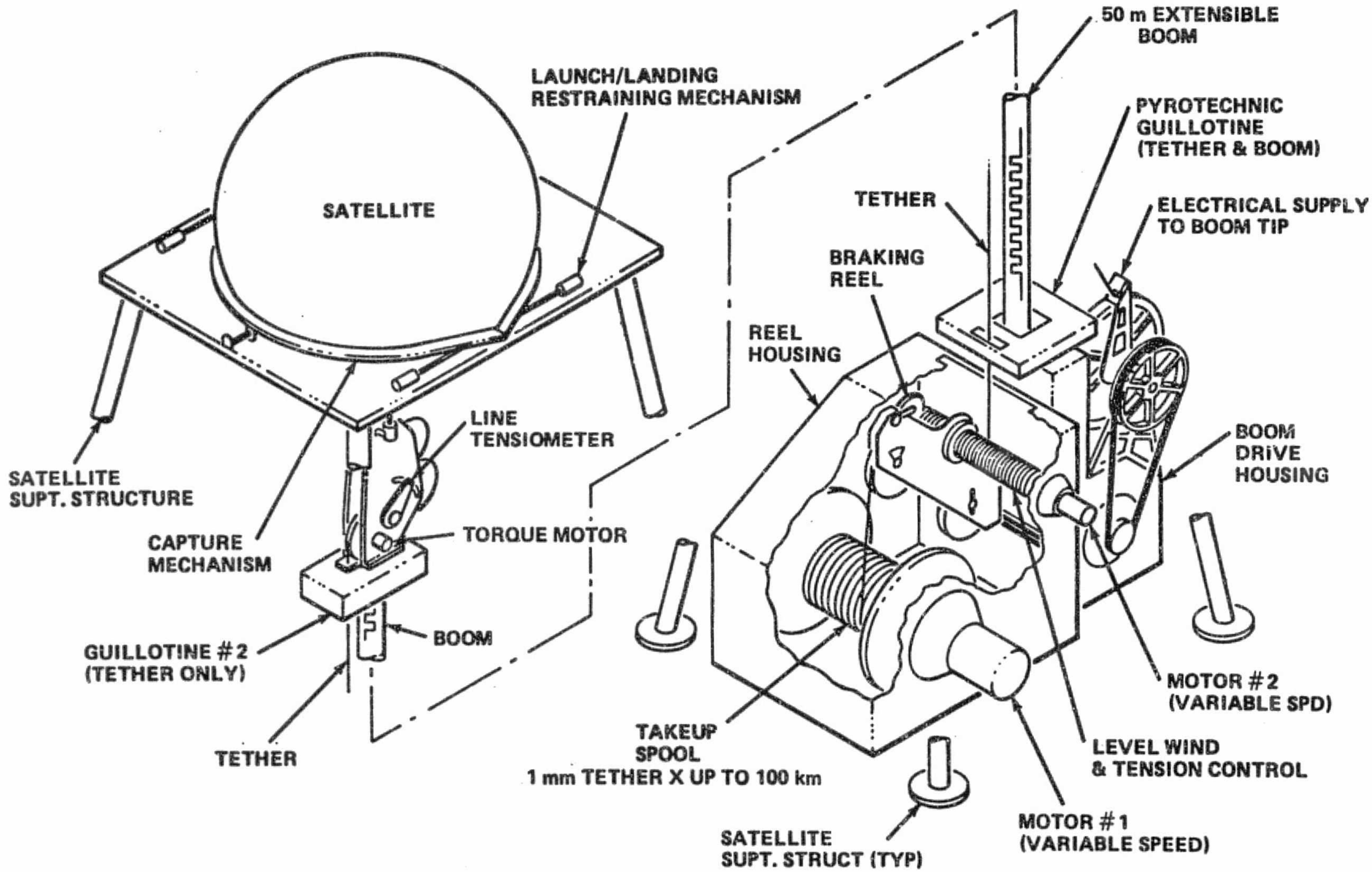


Figure 4-3. Operational configuration: boom and reel with satellite.

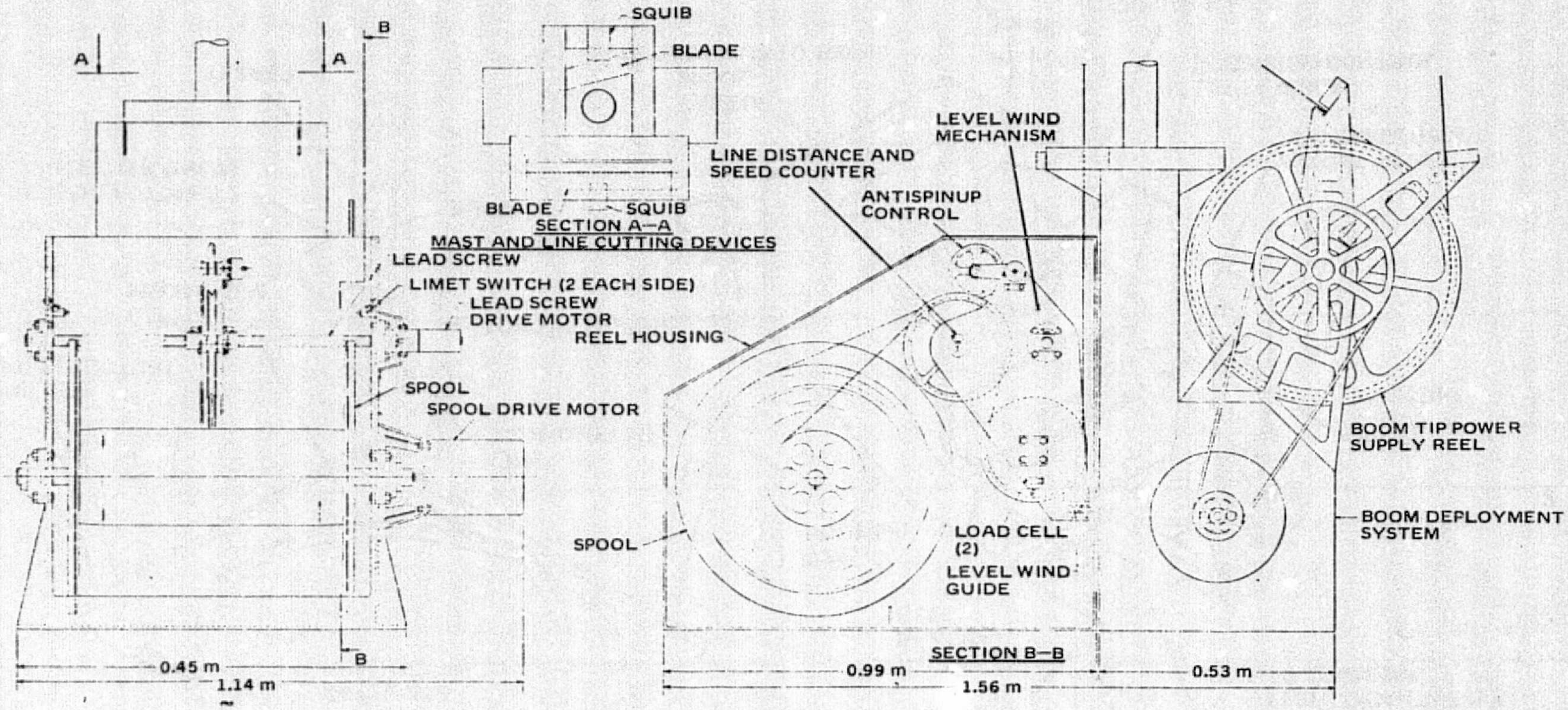


Figure 4-4. Reel mechanism.

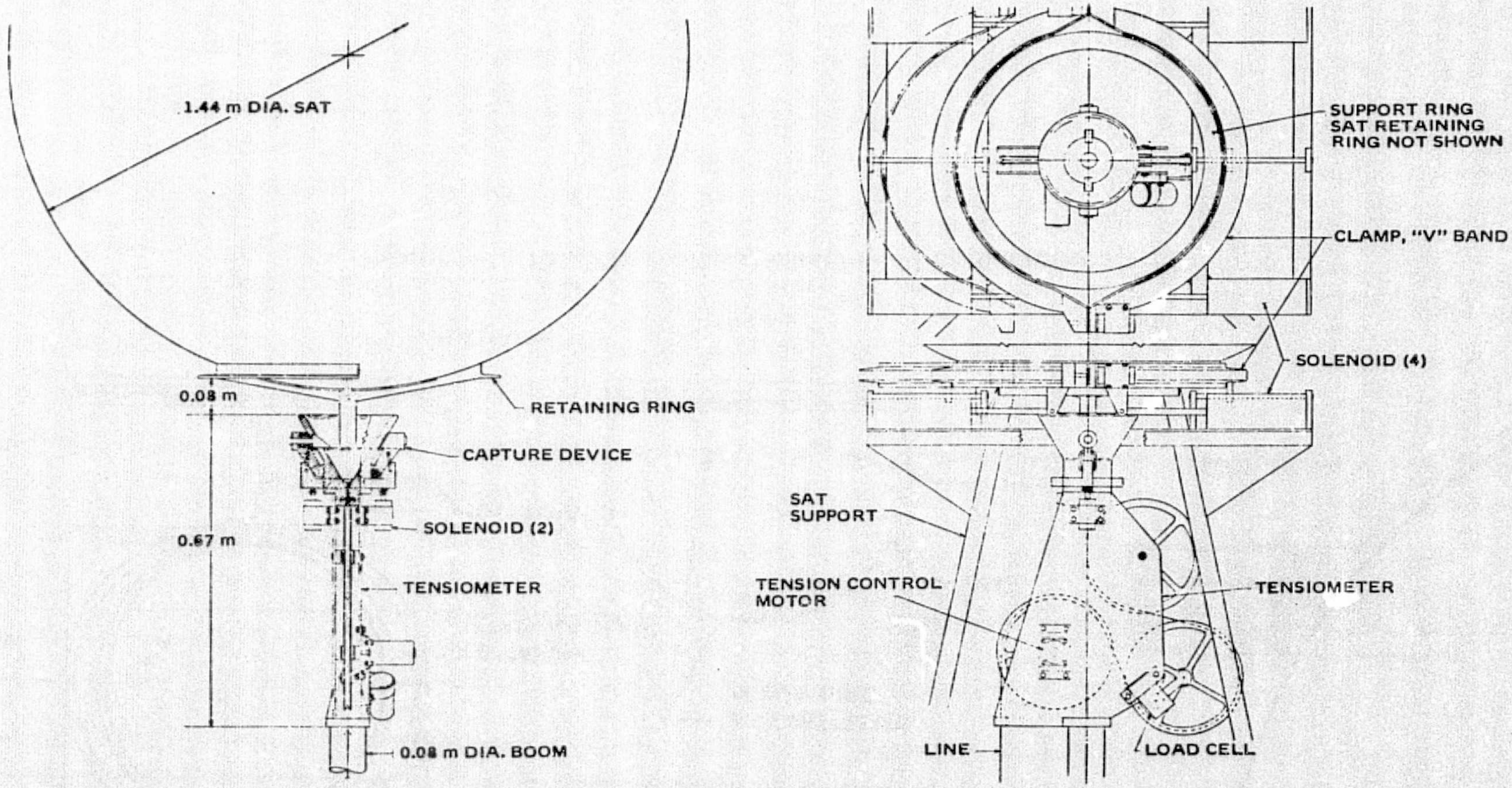


Figure 4-5. Boom tip and restraining mechanism.

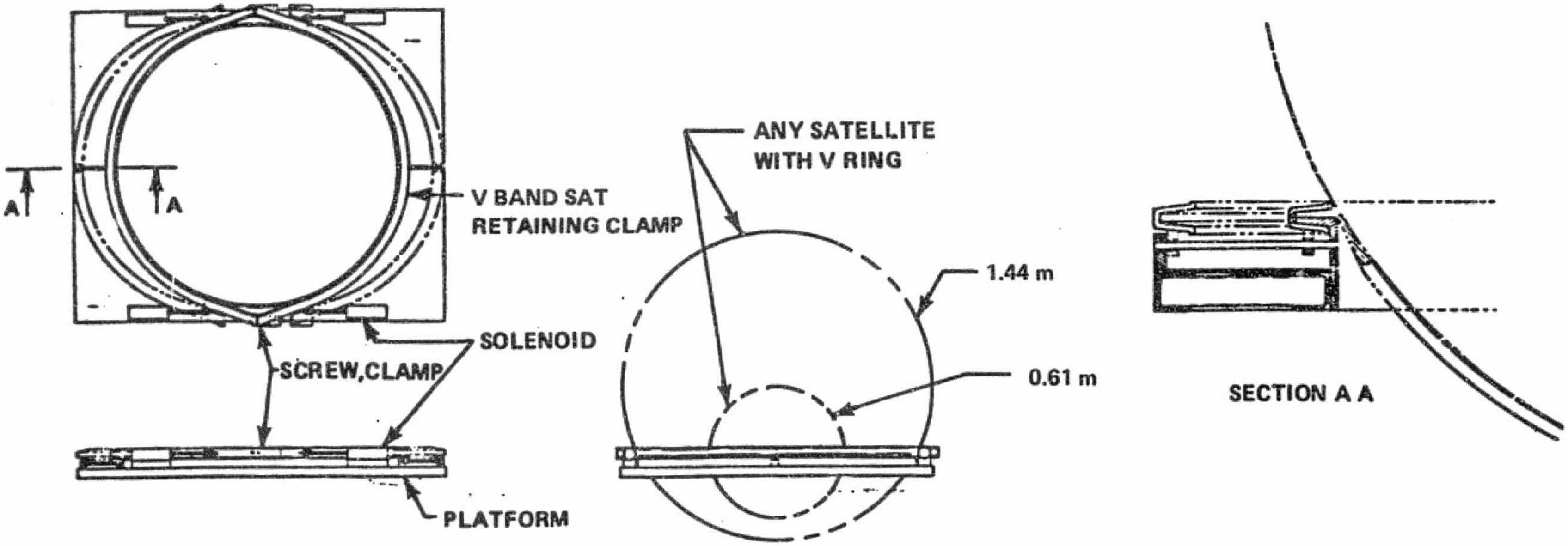


Figure 4-6. Satellite retaining clamp — all configurations.

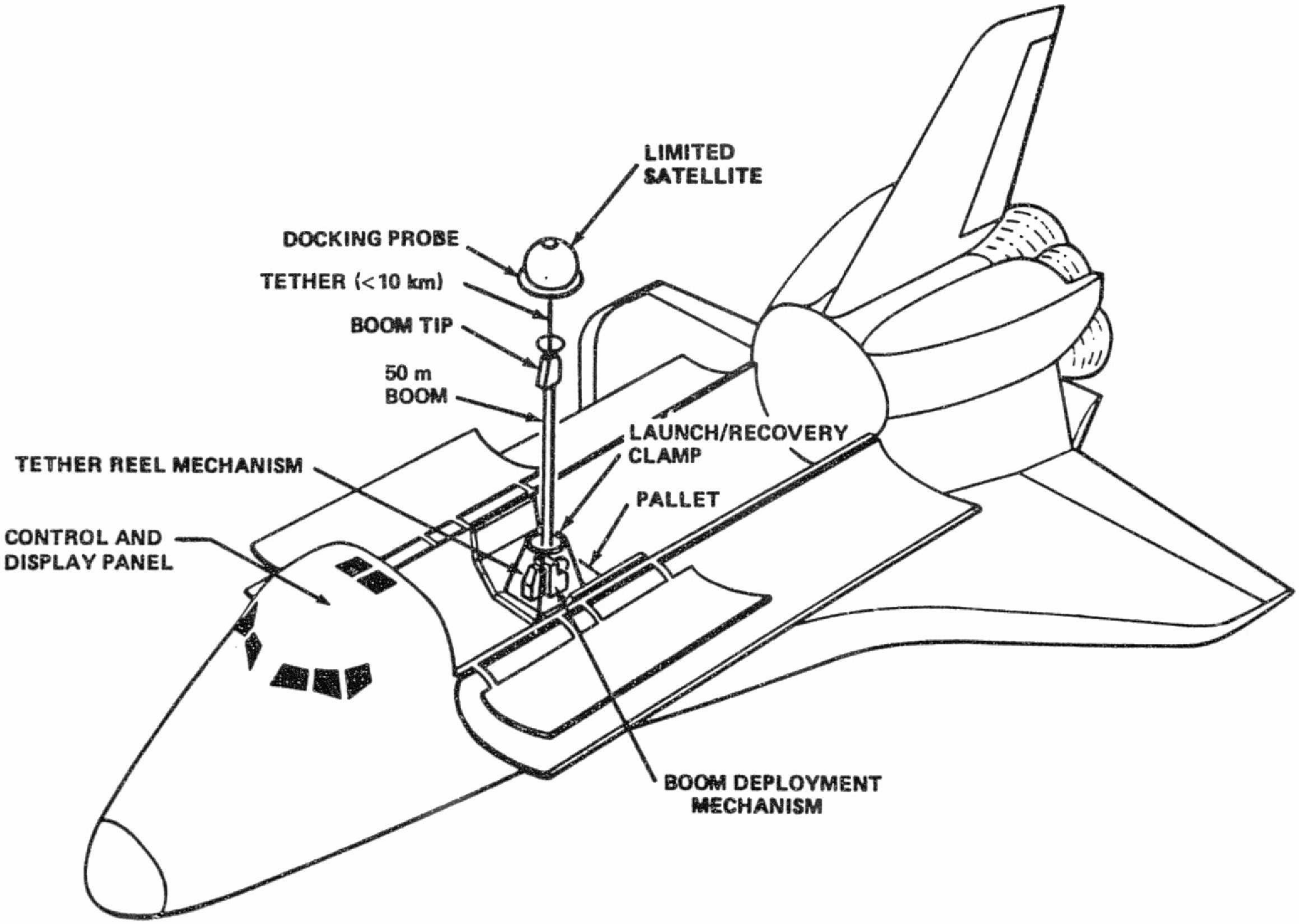


Figure 4-7. Limited operational system.

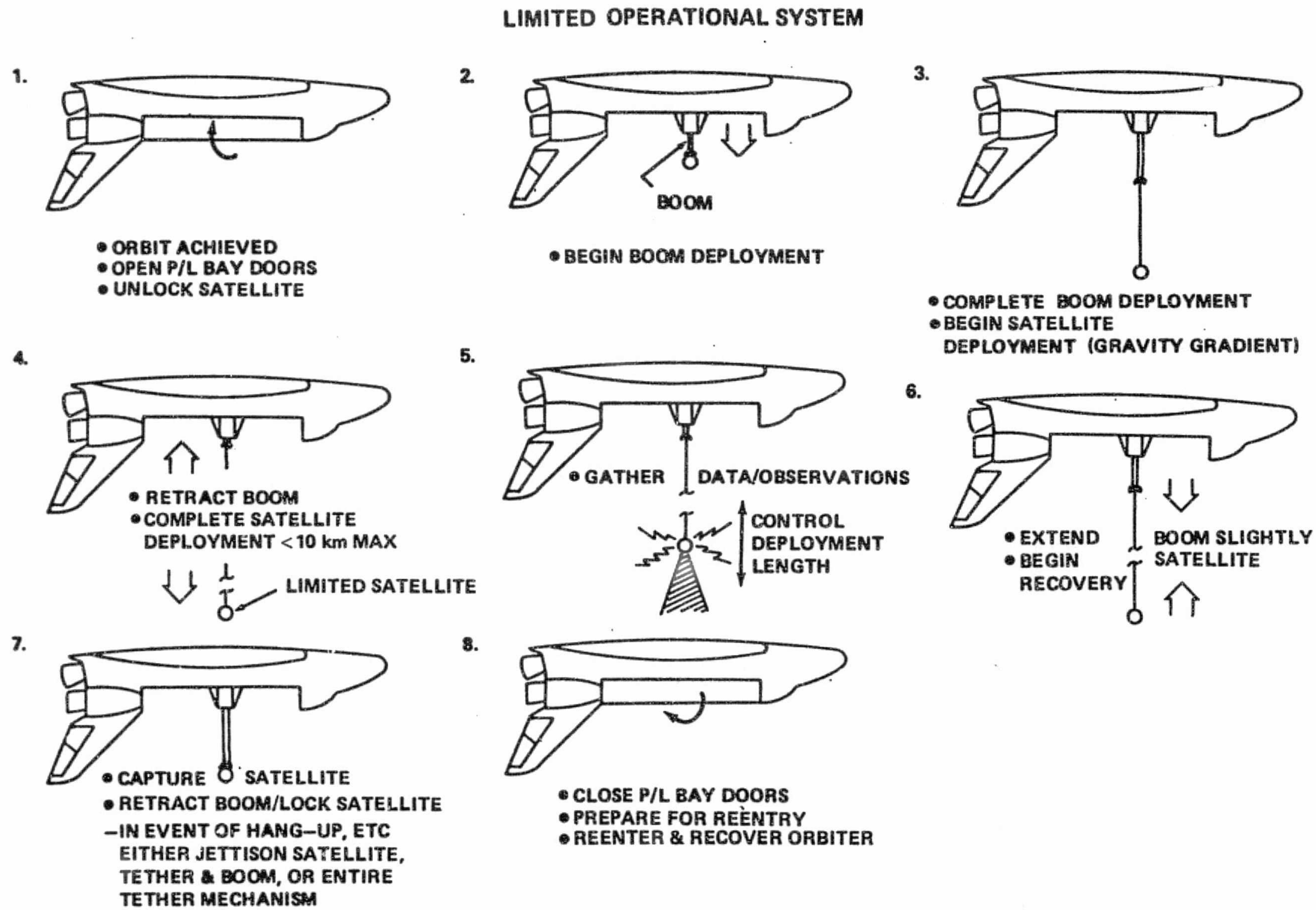


Figure 4-8. Operational sequence for the limited operational system.

4.4 Concept Demonstration System Configuration and Layout

The concept demonstration system, illustrated by Figure 4-9, is the minimum system with which it is possible to adequately demonstrate the tether concept. It consists of an uninstrumented 0.61 m (24 in.) diameter satellite which is a surplus Lageos engineering test model, a 0.366 mm (0.0144 in.) diameter tether with length limited to 10 km (6.2 miles) or less, a smaller reel mechanism, and with the boom replaced by the Shuttle RMS. The operational sequence of this system is shown in Figure 4-10.

Some details of the reel mechanism and satellite support structure are shown in Figure 4-11. More detailed assembly drawings are not provided for the concept demonstration system as such; however, portions of the alternate design configurations presented in Section 4.5 are applicable. Specifically, the concept demonstration system would use only the reel portion of Figure 4-12 and support structure and clamp portion of Figure 4-13.

4.5 Alternate Design Configuration

As mentioned in Section 4.1, two different diameter tethers, and therefore two reel sizes, were considered during the study. Some applications might appropriately use the smaller tether. Figure 4-12 shows the reel mechanism which is capable of accommodating a 0.366 mm (0.0144 in.) diameter solid stainless steel or woven Aramid yarn tether 100 km (62.14 miles) long. Smaller diameter pulleys can be used at the boom tip as shown in Figure 4-13.

Two different satellite clamping mechanisms were considered. The V-ring and band clamp shown in Figure 4-13 is preferred over the restraining latches shown in Figure 4-14.

Finally, two different methods for driving the level wind mechanism were considered. The drive screw motor that is slaved to the spool drive motor (Fig. 4-12) is preferred over the mechanical harmonic drive mechanism shown in Figures 4-15 and 4-16. The harmonic drive mechanism uses the self reversing cam lead screw shown in Figure 4-17.

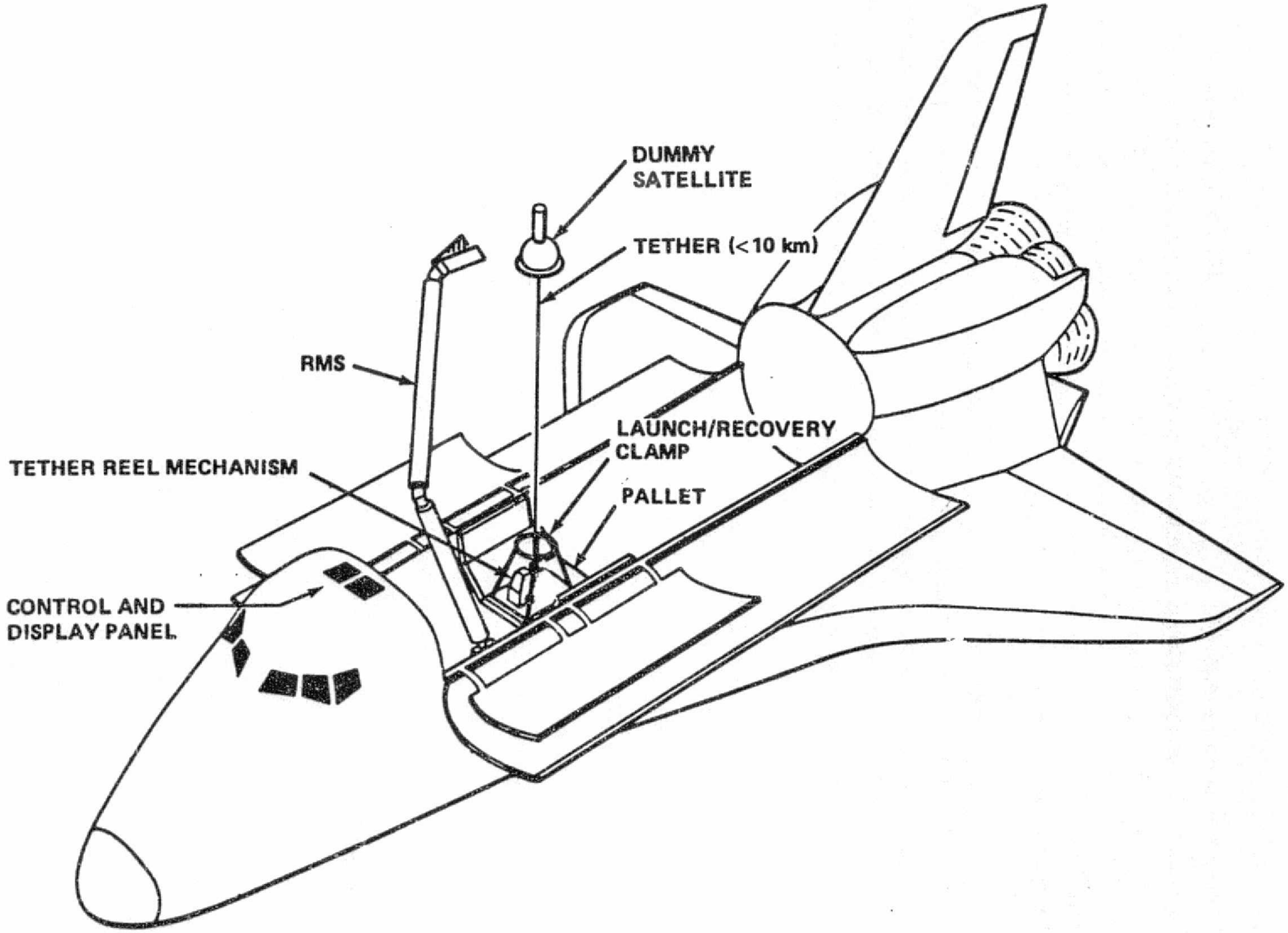


Figure 4-9. Concept demonstration system.

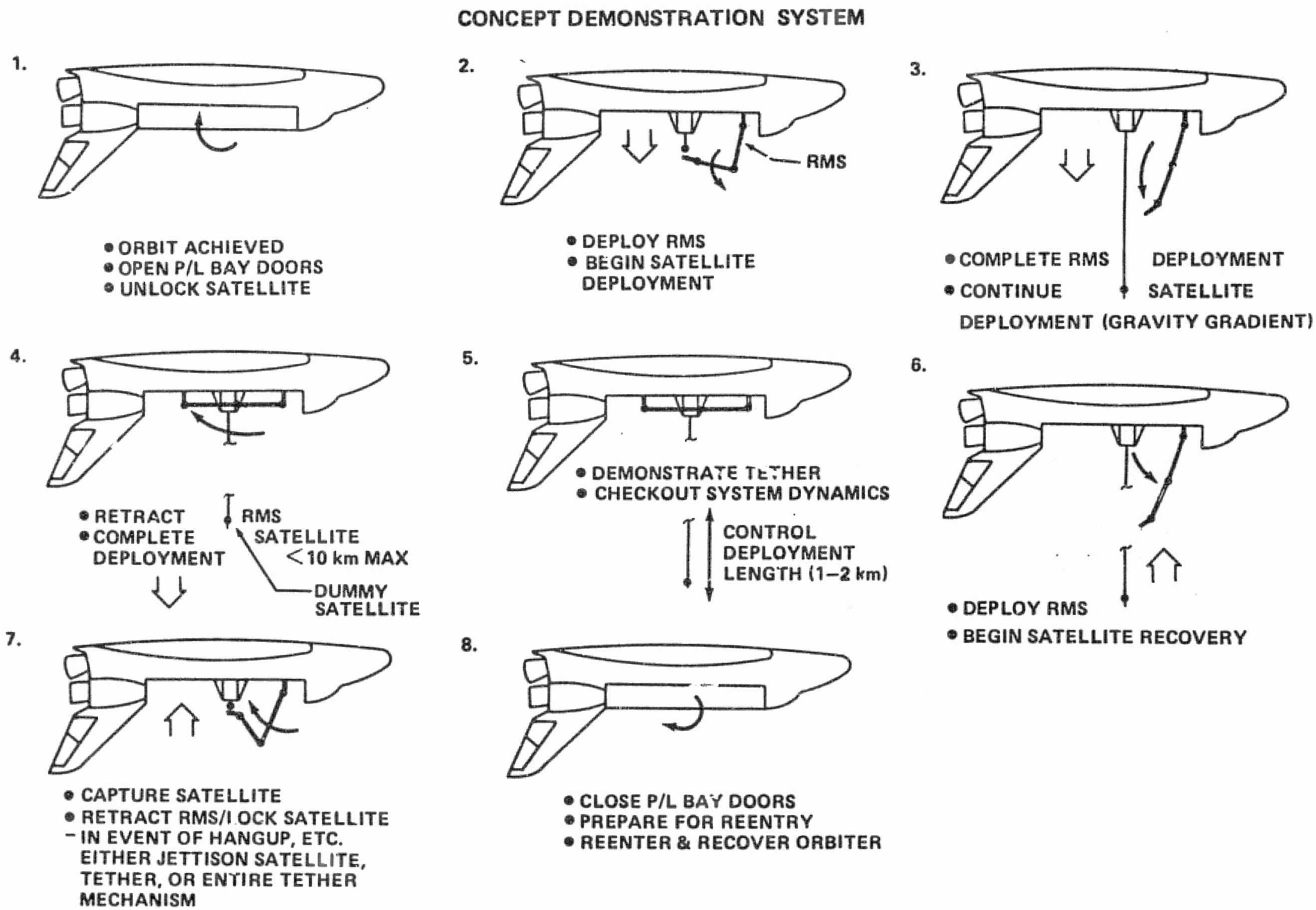


Figure 4-10. Operational sequence for the concept demonstration system.

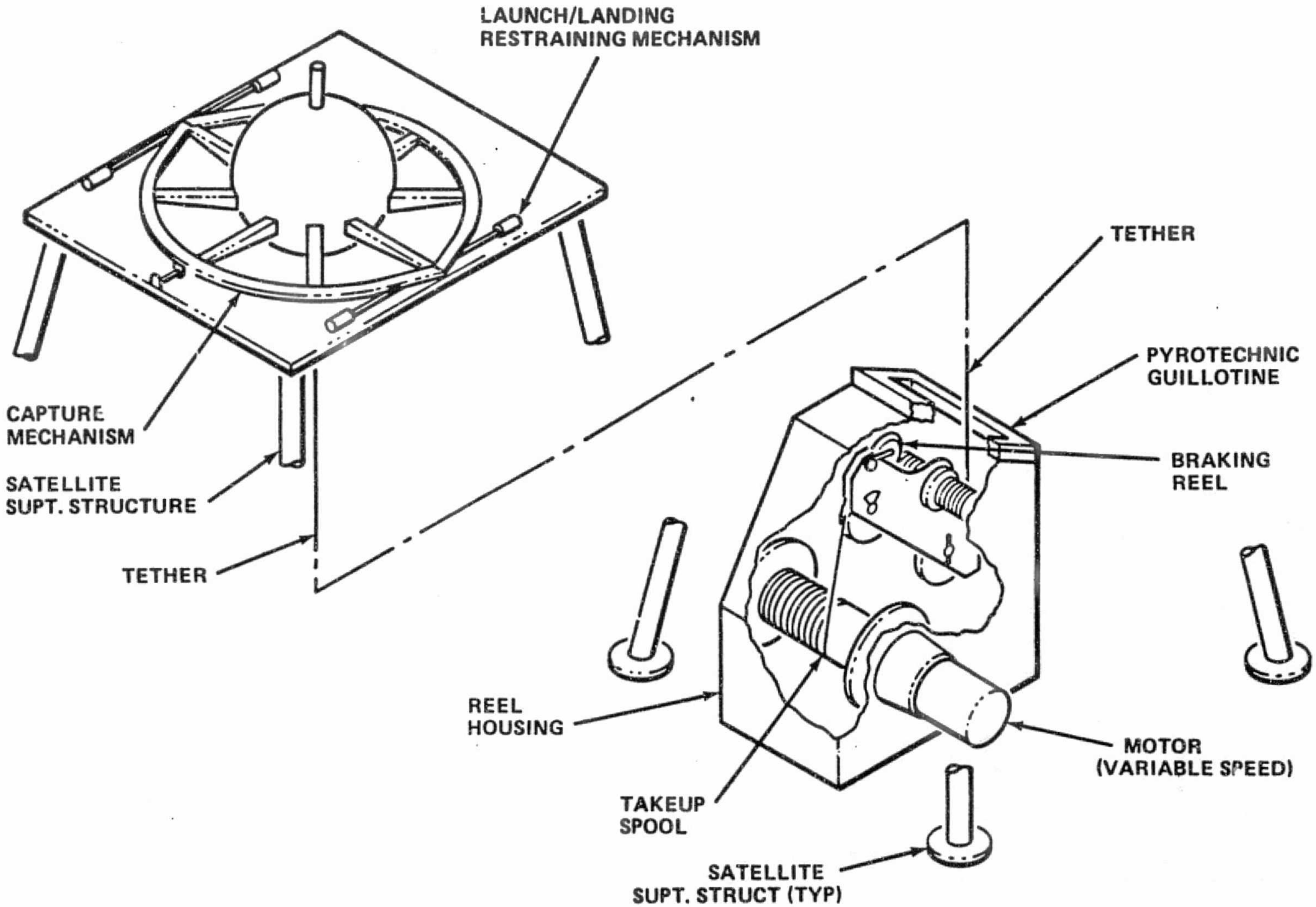


Figure 4-11. Concept demonstration tether system reel mechanism and support structure.

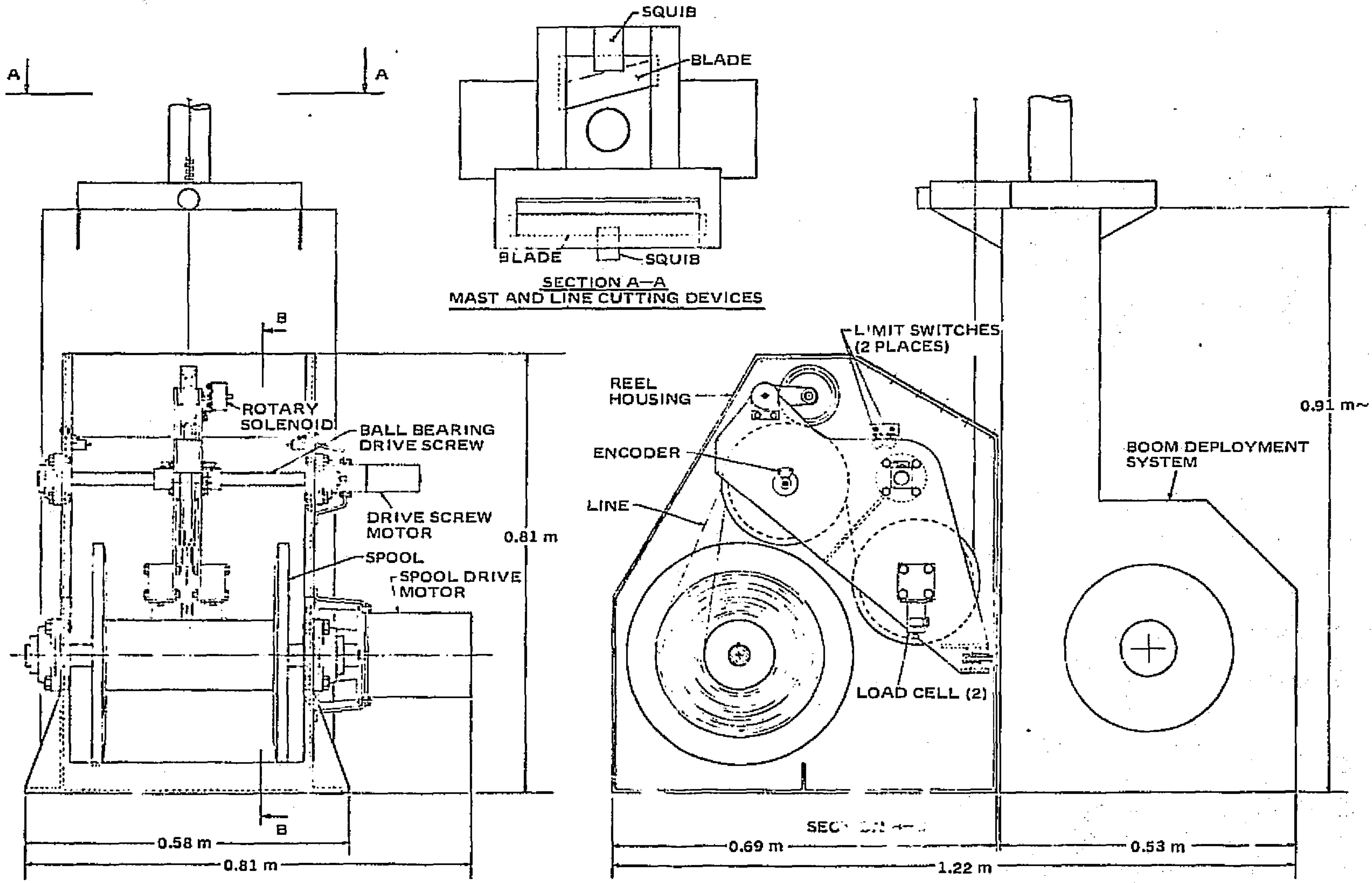


Figure 4-12. Reel mechanism.

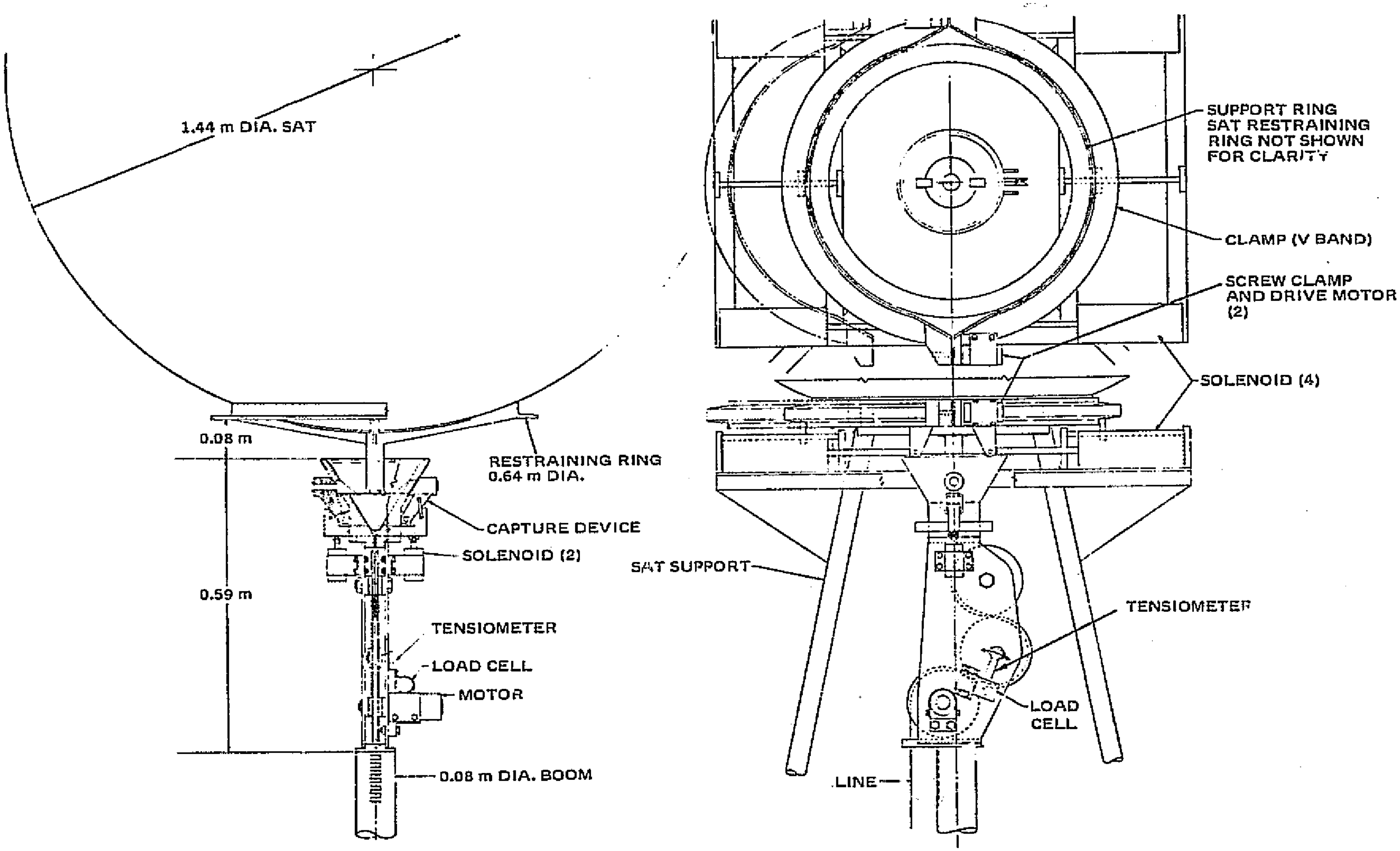


Figure 4-13. Boom tip and support structure.

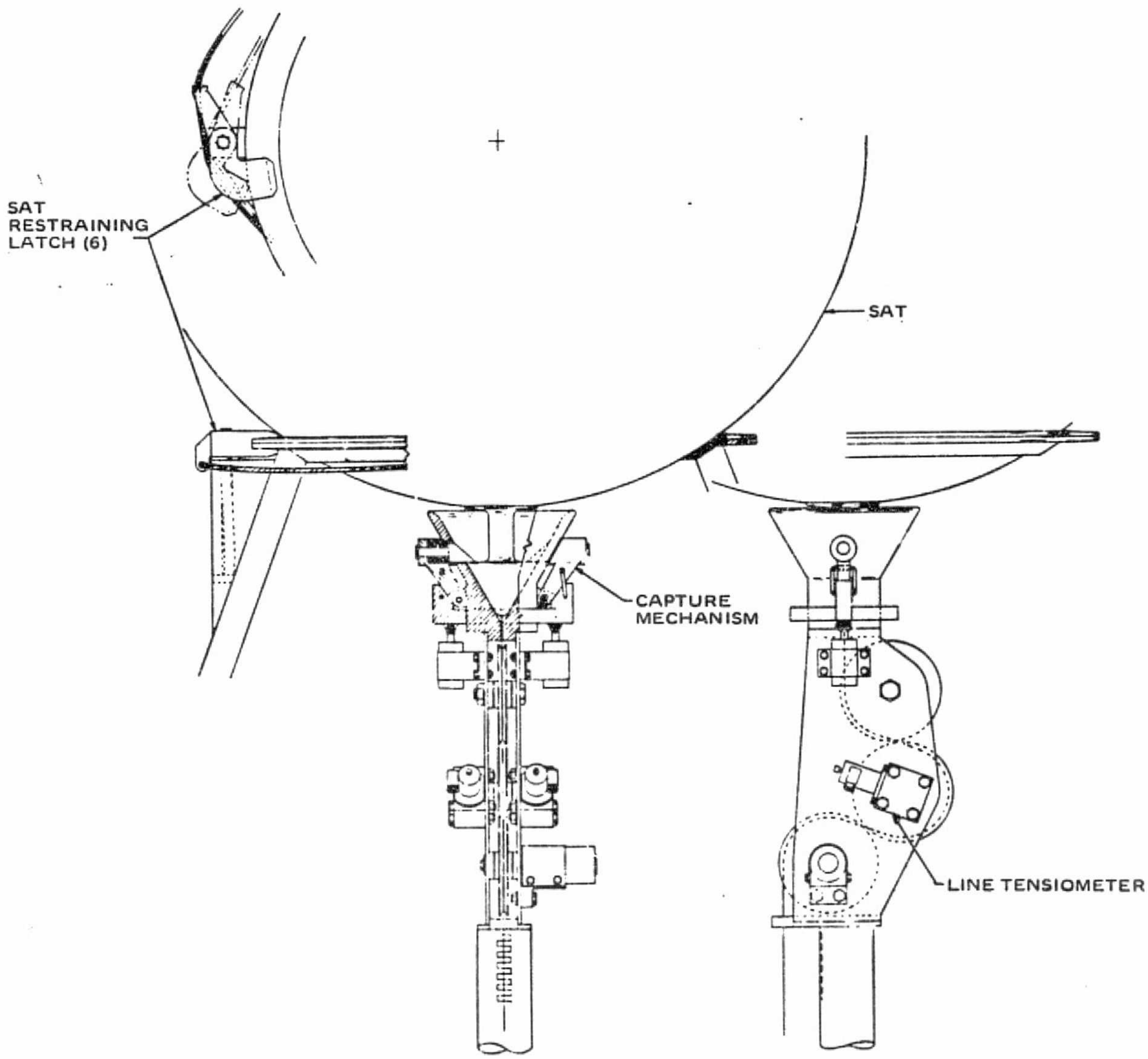


Figure 4-14. Boom tip.

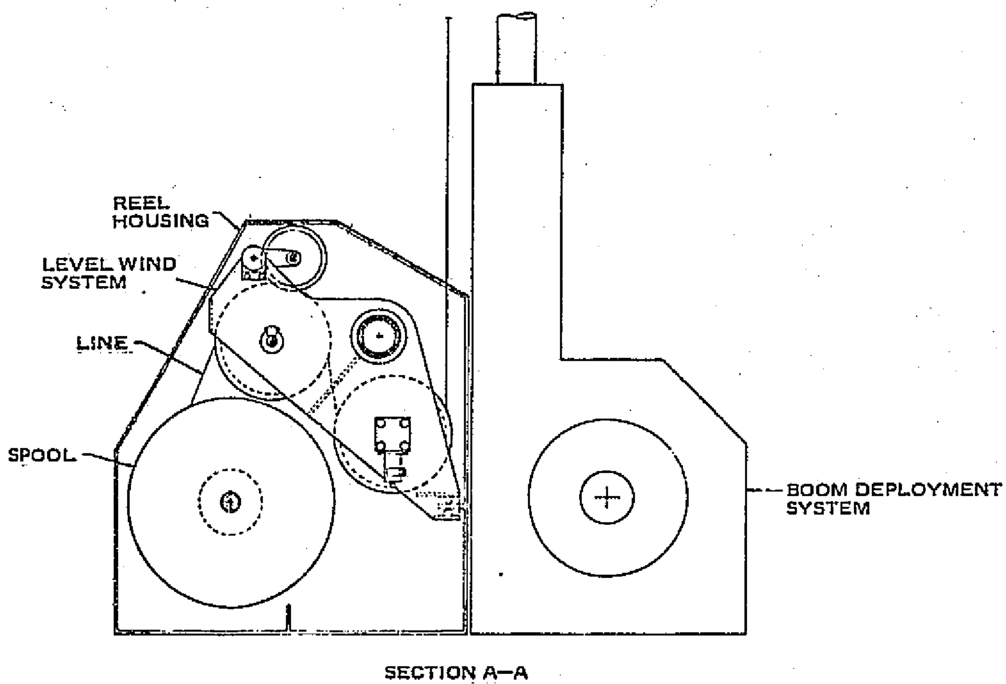


Figure 4-15. Reel mechanism (side view).

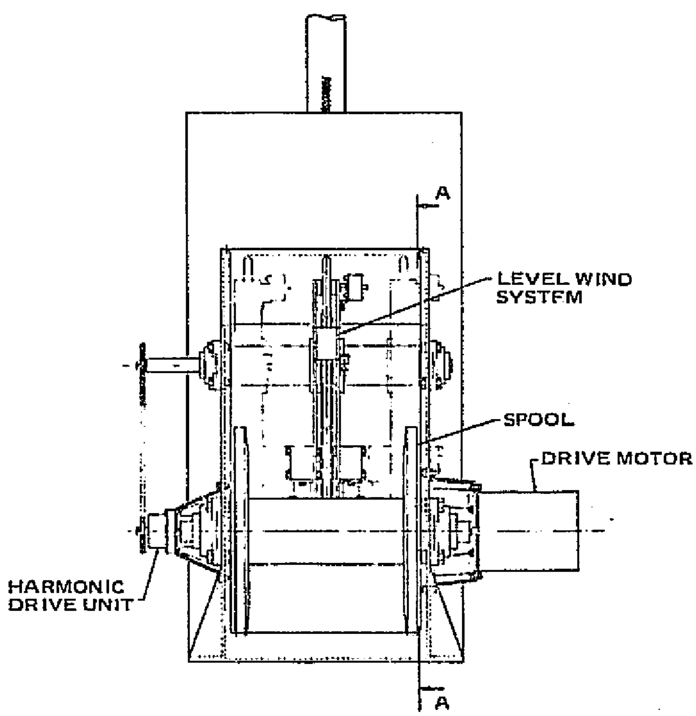


Figure 4-16. Reel mechanism (front view).

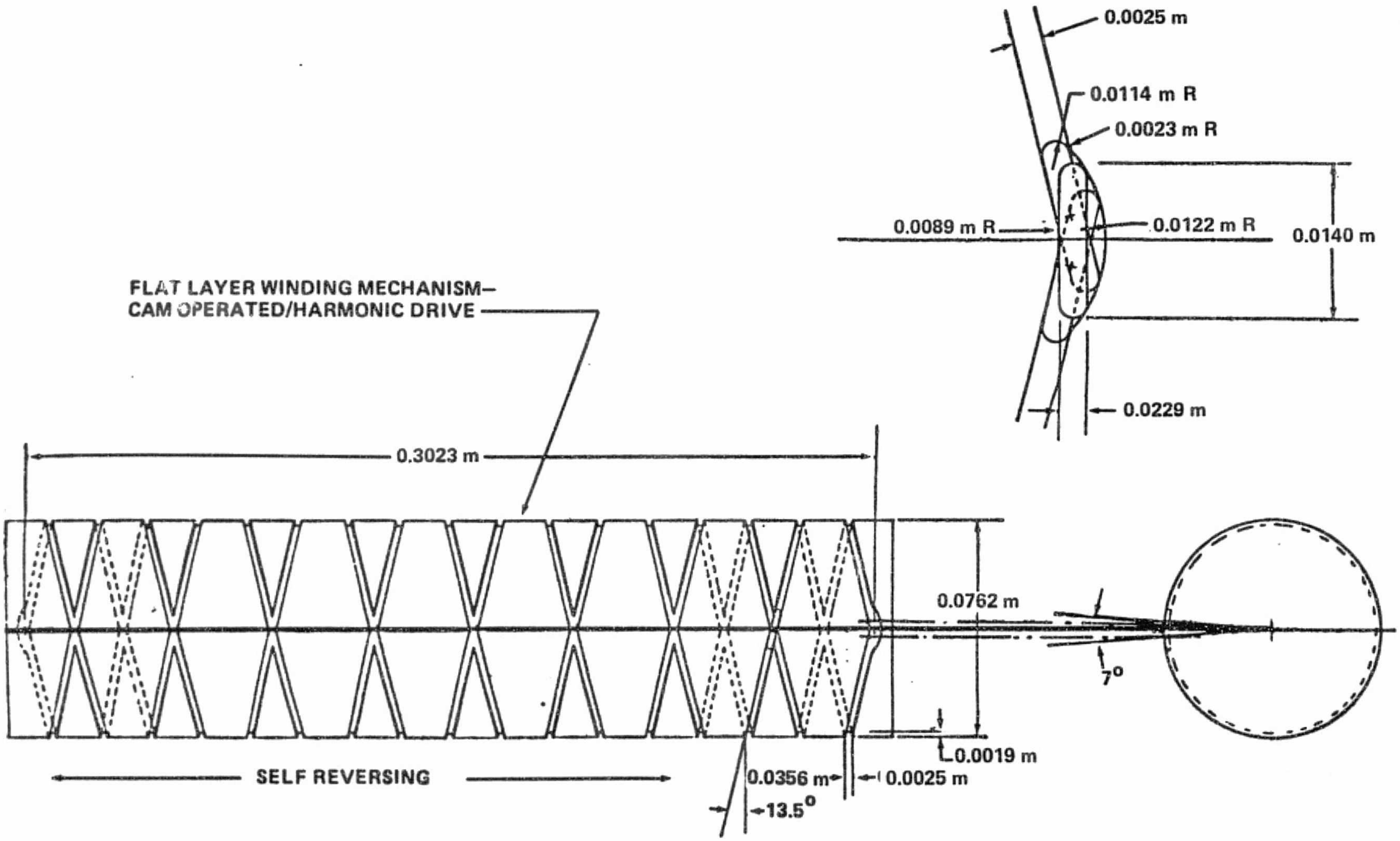


Figure 4-17. 0.366 mm (0.0144 in.) diameter tether – alternate reel mechanism – cam lead screw.

5.0 SUBSYSTEMS DESCRIPTION

This section describes the following subsystems for the three tether system configurations described in Section 4:

a. Reel and Boom Mechanism

b. Thermal Control

c. Electrical Power

d. Communications and Data Handling

e. Control and Display

f. Control

g. Tracking and Data Acquisition

h. Tether

i. Satellite

One exception was that the boom mechanism was not used for the concept demonstration system. The most complete system (operational) is described first and then the limited operational and concept demonstration systems are described, referring to the operational system where possible.

5.1 Operational Tether System

The operational system is configured to deploy a satellite to an altitude of 120 km (64.8 n. mi.) from an Orbiter at 200 km (102 n. mi.) altitude.

5.1.1 Reel and Boom Mechanism

While the three tether configurations are quite different in capability and mission, their mechanical functions are similar. The reel assembly consists of a motor-driven take-up spool on which the tether is wound, a level wind and tension control mechanism mounted on a motor-driven lead screw, various tension sensors, revolution counters, a 50 m (164 ft) motor-driven extensible boom (repla[illegible] by the Shuttle RMS in the concept demonstration system), a flat conducto[illegible] electrical supply line to the boom tip on a large spool (not required in the concept demonstration system), and various pyrotechnic guillotines.

The level wind mechanism is probably the only device that requires additional explanation. The level wind mechanism will assure that the tether is wound evenly on the take-up spool. An "autopilot," which uses revolution counter (s), tension-sensors, braking reel, and control electronics, varies and reverses the relative speeds of the spool drive-motor and lead screw drive-motor such that the assembly moves back and forth on the lead screw laying an evenly spaced row of tetherline on the spool. To assure free winding/unwinding, the spool is dished and wasp-waisted on both ends.

The large reel is designed to accept up to 100 km (62.1 miles) of 1 mm (0.039 in.) diameter solid 304 stainless steel wire. This requires a minimum diameter of 228 mm (9 in.) for the spool and idler pulleys. These pulleys are lightened as much as possible to minimize inertia effects so that the tether rate of pay-out or retrieval can be quickly changed. The 228 mm (9 in.) minimum diameter pulley, designed to accommodate the 1 mm (0.039 in.) solid steel line, can also accommodate more flexible or smaller diameter tethers.

The whole mechanism can be easily and economically mocked-up. A flight system can also be easily functionally tested on Earth. As shown, the reel/boom mechanism is a relatively simple device of high reliability with numerous fail-safe features.

5.1.2 Thermal Control

5.1.2.1 Introduction

A thermal investigation was conducted to determine the environmental temperatures for a tethered satellite and equipment mounted in the Orbiter payload bay. The study encompassed the thermal effects of aerodynamics, solar, Earth radiation and albedo, and internal heat upon a tethered subsatellite and its tether. Passive thermal control techniques to maintain acceptable temperature limits were first studied and when the temperatures exceeded these limits, an active system was employed to maintain satisfactory temperature control.

5.1.2.2 Analysis

The aerodynamic heating on the satellite and stainless steel tether was determined and is examined in this section.

5.1.2.2.1 Tether (Stainless Steel). The tether (stainless steel) is so small (diameter ≤ 1.0 mm) that the tether is in the free molecular flow regime over the entire altitude range (altitude ≥ 110 km). The Knudson number (molecular mean free path/tether diameter) of the tether at 120 km altitude is approximately 400, and it increases with altitude. A Knudson number of 10 or greater is sufficient to guarantee free molecular flow.

The equilibrium temperature of the tether was found by balancing the aerodynamic and solar heat input against the emitted radiant energy assuming that the absorbtivity and emissivity are equal to 0.85. Figure 5-1 gives the equilibrium tether temperature as a function of altitude with and without solar heating. At 120 km altitude, the equilibrium tether temperature will be 200°C (392°F) with aerodynamic and solar heating and 170°C (338°F) with aerodynamic heating only.

5.1.2.2.2 Nonaerodynamically Stabilized Satellites. If there is no requirement for aerodynamic altitude control, simple external shapes such as spheres and cylinders can be used. For the thermal investigation, a sphere with a diameter of 1.4 m was studied. The nominal expected external temperature of a spherical uncooled shell satellite versus orbital altitude was computed and the results are shown in Figure 5-2.

5.1.2.3 Results and Graphs

The largest increase in temperature of the tether per unit altitude occurs below 120 km. The maximum anticipated tether temperature will be 200°C at 120 km (Fig. 5-1). Below this altitude, the temperature begins to rise very rapidly. At 200°C, the tether does not present any thermal related problems.

At an altitude of 120 km and assuming two internally controlled temperatures of 30°C and 50°C with an external temperature of 190°C (Fig. 5-2), the heat transfer rates through the satellite wall were calculated for varying internally generated heat to obtain maximum orbit time (Fig. 5-3). This curve

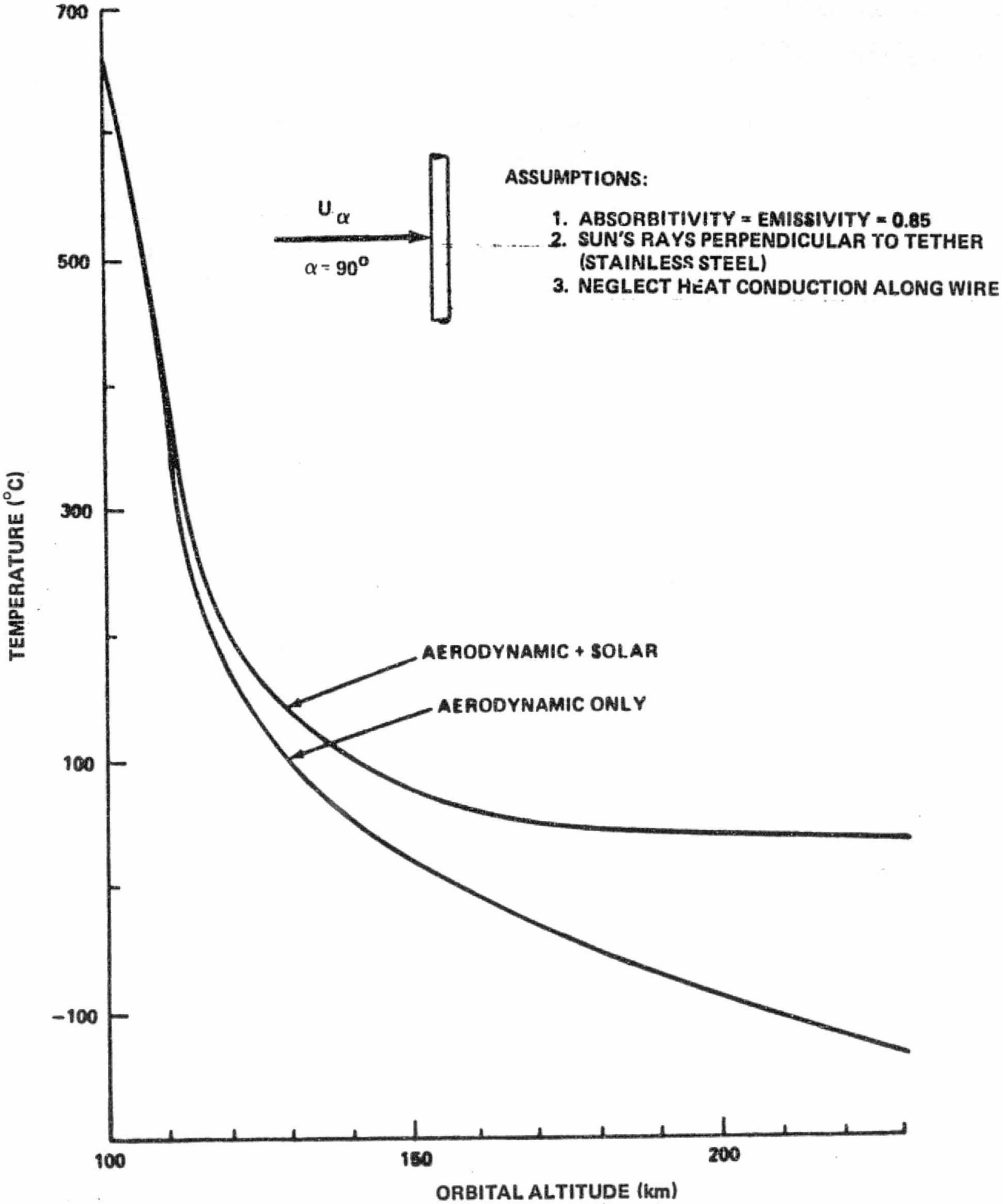


Figure 5-1. Equilibrium tether temperature versus altitude.

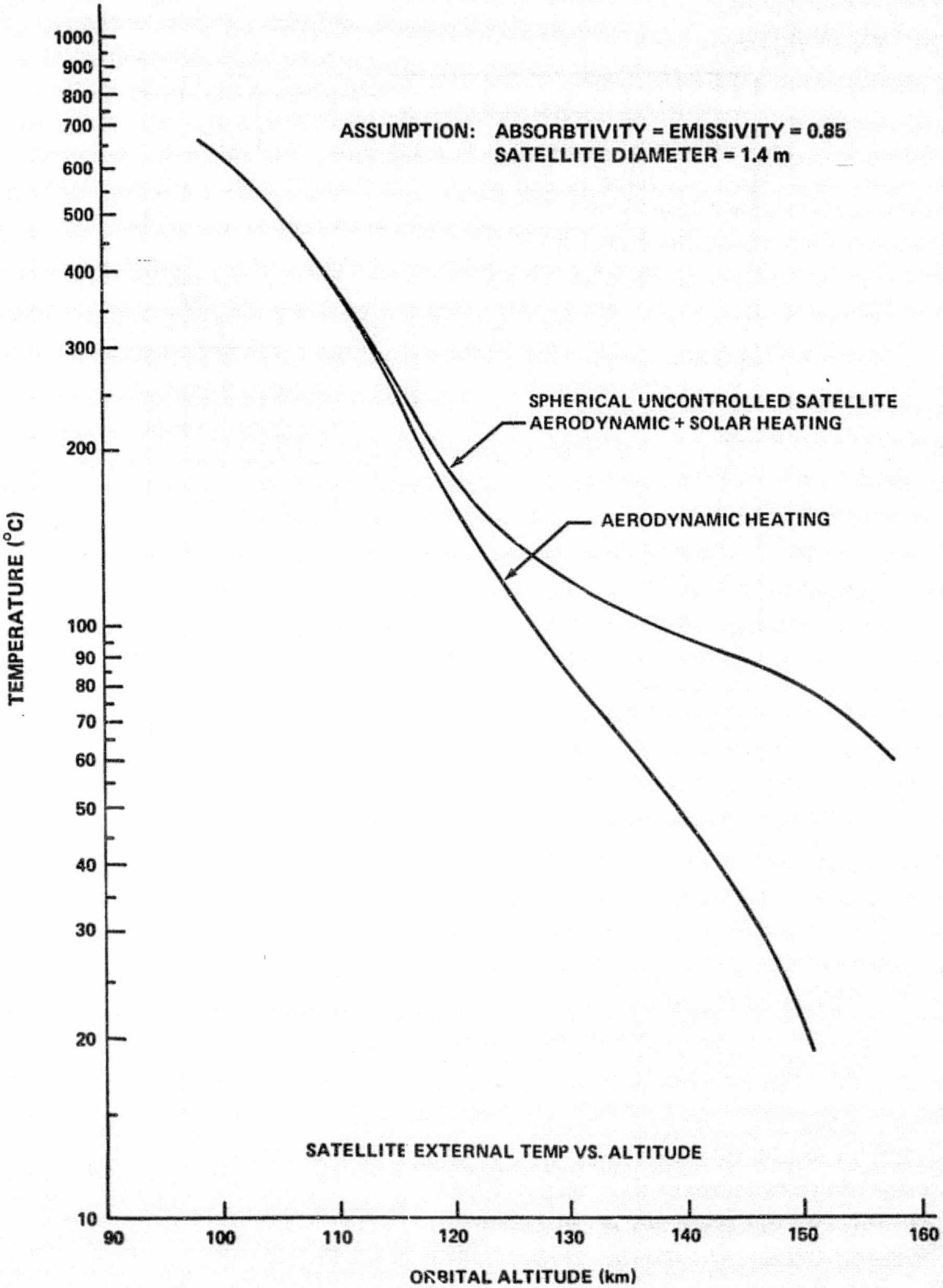


Figure 5-2. Equilibrium surface temperature for spherical satellite (diameter = 1.4 m).

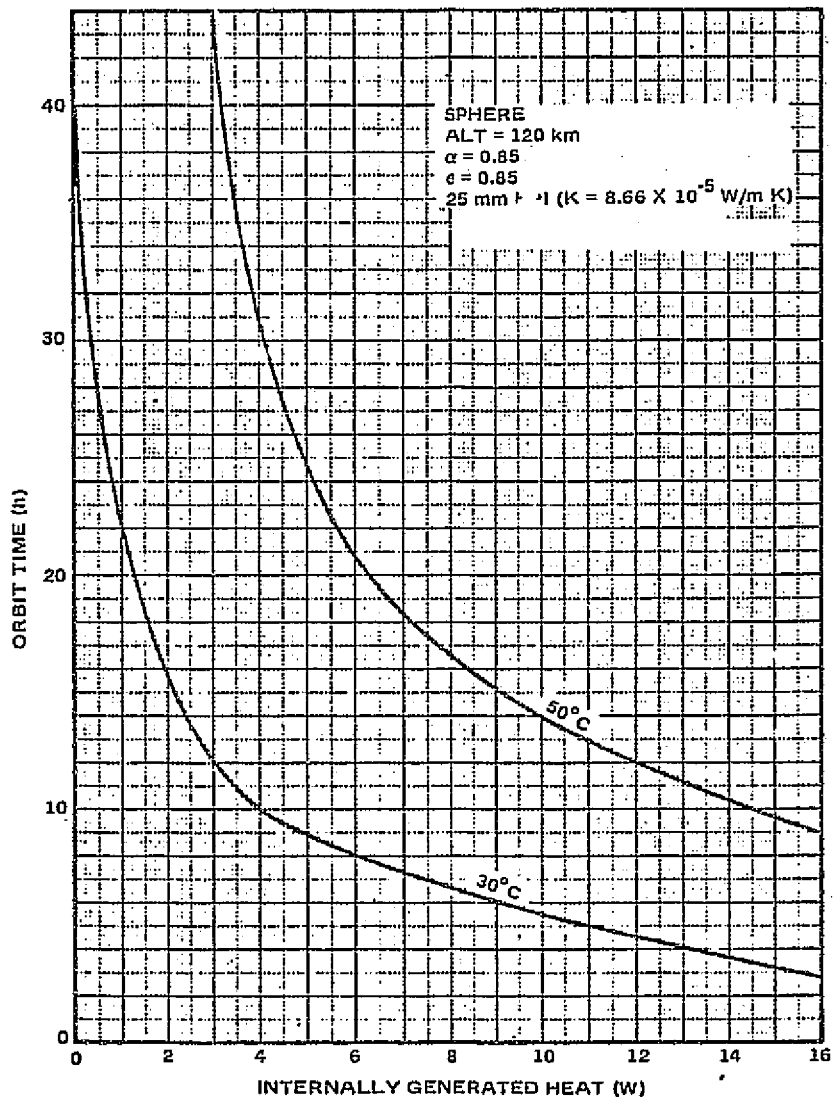


Figure 5-3. Tethered skyhook payload orbit time as a function of temperature and generated heat.

was produced using 25 mm of externally applied high performance insulation (HPI) to reduce the rate of heat flow internally and still not allow the internal temperature to increase too rapidly. Figure 5-3 indicates that an active system will be required if any large quantity of internal heat generation is anticipated.

The internal heat load that can be expected for scientific equipment when using a battery was determined to be 125 W. For this internal heat load, a parametric curve was produced to determine the amount of liquid nitrogen required to hold the internal temperature at 30°C and 50°C for a given time in hours (Fig. 5-4). Two tanks of 533 mm (21 in.) ID will be required for a 150 h duration. The tanks will require 25 mm (1 in.) of HPI to maintain the 105 kg liquid nitrogen for this duration in a 30°C environment.

Electrical components mounted in the Orbiter payload bay have a temperature range of -12° to 55°C (10° to 131°F). The anticipated temperature of the pallet bottom for an Orbiter orientation with the Z axis along the local vertical is -6° to -11°C. Therefore, the actual temperature of any equipment mounted in the payload bay will be a result of internal heat loads. A range of 90 to 145 W/m^2 can be handled passively. If this range is exceeded, then an active system will have to be used which will mean using the Orbiter cooling system, i.e., cold plates on the pallet.

Nothing was discovered from the thermal analyses that indicated any insurmountable problems. The tether temperature, satellite, pallet mounted equipment, and mechanical devices in the payload bay present no problems. An active cooling system using liquid nitrogen was sized for the satellite. Other active systems for the satellite should be investigated to determine any potential benefits with respect to cost and weight effectiveness.

5.1.3 Electrical Power

5.1.3.1 Introduction

The electrical power subsystem is composed of two parts: the Orbiter mounted and the satellite mounted. Orbiter power is used as the primary source for operation of the equipment associated with deployment and retrieval equipment as well as power for the communications and data handling system.

A silver-zinc secondary battery provides power for caution and warning and for safing operations in the event of the loss of primary power.

5.1.3.1.1 Orbiter Mounted. The major load for the power system is the reel drive motor. Approximately 950 W peak is required for operation of the motor. The extendable boom requires 10 W to extend and 15 W to retract. A closed-loop

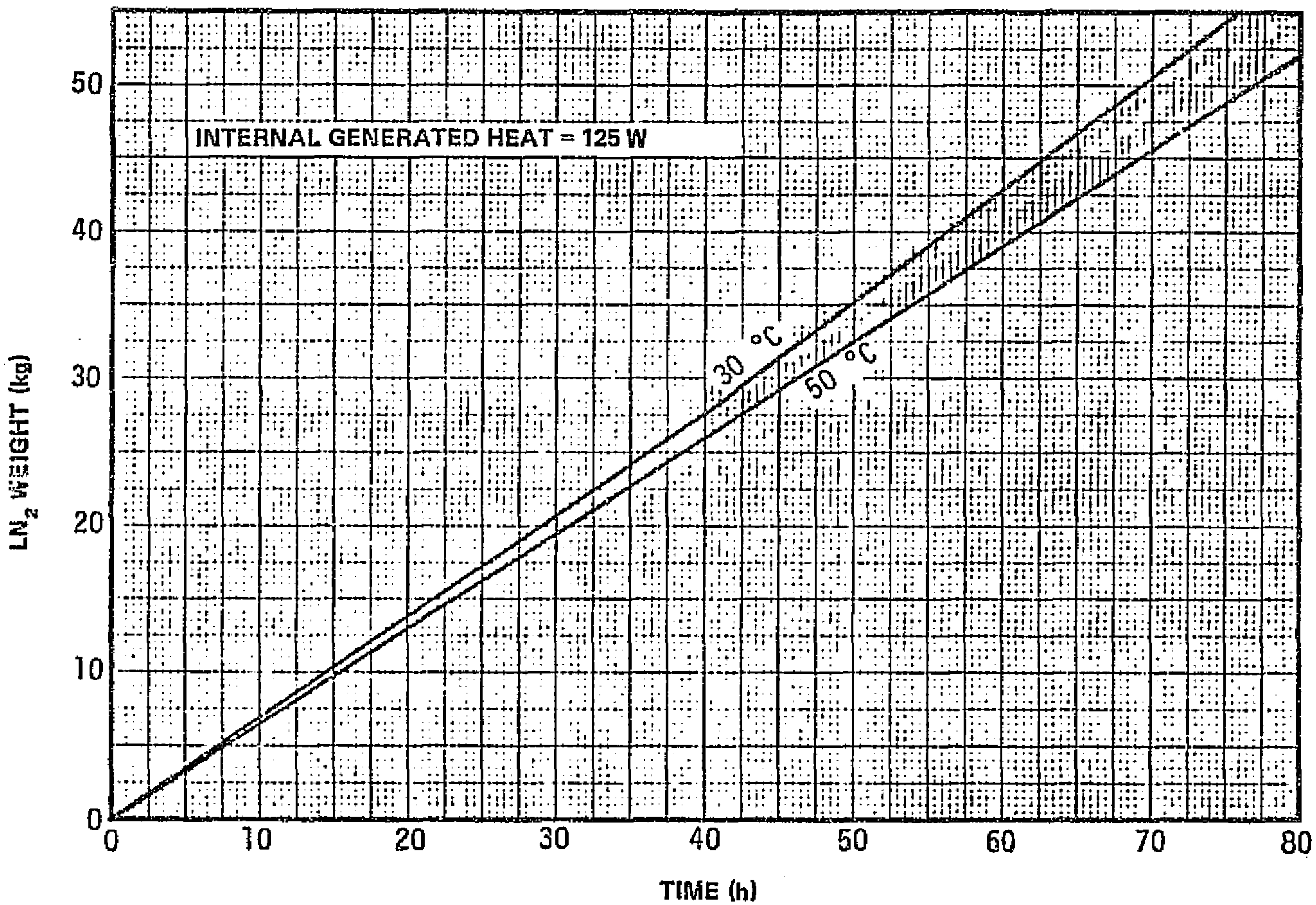


Figure 5-4. Tethered skyhook payload LN_2 requirement as a function of time and temperature.

control system is utilized for deployment and retraction and tether tension control. The digital processor, the input/output unit, and the servo amplifier require a total of 120 W. Peak power level required is 1128 W. The average power required for a 35.22 h test is 120.8 W representing 4255 W-h of energy required from the Orbiter. An average of 35 W is needed for backup purposes if main power is lost. This power is required only for short periods. If total main power loss occurs with the satellite deployed, the backup power source provides power for jettisoning of the tether system components. The total backup energy required is estimated to be approximately 18 W-h.

5.1.3.1.1.1 Orbiter Considerations. During satellite deployment, the forces will be such that the reel drive motor will be operating as a generator. This must be considered in the design of the motor control circuitry as well as an interface consideration.

5.1.3.1.2 Satellite System. The electrical power subsystem associated with the satellite is merely a battery and associated distribution equipment. Assuming minimum instrumentation, a total of 590 W-h are required for a 35.22 h mission (Fig. 5-5). Low cost lithium primary batteries are recommended because of their energy density. A single 2.2 kg (5 lb) battery will accommodate the mission. The average load is 14 W (1/2 amp at 28 V). The electrical power subsystem would have to be resized to accommodate additional instrumentation, if required.

5.1.4 Communications and Data Handling

5.1.4.1 Introduction

The primary purpose of the communications and data handling subsystem of the operational tether system is to provide attitude and position information of the satellite relative to the Orbiter and to provide instrumentation and recording of tether housekeeping and engineering measurements. Real time data for scientific or performance analysis are not required by the tether system, but might be required for some applications. Figure 5-6 presents a block diagram of the communications and data handling subsystem.

5.1.4.2 Satellite Position and Attitude Instrumentation

The fact that real time attitude and position information is not required for the tether system allows rather simple and potentially low cost instrumentation to determine the payload attitude and position relative to the

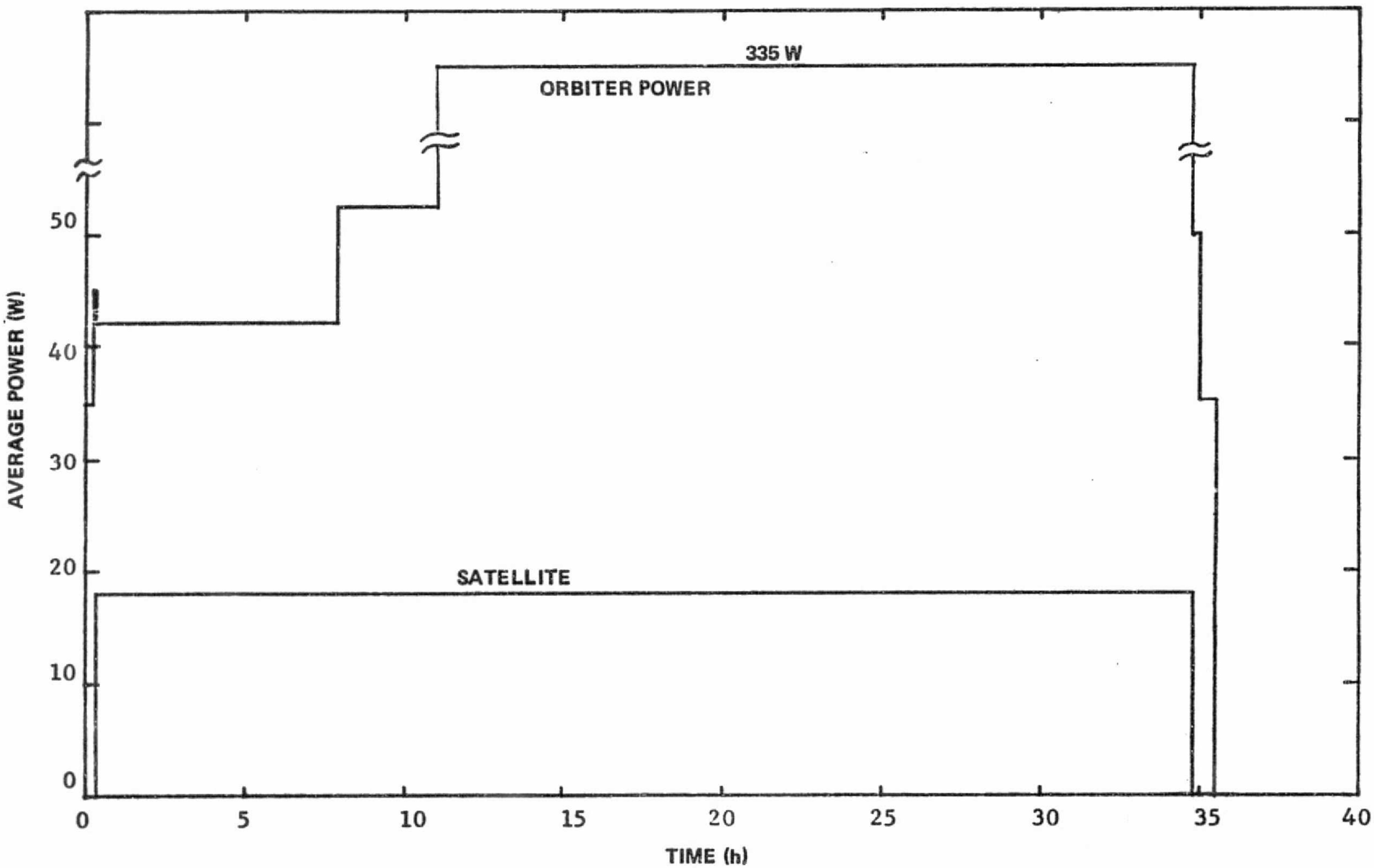


Figure 5-5. Operational system power profile (power averaged for each phase).

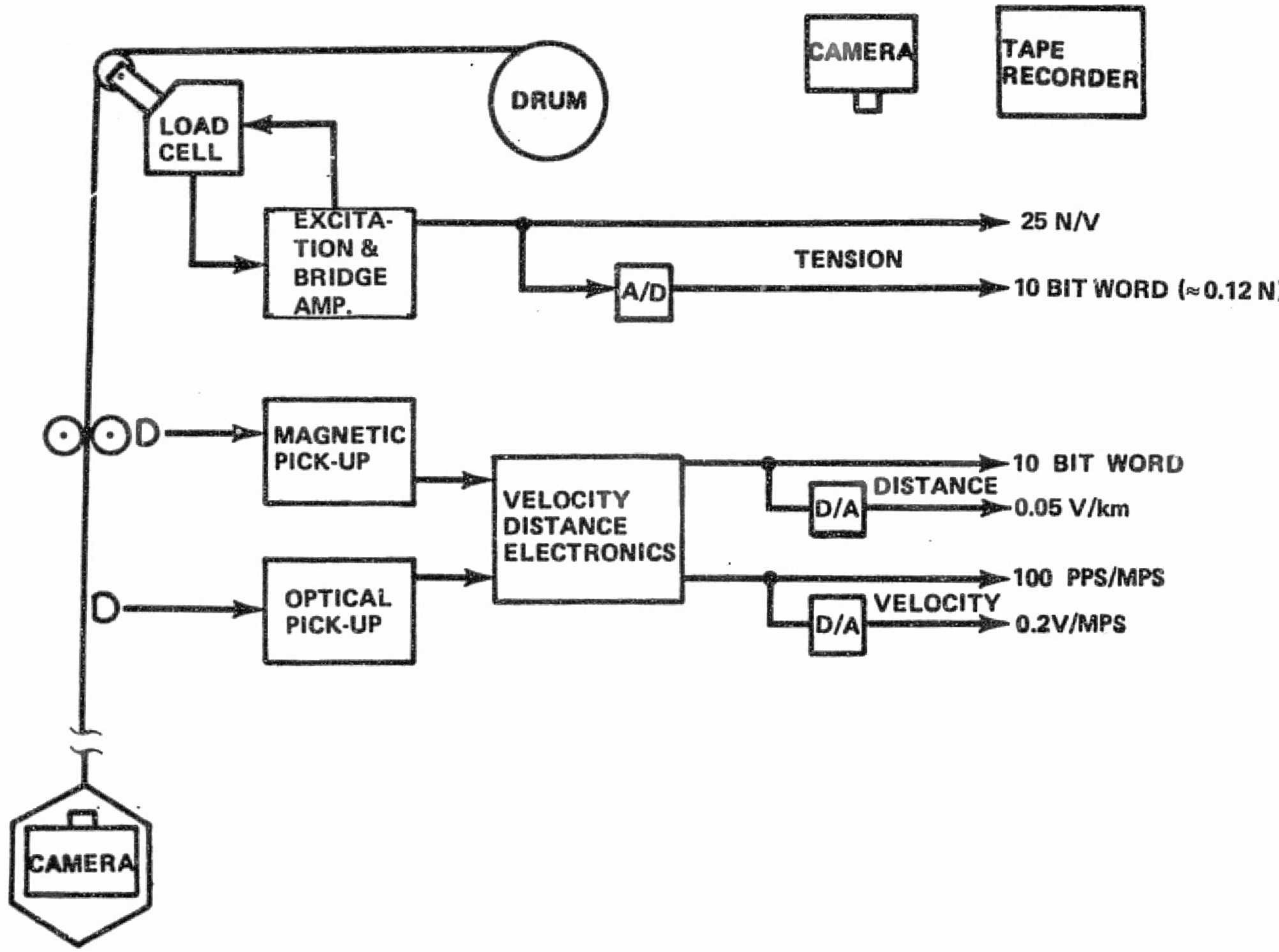


Figure 5-6. Communications and data handling subsystem.

Orbiter. Radio frequency and optical (laser) tracking concepts were eliminated due to their relatively high cost and complexity. Two concepts involving still photographic cameras were evaluated. One concept utilized a camera in the satellite which would photograph the Orbiter against the star field. This concept was discarded because of the rather stringent requirement to photograph the star field with a short exposure time. A short exposure time would be required because of the velocity of the Orbiter relative to the star field. The concept chosen has a camera located in the Orbiter and the satellite. Both cameras take photographs of the Earth and are time referenced. The attitude of the satellite can then be determined by the ephemeris data of the Orbiter and comparison of Earth landmarks on the two photographs. A film camera is included in the design of the operational satellite to obtain position and attitude of the satellite and is a potential candidate for a science payload.

Additional satellite instrumentation can be accommodated in the satellite to determine attitude, attitude rate, and acceleration information. The list, shown in Table 5-1, is typical of some of the available instrumentation sensors for this purpose.

5.1.4.3 Tether Instrumentation

The tether is instrumented to determine the tether tension, length, and rate of deployment or retrieval.

A dual range load cell measures the tension. Dual range is employed to obtain a 0.12 N resolution and a 120 N dynamic range. An analog to digital (A/D) converter provides the tension in digital format for recording and caution and warning, if required.

The velocity and distance are measured by an electronics package that has magnetic and optical sensor inputs. The magnetic pickup is a reluctance type that generates pulses as an idler wheel rotates as driven by tether deployment or retrieval. The electronics package counts the pulses to provide a digital output that represents tether distance. The package also measures the time rate of the pulses and provides a digital output that represents tether velocity. Two digital-to-analog (D/A) converters convert the digital measurements to analog levels for control and display and caution and warning, if required.

TABLE 5-1. CANDIDATE SATELLITE INSTRUMENTATION

Attitude:	Triaxial Magnetic Aspect Sensor Manufactured by Schonstedt Model RAM53C-2 Range — ±600 milliorsted Weight — 0.6 kg (21 oz) Voltage Required — 24 to 32 Vdc
Rate:	3 Axis Rate Package Manufactured by Nortronics Using GRG5 Gyros Range — Is Adjustable Weight — Approximately 0.9 kg (2 lb) Power Required — 10 to 12 W
Accelerometer:	Spar Accelerometer System Using Kearfott 2412 Accelerometer with In-House Electronics Range is Adjustable, Typically 10^{-5} to 18 g Power Required — 5 W Plus Heaters
Temperature Sensors:	Many Types Available Twelve Channels will Probably be Required

An optical sensor provides distance calibration, if required. At selected intervals, the tether will be encoded by paint or an other suitable medium. The coding will be related to the length of tether deployed. This will provide compensation for any slippage in the idle wheel or potential dropouts of reluctance pulses.

5.1.4.4 Computation and Data Handling

A NASA Standard Spacecraft Computer (NSSC) will be utilized for all computation and data handling. The services performed by the NSSC will be storage and execution of the tether control algorithms, data scaling and formatting, assistance in the checkout, caution and warning, command, and display functions.

An interface unit will provide input-output functions for the computer. In addition, it contains such elements as level shifters, A/D and D/A converters, and signal buffering.

5.1.4.5 Payload Data Transmission

A link for RF transmission of payload data to the Orbiter (and thus, to the ground) is included in the present cost estimates. A link to provide this service will not be major provided the data rate is not excessive. The Oribter payload interrogator has the capability of receiving data from free flying payloads at S-band. The maximum data rate is limited to 16 kbs if the data must be displayed and/or manipulated by the Orbiter. Should it not be desirable to process payload data by the Orbiter, then the Orbiter will provide "bent-pipe" data up to 1 MHz bandwidth directly to ground. If these two services are not adequate, the payload will have to provide its own link and interfaces with the ground network.

5.1.4.6 Data Recording

A NASA small standard tape recorder will be used for engineering data recording. The recorder has a bit capacity of 4.5×10^8 bits with a maximum record rate of 1 Mbs. The 30 h average recording rate capability will be approximately 4200 bps.

Table 5-2 lists the estimated power, weight, and volume for the computer, tape recorder, and interface unit. The block diagram for the overall communications and data handling subsystem is illustrated in Figure 5-6.

5.1.5 Control and Display

A control and display panel will be provided in the Orbiter aft cabin for the tether system utilization (see Fig. 5-7 for a typical layout). The panel displays tether length, velocity, and tension as the primary system control parameters. The controls include power on/off, arm/safe, latch/unlatch, deploy/retract, and emergency eject switches. Status lights are provided to show mission phases and other pertinent information such as latch conditions and boom position.

TABLE 5-2. HARDWARE ESTIMATES

	Power (W)	Weight		Volume	
		kg	(lb)	m^3	$(in.^3)$
Computer	17	6.6	(14.5)	1.01×10^{-2}	(618)
Tape Recorder	10	4.5	(10)	5.46×10^{-3}	(333)
Interface Unit	5	2.3	(5)	1.57×10^{-3}	(96)

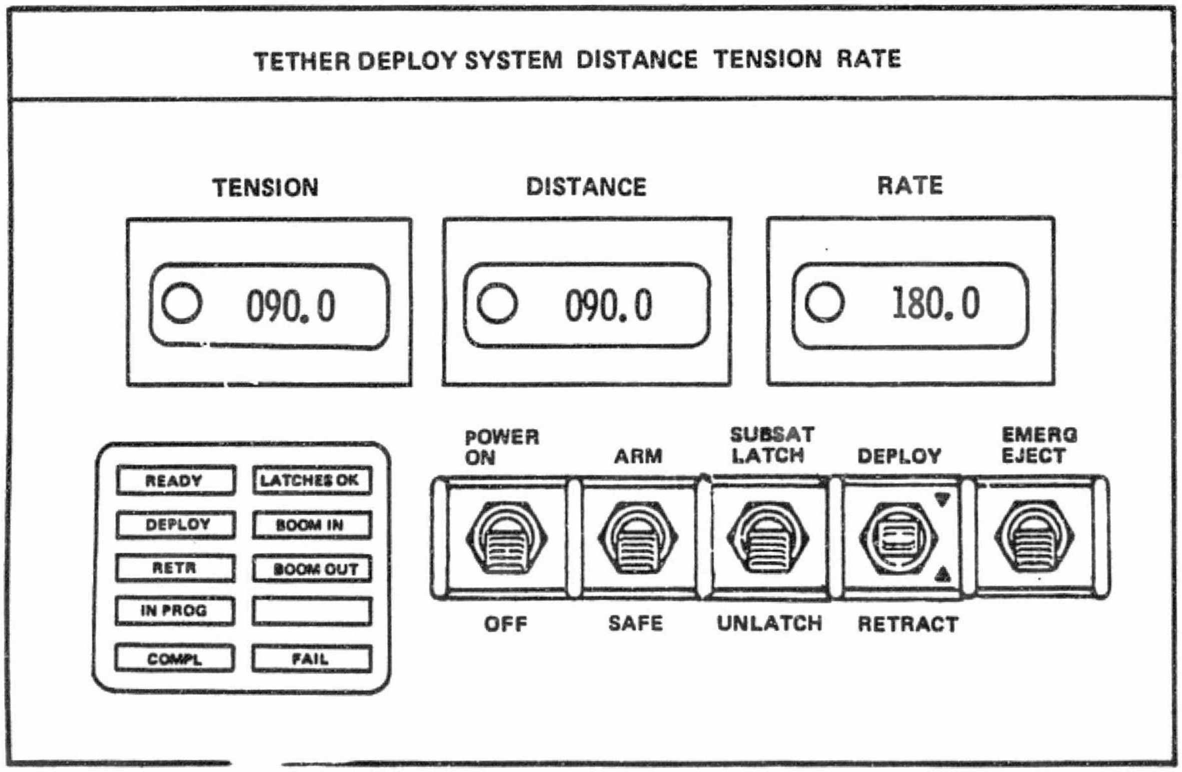


Figure 5-7. Control and display panel.

5.1.6 Control

5.1.6.1 Introduction

5.1.6.1.1 Tether Control System. The tether control system consists of a reel mechanism (for controlling the tension in the tether), the tether, the satellite, a control computer, and a control and display panel. Additional control system components contained within the reel mechanism consist of a servo amplifier, a reel drive motor, a tensiometer at the reel, a tether length and rate sensor, a tensiometer at the boom tip, and a motor at the boom tip. The whole system must work together, but for analysis purposes, the system can be thought as having three control loops. A block diagram of the entire system is shown in Figure 5-8.

The innermost control loops consist of the components required to provide the tether tension as commanded by the control computer. A block diagram of these control loops is shown in Figure 5-9. The control loop consists of a reel drive torque motor with a tensiometer mounted at the reel. Another torque motor and tensiometer are located at the tip of the boom. The reel drive motor functions as a brake during deployment of the tether system and as a motor by rewinding the tether on the reel during retrieval of the tether. When the tension is low in the tether deployed from the boom tip to the satellite, the motor at the tip of the boom serves to keep the tether taut through the reel mechanisms by pulling in opposition to the reel motor at the base of the boom. A very low torque is required of the boom tip motor — only enough to maintain about 0.1 N tension. The boom tip motor and reel drive motor are controlled in a closed-loop fashion using information derived from the two tensiometers.

An alternate implementation of the innermost control loops is shown in Figure 5-10. With this approach, the boom tip motor is driven in an open-loop fashion so that the tension applied is essentially constant over a wide range of tether speed. This motor could be a small direct current motor driven by a constant current generator. The accuracy need not be great at the higher speeds. However, the constant current generator must be capable of acting as a source during deployment and a load during retrieval of the tether. The tensiometer can be placed at the boom tip or at the reel as long as proper compensation is made to account for the tension supplied by the boom tip motor.

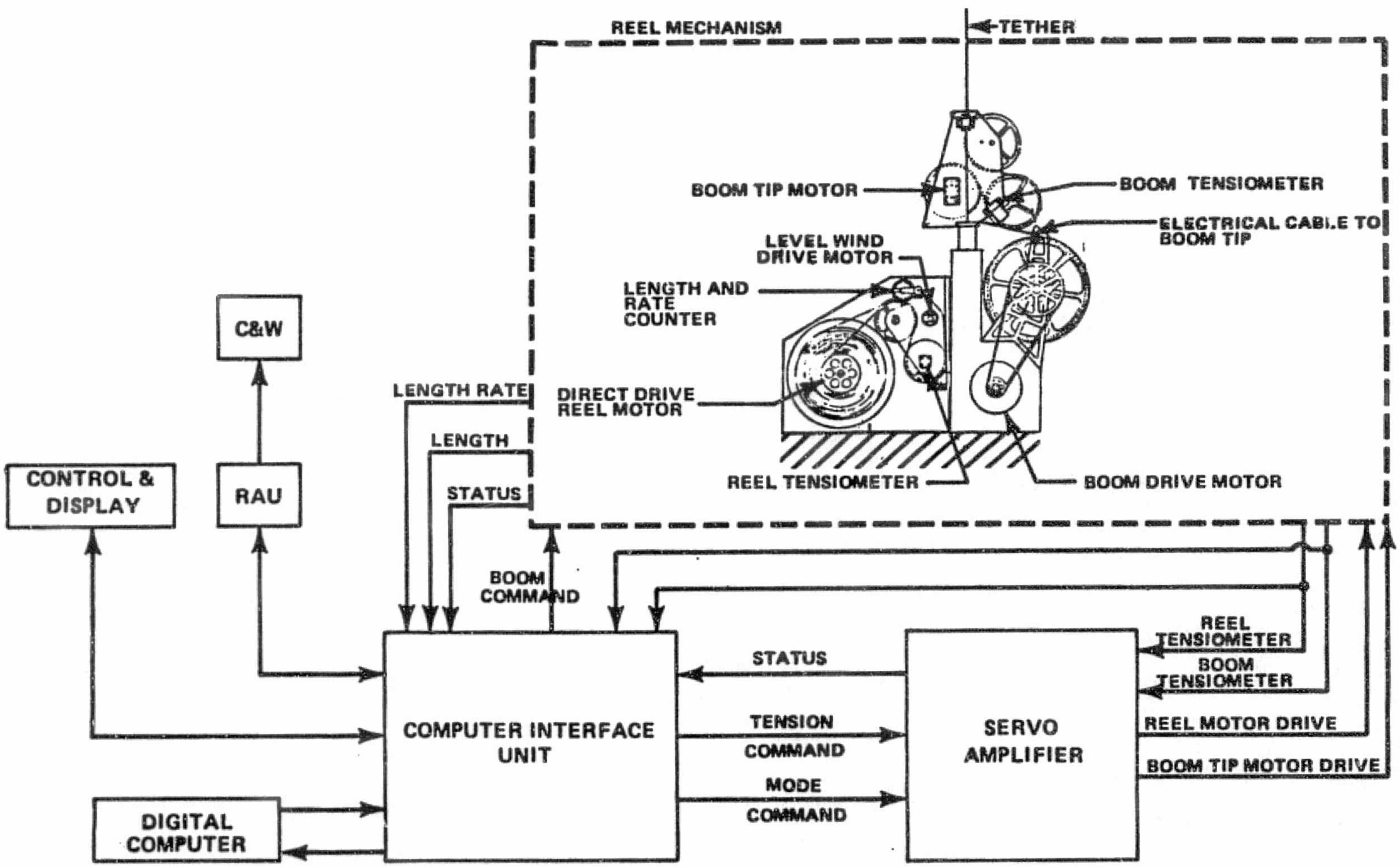
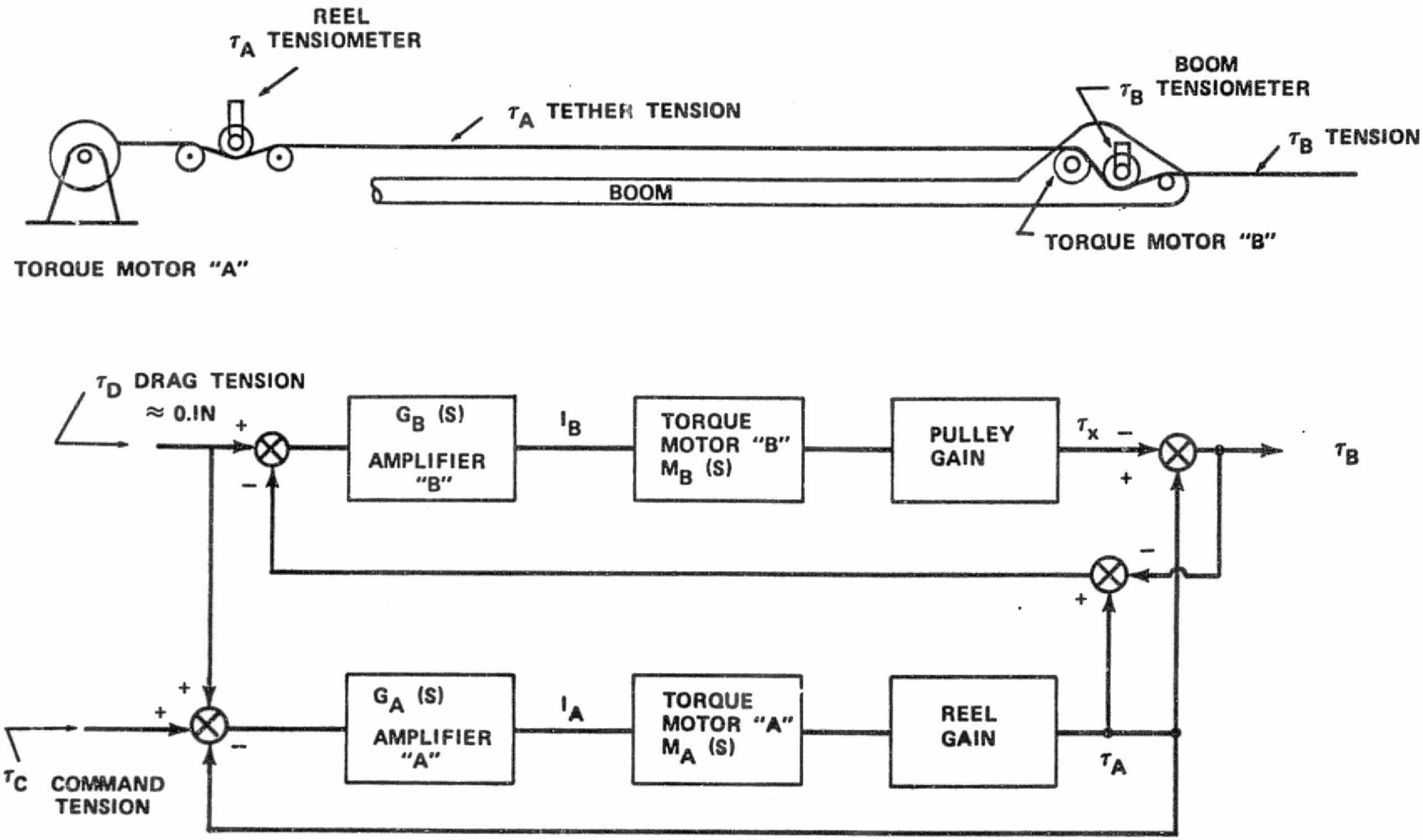


Figure 5-8. Tether control system block diagram.



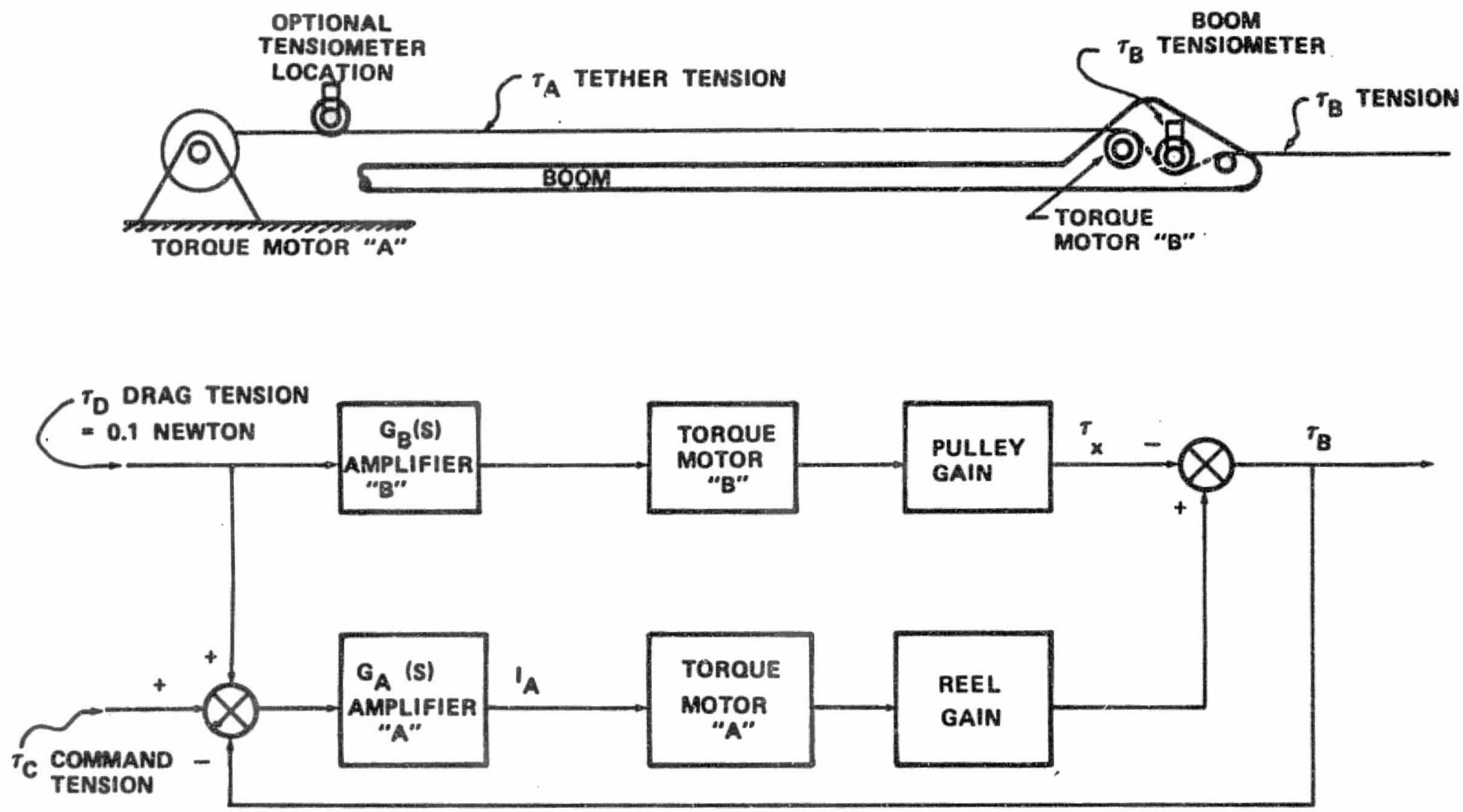
$$\tau_A = M_A(S)\, I_A$$

$$\tau_B = M_A(S)\, I_A - M_B(S)\, I_B$$

$$I_A = G_A(S)\,(\tau_c + \tau_D - \tau_A)$$

$$I_B = G_B(S)\,(\tau_D + \tau_B - \tau_A)$$

Figure 5-9. Control system inner control loop.



$$\tau_A = M_A(S)\ I_A$$

$$\tau_x = \tau_D$$

$$I_A = G_A(S)\ (\tau_D + \tau_c - \tau_B)$$

$$\tau_B = \tau_A - \tau_x$$

Figure 5-10. Alternate configuration of control system inner control loop.

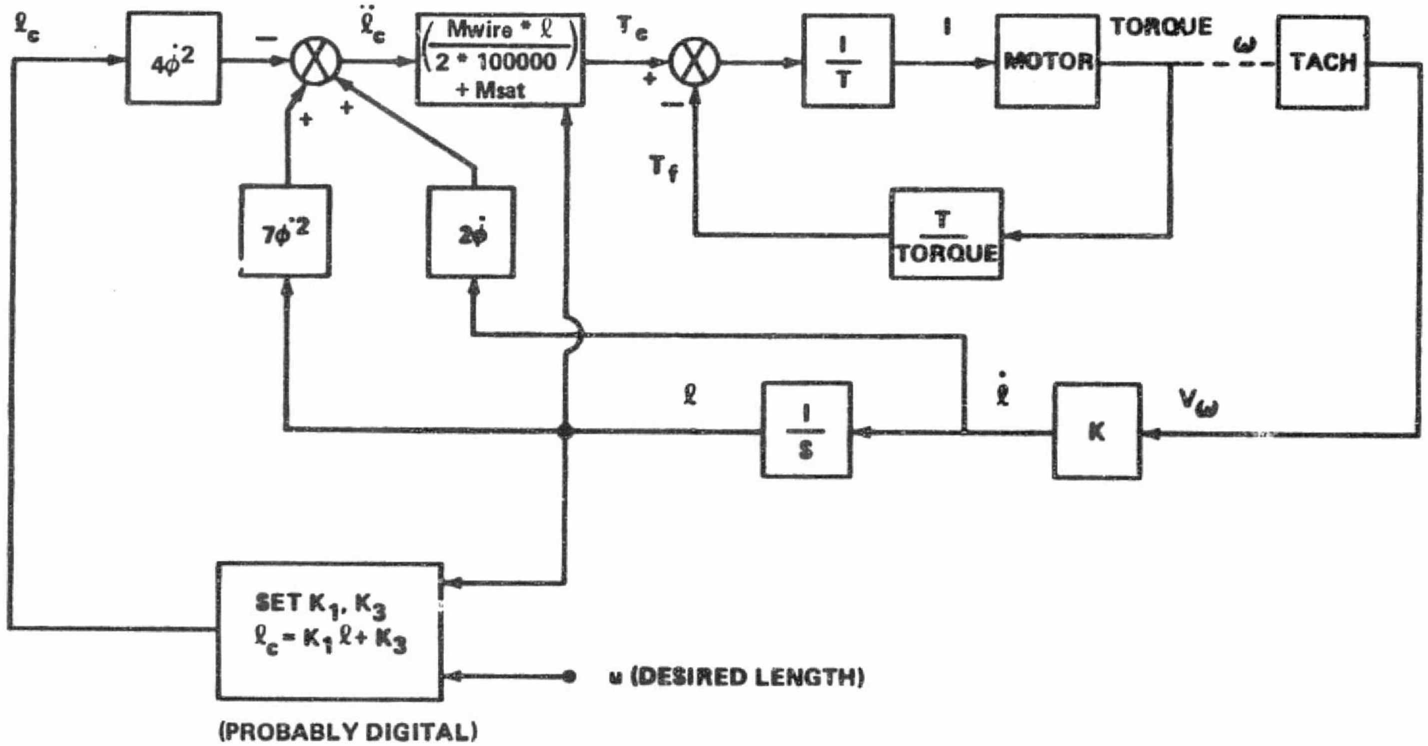
The outermost control loop consists of the components to cause the satellite to deploy, stationkeep, and retrieve according to commands from the control and display panel and the control computer. A block diagram of this control loop is shown in Figure 5-11. A control algorithm contained in the control computer is described in References 6 and 7. The control computer receives measurements from a tether length meter, a tether deployment and retrieval rate sensor, and the tensiometer and computes the tension required to cause the satellite to follow the desired trajectory. The tension command, so calculated, is sent to the inner control loop to cause the reel motor to apply the correct tension to the tether.

5.1.6.1.2 Satellite Attitude Control System. Aerodynamic control of the satellite attitude is addressed in Reference 7. The conclusion is that aerodynamic control is not feasible because of insufficient control authority, but aerodynamic stability is desired to minimize disturbance torques. Further work in attitude control was not pursued to allow study resources to be focused on tether system analysis. The attitude control requirements depend on the particular application; however, several concepts appear to be feasible including reaction control systems, momentum exchange systems, and possibly a single axis momentum exchange system using gyrocompassing. The subject of satellite attitude control will be addressed in a future study. Preliminary consideration of an engineering data package for sensing attitude, rate, and acceleration is provided in Section 5.1.4.

5.1.6.2 Tether Control System Operation

Deployment of a tethered satellite is initiated by placing the satellite a short distance from the Orbiter with an extendable and retractable boom. Alternatively, the satellite can be given an initial velocity along local vertical with a catapult or other launcher. Either method, or a combination of the two, causes the satellite trajectory to move ahead of and down from the Orbiter for deployment down towards the Earth. If the satellite is desired to be deployed upward, away from the Earth, the initial velocity or position should be in an upward direction and the resulting trajectory moves upward and behind the Orbiter.

As the satellite moves away from the Orbiter, the tether is unwound from the reel. The reel drive motor operates as a brake applied against the gravity gradient force which is acting on the satellite mass. This braking action provides tension on the tether which causes the satellite to move



DEFINITION OF SYMBOLS:

ℓ_c	=	COMMANDED TETHER LENGTH
ℓ	=	TETHER LENGTH
$\dot{\ell}$	=	RATE OF TETHER DEPLOYMENT
$\ddot{\ell}_c$	=	COMPUTED TETHER ACCELERATION
T_c	=	COMPUTED TETHER TENSION COMMAND
T_f	=	SENSED TETHER TENSION
$\dot{\phi}$	=	ORBITAL RATE
Mwire	=	MASS OF TETHER
Msat	=	MASS OF SATELLITE
K_1, K_3	=	CONTROL LAW GAINS
I/S	=	LAPLACE OPERATOR INDICATING INTEGRATION
K	=	CONVERSION FACTOR FROM REEL ANGULAR RATE TO TETHER LINEAR RATE
I/T	=	SERVOAMPLIFIER GAIN
I	=	MOTOR CURRENT
ω	=	REEL ANGULAR RATE
V_ω	=	SENSED ANGULAR RATE OF REEL
u	=	DESIRED TETHER LENGTH

Figure 5-11. Block diagram of the tether control loop.

further downward (or upward as the case may be). This action can be explained by observing that the tether tension can be thought of as having a vertical and a horizontal component. The horizontal component is of the proper direction to further deorbit the satellite for downward deployment, thus causing the satellite trajectory to move further downward (or upward).

The tension required to accomplish the deployment is computed by the control computer which, in turn, commands the reel torque motor. The tether tension, deployment rate, and length sensors provide the information which the control computer requires to compute the required tether tension. For the deployment case, the tension which is commanded is slightly lower than the gravity gradient force which would be applied to the satellite mass if the satellite were in equilibrium at its present distance. A tension greater than this amount will cause retrieval (to be discussed later). The control ιw has a deployment rate limit which limits the speed of the reel and motor to a safe value.

Stabilization of the satellite at the desired deployment distance is accomplished by changing the gains in the control law within the control computer. Any residual swinging motion is damped by the control law acting through the motor and reel. The damping action can be explained by considering the tether as a spring. As the satellite swings, variation in the gravity gradient force causes the spring to stretch and contract. These oscillations in the spring stretching motion are damped by any structural or viscous damping which might be present. The control law functions to make the tether appear to have any spring constant and damping desired by controlling the tension in the tether through the action of the motor and reel. In summary, the swinging motion is converted to stretching motion and then damped by the reel control system. The gains in the control law have been optimized to enhance this damping action which is active during deploymen and retrieval as well as the stabilization phase.

Retrieval of the tethered satellite is similar to the deployment except that the control law commands a tension greater than the equilibrium tension at the present satellite distance. This causes the motor and reel to retrieve the tether. The retrieval trajectory lies slightly behind local vertical for the downward deployed system or slightly ahead of local vertical for the upward deployed system. Another difference between retrieval and deployment is that deviations from the desired trajectory are more difficult to damp during retrieval; therefore, more time is required to effect the retrieval. The terminal phase of the retrieval occurs very slowly, and capture of the satellite is accomplished using the deployment boom fitted with a docking adapter. In this terminal phase, the satellite is slowly drawn into the docking adapter and then the boom is retracted, drawing the satellite back into the Orbiter.

5.1.6.3 Control System Requirements

5.1.6.3.1 Tether Tension. The tension in the tether consists of a horizontal component due to atmospheric drag and a vertical component due to gravity gradient. Figure 5-12 presents a plot of the total atmospheric drag along tether as a function of the length and diameter. The drag on the satellite should be added to the drag on the tether, if known. For a very short tether, the drag on the satellite predominates. Figure 5-13 shows a plot of the drag on a 1 mm diameter tether with various sizes of spherical satellites versus altitude assuming the Orbiter altitude is 200 km.

The gravity gradient force is shown in Figure 5-14 and is a function of the effective mass of the satellite and tether and the distance from the Orbiter. Figure 5-15 shows the same information for the short lengths. The effective mass is calculated by the following formula:

$$\text{Effective Mass} = \text{Satellite Mass} + \frac{\text{Tether Mass}}{2} .$$

The tension in the tether must be calculated for the various phases of operation to determine the tether load requirements. The static, or steady state tension, can be found by vectorially adding the drag force exerted by the atmosphere to the gravity gradient force acting on the combined effective mass of the satellite and tether. The peak tension requirements must be found from simulations where system dynamics are included. Reference 7 shows that a peak tension of 125 N is required for retrieval of a tether system with a mass configuration similar to the operational tether system compared to approximately 90 N in the steady state. The ratio of peak to steady state tension can be used to approximate peak tensions for tether systems with different mass configurations once the static tension has been found.

For example, consider the operational system with a 1.44 m diameter satellite with a mass of 175 kg and a tether length of 80 km with a deployed mass of 90 kg and 1 mm diameter. The effective mass is 220 kg; therefore, the gravity gradient force from Figure 5-12 is approximately 80 N.

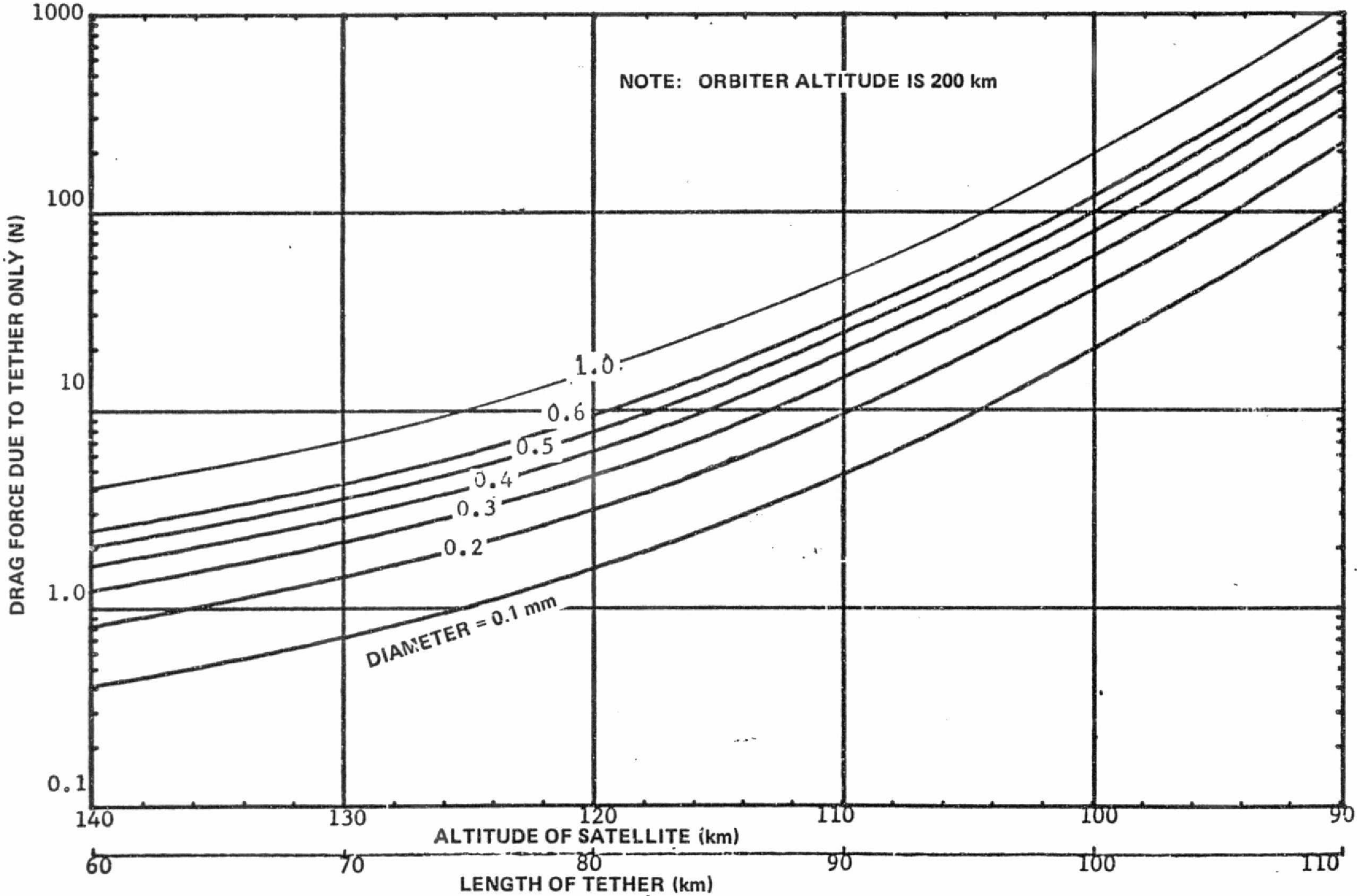


Figure 5-12. Atmospheric drag on the tether as a function of tether length.

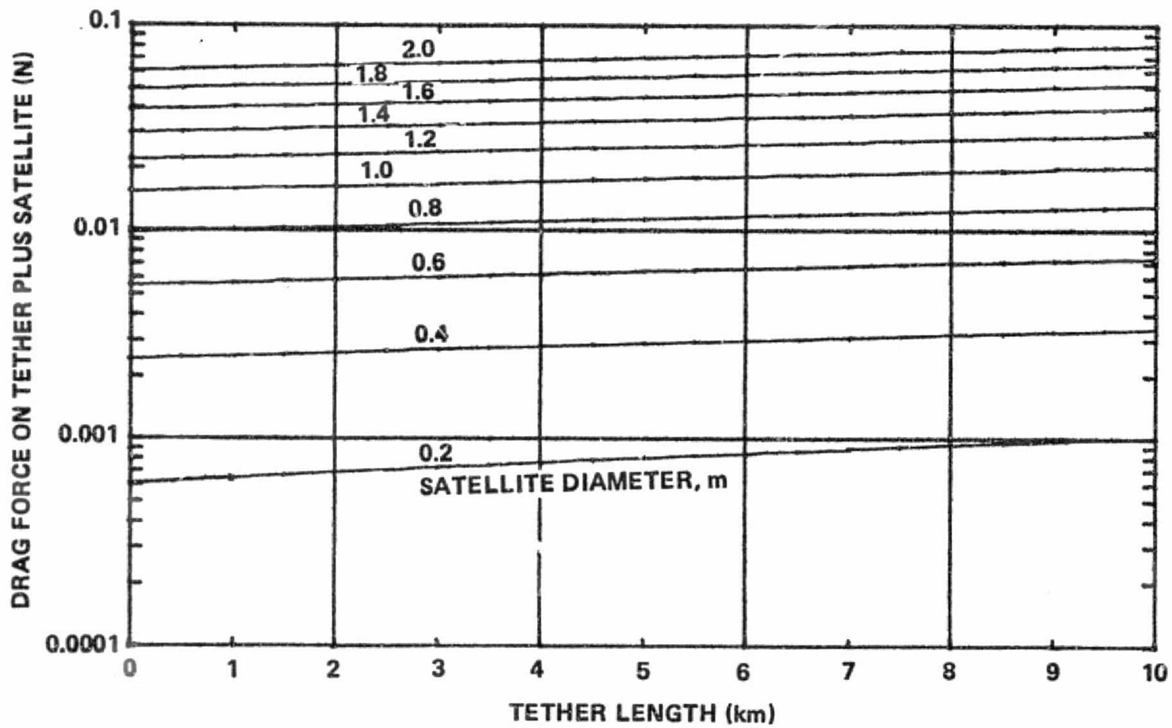


Figure 5-13. Aerodynamic drag on tether system as a function of length and satellite diameter (tether diameter is 1 mm, Orbiter is at 200 km, deployment is toward the Earth).

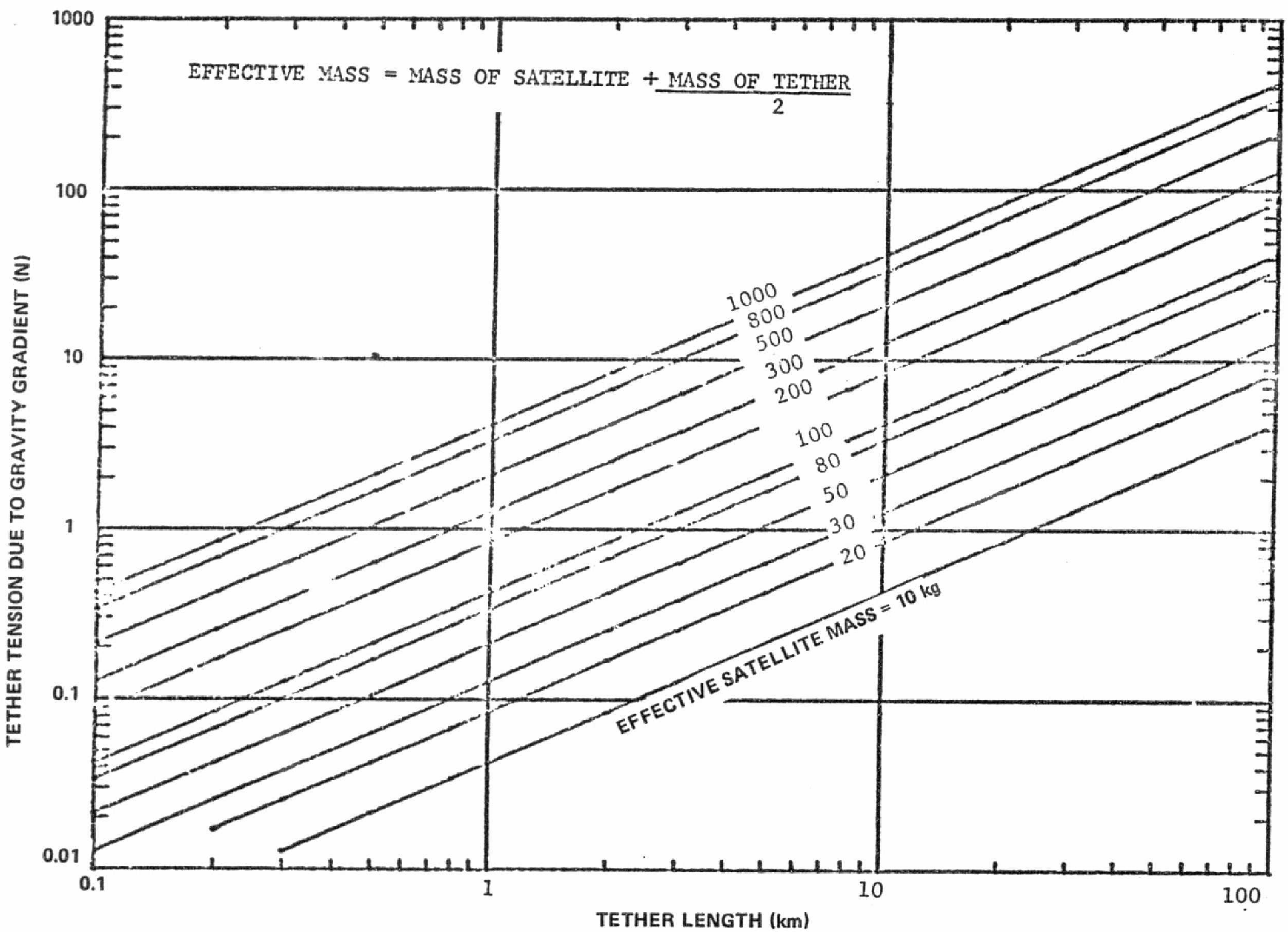


Figure 5-14. Tether tension due to gravity gradient versus tether length and effective satellite mass.

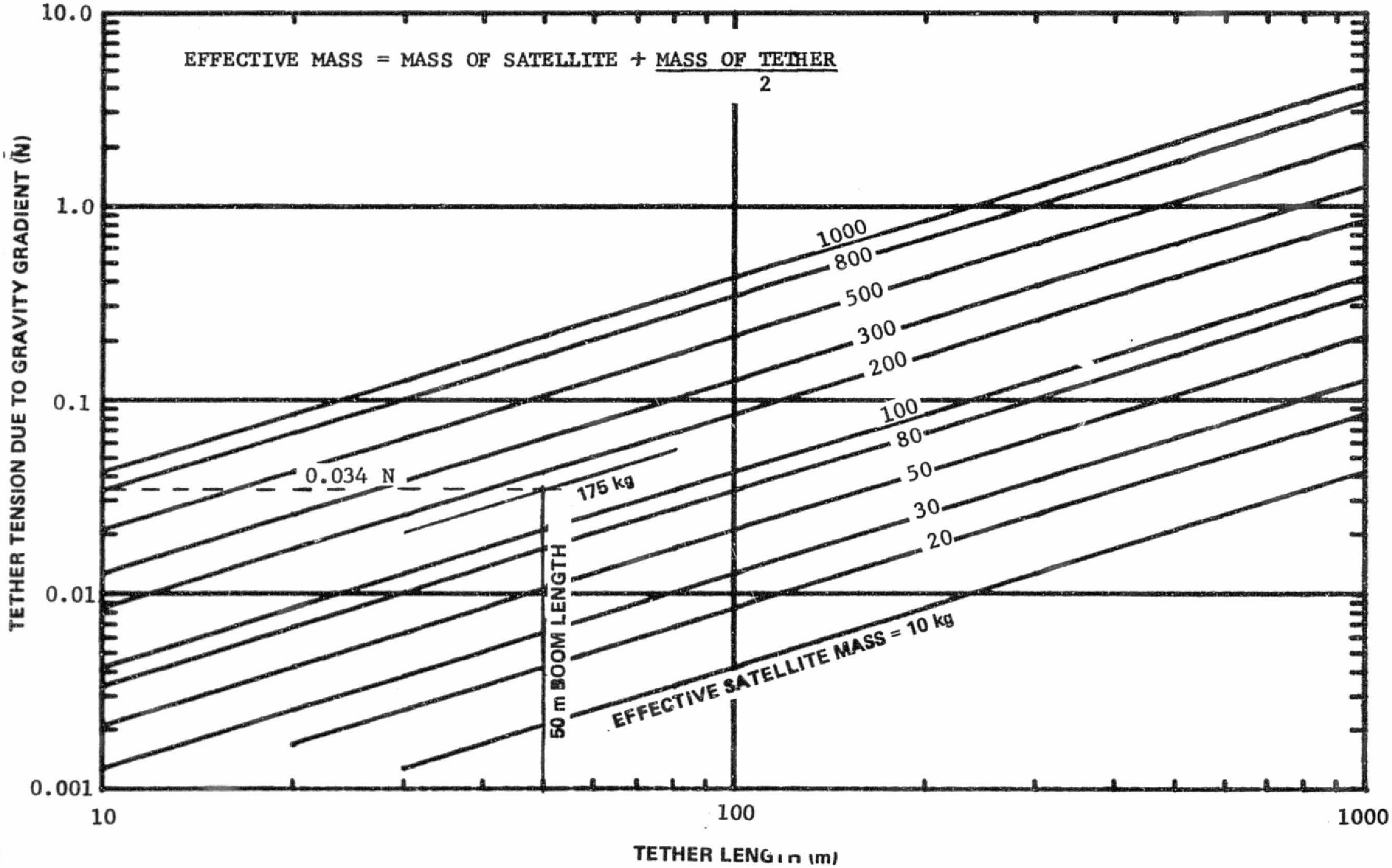


Figure 5-15. Tether tension due to gravity gradient versus tether length and effective satellite mass (short tether lengths).

The drag on the satellite can be approximated by adding a length of tether which has the same area as the satellite. The area of the satellite is 1.6 m^2 which is equivalent to 1.6 km of 0.001 m diameter tether. From Figure 5-12, the total drag for 81.6 km tether is approximately 17 N. The vector sum of 17 and 80 is 82 N for the static or steady state tension. Multiplying this static tension by the previously determined peak to static tension ratio gives a peak tension of 114 N for the operational system.

The peak tension for a 1 mm stainless steel tether is estimated at 250 N if deployed under these same conditions.

5.1.6.3.2 Relative Acceleration Due to Aerodynamic Drag. To evaluate the motion of the tether system relative to the Orbiter, the relative acceleration, which involves drag and mass of the tether system and the Orbiter, must be determined. Figure 5-16 presents a plot of the acceleration of the tether system with respect to the Orbiter due to aerodynamic drag. Two satellite designs are shown. The 175 kg mass satellite has a diameter of 1.44 m and represents the satellites designed for the operational and limited operational systems. The 113 kg mass satellite has a diameter of 0.609 m and represents the satellite designed for the concept definition system. The Orbiter attitude affects the relative drag. The range of drag between the Orbiter Y-axis perpendicular to the orbital plane (Y-POP, Z-LV) and X-axis perpendicular to the orbital plane (X-POP, Z-LV) attitudes can be obtained by biasing the Orbiter attitude. The relative drag is of importance in determining the equilibrium position of the satellite, as will be shown in a latter section.

5.1.6.3.3 Equilibrium Position of the Tethered Satellite. For the operational system composed of a 1 mm diameter aramid fiber tether and a 175 kg 1.44 m diameter spherical satellite, the relative drag force causes the equilibrium position of the satellite to trail the Orbiter. The same can be true of the concept definition system flying a 113 kg, 0.609 m diameter satellite. Figures 5-17 and 5-18 present plots of the angle the satellite equilibrium position makes with local vertical for the two systems of various lengths and for the Orbiter in either the Y-POP, Z-LV or X-POP, Z-LV attitudes at the 200 km altitude.

For short tether lengths, the ballistic coefficient of the satellite can cause the equilibrium position to deviate from local vertical. Large deviations could result in problems for deployment and retrieval of certain classes of satellites. To minimize the problem, the ballistic coefficient of the satellite

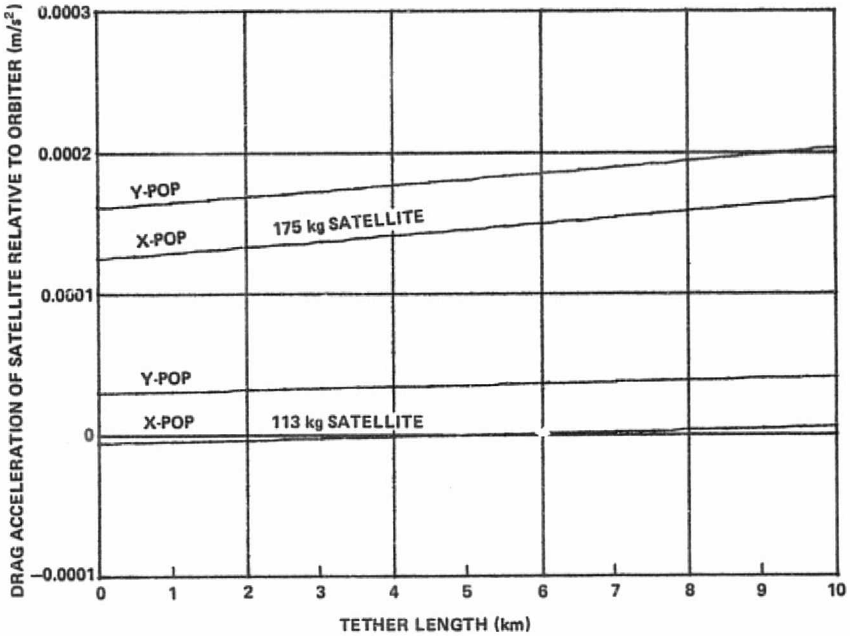


Figure 5-16. Drag acceleration of satellite relative to Orbiter as a function of tether length (Orbiter is at 200 km with Z-axis along local vertical and Y-axis perpendicular to orbital plane (Y-POP) or X-axis perpendicular to orbital plane (X-POP) as noted. The 175 kg satellite has a diameter of 1.44 m and the 113 kg satellite has a diameter of 0.609 m.).

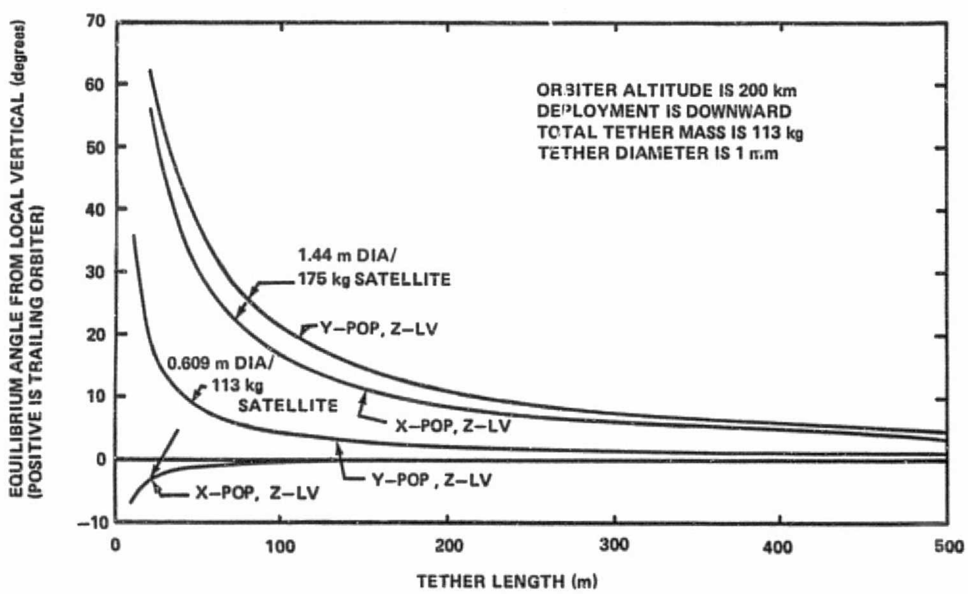


Figure 5-17. Plot of equilibrium angle measured from local vertical as a function of tether length, Orbiter attitude, and satellite design.

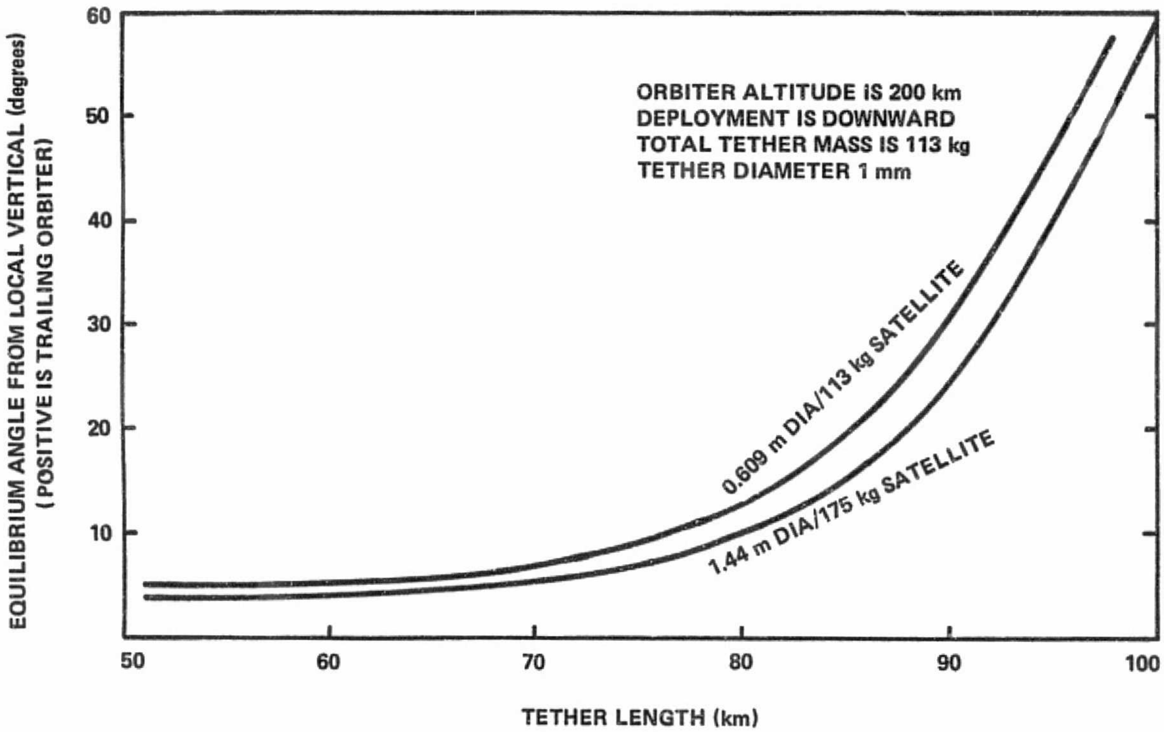


Figure 5-18. Plot of equilibrium angle measured from local vertical as a function of tether length and satellite design (Orbiter attitude is not significant).

should match that of the Orbiter close enough that the equilibrium position is within a nominal angle, such as 10°, from local vertical at the boom or remote manipulator system deployment distance. It must be emphasized that only the very near Orbiter distances might be of concern (200 m and shorter). In practice, the dynamics of a satellite approaching the Orbiter should predominate over the static equilibrium consideration as demonstrated by Kulla [8].

5.1.6.3.4 Reel Mechanism Design Criteria. Figures 5-14 and 5-15 illustrate the dynamic range requirements for the control devices and the tether strength requirements. For example, the tensiometer must be capable of measuring forces ranging from hundredths of a Newton to over a hundred Newtons.

Figures 5-14 and 5-15 can also be used to estimate the horsepower required for the reel motor. The estimate can be found by using the product of the maximum tension and retrieval rates. This estimate might result in over-design because the peak tether tension does not occur at the same time as the peak retrieval rates. A more accurate estimate can be determined by simulation. Figure 5-19 presents a plot of the horsepower required as a function of time for retrieval of the operational system using a tether mass equivalent of 1.0 mm diameter aramid fiber material. Horsepower requirements for other system sizes can be estimated using this figure and multiplying the horsepower by the ratio of their effective masses. For example, the operational system using 1 mm stainless steel tether would have an effective mass of 502 kg versus 232 kg for the system with the aramid fiber tether. The resulting horsepower required for the steel tether system would be approximately 2.1 horsepower compared to 0.94 horsepower for the system using an aramid fiber for the tether.

5.1.6.3.5 Computer Software and Hardware. Computer software must be provided to compute the control algorithm, to service the control and display panel, to detect system failures, to provide for telemetry, and to service the input and output to the control system. A flight type general purpose computer 8K memory should be sufficient to provide these functions. A preliminary estimate of core requirements is 2K. Control speed is relatively slow and should not be a factor in computer sizing.

5.1.6.3.6 Boom Design. The side loads imparted to the deployment boom are important in sizing the boom. The side load was calculated in a simulation of a 175 kg satellite deployment. Figure 5-20 shows the result of this calculation.

5.1.6.3.7 Orbiter Constraints. An Earth pointing attitude is required to operate the tether system. This could be X-POP, Z-LV or Y-POP, Z-LV with the payload bay toward the Earth. Tether system operation longer than 30 h will probably require the Orbiter attitude to be biased from Z-LV to allow the Orbiter thermal radiators to see some deep space. This biased attitude can be accommodated by the tether system by canting the deployment arm such that the tether tension vector passes through the Orbiter center of gravity while the Orbiter is in the biased attitude. This cant angle prevents the tether tension from torquing the vehicle, causing the thrusters to fire unnecessarily. Operation of short tether (less than 10 km) or a long tether for a short time could

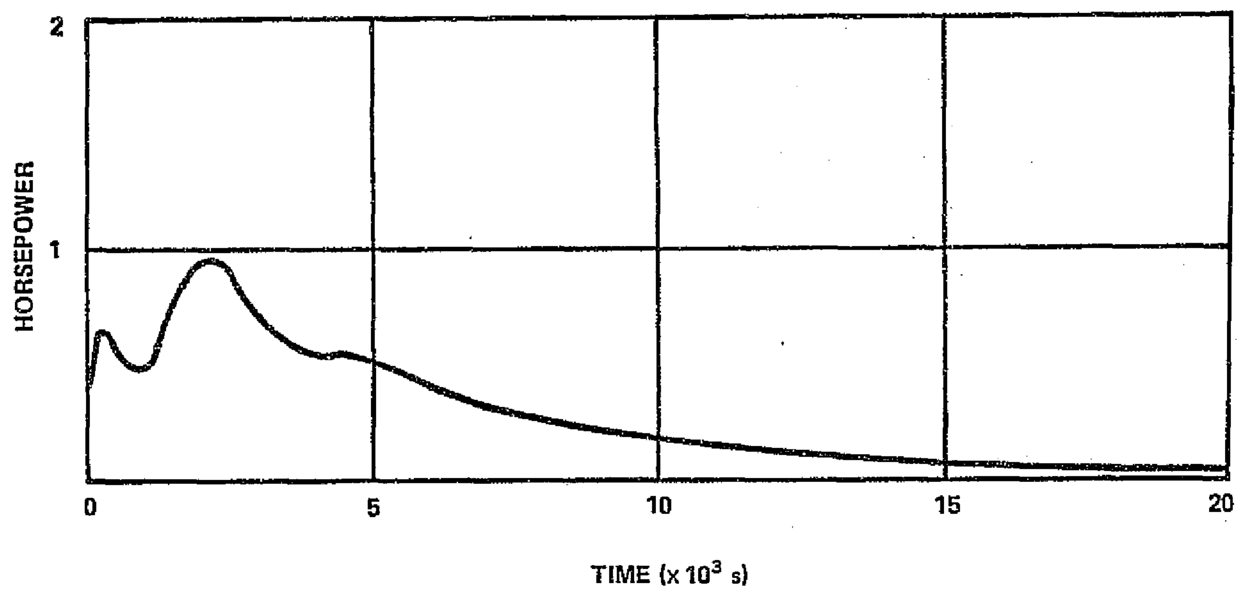


Figure 5-19. Horsepower versus time for retrieval of a 175 kg satellite deployed a distance of 100 km on a 1 mm diameter Aramid tether.

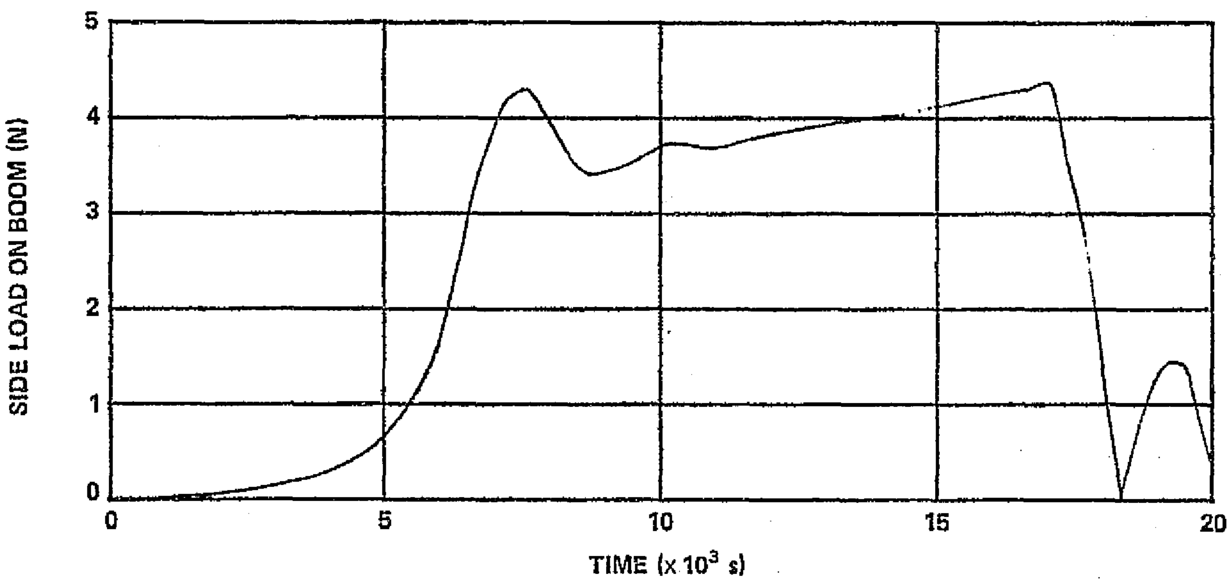


Figure 5-20. Side load on 50 m deployment boom as a function of time for a 175 kg satellite being deployed to a distance of 100 km on a 1 mm diameter Aramid tether.

probably be accommodated without a cant of the boom. If a cant is included, the side load would have to be calculated to determine if the load exceeds the boom capability.

5.1.6.3.8 Orbit Constraints. The tether system can be deployed upward or downward. The operational system deployed downward for a test of a satellite of 120 km altitude requires that Orbiter altitude be 200 to 210 km in a circular orbit because the tether maximum length is 100 km.

The inclination of the orbit is not critical from a tether system viewpoint but will probably be determined by other mission requirements.

5.1.7 Tracking and Data Acquisition

The tracking and data acquisition requirement will be held to a minimum during the tether demonstration flights. During operational flights, tracking and data acquisition will be consistent with the system's user requirements. Since there will be no direct communication link between the subsatellite and the ground, the Orbiter will be required to provide communications and data storage. Figure 5-21 illustrates a typical day's tracking opportunities for four ground stations. Table 5-3 presents the tracking summary for individual stations and their union. Tracking data presented are typical for the 200 km tether design altitude. Tracking data typical for the 81-2 mission is presented in Section 5.2.7. Figure 5-22 presents the Orbiter ground tracks for 1 day at 200 km altitude and 38° inclination.

5.1.7.1 Orbiter Navigation Accuracy

Representative navigation accuracies achievable by the Orbiter using the Spaceflight Tracking and Data Network (STDN) or the Tracking and Data Relay Satellite System (TDRSS) are given in Table 5-4. The actual accuracies achievable for a mission will depend on orbit altitude, inclination, and other mission peculiar parameters. Data are given for the accuracy at the end of a tracking period and propagated one orbital revolution later. The TDRSS accuracy is based on using a single TDRSS; if two TDRSS's are available, the navigation accuracy may be improved.

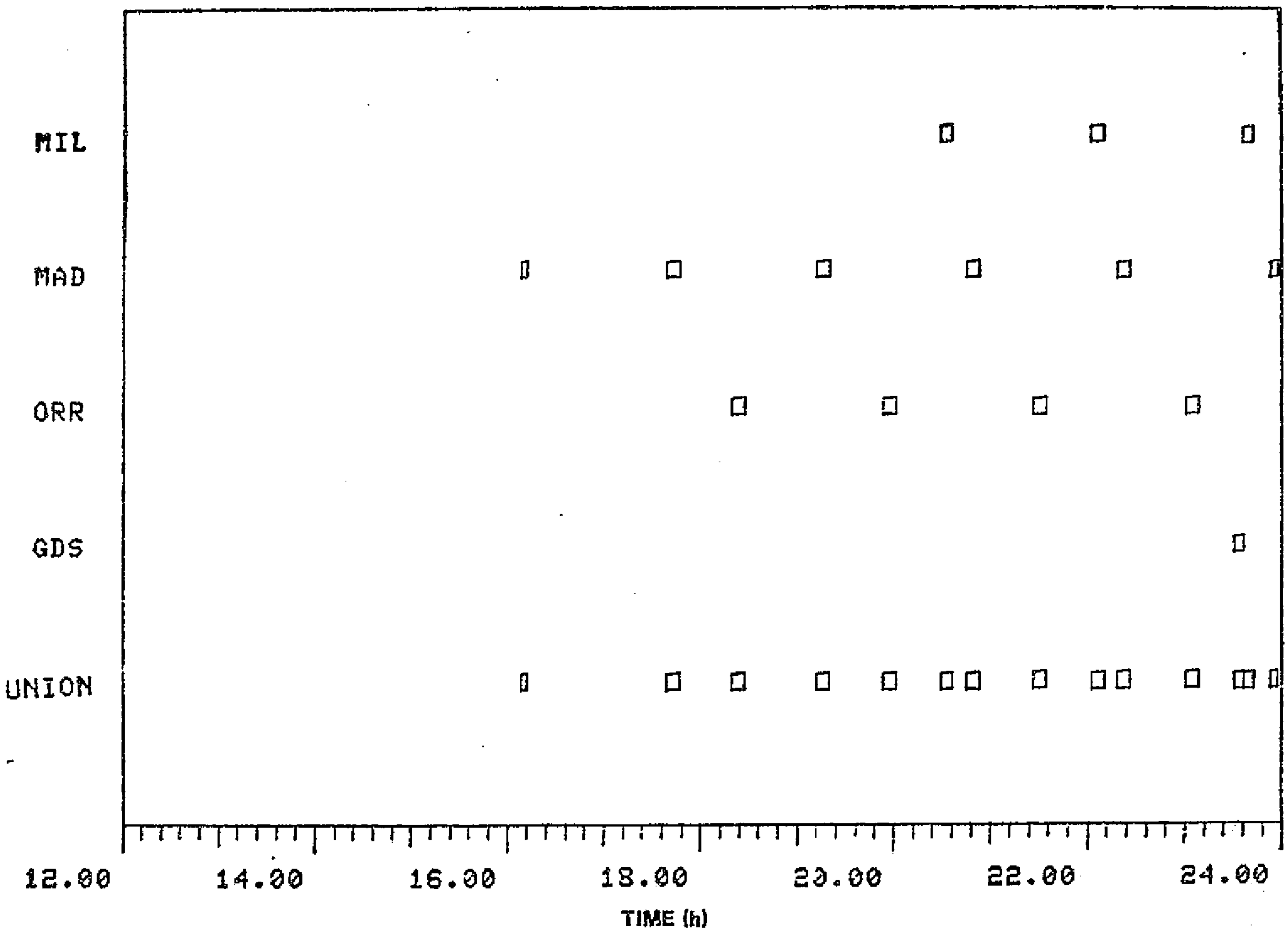


Figure 5-21. Typical tracking opportunities for 200 km orbit.

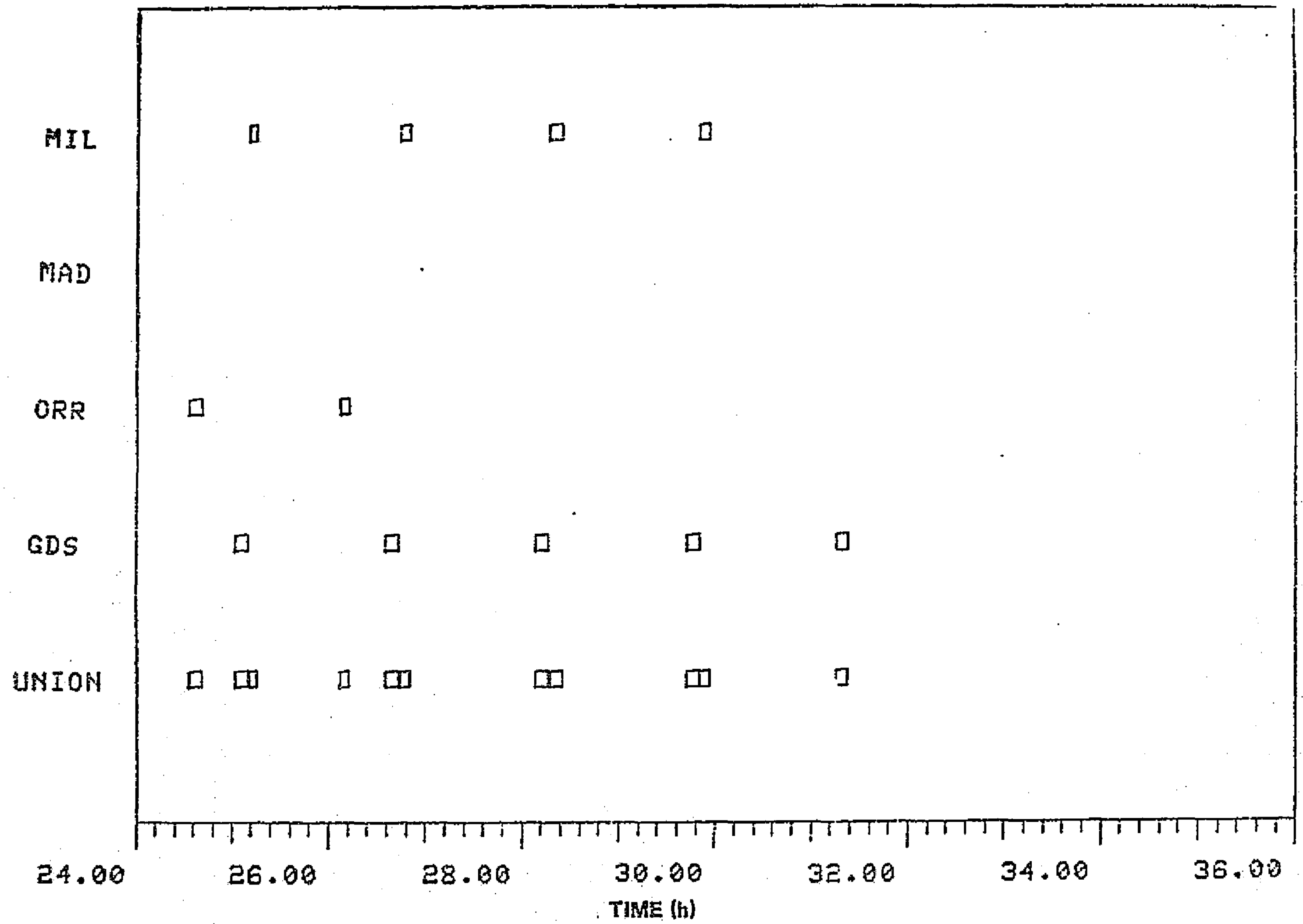


Figure 5-21. (Concluded).

TABLE 5-3. TRACKING SUMMARY FOR 200 km ORBIT

INDIVIDUAL TRACKING STATION SUMMARY TABLE AND TRACKING UNION TABLES

TARGET	AVERAGE TIME OVER TARGET/DAY MINUTES			AVERAGE NUMBER OF PASSES/DAY			AVERAGE TIME/PASS MINUTES		
	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0
MIL	7.65	19.04	19.04	1.43	4.43	4.86	5.36	4.30	3.92
BDA	13.60	19.90	19.90	2.57	4.00	4.71	5.36	4.98	4.22
MAD	8.40	13.07	13.33	1.57	2.71	3.71	5.35	4.81	3.59
ORR	15.20	19.27	19.27	2.71	3.71	4.14	5.60	5.19	4.65
GDS	14.13	18.07	18.07	2.57	3.57	4.14	5.50	5.06	4.36
TAN	7.71	10.74	10.85	1.43	2.29	4.29	5.40	4.70	2.53
FRB	.00	.00	.00	.00	.00	.00	.00	.00	.00
ROS	15.67	18.34	18.61	2.86	3.57	4.43	5.48	5.14	4.20
RS1	540.24	540.24	540.24	11.57	11.57	11.57	46.69	46.69	46.69
RS2	544.80	544.80	544.80	11.57	11.57	11.57	47.08	47.08	47.08

UNION OF TRACKING TIME FOR TDRS

AVERAGE TRACK TIME/ORBIT (HRS) = 1.31290
AVERAGE TRACK TIME/DAY (HRS) = 15.19210
TOTAL TRACK TIME/MISSION (HRS) = 106.34472
AVERAGE GAP TIME (HRS) = .17967
MAXIMUM GAP TIME (HRS) = .77167

UNION OF TRACKING TIME FOR GROUND STATIONS

AVERAGE TRACK TIME/ORBIT (HRS) = .14354
AVERAGE TRACK TIME/DAY (HRS) = 1.66091
TOTAL TRACK TIME/MISSION (HRS) = 11.62639
AVERAGE GAP TIME (HRS) = [illegible]
MAXIMUM GAP TIME (HRS) = [illegible]

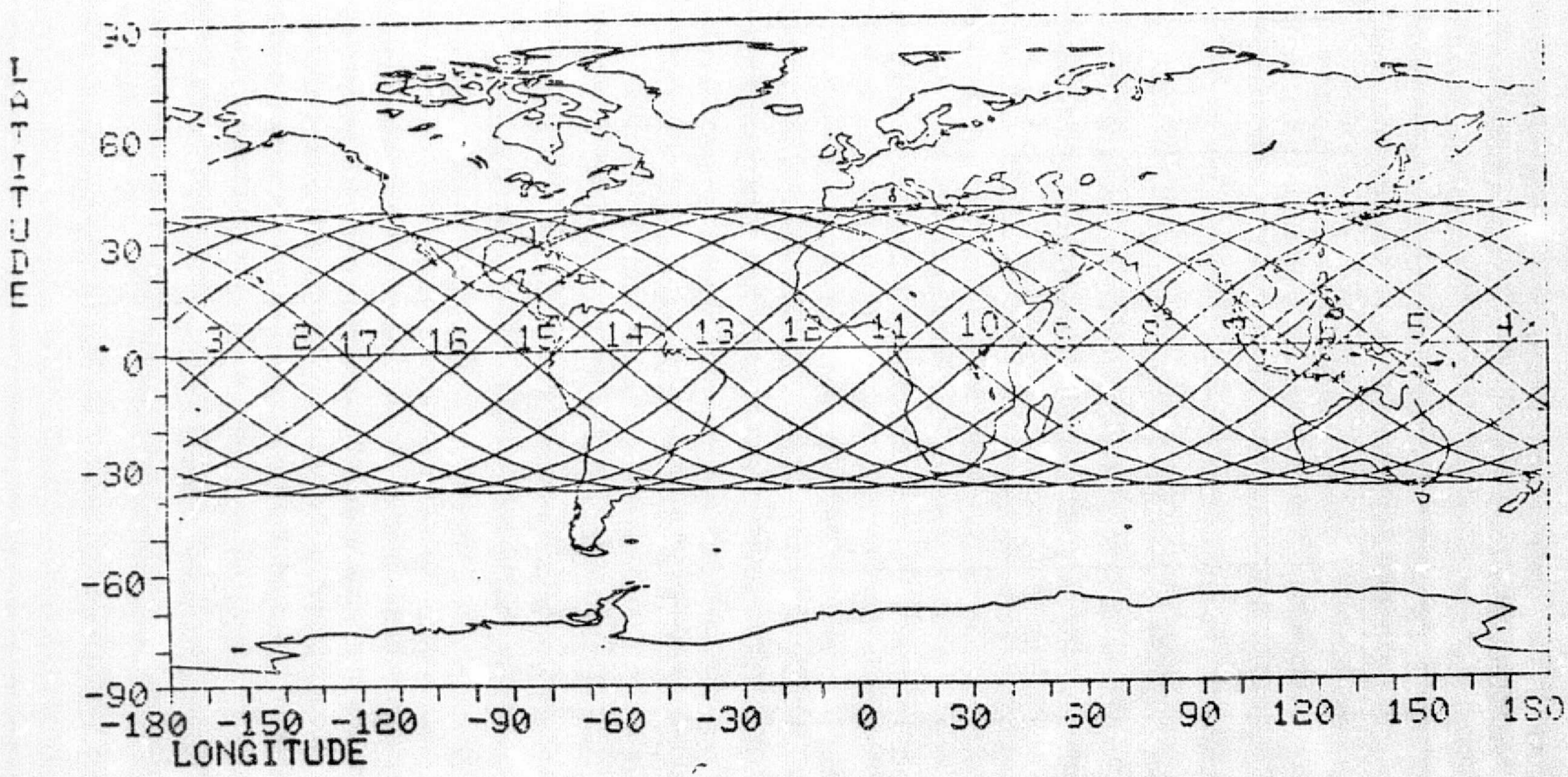


Figure 5-22. Typical day ground tracks for 200 km orbit.

TABLE 5-4. EXPECTED ON-ORBIT NAVIGATION ACCURACIES (3σ) FOR 185.2 km ALTITUDE

Navigation System	Position Accuracy (m)			Velocity Accuracy (m/s)		
	Altitude	Down Track	Cross Track	Altitude	Down Track	Cross Track
STDN						
End of Tracking	130	110	130	1.2	0.15	0.6
Propagated One Revolution	150	260	130	1.3	0.15	0.6
TDRSS						
End of Tracking	90	430	460	0.5	0.11	0.15
Propagated One Revolution	90	610	460	0.7	0.1	0.15

5.1.7.2 Orbiter Pointing Accuracy

The Orbiter has the capability of pointing any vector defined in the Orbiter navigation base axis system at any desired inertial, local vertical, or Earth fixed target to within a ±0.5° (3σ) angle. The duration of continuous pointing is dependent upon thermal constraints and inertial measuring unit drift.

For payloads mounted in the Orbiter bay, there is an orientation alignment uncertainty of approximately 2° between the Orbiter navigation base and the payload. To compensate for this, the Orbiter will accept attitude information from a payload supplied sensor and will point a vector defined in the sensor reference axis system to within ±0.5°. The pointing duration will be dependent on sensor drift characteristics.

Stability for all pointing is ±0.1°/axis. When using the vernier RCS, the stability rate is ±0.01°/axis and when using the large RCS thrusters, the rate is ±0.1°/s/axis.

For local vertical and Earth target pointing, additional errors are accrued because of the Orbiter navigation accuracy error. The contribution of these errors is shown in Table 5-5.

TABLE 5-5. POINTING ERROR DUE TO NAVIGATION ERROR

	Orbital Altitude		
	185.2 km (100 n. mi.) (degree)	370.4 km (200 n. mi.) (degree)	555.6 km (300 n. mi.) (degree)
Local Vertical			
STDN	0.16	0.16	0.16
TDRSS	0.16	0.16	0.16
Earth Target			
Looking Vertical			
STDN	0.18	0.16	0.16
TDRSS	0.28	0.20	0.18

5.1.8 Tether

An assessment of candidate tether materials was conducted including metallics listed in Table 5-6 and nonmetallics listed in Table 5-7. Additional comparative data are presented in Figures 5-23 and 5-24. The materials were evaluated based on the requirements for the operational tether system and used for applications such as magnetometer and gravity gradient platforms deployed at an altitude of 120 km. The selection of materials for further evaluation was based primarily on the following:

a. Capability to survive in a micrometeorite environment

b. Operate at temperatures of 150° to 200°C

TABLE 5-6. PROPERTIES OF METALLIC TETHERS

Material		Mechanical and Physical Properties						
		Ultimate Tensile, F_{tu} $\times 10^9$ N/m^2	(k lb/in.^{2})	Density $\times 10^3$ kg/m^3	(lb/in.^{3})	Elongation (%)	Thermal Expansion m/m/°C	(in./in./°F)
Stainless Steel	302	1.28	(185)	8.03	(0.29)	6	4.7×10^{-6}	(8.5×10^{-6})
	17-7 PH High Temp.	1.28	(185)	8.03	(0.29)	8	4.6×10^{-6}	(8.2×10^{-6})
	430 CRESS	1.24	(180)	8.03	(0.29)	8	4.6×10^{-6}	(8.3×10^{-6})
Carbon Steel	Music Spring	2.76	(400)	7.75	(0.28)	7	4.4×10^{-6}	(8.0×10^{-6})
Aluminum	7075	0.48	(70)	2.80	(0.101)	7	7.2×10^{-6}	(12.9×10^{-6})
Copper Alloy	Beryllium Copper	1.10	(160)	8.22	(0.297)		5.1×10^{-6}	(9.2×10^{-6})

TABLE 5-7. PROPERTIES OF NONMETALLIC TETHERS

Fiber	Physical Properties						
	Ultimate Tensile, F_{ut} $\times 10^6$ N/m^2	(k lb/in.^{2})	Density $\times 10^3$ kg/m^3	(lb/in.^{3})	Elongation (%)	Thermal Expansion m/m/°C	(in./in./°F)
Polyester	731–848	(106–123)	1.13	(0.041)	10	2.17×10^{-5}	(3.9×10^{-5})
Nylon	751–862	(109–125)	1.38	(0.050)	16	2.67×10^{-5}	(4.8×10^{-5})
Silk	310–572	(45–83)	1.30	(0.047)	18		
Teflon	324	(47)	2.08	(0.075)	12	3.06×10^{-5}	(5.5)
Aramid	2340	(340)	1.47	(0.053)	1.8		
Fiberglass	2760	(400)	2.55	(0.092)	2.4		

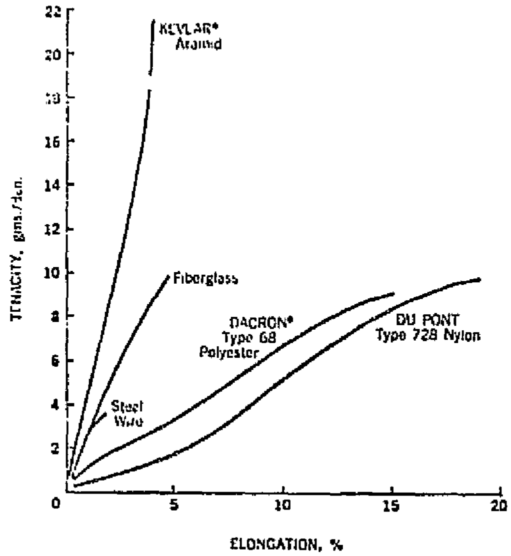


Figure 5-23. Stress-strain curves.

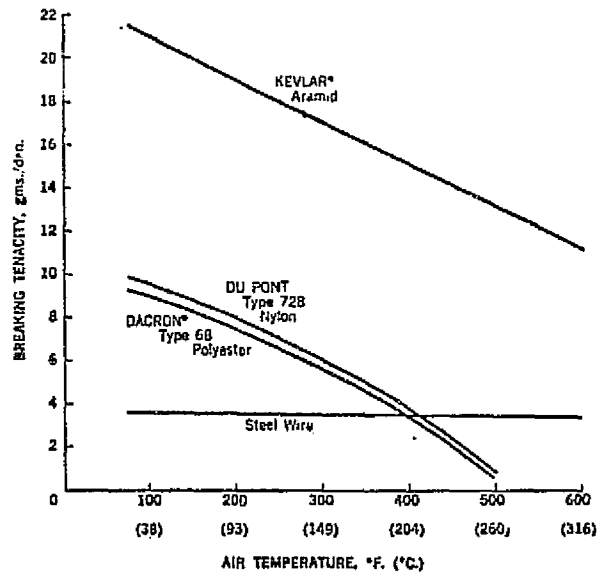


Figure 5-24. Breaking tenacity in air at elevated temperatures.

c. Load carrying capability

d. Weight.

Other desirable tether characteristics considered to a lesser degree included:

a. Flexibility

b. Low torque

c. Minimum elongation

d. Abrasion resistant

e. High packing density.

Finally, commercial availability and cost were considered.

Temperature compatibility and low tensile strength considerations eliminated all the nonmetallic materials considered except Aramid and fiberglass. Aramid was selected over fiberglass primarily because of weight advantages. Fiberglass could be considered for future applications requiring higher operating temperatures. The best metallic tether candidate for the applications evaluated appears to be 302 stainless steel. Based on the data presented in Table 5-6, it appears at first that carbon steel, which has greater tensile strength than stainless steel, is the best metallic material candidate. However, this is not the case, because the diameter of the wire is dictated by the requirement to survive in the micrometeorite environment and not by the load on the tether. Stainless steel is approximately the same weight as carbon steel, is nearly as flexible, and is available at lower cost.

Table 5-8 presents data showing the probability of being hit by a micrometeorite that is estimated to sever the tether. These data are based on a mathematical model developed using data in NASA TM X-64627 dated November 19, 1971, "Natural Environment Criteria for the NASA Space Station Program (Third Edition)." This mathematical model is designed for evaluation of the environment near Earth and encompasses particles of only cometary origin. The environment is composed of sporadic meteoroids in the mass range of 1 to 10^{-12} gms and stream meteoroids in the mass range of 1 to 10^{-6} gms.

TABLE 5-8. MICROMETEORITE IMPACT PROBABILITIES

Length: 80 km	Time in Orbit: 144 h		
Case I		Launch Date	Number Particles
Wire Size	0.1 mm	Feb 2	0.7
Effective Surface Area	25 m^2	Aug 4	3.14
Minimum Mass of Meteoroid	2×10^{-6}	May 6	2.12
Case II			
Wire Size	0.366 mm	Feb 2	0.104
Effective Surface Area	92 m^2	Aug 4	0.427
Minimum Mass of Meteoroid	7×10^{-5} gms	May 6	0.223
Case III			
Wire Size	1 mm	Feb 2	0.0166
Effective Surface Area	252 m^2	Aug 4	0.0688
Minimum Mass of Meteoroid	2×10^{-3}	May 6	0.0467

The model gives the meteoroid flux of mass (m) or larger. For sporadic meteoroids, this flux is a function of minimum mass (m), the defocusing factor for the Earth (G), the distance from the center of the Earth in units of the Earth's radius (R), and a seasonal meteoroid frequency factor (F). For stream meteoroids, this flux is a function of the minimum mass (m), the geocentric velocity of each stream (V), and the ratio of the cumulative flux of the stream to the average cumulative sporadic flux (K).

The particle density of the meteoroids is taken as 0.5 gm/cm^3. Average particle velocity is 20 km/s. Flux (sporadic) is determined by

$$\log Nsp = -14.41 - 1.22 \log m + \log G$$

$$+ \log \frac{1 + \sqrt{1 - 1/R^2}}{2} + \log F$$

where

Nsp = flux

G = defocusing factor = 0.568 + 0.432/R

F = seasonal factor.

Flux (stream) is determined by:

$$\log Nst = -14.41 - \log m - 4.0 \log (V/20) + \log K \quad .$$

These equations are valid for $10^{-6} \leq m \leq 1$ and do not take into account any shielding. Data on commercially available cables made of stainless steel and Kevlar are presented in Table 5-9.

Based on this comparison, Aramid is baselined as the tether material for the operational system. Further supporting research and technology is, however, recommended prior to final selection of the tether material. Table 5-10 provides detailed data on the Aramid registered by Dupont under the trade name of Kevlar®.[3]

3. Dupont Technical Information Bulletin K-1, December 1974.

TABLE 5-9. COMMERCIALLY AVAILABLE CABLE
(STANDARD SIZES NEAREST 1 mm)

Stainless Steel	
Nominal Diameter — 0.923 mm Construction — Concentric Stranded 7 × 7 Minimum Break Strength — 712 N (160 lb) Weight/km — 3.28 kg (7.23 lb)	Nominal Diameter — 1.13 mm Construction — Concentric Stranded 7 × 49 Minimum Break Strength — 756 N (170 lb) Weight/km — 5.25 kg (11.75 lb)
Aramid	
Nominal Diameter — 0.90 mm Construction — Braided Minimum Break Strength — 890 N (200 lb) Weight/km — 0.59 kg (1.3 lb)	Nominal Diameter — 1.4 mm Construction — Braided Minimum Break Strength — 1690 N (380 lb) Weight/km — 0.93 kg (2.05 lb)

Note: Load on Tether
Stainless Steel — Approximately 267 N maximum (safety factor — 2.6)
Aramid — Approximately 133 N maximum (Safety factor — 7.0)

5.1.9 Satellite

The operational system satellite (Fig. 5-25) consists of a 1.44 m (56.7 in.) diameter insulated sphere with a probe and V-ring fitting which are used to capture and lock-down the satellite during launch and landing. The all-up system of the operational system includes two LN_2 spherical tanks for thermal control, a camera, batteries, and other instrumentation as previously discussed in this section.

TABLE 5-10. TYPICAL PROPERTIES OF KEVLAR® ARAMID YARN

Denier	1500[a]
Number of Filaments	1000
Specific Gravity	1.44
Moisture Regain (commercial), %	7.0
Stress-Strain Properties	
Straight tests on conditioned yarn	
Breaking strength, N	324
Breaking tenacity, gpd	22.0
Elongation at break, %	4.0
Initial modulus, gpd	475
Loop tests on conditioned yarn	
Breaking strength, N	311
Breaking tenacity, gpd	10.5
Elongation at break, %	2.3
Thermal Properties	
Strength loss, %, after 48 hours in dry air at 180°C (350°F)	16
Shrinkage, %, in dry air at 160°C (320°F)	0.2
Zero-strength temperature[b], °C (°F)	455 (850)
Half-strength temperature, °C (°F)	400 (750)
Specific heat cal/gm/°C at 25°C	0.4
Thermal conductivity J/h/m^2/°C per m of thickness	230

a. Other deniers are also produced.

b. Temperature at which the yarn breaks under a load of 0.1 gm/denier.

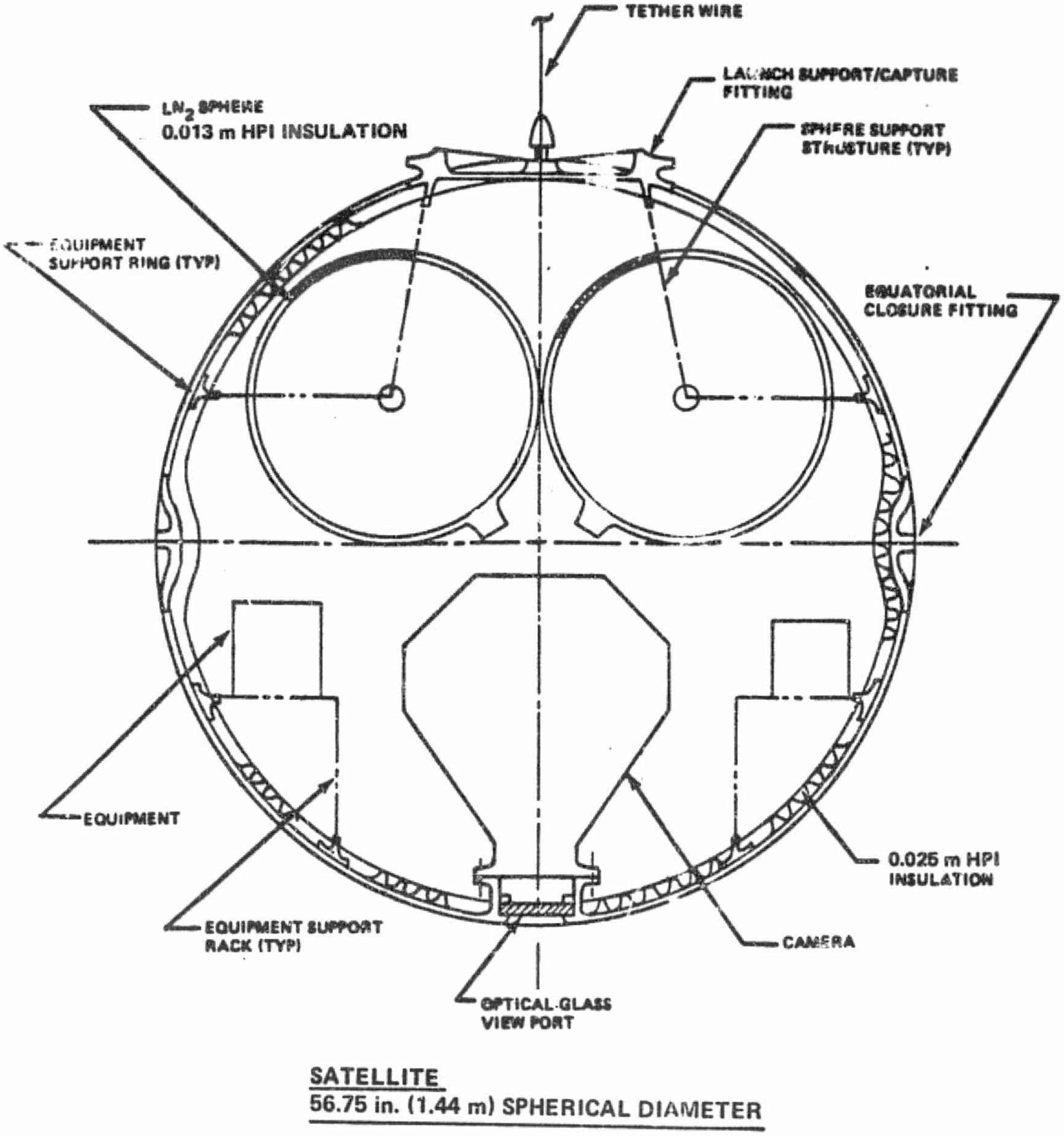


Figure 5-25. Operational system satellite.

5.2 Limited Operational System

Most of the characteristics of the limited operational system are identical to those of the operational system. Reference is made to the prior sections when applicable. Major changes appear in the communications and data handling section predominately because of the television link which was added to the satellite. The intent in including the television is to provide a vivid demonstration to ground viewers which is desired on early Orbiter flights.

5.2.1 Reel and Boom Mechanism

The reel and boom mechanism described for the operation system is also used for this limited operational system. Refer to Section 5.1.1 for details.

5.2.2 Thermal Control

For the demonstration 1-2 km system, only insulation and thermal control coatings are required. Heaters may be required for the higher altitudes during descent and return, depending on the minimum temperature required.

The equipment in the payload bay can easily be controlled by thermal coating and insulation, depending on the altitude of the Orbiter. The active system is required only when the Orbiter bay faces the Sun and then reorients to deep sapce. The coating selection is a function of the temperature requirements of individual black boxes and can be determined in a more detailed study at a later date.

5.2.3 Electrical Power

5.2.3.1 Introduction

The electrical power system is composed of two parts: Orbiter mounted and satellite mounted. Orbiter power is used as the primary source for operation of the equipment associated with deployment and retrieval equipment as well as power for the communication and data handling system.

A silver-zinc secondary battery provides power for caution and warning and safing operations in the event of the loss of primary power.

5.2.3.1.1 Orbiter Mounted System. The major load for the power system is the reel drive motor (Fig. 5-26). Approximately 45 W peak is required for operation of the motor. The extendable boom requires 10 W to extend and 15 W to retract. A closed-loop control system is utilized for deployment/ retraction and tether tension control. The digital processor, the input/ output unit, and the servo amplifier require a total of 30 W. Peak power level required is 90 W.

The average power required for a 5.49 h test is 70 W representing 383 W-h of energy required from the Orbiter. An average of 35 W is needed for backup purposes if main power is lost. This power is required only for short periods. If total main power loss occurs with the satellite deployed, the backup power source provides power for boom jettison components. The total energy required is estimated to be approximately 18 W-h.

During satellite deployment, the forces will be such that the reel drive motor will be operating as a generator. This must be considered in the design of the motor control circuitry as well as an interface consideration.

5.2.3.1.2 Satellite System. The electrical power system associated with the satellite is merely a battery and associated distribution equipment. A total of 7576 W-h is required for a 5.49 h mission. Low cost lithium primary batteries are recommended because of their energy density. A 27 kg (60 lb) battery package will accommodate the mission. The average electrical load is 138 W.

5.2.4 Communications and Data Handling

5.2.4.1 Introduction

The limited operational communications and data handling system differs from the operational system in that the two film cameras are replaced by a television link from the satellite to the Orbiter. The tether instrumentation, computation, and recording will be the same as the operational system.

5.2.4.2 Television Link

The television link consists of a TV camera, transmitter, and antenna located in the satellite and a TV receiver and antenna located in the Orbiter. The satellite camera is mounted in a fixed position so that it will be

C-2

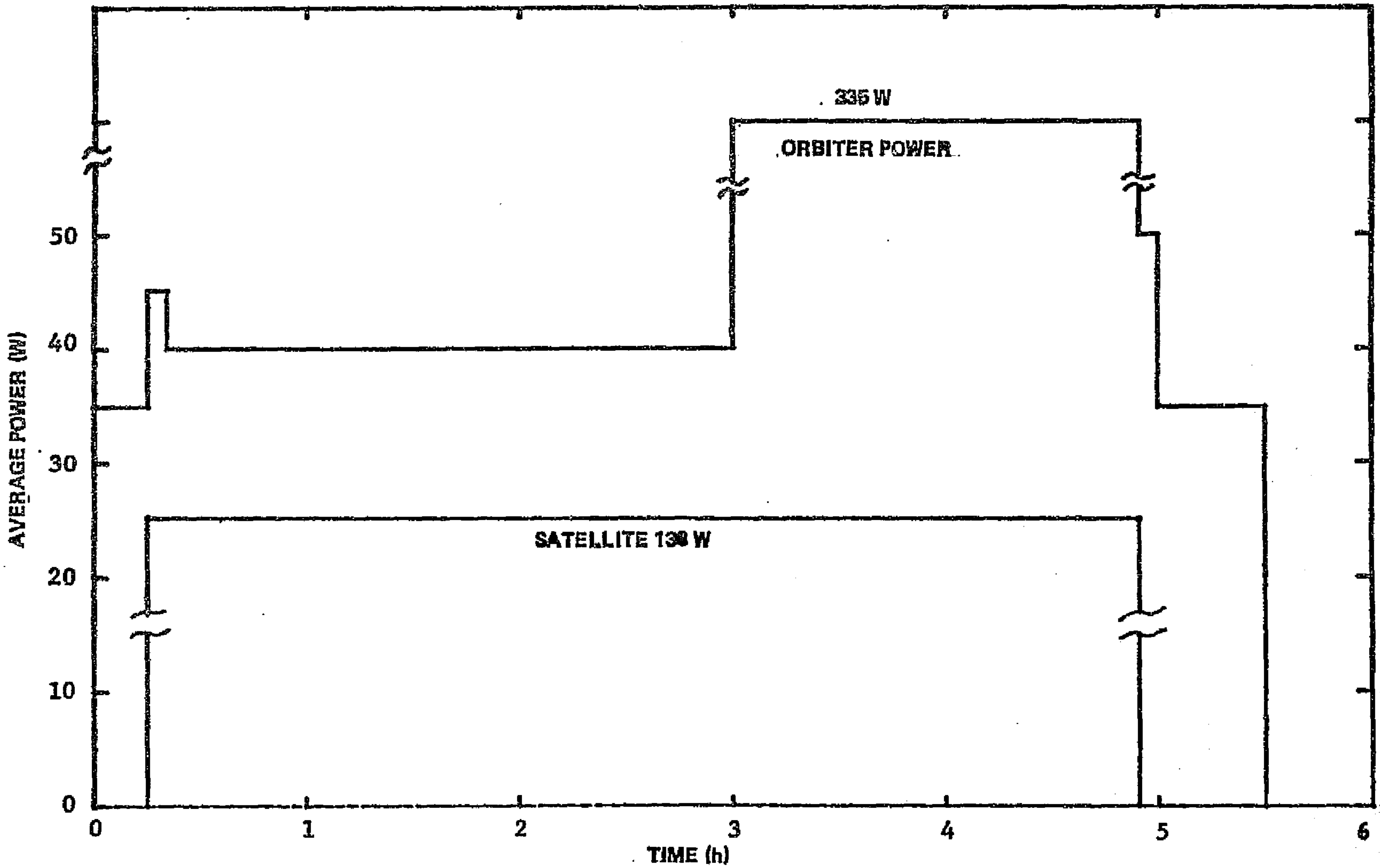


Figure 5-26. Limited operational system power profile (power averaged for each phase).

pointing toward the Orbiter during the deployment and retrieval phases. The receiver demodulates the video which is interfaced with the Orbiter for display on the Orbiter closed circuit television (CCTV) and for recording.

VHF transmission will be employed for the TV receiver and transmitter. VHF appears to be the most efficient choice of frequencies assuming omni TV receiver and/or near onmi antennas. The Orbiter payload interrogator which includes an S-band receiver and antenna is a possible alternate to a tether system supplied. The bandwidth of the Orbiter payload interrogator receiver will allow medium quality TV. Employing this receiver (should it be allowed) will eliminate the requirement for the tether system to supply them. A video demodulator would have to interface with the payload interrogator receiver because it presently does not have video demodulation capability. This system operates at S-band and requires more power from the payload. The payload batteries have been sized for an S-band transmitter. Table 5-11 presents power, weight, and volume for components of television link.

TABLE 5-11. HARDWARE ESTIMATES FOR TELEVISION LINK

	Power (W)	Weight kg	(lb)	Volume 10^{-3} m^3	(in.^{3})
TV Camera	8	0.45	(1)	1.6	(94)
TV Transmitter	112	0.90	(2)	0.7	(42)
Transmitter Antenna	-				
Receiver Antenna	-				
TV Receiver	10	2.3	(5)	1.0	(64)
Interface Unit	5	2.3	(5)	1.6	(96)

5.2.5 Control and Display

The control and display of the limited operational tether system will be similar to that described for the operational system. The ranges on the parameters to be displayed will be different but this should not impact the hardware design.

5.2.6 Control

The control subsystem for the limited operational system is essentially the same as the operational system. Sufficient data were presented in parametric form in Section 5.1.6 covering the operational system to describe the operation of the limited operational system. Although the power to drive the reel motor is less in the demonstration flight than in the operational flight, the same hardware design is maintained so that the limited operational system can easily be retrofitted to operational system specifications. Although studies have not yet indicated the need, some optimization of the control algorithm for the 1 km deployment and retrieval might be required. Trajectories of a 1 km deployment and retrieval are presented in Section 7.2.1.

Previous comments regarding the satellite attitude control are also applicable for the limited operational system satellite. Aerodynamic torques on the satellite are negligible because of the high altitude and spherical shape of the satellite used in the demonstration test.

5.2.7 Tracking and Data Acquisition

The tracking requirements for the limited and concept demonstrations systems are essentially the same as the operational system. Figure 5-27 illustrates a typical day's tracking opportunities which is representative of the tracking coverage available for OFT-5, OFT-6, and 81-2 missions. Table 5-12 presents the tracking summary for individual stations and their union. Relay satellite (RS 1 and RS 2) opportunities are also shown. Figure 5-28 illustrates 1 day of Orbiter ground tracks typical of the OFT-5, OFT-6, and 81-2 missions.

5.2.8 Tether

Many of the concerns regarding tether strength and environmental survivability disappear when the short length (1 to 10 km) and short time exposure (5 to 10 h) are considered in the demonstration test with the limited

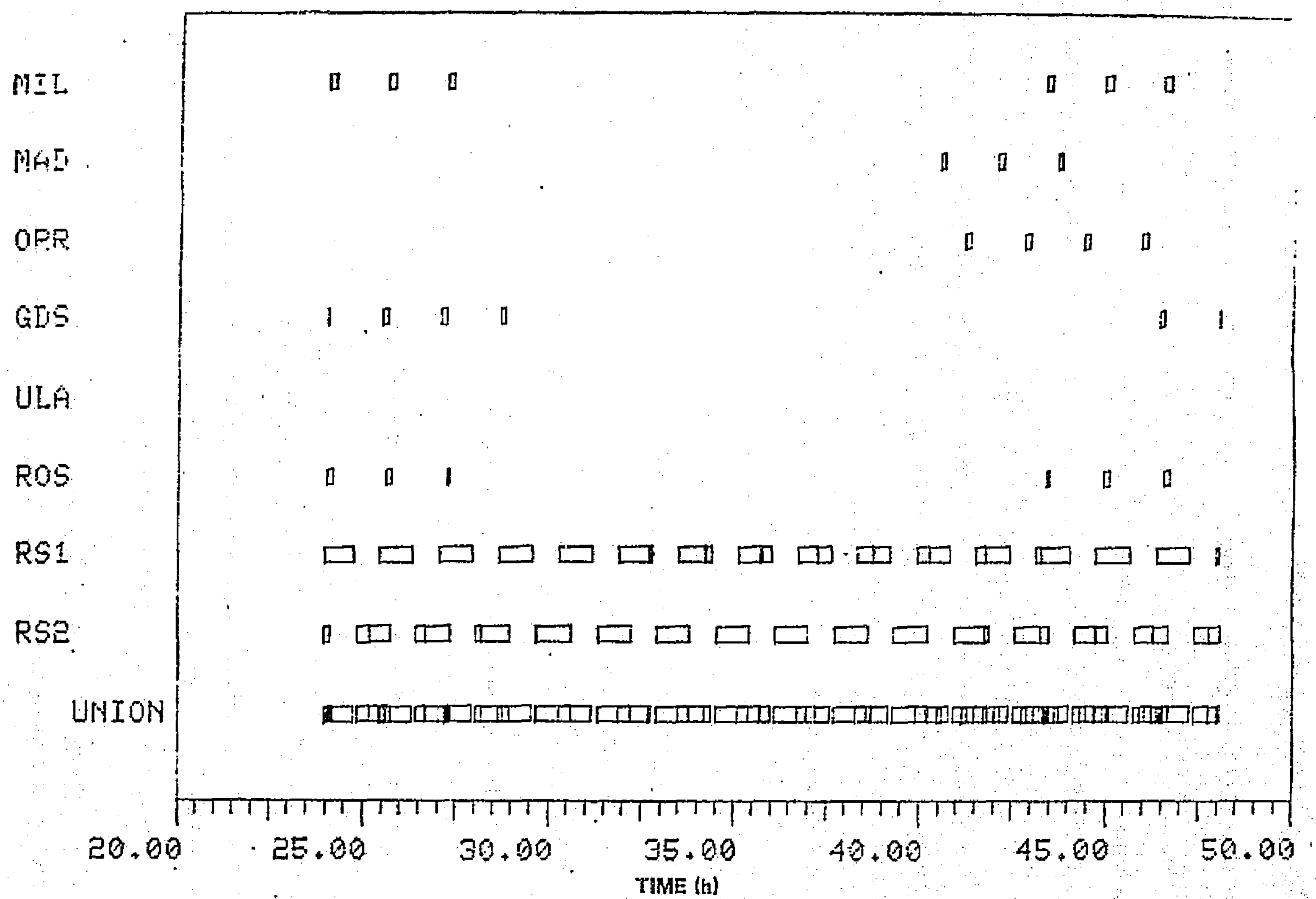


Figure 5-27. Typical tracking opportunities.

TABLE 5-12. TYPICAL MISSION TRACKING SUMMARY

INDIVIDUAL TRACKING STATION SUMMARY TABLE AND TRACKING UNION TABLES FOR 5. DEGREES ELEVATION ANGLE

TARGET	AVERAGE TIME OVER TARGET/DAY MINUTES			AVERAGE NUMBER OF PASSES/DAY			AVERAGE TIME/PASS MINUTES			TIME OVER TARGET/MISSION MINUTES			NUMBER OF PASSES/MISSION		
	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0	> 5.0	>2.0	> .0
MIL	29.06	32.35	32.35	4.46	5.33	6.19	6.51	6.07	5.23	201.87	224.72	224.72	31	37	43
MAD	.00	3.03	3.56	.00	1.15	3.46	.00	2.63	1.03	.00	21.02	24.72	0	8	24
ORR	11.15	16.56	16.58	2.02	3.46	4.61	5.53	4.79	3.60	77.44	115.07	115.16	14	24	32
GDS	13.73	18.24	18.41	2.45	3.60	4.90	5.61	5.07	3.76	95.36	126.72	127.90	17	25	34
ULA	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	.00	0	0	0
ROS	12.27	18.23	18.56	2.16	3.74	5.18	5.68	4.88	3.58	85.20	126.97	128.93	15	26	36
RS1	781.39	781.39	781.39	15.98	15.98	15.98	48.90	48.90	48.90	5427.39	5427.39	5427.39	111	111	111
RS2	779.26	779.26	779.26	15.98	15.98	15.98	48.76	48.76	48.76	5412.64	5412.64	5412.64	111	111	111

UNION OF TRACKING TIME FOR TDRS

AVERAGE TRACK TIME/ORBIT (HRS) = 1.35617
AVERAGE TRACK TIME/DAY (HRS) = 21.67266
TOTAL TRACK TIME/MISSION (HRS) = 150.53472
AVERAGE GAP TIME (HRS) = .15544
MAXIMUM GAP TIME (HRS) = .18250

UNION OF TRACKING TIME FOR GROUND STATIONS

AVERAGE TRACK TIME/ORBIT (HRS) = .07375
AVERAGE TRACK TIME/DAY (HRS) = 1.17852
TOTAL TRACK TIME/MISSION (HRS) = 8.18583
AVERAGE GAP TIME (HRS) = 1.60601
MAXIMUM GAP TIME (HRS) = 13.89028

UNION OF TRACKING TIME FOR ALL STATIONS

AVERAGE TRACK TIME/ORBIT (HRS) = 1.35617
AVERAGE TRACK TIME/DAY (HRS) = 21.67266
TOTAL TRACK TIME/MISSION (HRS) = 150.53472
AVERAGE GAP TIME (HRS) = .15544
MAXIMUM GAP TIME (HRS) = .18250

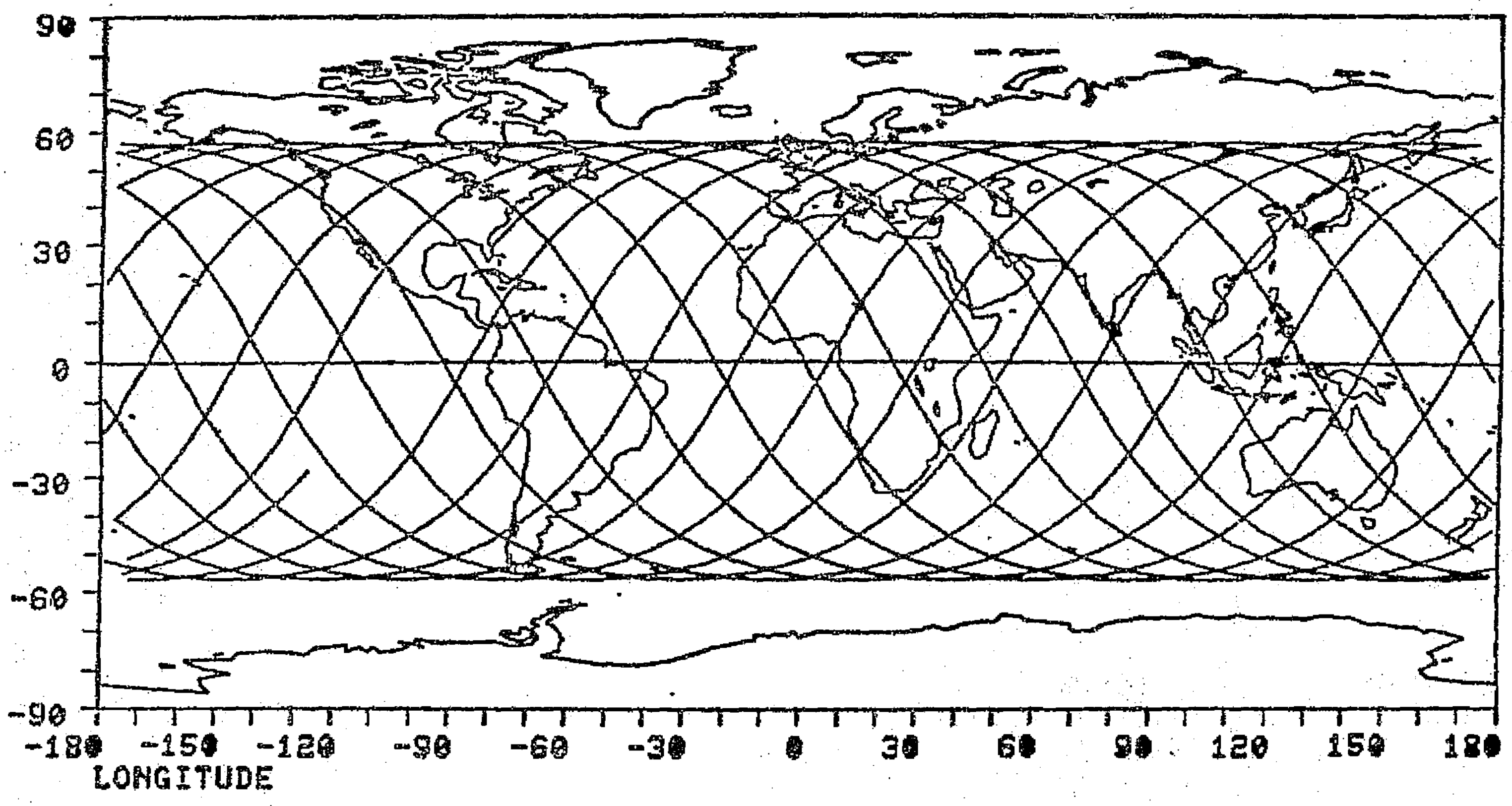


Figure 5-28. Typical day ground tracks.

operational system. Also, the short length reduces the manufacturing problems associated with long continuous metallic cables. Although the material selection criteria are eased for the demonstration test, there is a desire to use the same material for the demonstration test that would be used for the operational flight. Another factor which might affect the material selection is the possibility that the demonstration test might include a science payload requiring different characteristics from the magnetic and gravity instrumentation. Until plans for the demonstration test are firm, both Kevlar and stainless steel options will be considered.

5.2.9 Satellite

The satellite is similar to that of the operational system satellite (Fig. 5-14) except that different instrumentation is provided as previously discussed. Less LN_2 for thermal control and less power are required.

5.3 Concept Demonstration System

The concept demonstration system was configured as the minimum tether system which could be flown to demonstrate the tether concept. The design is risky from several aspects. The Shuttle RMS instead of the tether system boom is used for initial satellite deployment and final capture and retrieval. End effectors for the RMS to accomplish this task will probably not be available in the OFT time frame. Also, there is concern regarding the positional accuracy of the RMS when retrieving satellites. The proposal to use the RMS instead of the boom was based purely on a possible cost savings in the early phase of system development. This cost savings might not materialize if a special end effector has to be fabricated and if the tether system has to pay for use of the RMS in hidden costs such as crew time, procedures, training, etc. A second problem is that the distance the RMS can place a satellite away from the Orbiter is limited to less than 15 m compared to 50 m for the boom. This means less gravity gradient force is available for control which will require resizing of the control system.

Another cost saving proposal is to use a surplus Lageos simulator as a satellite. The simulator is a hollow 0.6 m (2 ft) diameter aluminum sphere 10 cm (4 in.) thick. Engineering information would be limited to the parameters measured in the control system. A further cost savings is obtained by storing this limited engineering data in unused computer core and eliminating the data tape recorder.

A smaller reel mechanism to house a smaller diameter (0.366 mm) tether is also described. The construction and flight of the smaller system cannot be recommended as a cost savings approach when the operational system will be developed later in the program.

5.3.1 Alternate Reel Mechanism

Initially a simpler reel mechanism was developed (Figs. 4-15, 4-16, and 4-17). In this case a harmonic drive is used to mechanically link the spool-drive motor with the lead screw (Fig. 4-16) which, in turn, is mechanically self-reversing (Fig. 4-17). Thus a much simpler control system is required. Since meteoroid damage to the tether had not been assessed at the time this configuration was designed, the smaller 0.366 mm diameter tether is shown. The larger 1 mm diameter tether could easily replace the smaller line by merely enlarging the spool, housing, and pulleys. Although the system is mechanically and functionally simpler than the later system, it does need extensive terrestrial testing to determine the more subtle points of failure and verification of the anticipated high reliability. Concept verification is also required. Selection of one of the two systems comes to a tradeoff to the mechanically more complex, electronically simpler alternate system versus the mechanically simple, electronically more complex (but with more redundancy) system.

5.3.2 Thermal Control

The satellite selected for the concept demonstration system is completely passive. The thermal control system for the pallet mounted system is essentially the same as that for the operational system (refer to Section 5.1.2).

5.3.3 Electrical Power

5.3.3.1 Introduction

The electrical power system is composed of two parts: Orbiter mounted and subsatellite mounted. Orbiter power is used as the primary source for operation of the equipment associated with deployment and retrieval equipment and for the communications and data handling system.

A silver-zinc secondary battery provides power for caution and warning and safing operations in the event of the loss of primary power.

5.3.3.1.1 Orbiter Mounted System. The major load for the power system is the reel drive motor (Fig. 5-29). Approximately 45 W peak is required for operation of the motor. A closed-loop control system is utilized for deployment/retraction and tether tension control. The digital processor, input/output unit, and servo amplifier require a total of 30 W. Average power level required for a 7.74 h test is 30 W representing 233 W-h of energy required from the Orbiter. An average of 35 W is needed for backup purposes if main power is lost. This power is required only for short periods. If total main power loss occurs with the satellite deployed, the backup power source provides power to jettison components. The total backup energy required is estimated to be approximately 18 W-h.

During satellite deployment, the forces will be such that the reel drive motor will be operating as a generator. This must be considered in the design of the motor control circuitry as well as an interface consideration.

5.3.4 Communications and Data Handling

The communications and data handling consists of tether instrumentation and data recording. No attempt will be made to determine the payload attitude except by vision. The tether instrumentation will be the same as the operational system. The housekeeping and the line data will be stored in the computer core.

5.3.5 Control and Display

The concept demonstration system controls and displays are similar to those of the limited operational system. Refer to Section 5.2.5 for appropriate comments.

5.3.6 Control

The tether control system for the concept demonstration system is similar to that for the operational systems. The important changes are described in this section.

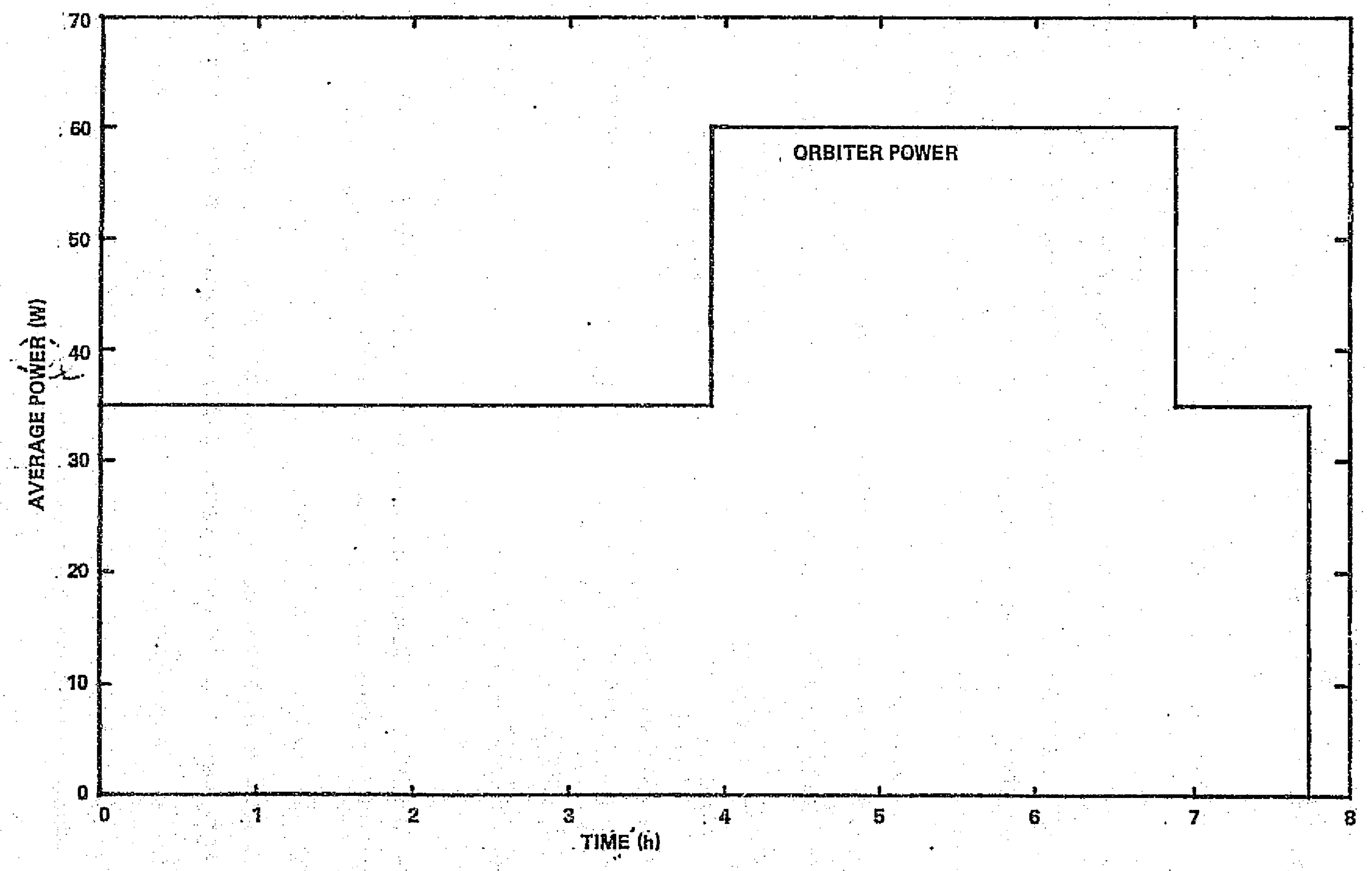


Figure 5-29. Concept demonstration system (power averaged for each phase).

5.3.6.1 Tether Tension

The range of tether tension over which the reel mechanism must function is different from the previously described system because the satellite mass is 113 kg versus 175 kg for the operational system and because the initial satellite deployment distance by the RMS is as small as 10 m compared to 50 m for the tether system boom. The tension ranges from 0.0045 N at 10 m distance to 0.45 N at 1 km. The extremely low minimum tension will probably require special consideration concerning tether kinking, force measuring fixture design, and satellite ballistic coefficient. The parametric studies in Section 5.1.6.3 illustrate the problems associated with short tether system lengths.

5.3.6.2 Reel Mechanism Design Requirements

Elimination of the tether system boom displaces some important elements of the control system. The tensiometer and torque motor (previously located at the boom tip) will have to be relocated. Two choices are possible. The reel mechanism design can be modified to incorporate the boom tip torque motor and, by using the alternate control system inner loop configuration described in Section 5.1.6.1, only one tensiometer will be required in the subsystem. An alternate choice would be to use the operational system boom tip mechanism connected directly to the reel mechanism case. A proper tradeoff of these two alternatives will have to be made if the concept demonstration system is to be pursued.

5.3.6.3 Satellite Attitude Control

By definition, no attitude control of the satellite and no engineering data package for measuring environmental data are provided.

5.3.7 Tracking and Data Acquisition

See Sections 5.1.7 and 5.2.7.

5.3.8 Tether

The comments in Section 5.2.8 also apply to the tether for the concept demonstration system. A smaller diameter material (0.366 mm) is sized and is consistent with the strength and survivability requirements associated with the short duration, short length demonstration flight.

5.3.9 Satellite

The satellite (Fig. 5-30) is a dummy system, modified from an uninstrumented Lageos simulator of 0.609 m (24 in.) diameter. A V-ring has been added to adapt the simulator to the lock-down assembly for capture, launch, and landing. A length of tubing has been added for the RMS end effector to grasp for deployment or capture. Either the grasping surface of the RMS end effector or the tubing should be knurled and the other surface coated with an elastomer.

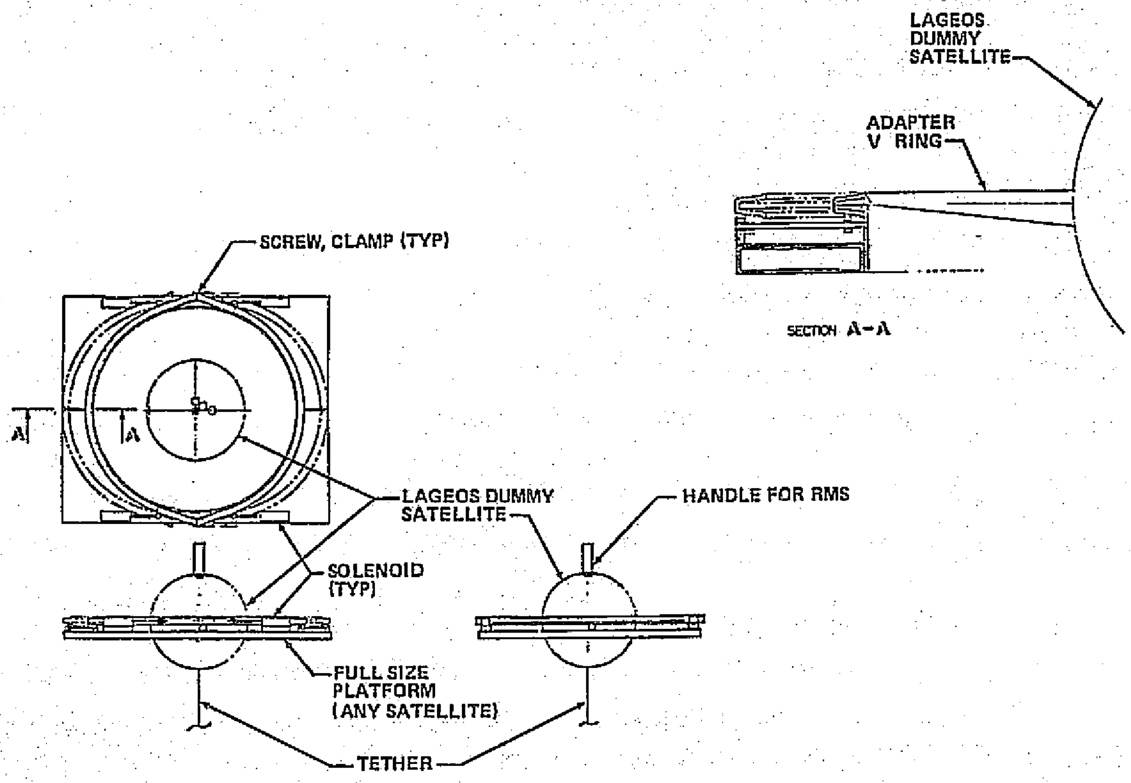


Figure 5-30. Lageos dummy satellite 0.61 m (24 in.) spherical diameter.

6.0 INTERFACE DEFINITION

Table 6-1 summarizes the Orbiter-Tether System interface for the operational, limited operational, and concept demonstration systems.

6.1 Mass Properties

Table 6-2 shows the weight summary for the operational, limited operational, and concept demonstration tether systems using an Aramid tether. An additional 540 kg (1190 lb) would have to be added to the operational system if a stainless steel tether is used instead of an Aramid. An additional 54 kg (120 lb) would have to be added to the limited operational and concept demonstration systems for the stainless steel option.

6.2 Volume Requirements

The overall sizes of the three tether system designs are shown in Table 6-3.

6.3 Power Requirements

Figure 6-1 compares the power requirements versus time for the three system designs. The charts are for the pallet mounted system. Power required by the satellite instrumentation is discussed in Sections 5.1.3 and 5.1.4.

As previously mentioned, the tether reel mechanism will act as a generator during deployment. Peak power generated will be on the order of 700 W. This power will have to be dumped into a dummy load or fed into the power bus for use. Preliminary analysis indicates that the Orbiter power system will accommodate this amount of power.

6.4 Energy Requirements

The energy required for the three system designs is shown in Table 6-4.

TABLE 6-1. ORBITER INTERFACE SUMMARY

Requirement	Demonstration of Operational System	Demonstration of Limited Operational System	Demonstration of Concept Demonstration System
System Availability	OFT-6	OFT-6	1981-2
Tether Length	100 km Max.	1-10 km	1-10 km
Launch and Return Mass	705 kg	607 kg	456 kg
Altitude	190 to 210 km Circular	>200 km Circular	>200 km Circular
Inclination	Any	Any	Any
Direction of Deployment	Down	Up or Down	Up or Down
Attitude	Z-LV; X-POP or Y-POP Payload Bay to Earth	Z-LV	Z-LV
Total Operating Time	36 h	6 h	8 h
Beta Angle	Any	Any	Any
Power, Peak/Average	1128/120.8 W	90/70 W	45/30 W
Energy	4255 W-h	383 W-h	233 W-h
Size	2.9 × 1.8 × 3.6 m	2.9 × 1.8 × 3.6 m	2.5 × 1.8 × 2.3 m
Data Storage	Provided by Tether System	Same	Same

TABLE 6-1. (Concluded)

Requirement	Demonstration of Operational System	Demonstration of Limited Operational System	Demonstration of Concept Demonstration System
Crewmen Required	One	One	Two
Orbiter TV Required	Yes	Yes	Yes
Remote Manipulator Required	No	No	Yes
Power Conditioning	28 Vdc Unregulated	Same	Same

TABLE 6-2. WEIGHT SUMMARY FOR THE TETHER SYSTEM USING AN ARAMID TETHER

	Operational		Limited Operational		Concept Demonstration	
	kg	(lb)	kg	(lb)	kg	(lb)
Reel Mechanism and Case	223	(492.6)	223	(492.6)	223	(492.6)
Boom and Drive Mechanism	83	(183.1)	83	(183.1)	0	(0)
Reel and Boom Attachment to Pallet	39	(86.4)	39	(86.4)	39	(86.4)
Jettison Device	20	(44.2)	20	(44.2)	20	(44.2)
Satellite Fitting and Capture Cone	3	(6.3)	3	(6.3)	3	(6.3)
Launch Lock and Support Structure	26	(57.8)	26	(57.8)	26	(57.8)
Tether (Aramid)	113	(248)	11	(24.8)	11	(24.8)
Satellite (Control Weight)	175	(386)	175	(386)	113	(250)
Computer and Interface Unit	9	(19.5)	9	(19.5)	9	(19.5)
Control and Display Panel	5	(10)	5	(10)	5	(10)
Recorder	2	(5)	2	(5)	0	(0)
Control Electronics	3	(6)	3	(6)	3	(6)
Distribution Equipment	4	(8)	4	(8)	4	(8)
RF Receiver	0	(0)	2	(5)	0	(0)
Total	705	(1552)	605	(1335)	456	(1005)

TABLE 6-3. SIZE OF TETHER SYSTEM DESIGNS

	L × W × H (m)	L × W × H (in.)
Operational System	2.92 × 1.76 × 3.60	115 × 70 × 142
Limited Operational	2.92 × 1.76 × 3.60	115 × 70 × 142
Concept Demonstration System	2.48 × 1.76 × 2.29	98 × 70 × 90

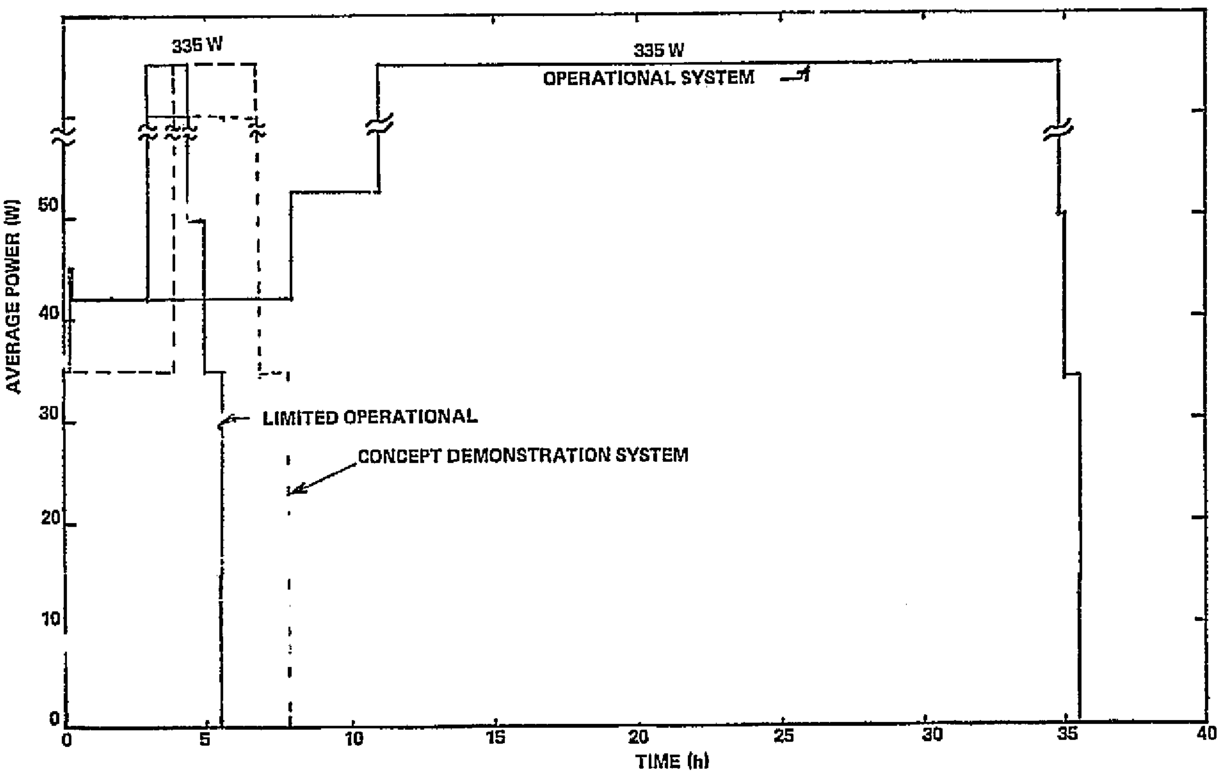


Figure 6-1. Power requirement versus time (power averaged for each phase).

TABLE 6-4. ENERGY REQUIRED FOR THE TETHER SYSTEM DESIGNS

	Energy Required/ (W-h)
Operational System	4255
Limited Operational System	383
Concept Demonstration System	233

6.5 Thermal System Requirements

Tether system components mounted in the Orbiter payload bay can operate with a temperature of -12° to 55°C (10° to 131°F). The anticipated temperature of the pallet bottom for an Orbiter orientation with the Z axis along the local vertical is -6° to -11°C. Therefore the actual temperature of any equipment mounted in the payload bay will be a result of internal heat loads. A range of 90 to 145 W/m^2 can be handled passively. If this range is exceeded, then an active system will have to be used which will mean using the Orbiter cooling system (i.e., cold plates on the pallet).

6.6 Layout of Tether System in Payload Bay

6.6.1 Layout — Operational System in Payload Bay

The operational system is shown in the Orbiter payload bay in Figure 6-2. The system may be mounted to the Spacelab pallet and still provide volume for additional requirements and adequate clearance around the payload bay (Fig. 6-2). It could also be mounted to a platform or even be custom mounted to the payload bay. While this alternate approach could possibly save volume enough to permit the use of the tether on more Orbiter flights, tradeoffs of these customized mounting costs versus increased flight opportunities and additional experiments remain to be assessed.

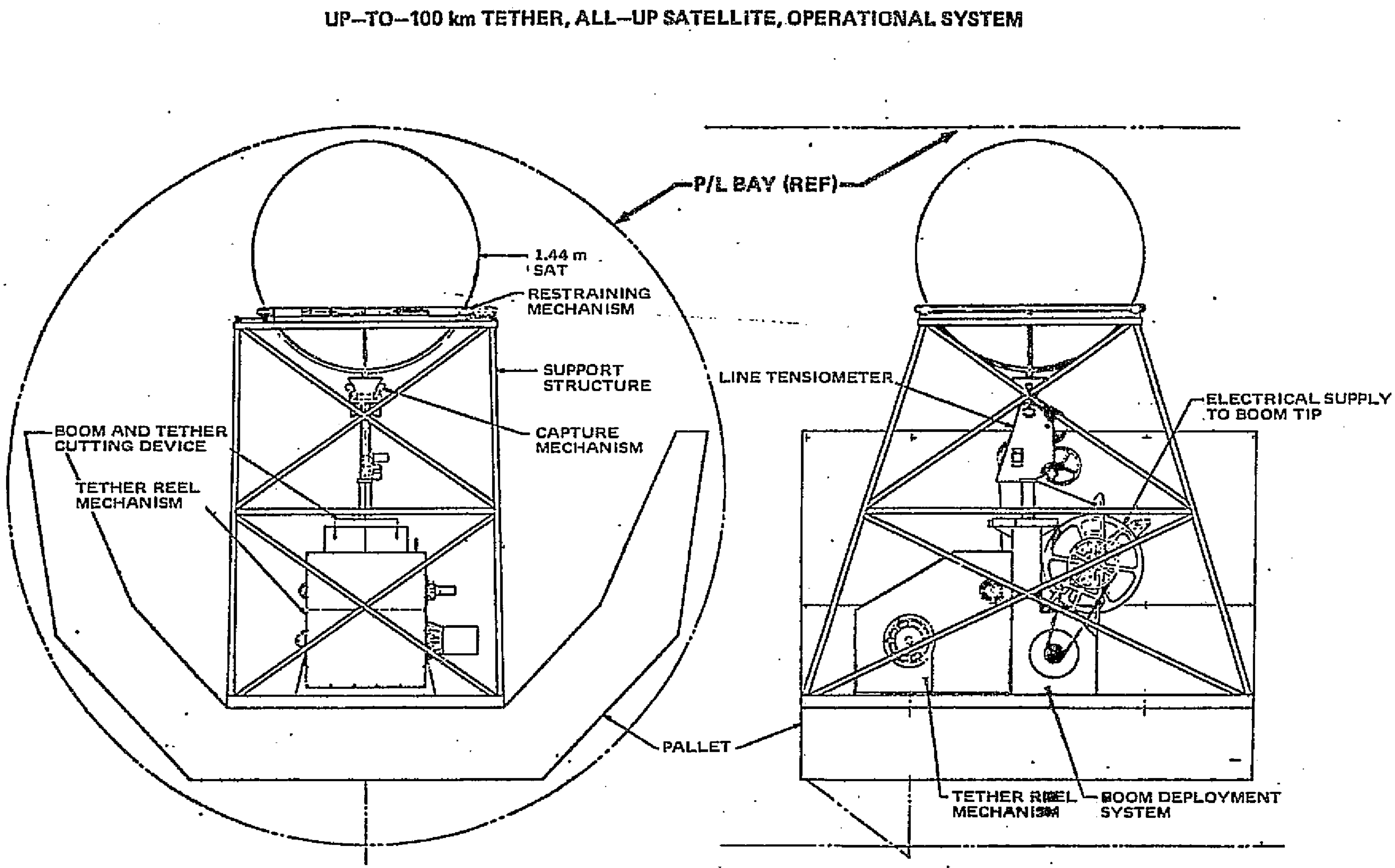


Figure 6-2. Operational system.

6.6.2 Layout — Limited Operational System in Payload Bay

The limited operational system is shown in Figure 6-2 (same as operational system). The comments in Section 6.6.1 apply with emphasis on earlier Orbiter flights.

6.6.3 Layout — Concept Demonstration System in Payload Bay

The concept demonstration system is shown in Figure 6-3. Again, the comments in Section 6.6.1 apply, especially to those concerning additional and earliest flight opportunities. Since the concept demonstration system uses the smaller systems, less volume is required. Also, the lighter/ smaller system could permit forward mounting on over-the tunnel platforms, etc. with minor center of gravity shift effects. Again, commonality and cost considerations may dictate the use of the same larger geometry of the operational system, although this system would still be lighter (no boom, slightly smaller satellite) than either the operational or limited operational systems.

6.7 Mission Requirements

6.7.1 Operational System

The operational tether system consists of a 100 km tether, a 1.44 m spherical satellite, a 50 m boom, reel mechanism, and other equipment. The total system weight is approximately 705 kg (1552 lb). The system requires one pallet for mounting in the cargo bay. Since the system weight and volume requirements are minimal, the tether system can potentially share Shuttle flights with many other payloads. However, the local vertical attitude requirement of (up to 6.5 days for subsequent operational flights) approximately 1.5 days for the first operational flight estimates sharing flights with some candidate payloads not eliminated by volume and weight requirements. In detailed mission planning phases, consideration must be given to providing a low to medium beta angle to prevent Orbiter thermal problems arising when in local vertical attitude.

The design orbit altitude of 200 km will probably require stationkeeping because the Orbiter lifetime at 200 km is only a few days. The acceleration experienced by the Orbiter at this altitude is approximately 10^{-5} g's. Since some applications of the tether system require global coverage, the tether system design must be compatible with any inclination orbit.

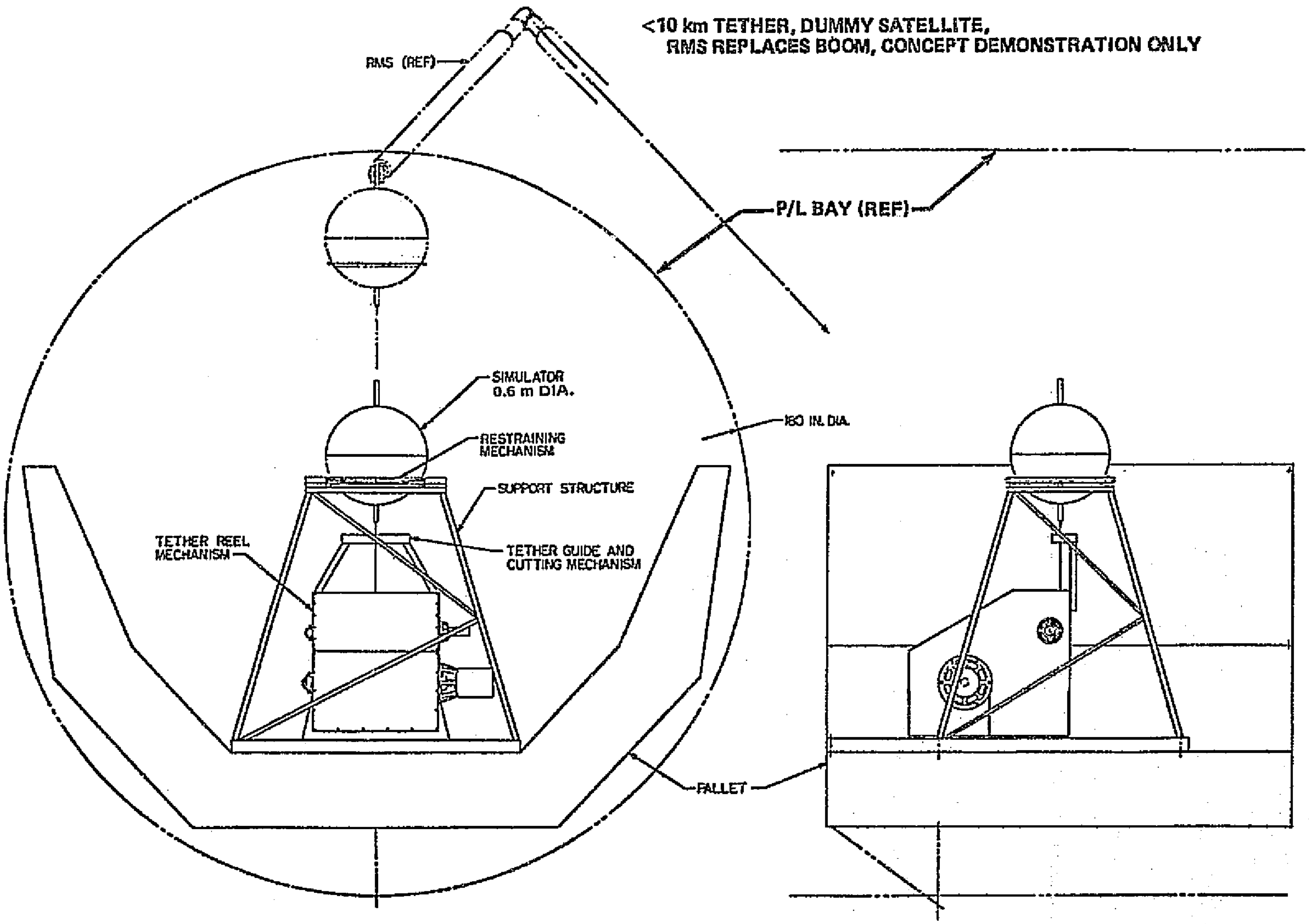


Figure 6-3. Concept demonstration system.

Energy requirements for deploying, stabilizing, controlling, and retrieving the tether will be provided by tension to the tether with an electrical motor. Thus, additional Orbiter propellant is not required for these operations. Energy requirements for Orbiter stationkeeping, attitude control, and the effect of Orbiter thruster firings on the tether were not addressed in this study.

Shuttle mission 81-2 is potentially a flight that the tether system could share. It is made up of Earth viewing experiments and thus requires the local vertical attitude compatible with the tether attitude requirement. However, the planned orbit altitude of 325 km would be well above the 200 km tether design altitude. The 200 km altitude orbit could, however, be provided early in the mission and the Orbiter transfer to a higher orbit upon completing the tether operations. The planned inclination of 57° is satisfactory for the tether system. Weight and volume requirements could also be accommodated with trades between potential payloads being performed in later studies.

6.7.2 Limited Operational System

The limited operational tether system is similar to the operational system except that the tether is only 10 km long and will be deployed only 1 to 2 km. The system weight and volume requirements are approximately equal to the operational system requirements. Since the differential effect of drag is negligible for altitudes above 150 km, the 1 to 2 km demonstration and test objectives can be satisfied from any orbit altitude and inclination. The local vertical attitude will be required less than 10 h.

The limited operational system and/or the concept demonstration system potentially can be flown as part of the OFT-5 or OFT-6 science payload. The primary objective of the OFT flights is Orbiter flight testing with remaining capability allocated to science payloads. Orbit requirements and flight summary data are presented in Table 6-5. The orbit requirements are compatible with the tether systems because tether operations can be scheduled when the Orbiter attitude is unconstrained. Also, a detailed study of payload packaging will be required for the OFT-5 because pallet space is critical for this flight. Figures 6-4 through 6-7 present typical beta histories and percent time in the Sun for the OFT 5 and 6.

TABLE 6-5. ORBIT REQUIREMENTS AND FLIGHT SUMMARY

Data	OFT-5 February 1980	OFT-6 March 1980
Inclination	50°–57°	28–1/2°–40°/TBD
Altitude	463 km	463 km
Flight Duration	7 Days	7 Days
Beta Angle	Medium	Medium
Crew Size	3–4 Men	4 Men
Landing Site	KSC	KSC
Up Weight	9980 kg	9980–24 948 kg
Down Weight	9980 kg	9980 kg
Time of Unconstrained Attitude	50 h	50 h

6.7.3 Concept Demonstration System

The concept demonstration system is made up of a simplified reel, an 0.6 m diameter satellite, and the RMS will be used for deploying the satellite instead of the boom. The tether will be deployed 1 to 2 km. The mission requirements and applicable Shuttle flights for this system are similar to the limited operational system.

Table 6-6 summarizes the timeline for testing the operational, limited operational, and concept demonstration systems.

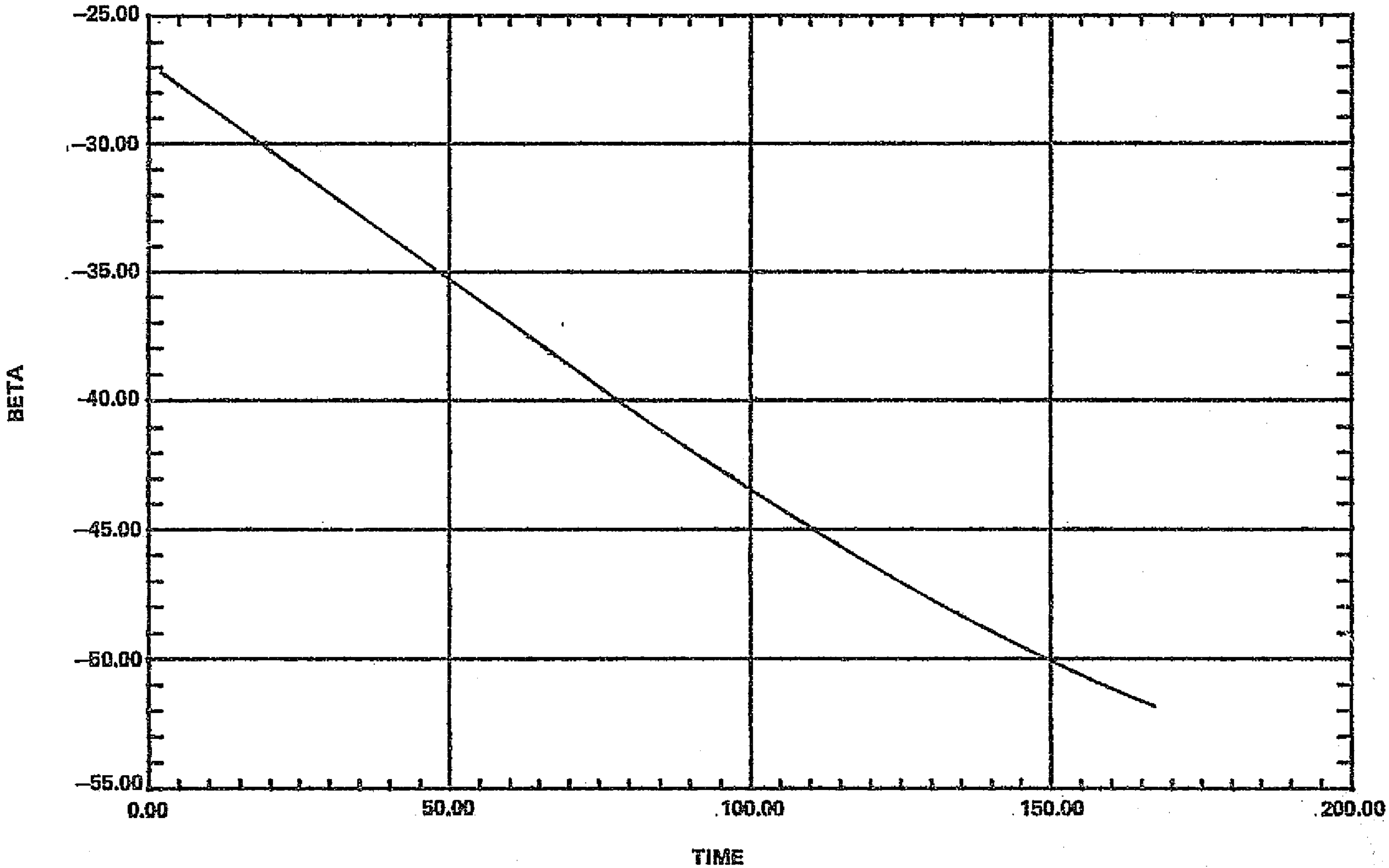


Figure 6-4. Typical OFT-5 beta history.

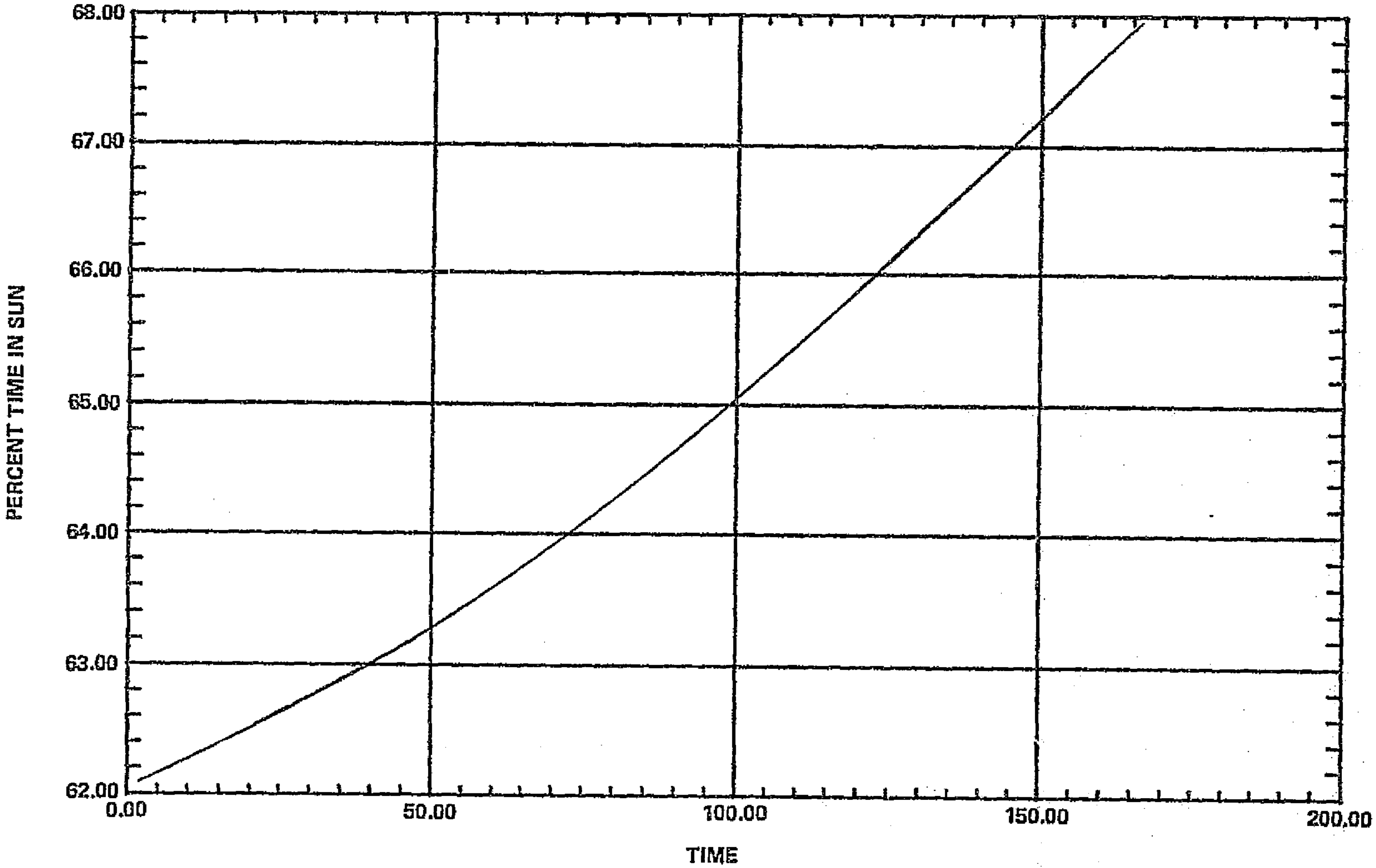


Figure 6-5. Typical OFT-6 time in Sun.

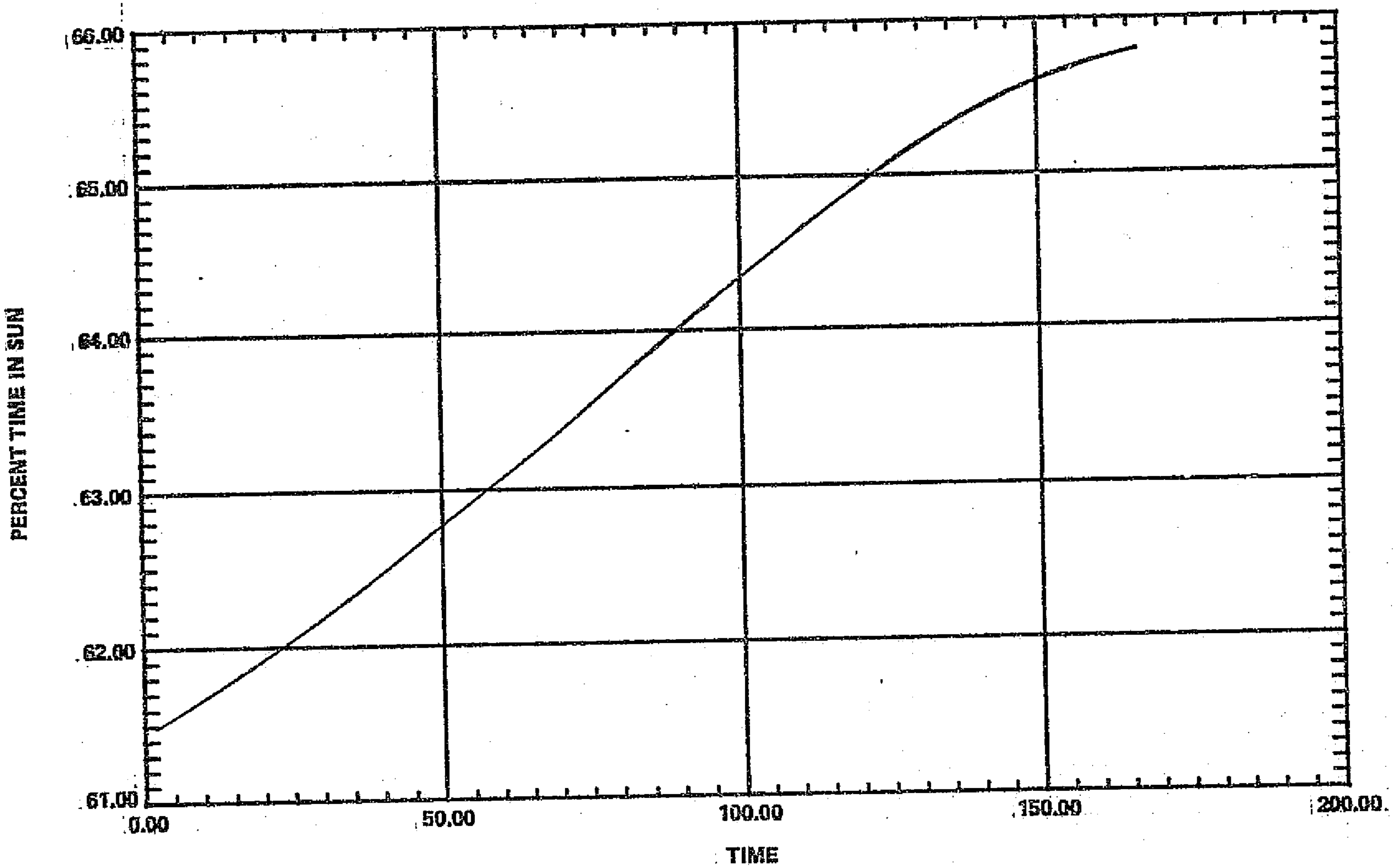


Figure 6-6. Typical OFT-6 beta history.

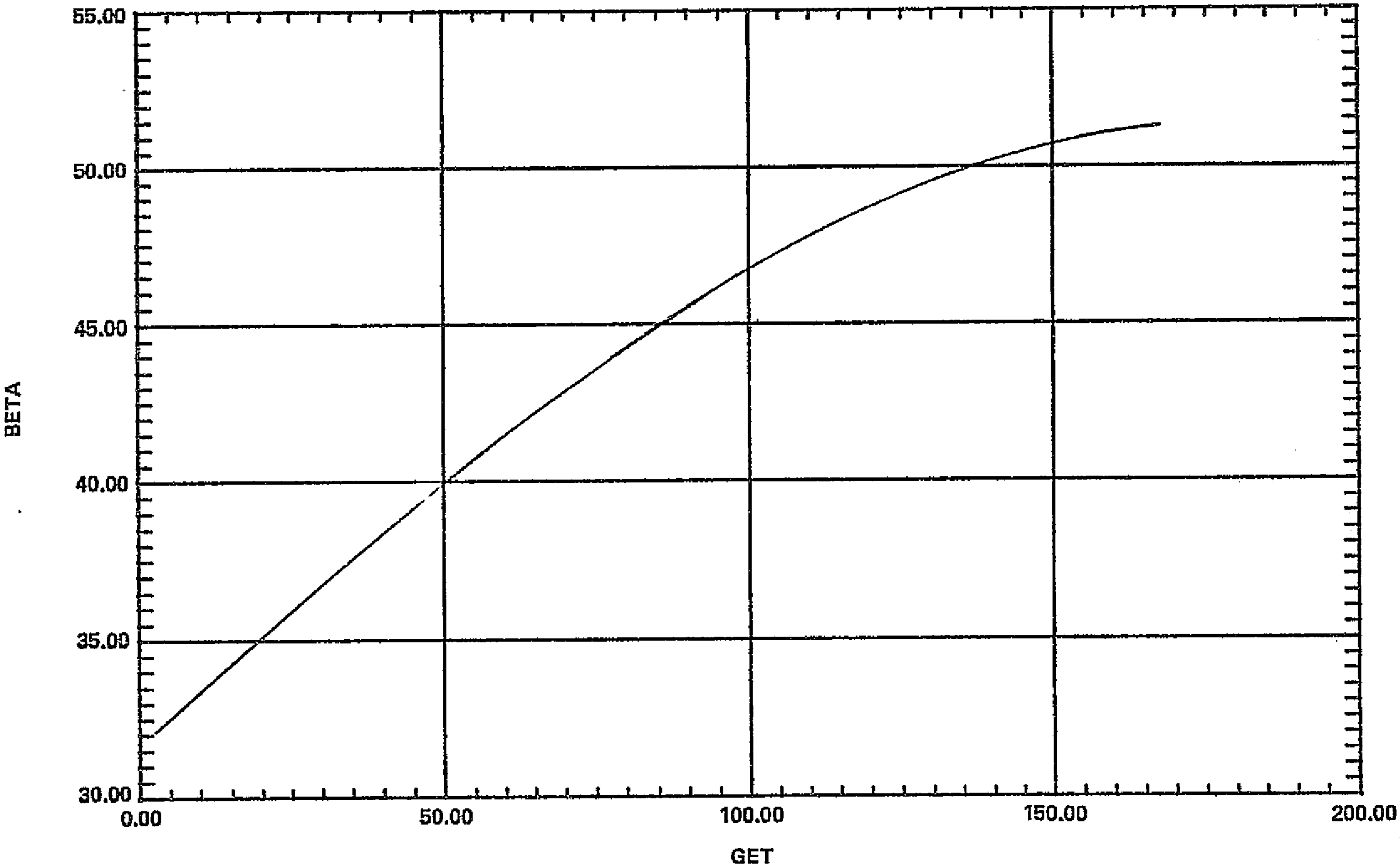


Figure 6-7. Typical OFT-6 time in Sun.

TABLE 6-6. TIMELINE FOR TESTING TETHER SYSTEMS

Function	Timeline (h)		
	Operational System	Limited Operational System	Concept Demonstration System
Subsystem Activation	0.25	0.25	0.25
Initial Deployment	0.1	0.1	0.52
Display Tether	7.5	1.4	1.96
Conduct Experiment	3.0	1.26	1.18
Retrieve Tether	23.8	1.91	2.97
Capture and Stow Satellite, etc.	0.58	0.58	0.86
Total	35.23	5.49	7.74

7.0 MISSION ANALYSIS

This section deals with the operation of the tether system and covers subjects which are highly mission dependent. These subjects are the operational timeline, crew support requirements, trajectories, and ground support requirements. Only two missions are recognized at this time: an early demonstration of the system concept possibly on OFT-6 and a demonstration of the operational system to an altitude of 120 km on mission 81-2. Science payloads or other applications might be added to these mission requirements at a later date.

7.1 Operational System

7.1.1 Timeline and Crew Support Requirements

Demonstration of the operational system will require approximately 35 h for the following major events:

Operational System 80 km Deployment

Event	Time (h)
Subsystem Activation	0.25
Extend Boom (50 m at 0.15 m/s)	0.1
Deploy Tether (80 km)	7.5
Conduct Experiment	3.0
Retrieve Tether	23.8
Retract Boom	0.08
Safe Satellite	0.50
Total Time Required	35.23

Subsystem activation will require approximately 0.25 h and consist of activation of the boom drive assembly coupled with a visual check of the reel case, jettison device, satellite, and capture cone. Any calibration or activation of the satellite might require additional time.

Retraction of the boom will require approximately 0.08 h, and safing of the satellite will require approximately 0.5 h. The satellite will then be placed in its stowed position for return to Earth.

7.1.2 Trajectories for the Operational System

Figure 7-1 shows the trajectory for the operational system deployed 80 km in length and then retrieved. Aerodynamic drag has not been included in this particular simulation. However, the simulation is useful to show the time required for the deployment and retrieval phases.

Further simulation results are shown in Appendix D. These simulations are for an open-loop deployment scheme as described in Chapter 2 of Reference 7. Since it was an open-loop control law, there was no attempt to retrieve the tether system in this simulation. Also, the simulation is for the earlier tether system design previously described (Section 5.3.1). The mass of this system is similar to that described for the operational system. However, the tether and satellite diameters are smaller in the simulation; therefore, the effects of aerodynamic drag will be more pronounced for the operational system than is presented in Appendix D. One further point needs to be clarified. Case A of the simulation results is for a circular orbit. Case B is designed to simulate the geometric effects of an oblate Earth by sinusoidally lowering the altitude of the satellite 10 km once per orbit to determine the atmospheric density. This geometric altitude variation in combination with the atmospheric rotation obtained in a polar orbit will excite an out-of-plane motion of the tether system (Fig. 9 of Reference 7). This motion will grow slowly in the absence of active damping or other compensation techniques.

7.1.3 Ground Support Requirements

7.1.3.1 Preflight

Preflight checkout of the operational tether system will be limited to functional checks of the power, communication and data handling, and portions of the control subsystems. No operation of the control system will be possible.

The satellite described in this report for use in the operational system contains cryogenic nitrogen for cooling. Sufficient capacity has been designed into the satellite tanks to allow for vents during Orbiter preparation. An external vent might be necessary if venting into the payload bay is not possible. Special handling of experiment payload film might be required.

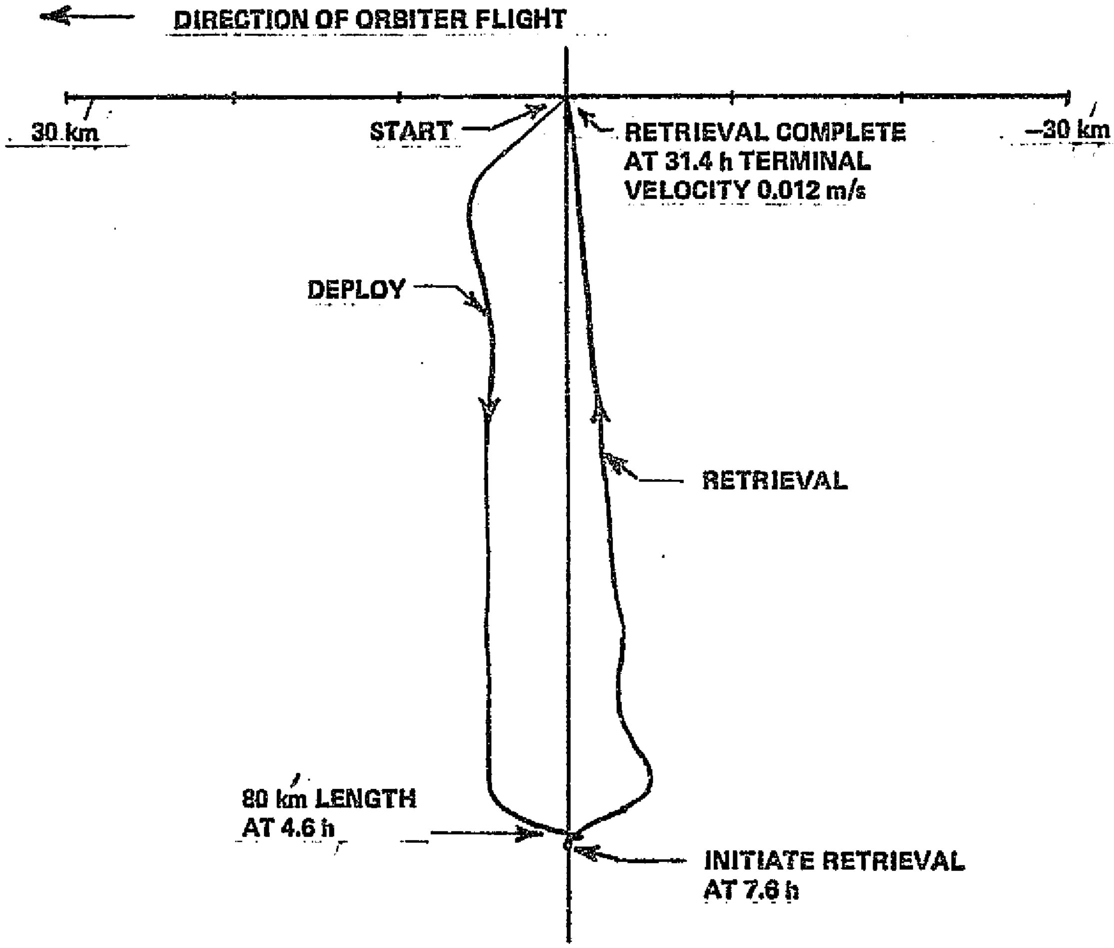


Figure 7-1. In-plane trajectory of satellite for deployment to 80 km and return.

7.1.3.2 Flight Support

No real time data for mission support are being planned at this time. This could change as user and mission operation requirements are determined. A tether system specialist will be available for mission support as required.

7.1.3.3 Post Flight

Tether system data will be recorded on a tape recorder supplied as part of the tether system. Data reduction will be accomplished at MSFC or a designated contractor facility. The tether system and satellite will be returned to MSFC for inspection. Detailed inspection of the tether will be accomplished to determine the effects of exposure to the space environment and reel operation. Processing of experimental data and film will be accomplished at MSFC.

7.1.4 Orbital Requirements

The orbit requirements are not dictated by any particular science requirements for the flights discussed in this report, because the study was directed toward designing the tether system rather than the satellite. The tether system is being designed to accommodate a satellite 100 km below an Orbiter at 200 km altitude. The low altitude capability would be particularly applicable to atmospheric, gravitational, and magnetospheric studies. Other potential applications are illustrated in Figures 1-1 and 1-2.

The altitude requirements for all applications of the tether system cover the entire Shuttle altitude capability. Considering the wide range of applications and global interest of same, the tether system will also be designed to operate at any orbital inclination.

Demonstration of the operational tether system is currently envisioned as part of mission 81-2. The altitude and inclination for this flight will probably be 325 km and 57°, respectively. To demonstrate the tether system on mission 81-2 at the 200 km design altitude would require the Orbiter to spend approximately 2 days at the lower altitude and then transfer to the higher altitude for the remainder of the mission.

7.2 Limited Operational System

7.2.1 Timeline and Crew Support Requirements

Demonstration of the limited operational system will require about 5.49 h. This system differs from the concept demonstration because this system utilizes the 50 m boom to extend the satellite out of the cargo bay, while the concept demonstration system uses the manipulator to perform this task. A timeline for the limited operational system is presented as follows:

Limited Operational System

Event	Time (h)
Subsystem Activation	0.25
Extend Boom (50 m)	0.10
Deploy Tether (1 km)	1.40
Conduct Experiment	1.26
Retrieve Tether	1.91
Retract Boom	0.08
Safe Satellite	0.50
Total Time Required	5.49

Subsystem activation will require approximately 0.25 h, followed by extending the boom/satellite out of the cargo bay to a distance of 50 m. The tether is then deployed a distance of 1 km and requires approximately 1.4 h duration. The experiment associated with the satellite is allotted 1.26 h after which the tether will be retrieved (1.91 h), the boom retracted (0.08 h), and the satellite safed (0.5 h) and placed in its stowed position.

7.2.2 Trajectories for the Limited Operational System

Figures 7-2 and 7-3 show the trajectory of the limited operational system deployed 1 km and retrieved. The initial position of the satellite was 50 m from the Orbiter, trailing and out of plane by 2.5°, simulating the effect of Orbiter pointing errors. Aerodynamics have not been included for this simulation because the effects are negligible for the expected Orbiter altitude of 463 km (250 n. mi.). The aerodynamic effects would have to be included if the altitude for the demonstration of the limited operational system were lowered significantly.

7.2.3 Ground Support Requirements

Ground support requirements for the limited operational system are similar to those of the operational system with the following exceptions.

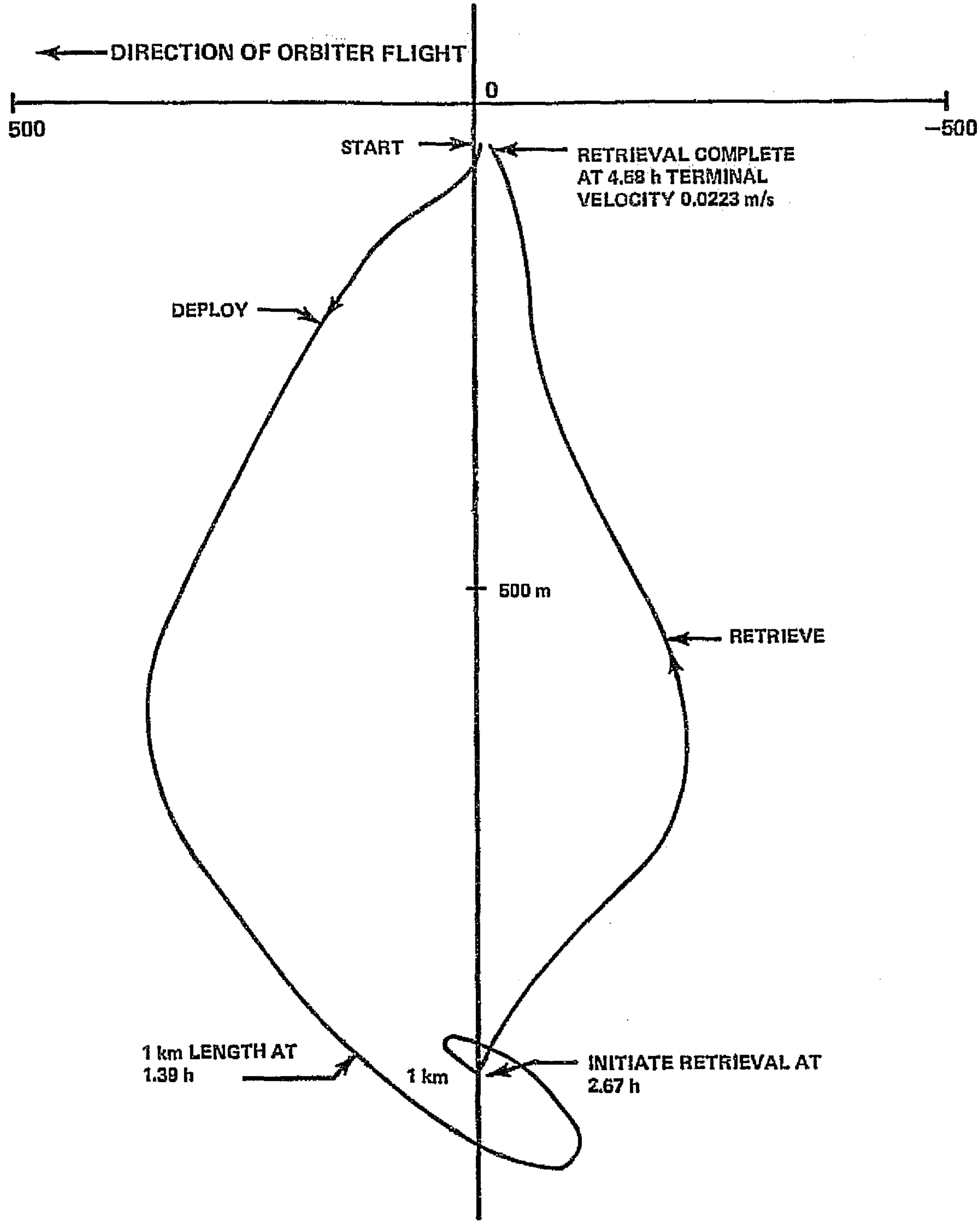


Figure 7-2. In-plane trajectory of satellite for deployment from 50 to 1000 m and return.

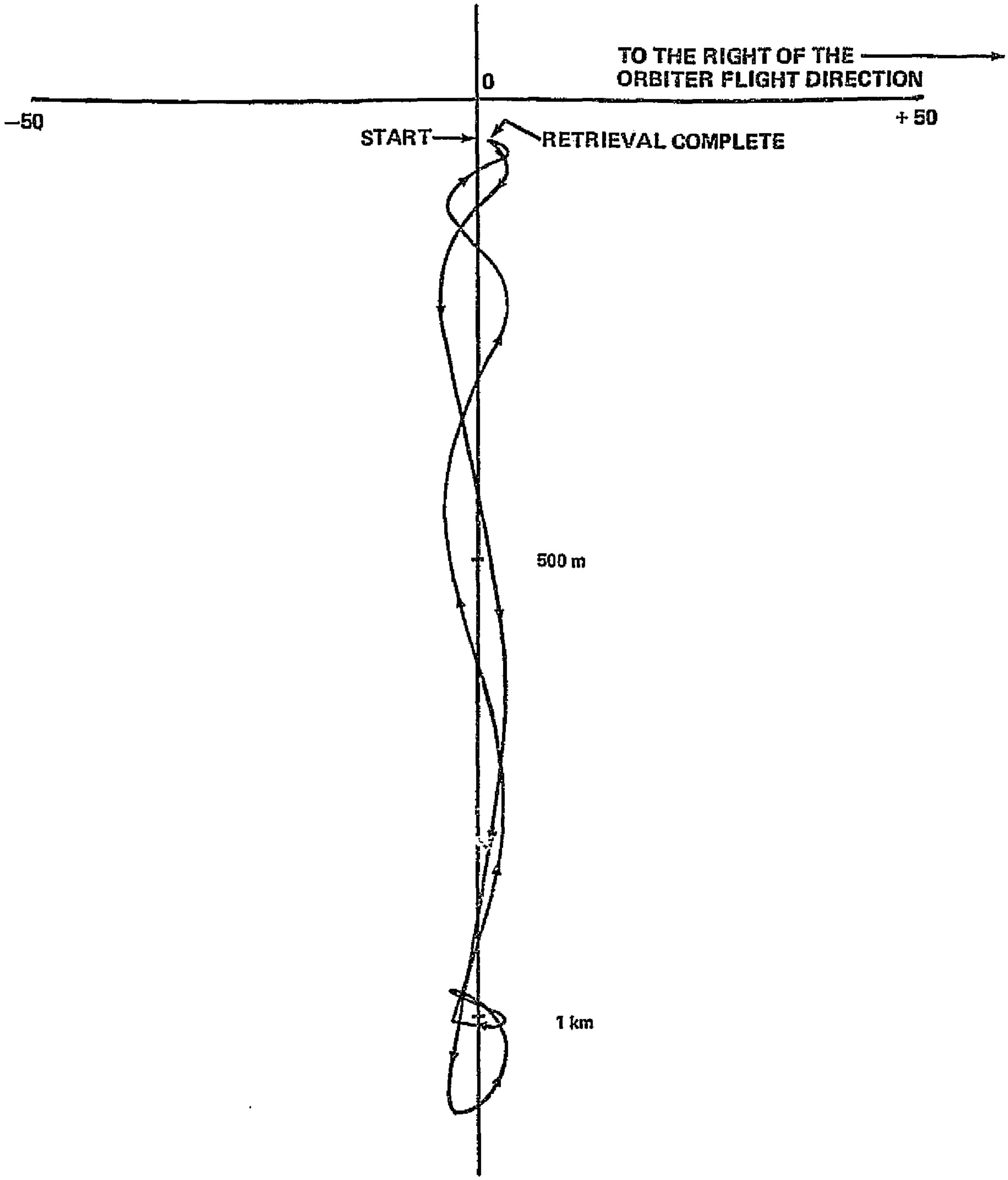


Figure 7-3. Out-of-plane trajectory of satellite for deployment from 50 to 1000 m and return.

The satellite is not expected to be subjected to a high temperature environment; therefore, cryogenic cooling is not required. Special handling of film and cryogenic nitrogen is not required.

7.2.4 Orbital Requirements

The limited operational and concept demonstration systems can be utilized and demonstrated from any orbit. OFT-5 and OFT-6 are possible flights on which these tether systems could be flown. Typical orbital inclination and altitude for these missions are 57° and 463 km.

7.3 Concept Demonstration System

7.3.1 Timeline and Crew Support Requirements

Demonstration of the concept demonstration system can be accomplished in approximately 7.74 h as follows:

Concept Demonstration System

Event	Time (h)
Subsystem Activation	0.25
RMS Activation and Checkout	0.25
Move RMS from Stowed Position and Grapple Satellite	0.13
Raise Satellite to 10 m above Orbital Horizontal Centerline (1.2 m/min)	0.14
Deploy Tether (1 km)	1.96
Conduct Experiment	1.18
Retrieve Tether	2.97
Grapple Satellite	0.1

Event	Time (h)
Move Satellite to Stowed Position	0.14
Safe Satellite	0.50
Move RMS to Stowed Position	0.12
Total Time Required	7.74

These activities will require a full time payload specialist to operate the tether system and a part time crewman to operate the RMS. Subsystem activation will require approximately 0.25 h and consist of activation of the boom drive assembly coupled with a visual check of the reel case, jettison device, satellite, and capture cone. Any calibration or activation of the satellite might require additional time.

Following subsystem activation, a crewman will transfer the RMS from its stowed position and use it to grapple the satellite. This activity will require approximately 0.13 h and, still using the RMS, will be followed by raising the satellite 10 m above the Orbiter horizontal centerline (1.2 m/min). Deployment of the tether to 1 km will require approximately 1.96 h followed by 1.18 h to conduct the experiment.

Retrieval of the tether will require approximately 2.97 h. After the tether has been retrieved, the satellite will be grappled (using the RMS), safed, and then moved to its stowed position.

7.3.2 Trajectories for the Concept Demonstration System

Figure 7-4 and 7-5 show the trajectories for the concept demonstration system deployed to 1 km and then retrieved. The trajectories are similar to those of the limited operational system except the initial deployment and final retrieval distances are 10 m which should be typical of Orbiter manipulator arm operation. A 2.5° Orbiter pointing error is assumed.

7.3.3 Ground Support Requirements

Ground support requirements for the concept demonstration system are similar to those of the operational system except no cryogenics or film are contained in the satellite, and only limited engineering data contained in the

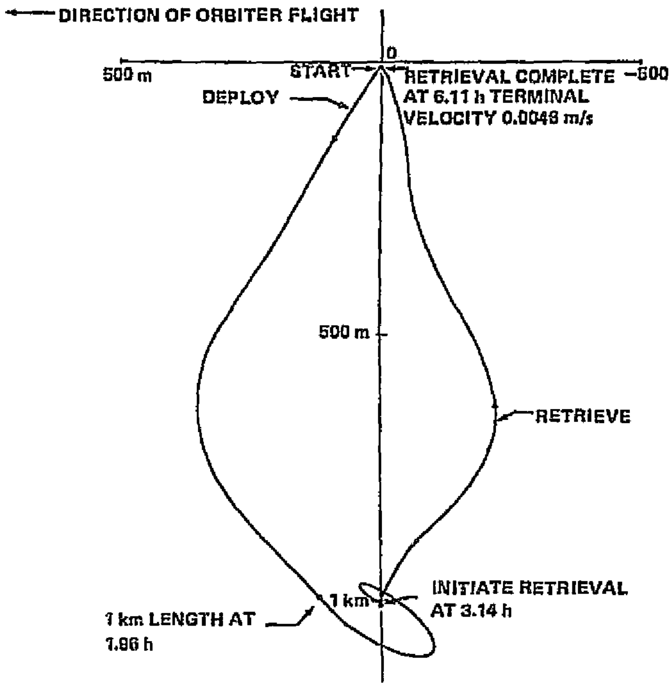


Figure 7-4. In-plane trajectory of satellite for deployment from 10 to 1000 m and return.

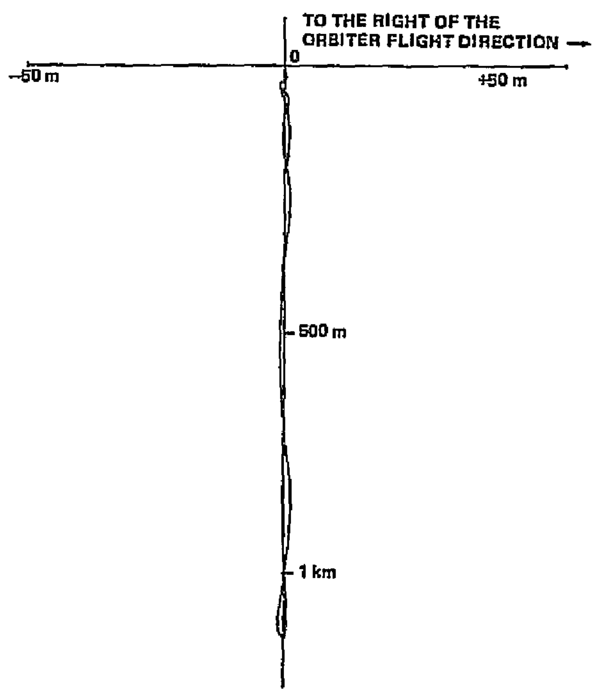


Figure 7-5. Out-of-plane trajectory of satellite for deployment from 10 to 1000 m and return.

computer core are returned for data reduction. Orbiter video tape of the satellite deployment and retrieval operations is required.

7.3.4 Orbital Requirements

See Section 7.2.5.

8.0 SAFETY

This section describes the concerns which have been identified regarding crew, Orbiter, mission, and ground safety. As these concerns have surfaced during analysis, presentations, and discussions throughout NASA, an attempt has been made to provide designs for equipment which would satisfy these concerns. Analysis is incomplete at this time, and safety reviews and discussions with appropriate people of other NASA Centers are planned to continue as future studies are pursued. The following discussion presents the concerns which have been identified, and the current status of each.

8.1 Requirements

The tether system is a new concept to be seriously considered for Orbiter flight. Previous manned tether flights in the Gemini program are generally regarded as successful. However, these flights have not eased the concern for safety regarding tether systems. The concern stems from the following questions:

a. What will prevent the tether from wrapping around the Orbiter in the event of a stuck thruster on the Orbiter?

b. What will prevent a slack tether from tangling in the Orbiter structure?

c. How can the tether system be quickly safed if the Orbiter payload bay doors have to be closed?

d. Will firing the Orbiter thrusters, RCS, or vernier control system (VCS) cause the tether to go slack or otherwise affect the dynamics of the tether system to the extent that a safety problem exists?

e. How can the Orbiter avoid recontact with jettisoned debris?

f. What will happen to the satellite and tether if it is jettisoned or if the tether is accidentally cut?

g. Will the satellite and tether burn up or reenter if the tether is severed?

These questions, and others, form the basis for safety requirements which the system must meet. The only requirement specifically stated thus far is a requirement to provide a means of jettisoning all items which protrude beyond the payload bay envelope. Future requirements for this system should be identified jointly between MSFC, JSC, and possibly some of the tether system users. Until the final requirements are formulated, these questions will be treated as requirements. The following section discusses tether system safety features which bear on these concerns.

8.2 Tether System Safety Devices

As shown in Figures 4-3, 4-4, and 8-1, a pyrotechnic guillotine is provided to sever the tether at the boom tip and another guillotine is provided to sever the boom, tether, and electrical cabling atop the boom housing. In this way there are provisions to rapidly jettison the tether (with or without the satellite) at any stage of deployment or retrieval. These two levels of safing will allow the system to be safed and, at the same time, provide for maximum return of hardware to Earth. In the event of a need to safe a properly operating tether system, only the tether at the boom tip needs to be severed. The guillotine is designed to grip the reel end of the cut tether so it does not become disengaged from the reel mechanism. This allows the boom to be retracted and returned to Earth. In the event that the guillotine fails, the entire tether mechanism can be jettisoned. In this situation, the umbilical line providing Orbiter power and data interface to the tether system must be severed by a pyrotechnic disconnect simultaneously as the tether mechanism platform is jettisoned. Springs impart an initial velocity to the mechanism after pyrotechnic actuated bolts have been fired.

Generally, the Orbiter might have to be maneuvered to avoid recontact with the debris that has been jettisoned. In some cases, probability of recontact would be negligible. A satellite with a severed tether is in a free flying trajectory. This trajectory is elliptical with a different period from that of the Orbiter. Also, the closest approach to the orbit of the Orbiter depends on the separation between the satellite and the Orbiter at the time the tether was severed.

8.3 Current and Future Safety Issue Studies

The Smithsonian Astrophysical Observatory (SAO) is currently studying the dynamics of a tethered satellite system under contract NAS8-32199. Part of their study activity is an analysis of abort dynamics. A finite element

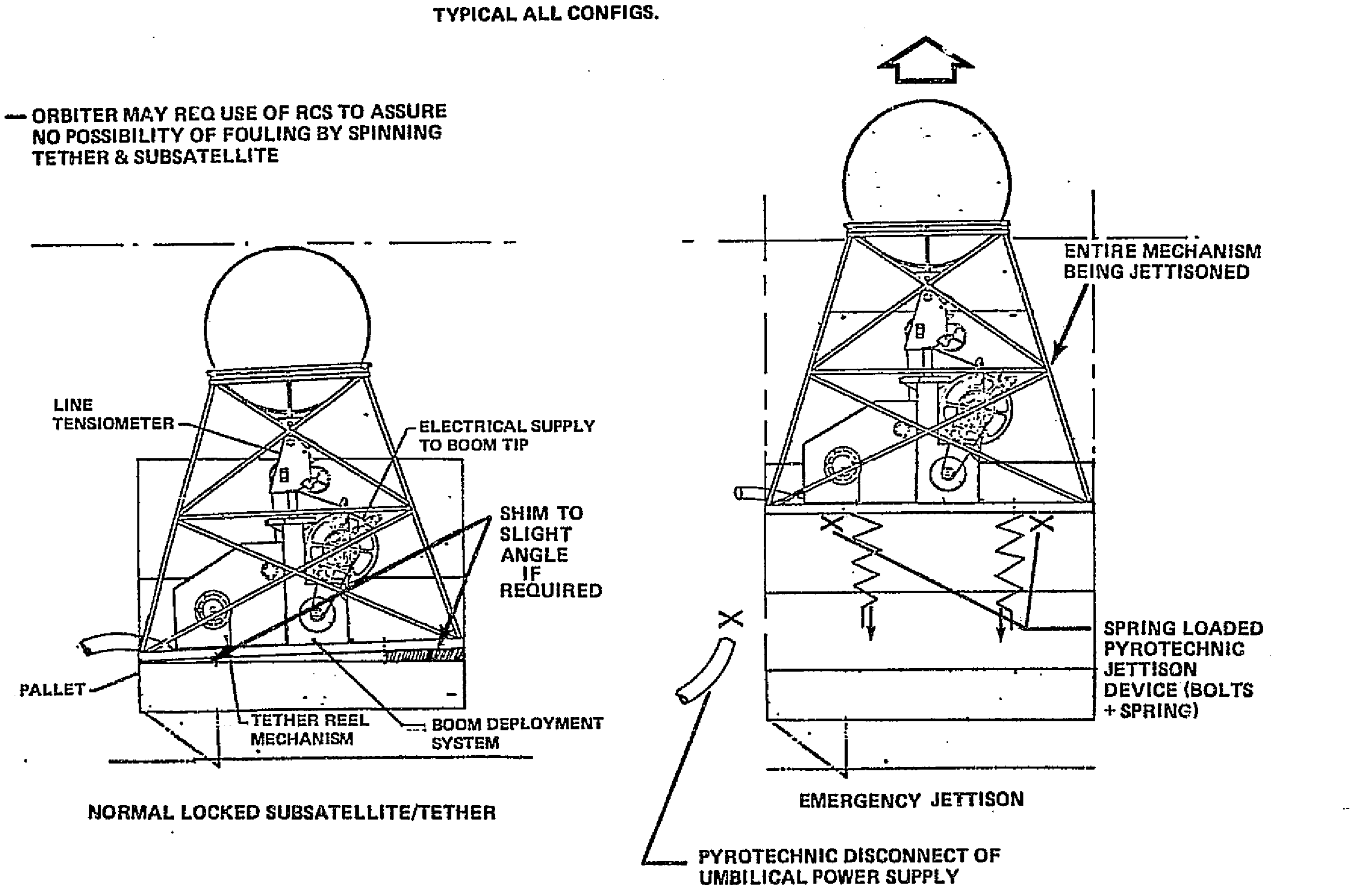


Figure 8-1. Tether study entire tether reel/boom jettison mechanism.

approach is being used in simulations which will calculate trajectories of the tether and satellite under various abort conditions. Trajectories of the Orbiter end and satellite end of a cut tether are determined in this study. This activity will also address stability and dynamics of a properly operating tether system concentrating initially on tether lengths up to 1 km.

Another study by WHF and Associates under a GSFC contract addresses perturbations of the tether system by possible disturbance forces. This study will characterize the disturbance forces and will address methods for minimizing their influence.

A third study to be accomplished is a modal analysis begun by the Dynamics and Control Laboratory at MSFC during the study reported in Reference 7. The modal analysis, combined with the finite element by SAO, will be complementary tools for addressing complex questions in dynamics and control.

These studies are part of the supporting design studies that will preceed and complement the planned definition and preliminary design studies of Phase B. Results will become available during 1977.

In addition to these studies, plans are being made to involve elements of the JSC concerned in crew and Orbiter safety. Joint effort between JSC and MSFC will insure that the system can perform with minimum risk.

9.0 TECHNOLOGY TRADE STUDIES

As discussed in the section covering tether materials, an Aramid yarn and 302 stainless steel have been selected as the best material candidates for the tether. Further research is recommended to aid in the final selection of tether materials to support predictions of tether survival probability in a micrometeorite environment and to identify alternate tether materials to satisfy particular application requirements in addition to magnetometry and gravity gradiometry.

9.1 Tether Materials

9.1.1 Aramid

Aramid is a very attractive candidate for the tether; it has strength characteristics superior to stainless steel, has low density, and can be braided to provide flexibility with low torque. It can be impregnated to provide good resistance to abrasion and protection from ultraviolet and can be manufactured in the desired lengths, thereby making splicing unnecessary. Testing is recommended to establish its performance at the expected temperature for an extended period of time (mission duration) and to evaluate its capability to be drawn over a pulley after being damaged by micrometeorites.

9.1.2 302 Stainless Steel

Based on present manufacturing techniques, the maximum length of stainless steel tether that can be produced is approximately 7.6 km. Therefore, to be an acceptable tether material, tooling must be developed to allow manufacturing much greater lengths of stainless steel tether or reliable splicing techniques must be developed. Developing and demonstrating the suitability of splicing techniques is a recommendation of this study.

A stranded stainless steel tether configuration appears to be a requirement to provide tether flexibility. Testing of various strand configurations, i.e. 1 by 7, 1 by 19, 7 by 7, etc., should be made to determine their torque characteristics and ability to be drawn over a pulley after being damaged by micrometeorites.

Coating materials should be tested for their ability to protect the stainless steel tether from abrasion or micrometeorite damage.

9.2 Survival in a Micrometeorite Environment for Aramid and 302 Stainless Steel

The predicted probability of tether survival is extremely sensitive to what assumptions are made concerning the micrometeorite, e.g. the density, velocity, shape, angle of impact on the tether, etc., and also to the assumption of what mass meteorite will fail the tether. Therefore, the following is recommended:

a. Perform a flight test on the tether.

b. Perform a three-dimensional analysis to define the effects of micrometeorites impacting the tether at various angles.

c. Investigate the feasibility of laboratory hypervelocity testing.

A statement of work for a possible hypervelocity test program has been proposed and is included in Appendix E. Spin-off of the test results will be useful to other programs such as large space structures which use large amounts of load bearing cables. An inexpensive flight test called DOTS is proposed in Appendix F and could provide flight experience regarding survivability of load bearing cables.

9.3 Low Altitude/High Temperature Materials

An area of great interest is the 120-80 km satellite altitude. At this altitude range, a slight change in altitude causes a rapid change in heating rate. There are two problems if the heating becomes excessive: the survivability of the satellite instrumentation and the potential severing of the tether (through high temperature charring). One proposed solution is the use of a leader made of a thicker material or of a different material such as a high temperature alloy which is attached to the satellite. The splice might prove to be a problem in reeling, particularly if the leader is a much greater diameter. Another possibility is to use a second tether reel housed in the satellite which operates with the high temperature portion of the tether, thereby, eliminating the need to run a splice through the reel. The leader could also be a length of the tether at the satellite end which has been coated with an ablative, or other, heat protective layer.

9.4 Reentry

The current spherical satellites designed with thick or insulated walls have a high probability of surviving reentry if the tether is severed. It is possible that expendable tethered satellites can be separated at points in the orbit which would reliably cause the satellite to impact in an ocean area. Accidental separation of the tethered satellite, such as by a micrometeorite, could cause the system to reenter over populated areas. The synthetic tether should burn up in the atmosphere if attached to a heavier object. However, a piece of tether, especially the synthetic materials, has a very poor ballistic coefficient and might slow down in velocity before burning up. It might then float to Earth and possibly be a nuisance. However, the satellite alone might cause damage if it fell in a populated area. Although these problems are not unique to tether systems, answers should be proposed and will be required as part of an environmental impact investigation.

10.0 CONCLUSIONS AND RECOMMENDATIONS

10.1 Conclusions

Based on the analysis performed to date, deployment, stabilization, and retrieval of a tethered satellite system are feasible from system design and Orbiter accommodation aspects. Accommodation of a wide range of science and other application payloads remains an open question until requirements for such payloads and detailed studies of tether system induced environment (i.e. satellite design, dynamics, and control) are completed.

The compatibility of the tether system and its associated payloads with other sortie payloads will depend on particular applications and missions. The system is apparently compatible, for example, with OFT-6 and the first multi-user sortie flight (81-2); however, tether system operations exceeding 30 to 35 h could experience timeline problems.

No new technology is required to build the tether system. All parts appear to be within the current state of the art. Additional technology will be required to extend the capabilities beyond those known for the Magsat and Gravsat programs. For example, some applications desire lower altitudes (95 to 100 km) to study reentry phenomenon which will require high strength, high temperature alloys, or some other protective means. Other applications require a conducting tether with the reel mechanism insulated from the Orbiter. Accommodation of some of the applications listed in Section 1.2 will surely require different kinds of tether systems from the current designs. However, this does not imply that such applications are infeasible.

10.2 Recommendations

Special studies in the control and dynamics of the tether system should be continued. This should be accomplished to answer the safety concerns and determine the ability to accommodate specific science applications. These studies should be timely to be able to impact subsequent design definition.

The definition of the design of the tether system should proceed within the budget and schedule constraints. Planning for such studies should include sufficient studies within the MSFC Science and Engineering Directorate to be able to manage and direct contractor efforts.

A tether system user group should be established within NASA to provide a point of contact by potential users outside of NASA, coordinate activities between Headquarters elements and elements at various NASA Centers, and provide a central point for coordinating user requirements and guidelines.

Plans should proceed to fly a proof-of-concept test of the tether system on an early Orbiter flight during the OFT series. This flight should be a reduced version of the tether system which requires nominal time to demonstrate but includes all the system elements (e.g. the tether system boom) required for the operational system.

A later demonstration of the operational system should be flown to prove the performance of the system to an altitude of 120 km. Engineering data should be taken on this flight to determine design requirements for future science on other application payloads.

Finally, the possibility of a standard module approach for the design of the satellite should be addressed. This could be the approach which would provide low cost accommodation of many science payloads.

11.0 REFERENCES

1. Eades, J. . and Wolf, Henry: Tethered Body Problems and Relative Motion Orbit Determination, Final Report. Contract NAS5-21453, Analytical Mechanics Associates, Inc., August 1972.

2. Straly, W. H. and Adlhoch, R. W.: Study of the Retrieval of an Astronaut from an Extra-Vehicular Assignment. The Marquardt Corporation, Van Nuys, California, TMC Report No. S-356, November 1, 1963.

3. Perrine, B. S.: A Method for Soft Tether Stationkeeping. NASA TM X-53643, Marshall Space Flight Center, July 31, 1967.

4. Lang, D. L. and Nolting, R. K.: Operations with Tethered Space Vehicles. NASA SP-138, Gemni Summary Conference, 1967.

5. Colombo, G., et al.: Shuttle-Borne "Skyhook": A New Tool for Low-Orbital-Altitude Research. Proposal by Smithsonian Institution Astrophysical Observatory, September 1974.

6. Rupp, C. C.: A Tether Tension Control Law for Tethered Subsatellites Deployed Along Local Vertical. NASA TM X-64963, Marshall Space Flight Center, September 1, 1975.

7. Baker, W. P., et al.: Tethered Subsatellite Study. NASA TM X-73314, Marshall Space Flight Center, March 1976.

8. Kulla, P.: Dynamics of Tethered Satellites. TMM/76-119/PK/AVS, European Space Agency, April 1976.

9. Payload Description, Volume I, Automated Payloads, October 1973.

10. Earth Physics Satellite Gravity Gradiometer Study, JPL X74-10307, May 17, 1972.

APPENDIX A

STUDY GUIDELINE SUMMARY

1. General

a. Launch Date: For schedule and costing purposes, a demonstration and proof of concept developmental flight will be scheduled for OFT-5 or OFT-6. The first flight opportunity for the operational system will be multi-mission multiapplication flight 81-2 in September 1981.

b. Objectives: Develop a tether system which will increase the potential for Shuttle utilization in a wide variety of applications but initially concentrating on Earth gravity and magnetic field mapping. An intermediate objective is to prove the system concept by means of a demonstration test flight.

c. Mission Duration: The operational system should be designed for a 6.5 day mission. The demonstration test flight should require between 5 and 10 h. The time required to test the operational system should require approximately 30 h.

d. Orbit Requirements: The operational system should be designed to operate with orbital inclination from 0° to 90°. The operational system will be designed to support a satellite at 120 km altitude or lower from an Orbiter at an altitude of 200 km.

e. Launch Site: The operational system will be capable of being launched from the optimum location to carry out the experiment program objectives for each mission. The two flights scheduled in this study will be launched from KSC.

f. Crew Size: Will be established based on operations to be accomplished for a 12 h shift, two-shift per day schedule.

g. Deployable Hardware: All items which deploy will be equipped with release/jettison mechanism to insure that the cargo bay doors can be closed.

h. Maintenance: The tether system will be designed so that planned or contingency maintenance can be performed by either one or two crewmen performing extravehicular activity (EVA) for safing operations. Flight-line and depot-type maintenance will be performed on Earth.

i. Number of Systems: One system will be scheduled for the demonstration flight and a second system for the operational flight. Two additional tethers will be procured for spares. The demonstration flight system should be designed to be refurbished for a second operational system.

j. Satellite Retrieval: The study goal is to determine if the satellite can be retrieved from 120 km attitude to the Orbiter at 200 km altitude in less than 24 h.

k. Payload Mass: The tether system will be designed to support a satellite mass of 200 kg at a distance of 100 km. The demonstration test flight system shall be capable of supporting 500 kg test satellite at least 2 km from the Orbiter.

l. Satellite Design: Normally, the operational satellite design will be provided by the user. However, this study will address the design of a satellite carrying minimum instrumentation for a demonstration of the system deployed to an altitude of 120 km. An additional two designs for the 1 to 2 km long demonstration system satellite will be considered — a specially designed satellite carrying a television camera and a Lageos simulator. The cost of these three demonstration satellite approaches will be determined.

m. Consumables: Will be sized for a 7 day mission for the operational flight system. The demonstration flight consumables will be sized for 30 h for the 120 km altitude mission or 10 h for the 1 to 2 km altitude mission.

n. Launch Mode: The launch mode for the tethered subsatellite is by the Shuttle, with all hardware to be carried in the Shuttle cargo bay. The tether system will be pallet-mounted with the control and display functions in the aft flight deck.

o. EVA: Baseline operation primary mode will be from inside the Orbiter crew compartments; however, contingency EVA capability will be provided.

p. Design Concepts: Maximum utilization of proven design concepts from previous programs will be used to enhance reliability of critical systems and to maintain lower overall costs.

q. Reliability: The design will be such that mission critical systems involving possible loss of life will not have single failure points.

r. Docking: There are no known docking requirements for operations. Retrieval of the tethered satellite is a function of the tether system and will not involve Orbiter docking systems.

s. Systems Test: Test requirements will be determined in this study. A test flight of the operational system is to be planned before the user operational flight. This operational flight of the tether system might accommodate a test flight of a user payload. A demonstration flight will be planned for OFT-5 or OFT-6 which will prove the system concept. Orbital operation verification capability will be incorporated in the hardware design.

t. Coordinate System: The coordinate system for defining mass locations will be the same as used on the Orbiter.

u. Units of Measure: The system of units to be used in this study will be the international system of units (SI) with customary units in parentheses, except for those measurements related to the Orbiter systems, where English units are used primarily.

v. Protection Devices: These will be incorporated where needed to avoid all credible hazards to assure safe termination of the mission.

w. Electrical/Electronics Components: The tether system electrical/electronics components shall meet the design intent of MIL-STD-461 and MIL-STD-462 and applicable portions of MIL-E-60510.

2. Scientific Equipment

a. Satellite Design Requirements: The satellites designed in the three approaches will accommodate science and/or engineering measurement requirements defined during the course of the study.

b. Science Payload: The Magsat and Geosat programs have been identified as early users of the tether systems. The design requirements for satisfying the objectives of these programs will be incorporated into the study as they are identified. In lieu of science requirements, strawman science requirements will be derived from earlier studies.

c. Any computer operation will be performed by the tether system computer facilities.

d. The tether system data management subsystem shall be capable of receiving data from the satellite and tether system simultaneously.

e. Satellites will be designed to communicate directly with the tether system. Satellite control will be primarily from the Orbiter aft flight deck.

3. Electrical Power System

a. Circuit Protection: Shall be provided on electrical circuits.

b. Primary Power: Primary electrical power shall be provided by the Space Shuttle Orbiter.

c. Secondary Power: The tether system shall provide peaking batteries and associated charge controls to meet transient or supplemental load requirements that exceed primary power source capabilities.

d. Emergency Power: Sufficient emergency electrical capacity shall be provided to assure safe shutdown of equipment and egress of personnel in the event of primary power failure. Emergency systems shall be protected from reverse current by the primary power source.

e. Power Conditioning: The tether system shall provide the necessary power conditioning and distribution equipment to satisfy the requirements for regulated dc power and single- and three-phase ac power.

f. Distribution: The tether system shall furnish the necessary connectors, cable distribution, and control equipment required for the mission equipment that may be located within external pressure modules or on pallets.

g. Grounding: The structure shall not be used for distribution independent of the Orbiter electrical power system.

4. Sensing and Control System

a. Full control of all tether systems and instruments will be accomplished from within the Orbiter crew compartment unless there are significant and specific advantages in performing functions at other locations.

b. Attitude stabilization of Orbiter in general will be sufficient for operation of the tether system. Additional pointing and stabilization requirements will be met with experiment and support system hardware as required.

5. Communication and Data Management

a. The Tracking and Data Relay Satellite System (TDRSS) is to be considered in this study but is not mandatory for tether system operations.

b. Communication between ground and orbit will be via the Shuttle communications system.

6. Structures

a. A safety factor of 1.4 will be applied to limit loads to obtain ultimate loads. A safety factor of 2.0 will be applied to the pressure in pressurized volumes.

b. Acceleration forces and crash landing load factors shall be in accordance with Johnson Space Center document JSC 07700, Vol. XIV, Rev. C.

7. Thermal/Environmental Control

Thermal control will be accomplished by insulation, reflective absorbtive coatings, and appropriate cold/heat plates. Coolant shall be obtained from the Orbiter.

8. Controls and Displays

a. Mission critical and safety critical payload parameters will be monitored on caution and warning displays in Orbiter.

b. Controls and displays for the tether system will be located at the Orbiter aft flight deck.

APPENDIX B

TETHERED MAGNETIC FIELD EXPERIMENT: PRELIMINARY DISCUSSION OF INSTRUMENTATION SPECIFICATIONS

A magnetic field experiment imposes a severe problem on any mission because of the limitation on extraneous fields at the sensor. For the tethered experiment, fields at the sensor from the Shuttle, the tether, and all electronics etc. must be less than 3 gamma. Normally this requires removal of the magnetometer sensor from the rest of the spacecraft, including the magnetometer electronics, via a boom. This may be necessary on the tether also. At present, sufficient nonmagnetic electronics is not available and would be expensive to develop.

The basic scalar magnetometer has approximately the following characteristics:

Weight: 4 kg

Power: 9 W

Temperature requirement:

a. Sensor must be controlled at 45° ± 2°C, while dissipating 4.5 W

b. Electronics must be less than 85°C.

Size:

a. Sensor: 30 cm diameter sphere

b. Electronics: 300 cc

Data Rate: 100 bps, minimum

Acceleration: Can withstand (while nonoperating) 22.5 g in the thrust axis and 6.0 g in the lateral axis for a duration of 3 min.

It is not clear that the magnetometer can be designed separate from the entire package at the tether's end. Temperature and magnetic cleanliness requirements seem to dictate an all-inclusive feasibility study and design. The first step in this process is contemplated under RTOP 681-01-01 in FY77.

At present the mission concept calls for obtaining data continuously during each Shuttle flight on which the experiment is flown. The position will need to be known to 60 m vertically and 300 m horizontally.

At present the feasibility of vector measurements at the tether's end is very low. This is unfortunate but the scalar data are primary data sources for anomaly studies and therefore are very desirable even apart from directional information. In the event it proves feasible to determine the attitude of the package accurately at the end of the tether, it would be desirable to include a vector magnetometer. The needed attitude accuracy is approximately 30 arc s at the sensor.

Typical vector magnetometer characteristics are:

Weight: 5 kg

Power: 6 W

Size:

a. Sensor: 30 cm sphere

b. Electronics: 20 by 15 by 15 cm.

Data Rate: 200 bps, minimum.

Note that data rates are for the magnetometers only. They do not include incidental housekeeping or, in the case of the vector measurements, attitude information.

APPENDIX C

GRAVITY GRADIOMETER INSTRUMENTATION SPECIFICATIONS

Gradiometer type — Rotating cruciform

Diameter — 3.66 m

Length — 1.5 m

Weight — 30 kg

Electrical power (operating) — 40 W

Data:

- Science — 200 bps
- Housekeeping — 200 bps
- Command input — 1024 bps

Thermal control — 303 ± 0.1 K

EMI limits — Adhere to MIL STD 461A

Cleanliness class — 100 000

Tolerable acceleration level at instrument — 10^{-11} cm/s^2

Position tracking accuracy (Shuttle reference) ±10 m x, y, z

Position tracking accuracy (geocentric reference) ±100 m, 2 s

APPENDIX D

DYNAMIC ANALYSIS OF A TETHERED SUBSATELLITE, SUPPLEMENTARY DATA

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of: ED15-76-40 October 8, 1976

TO: PD12/C. Rupp

FROM: ED15/Z. Galaboff, Systems Dynamics Laboratory

SUBJECT: Dynamic Analysis of a Tethered Subsatellite, Supplementary Data

Reference is made to NASA TM X-73314, Tethered Subsatellite Study, dated March 1976.

In accordance with your request for supplementary data to Section II of the reference, additional data are supplied. The figures in Section II present the dynamic behavior of a tethered subsatellite for a deployed length of 100 km. In particular the figures showed the pitch angle, θ, versus time, roll angle, ϕ, versus time, in-plane motion (x-z plot), out-of-plane motion (y-z plot), equatorial motion (x-y plot), tether elongation versus time, and tether tension versus time. Each figure contained two plots: one for a circular orbit (case A) and one for an eccentric orbit with ΔH=10 km (case B). Corresponding plots for tether lengths of 90 km and 80 km are presented in figures 1-7 and 15-21, for equatorial orbits, and, figures 8-14 and 22-28, for polar orbits, respectively.

Zachary J. Galaboff

Zachary J. Galaboff

APPROVAL:

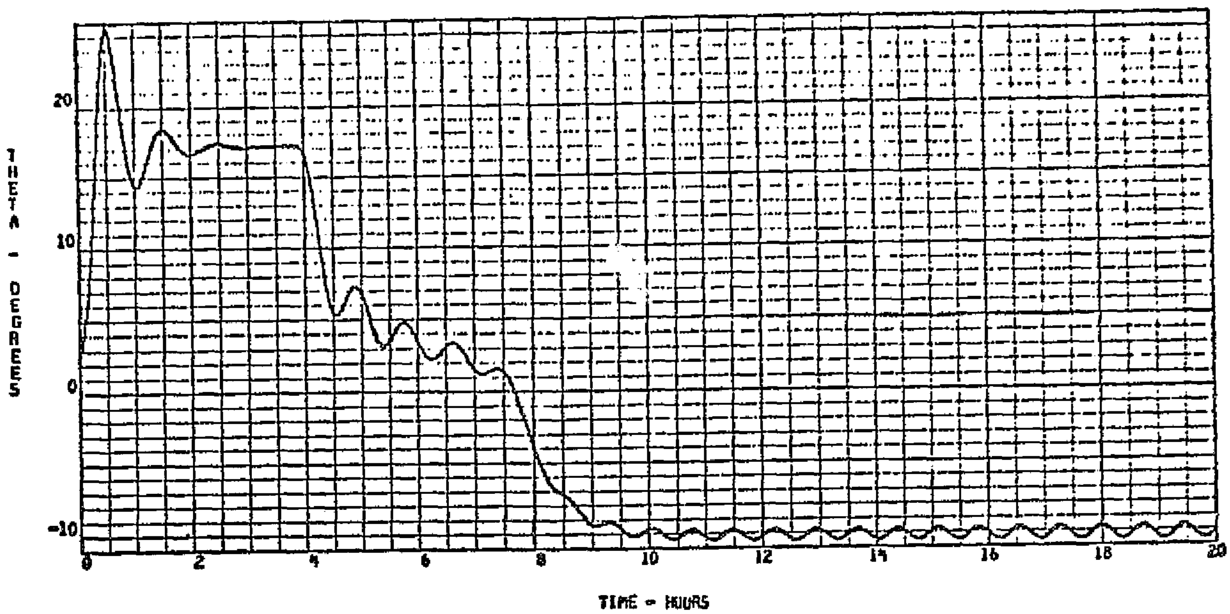
J C Blair

James C. Blair
Chief, Control Systems
Division

Enclosures

cc:
ED01/Mr. Sisson
ED01/Mr. Rheinfurth
ED11/Dr. Blair
ED15/Mr. Buchanan

CASE A



CASE B

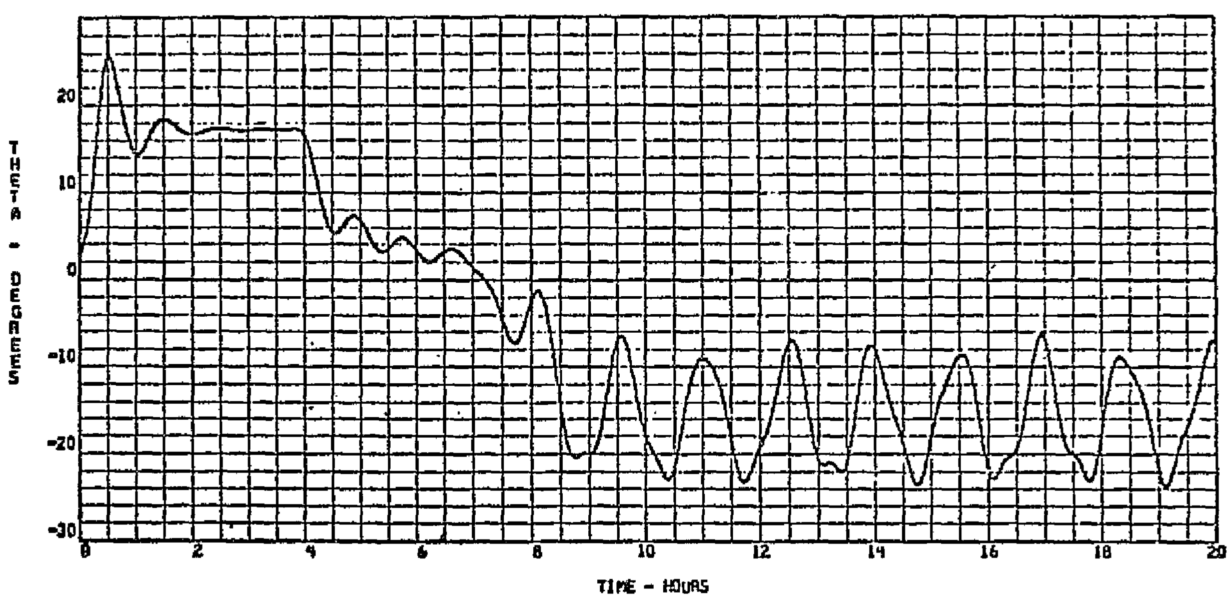
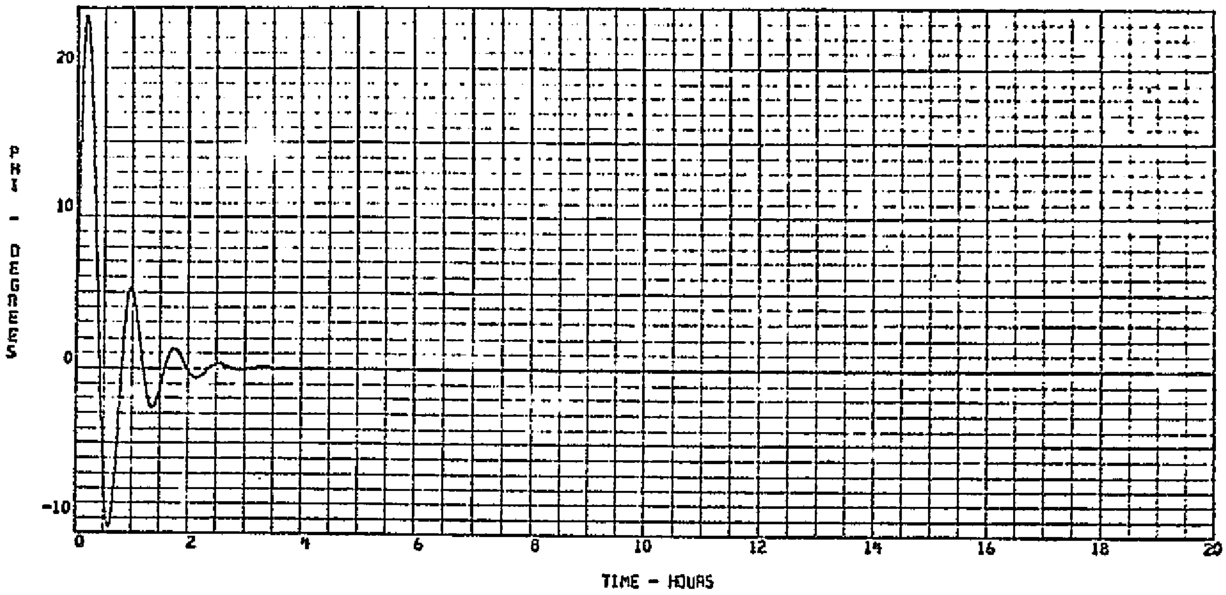


Figure D-1. Pitch angle versus time, equatorial orbit, 90 km tether.

CASE A



CASE B

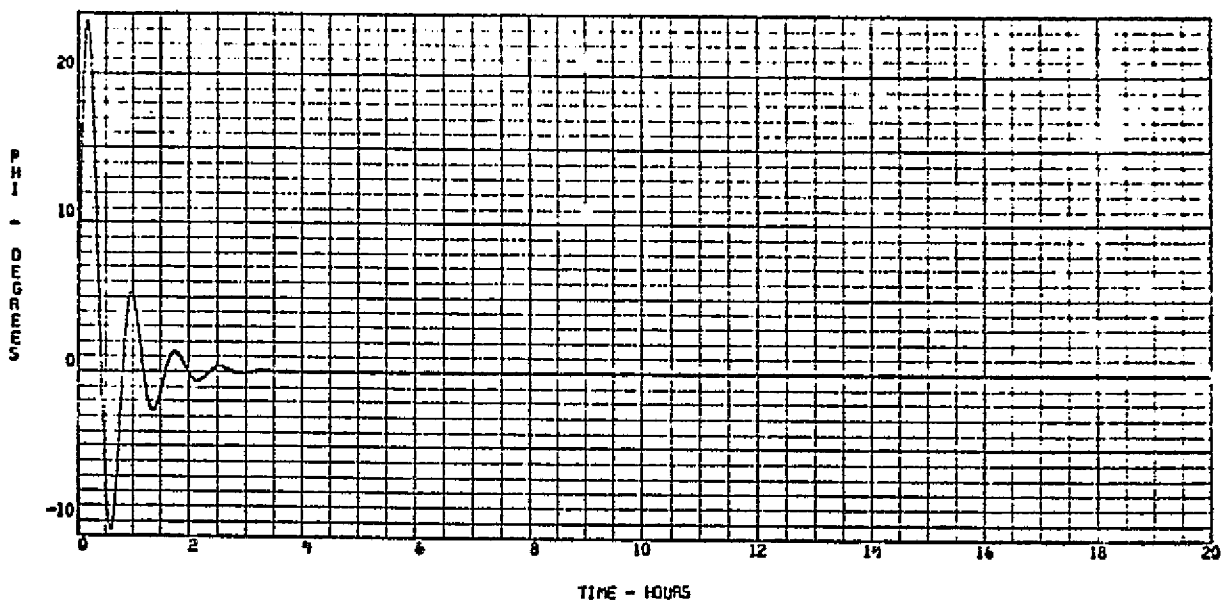
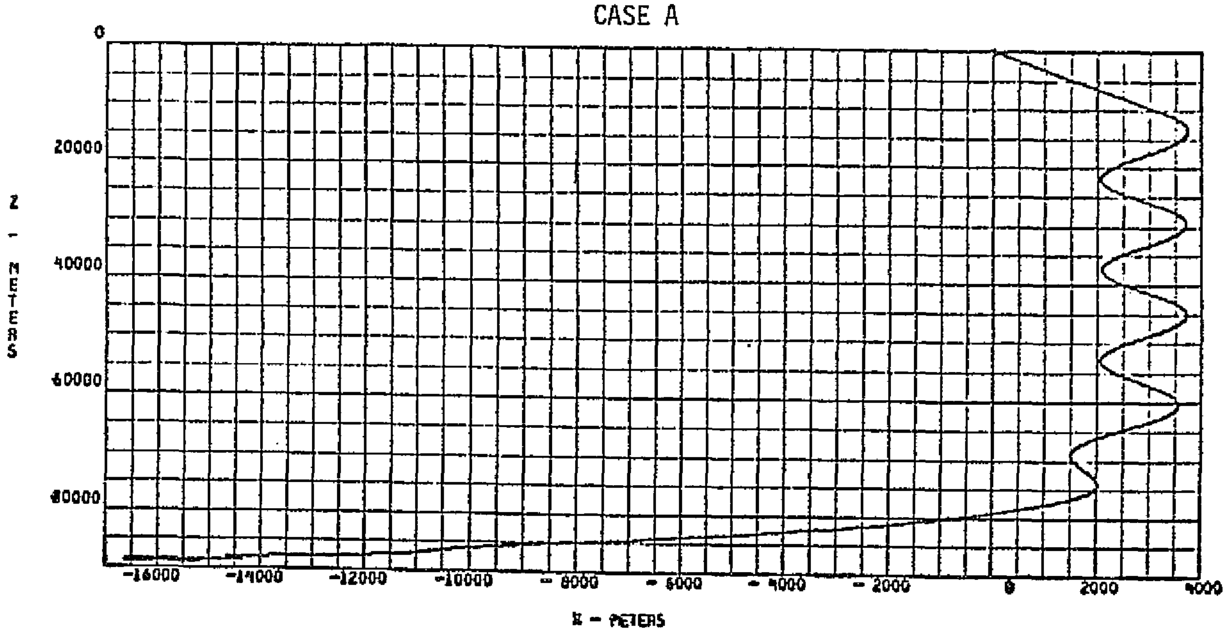


Figure D-2. Roll angle versus time, equatorial orbit, 90 km tether.

CASE A



CASE B

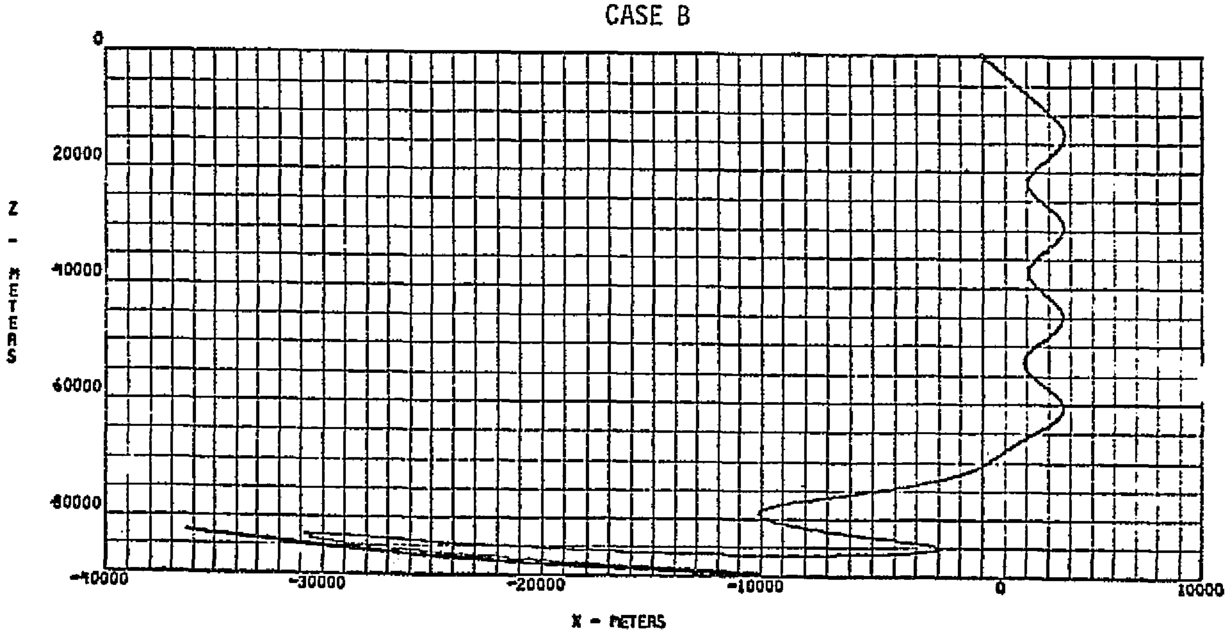
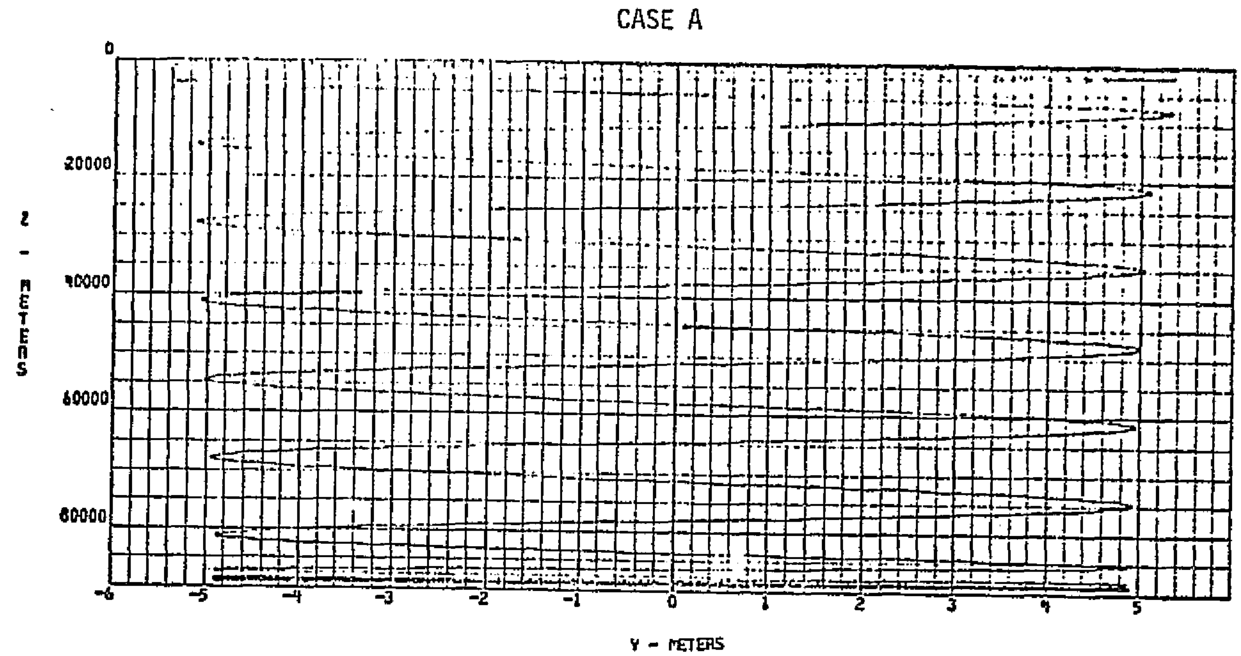


Figure D-3. In-plane motion (X-Z plot), equatorial orbit, 90 km tether.



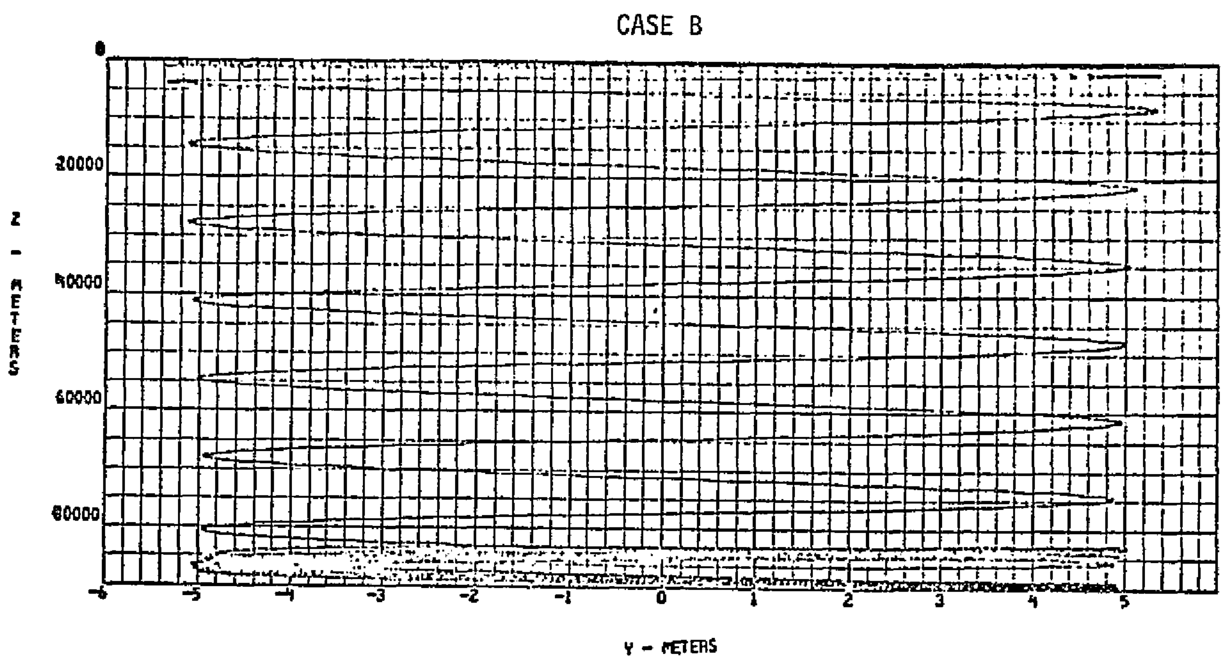
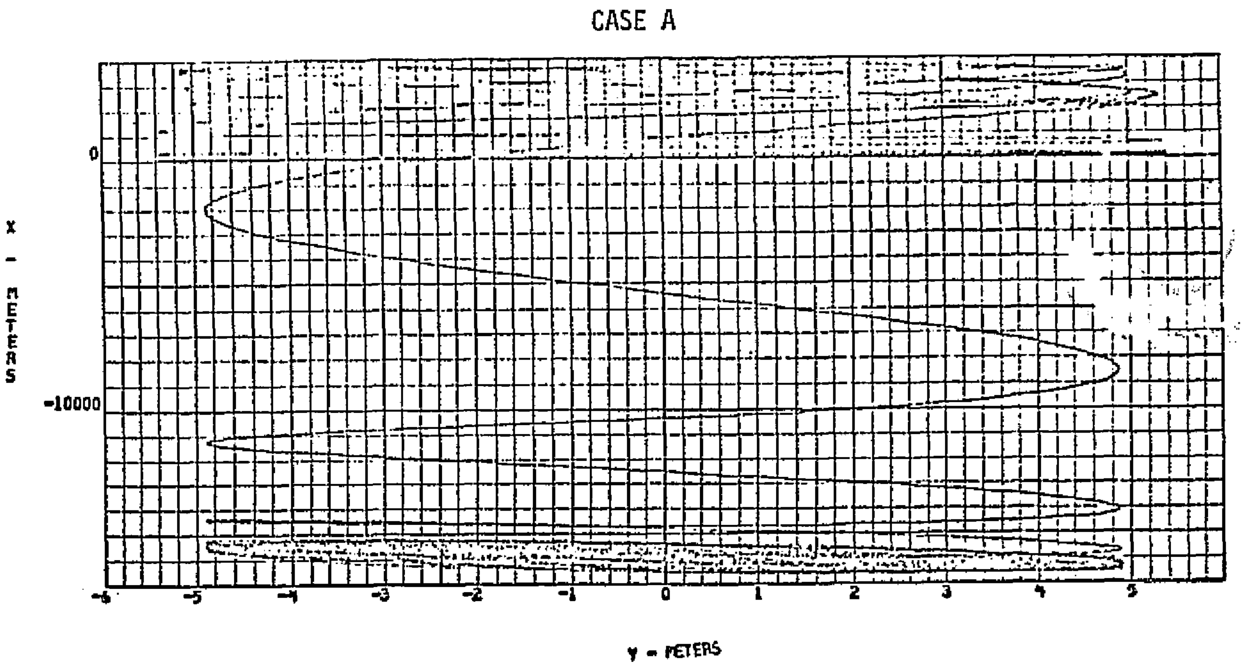


Figure D-4. Out-of-plane motion (Y-Z plot), equatorial orbit, 90 km tether.



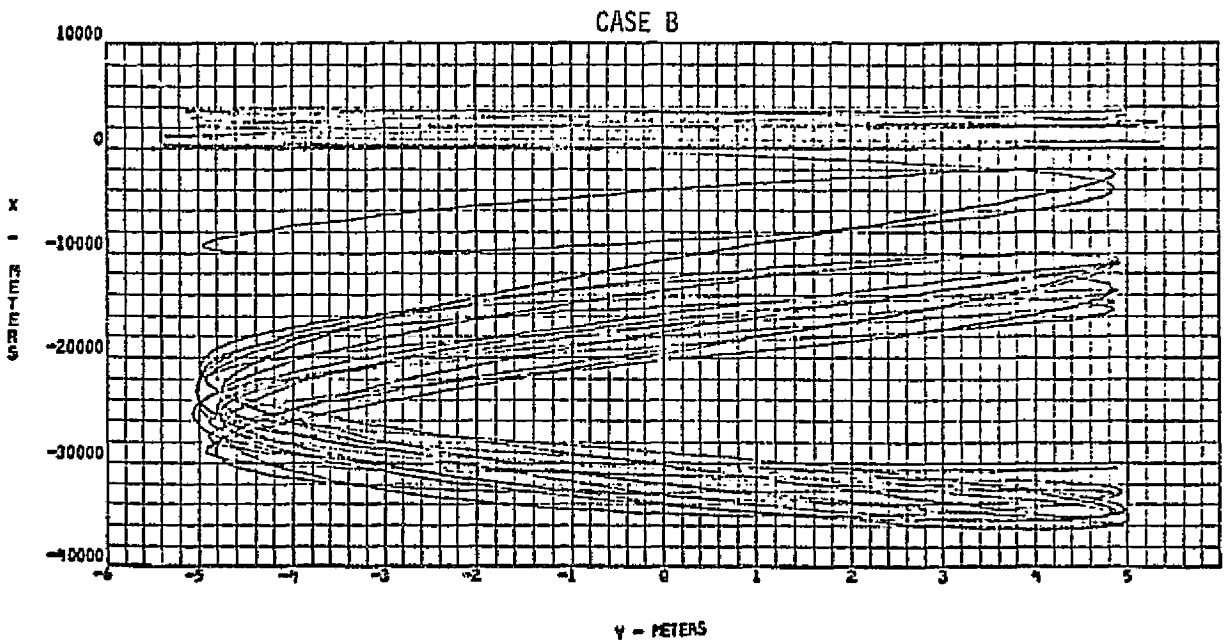
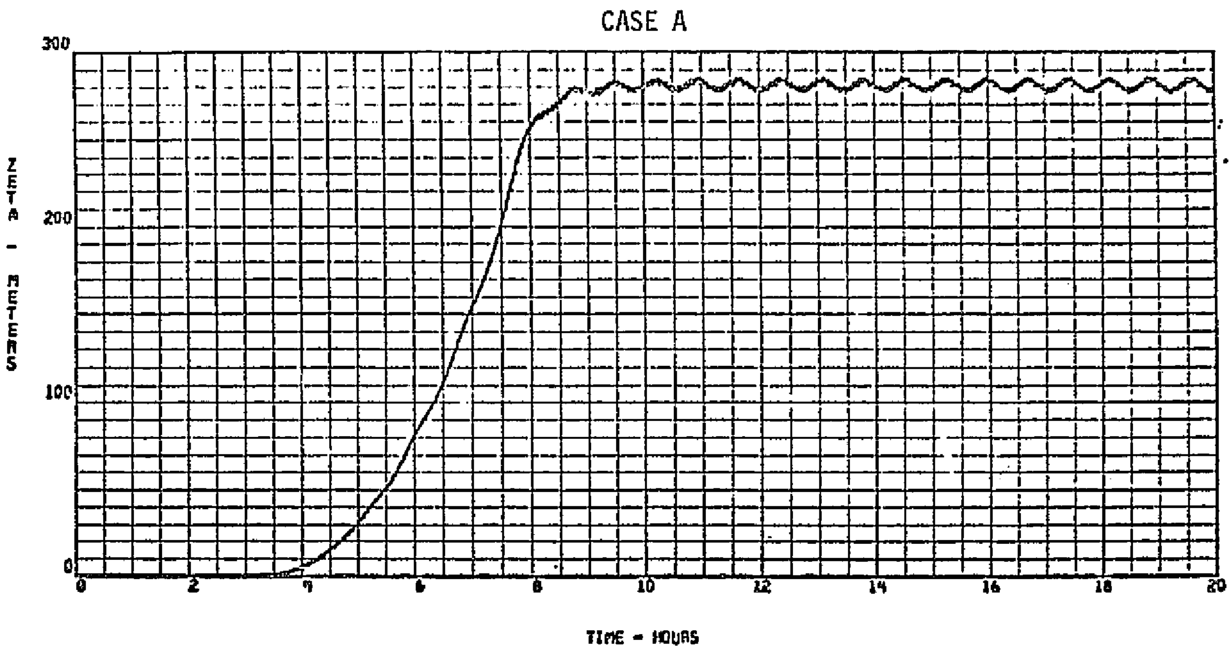


Figure D-5. Equatorial motion (X-Y plot), equatorial orbit, 90 km tether.



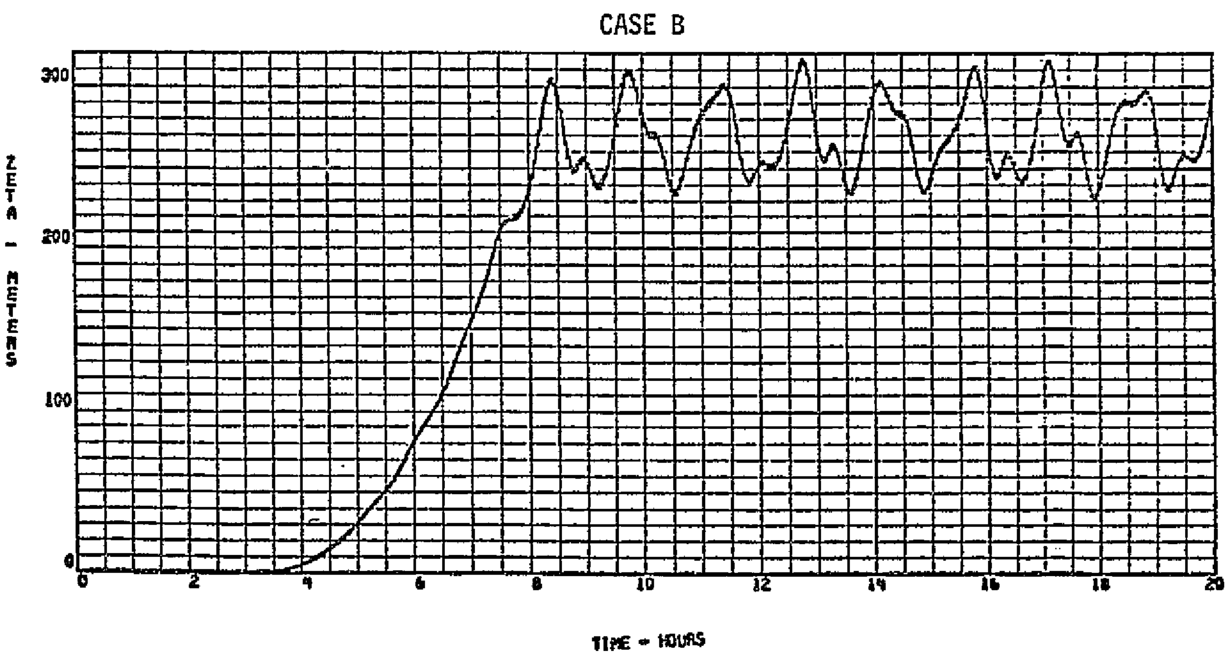


Figure D-6. Tether elongation versus time, equatorial orbit, 90 km tether.

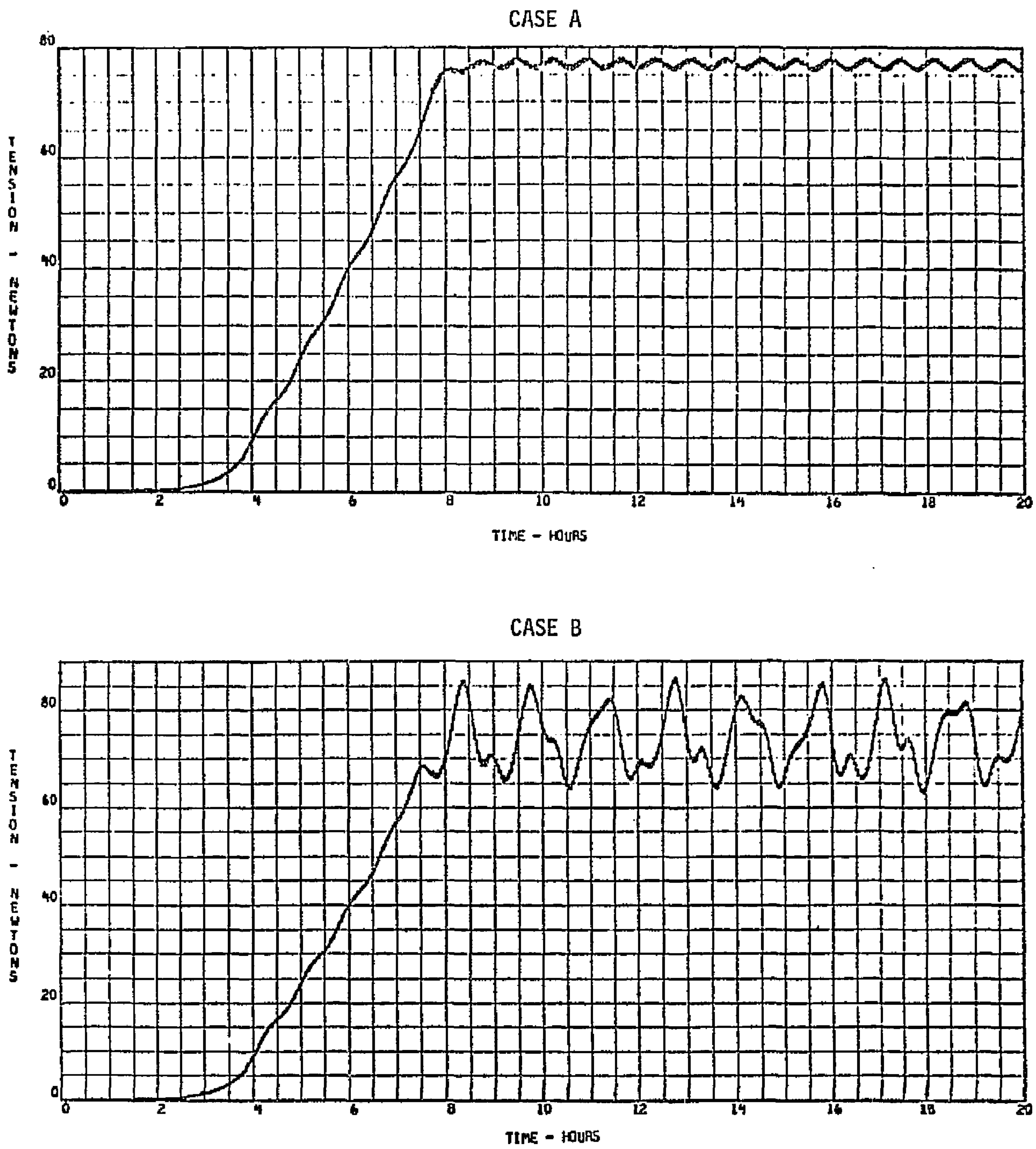
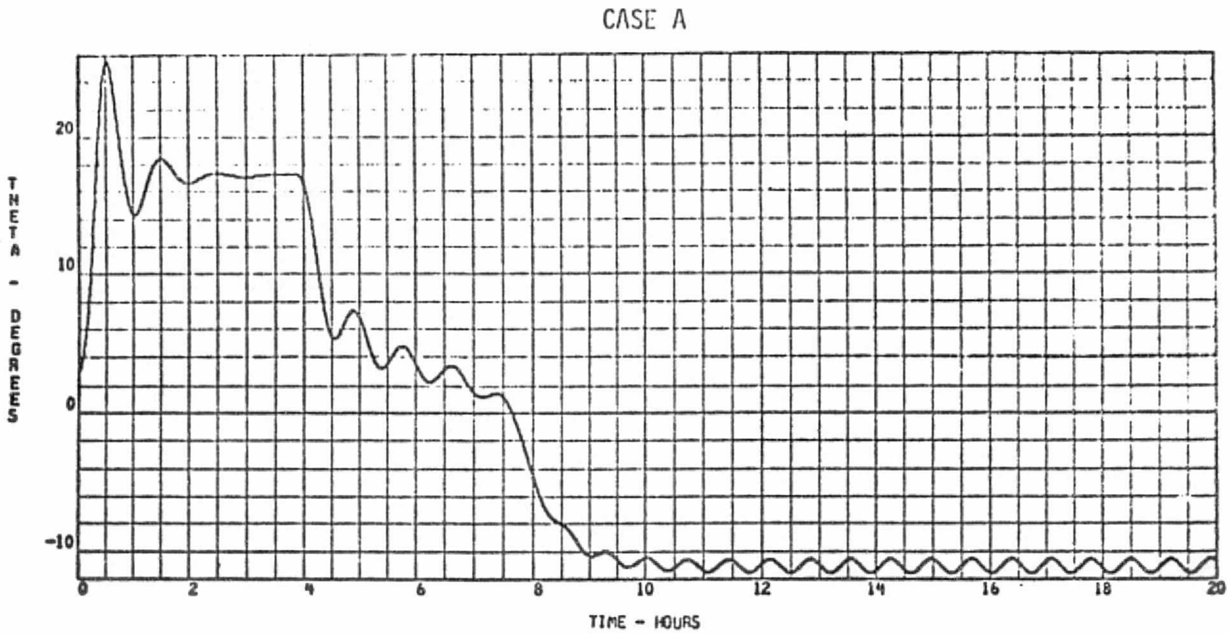


Figure D-7. Tether tension versus time, equatorial orbit, 90 km tether.



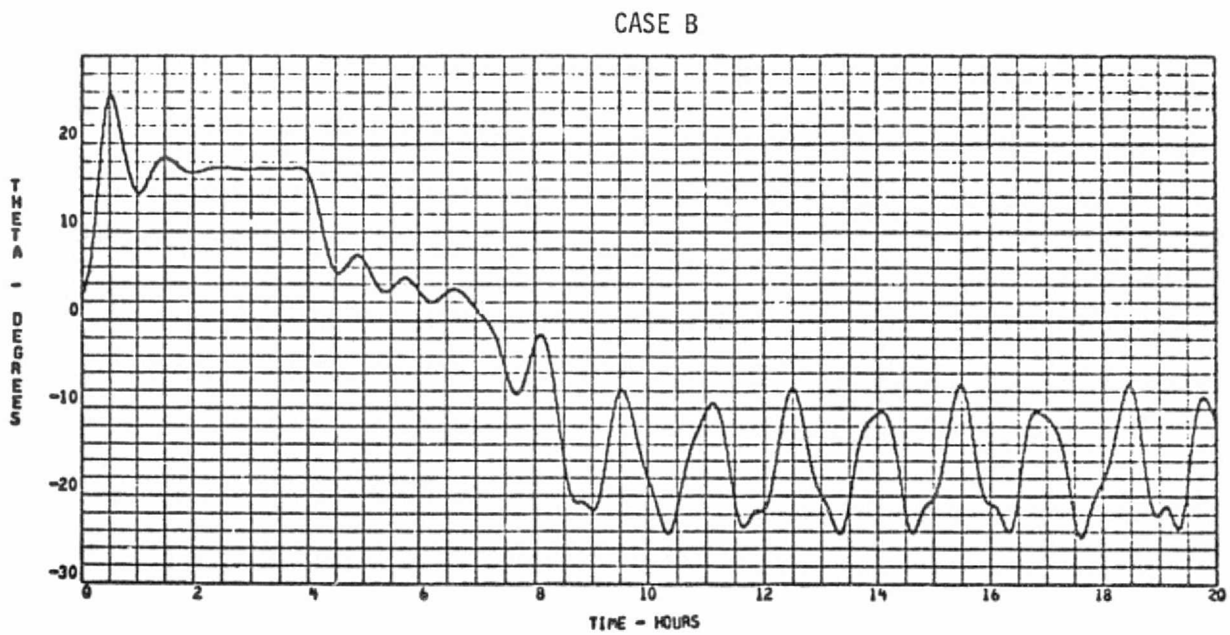
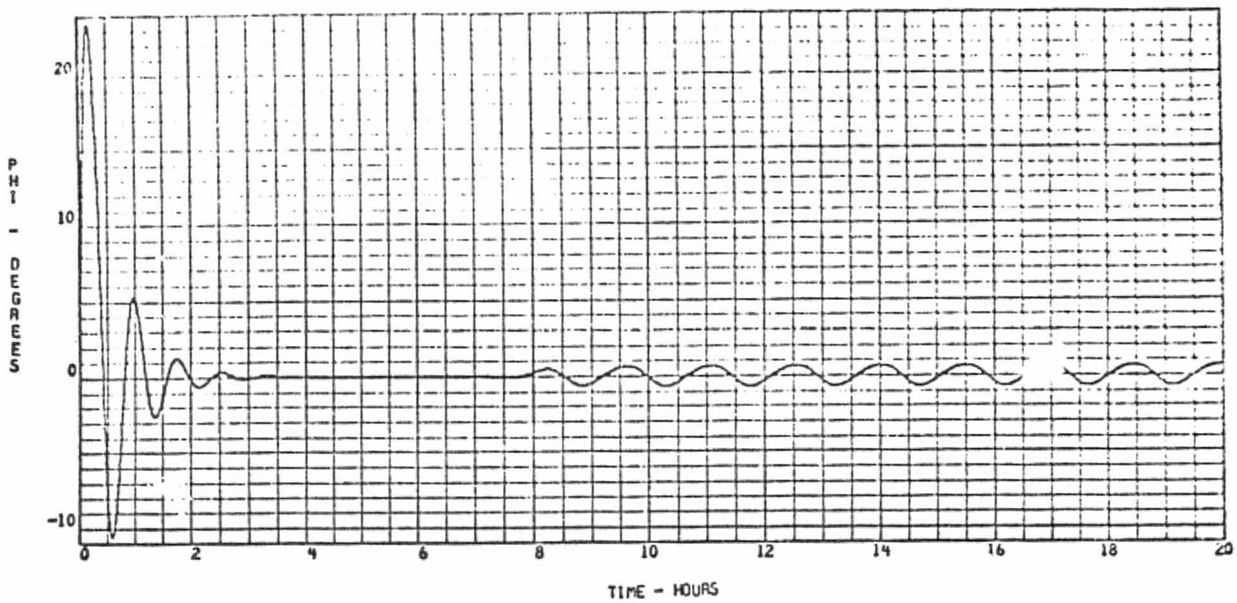


Figure D-8. Pitch angle versus time, polar orbit, 90 km tether.

CASE A



CASE B

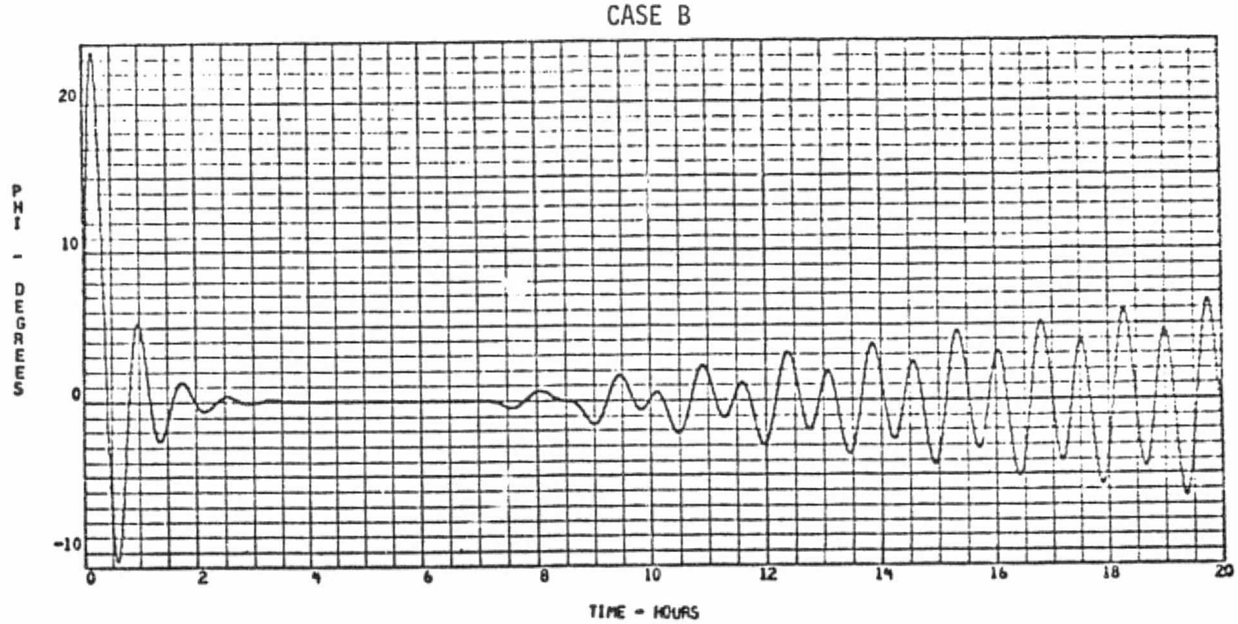


Figure D-9. Roll angle versus time, polar orbit, 90 km tether.

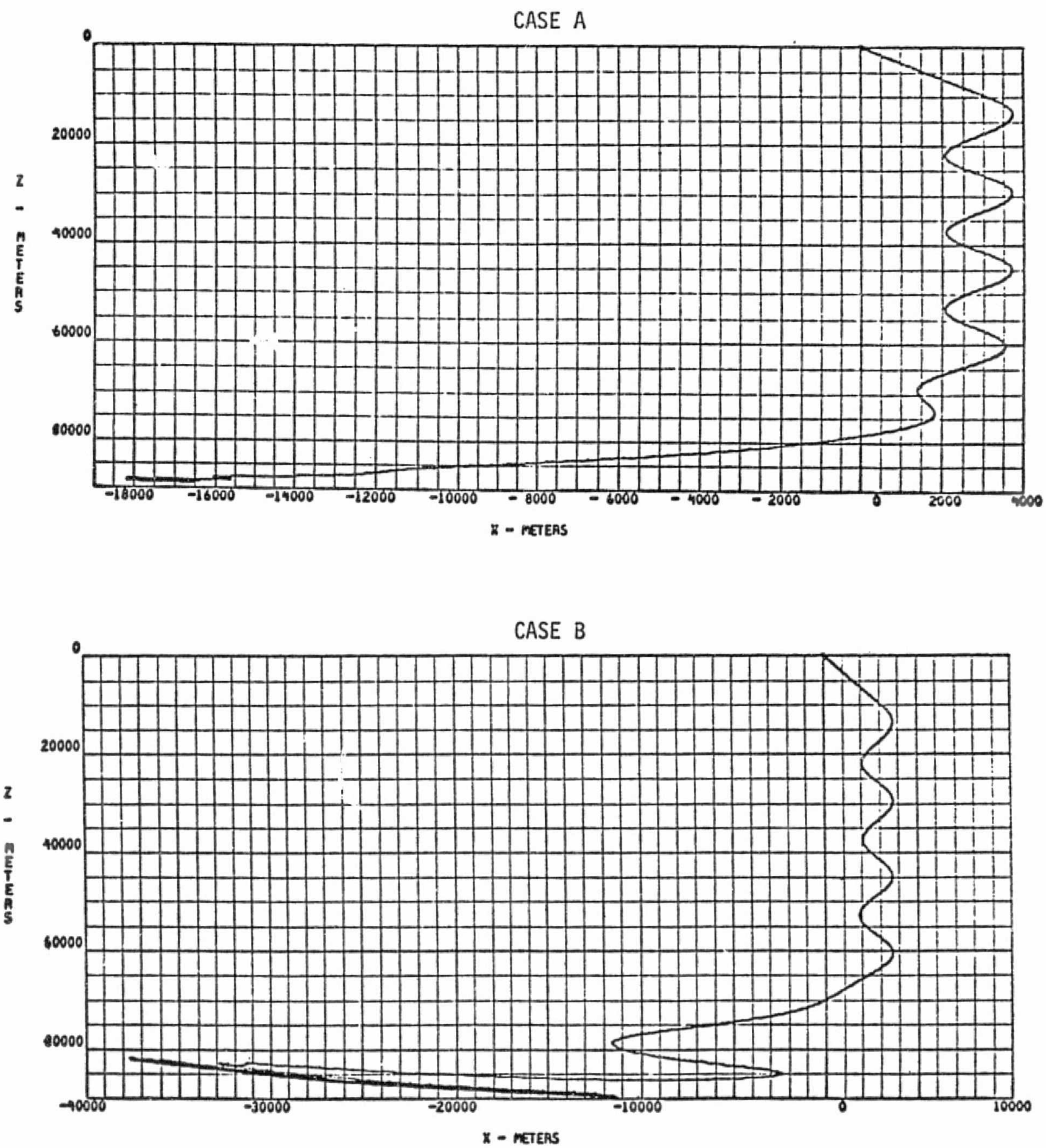


Figure D-10. In-plane motion (X-Z plot), polar orbit, 90 km tether.

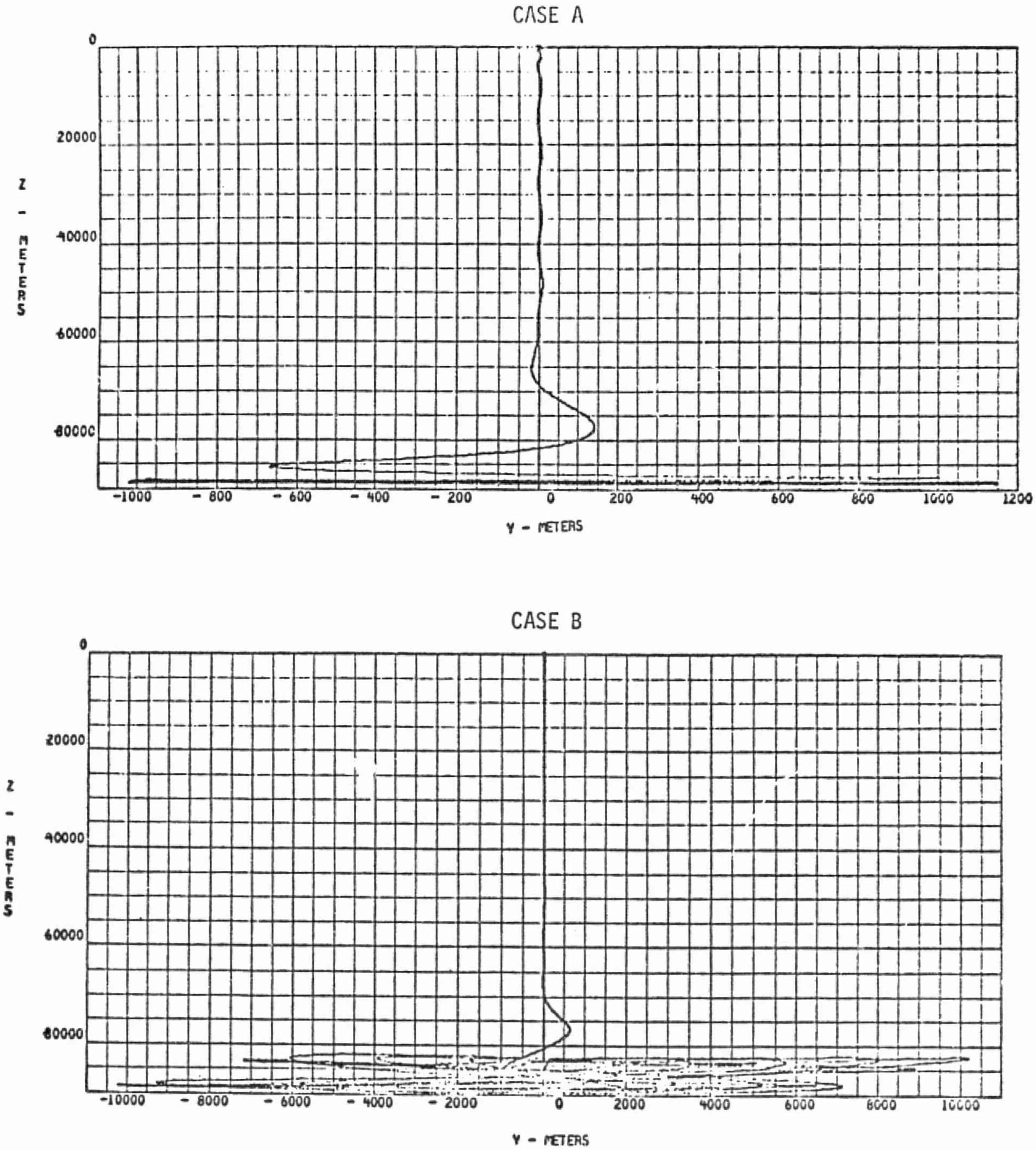
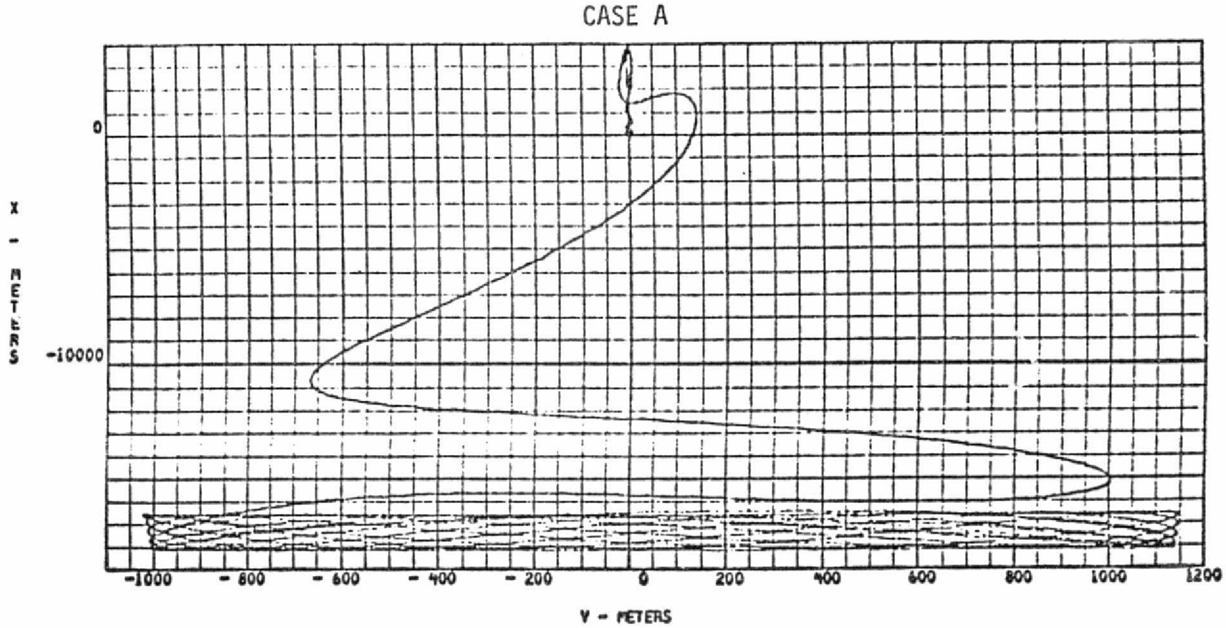


Figure D-11. Out-of-plane motion (Y-Z plot), polar orbit, 90 km tether.



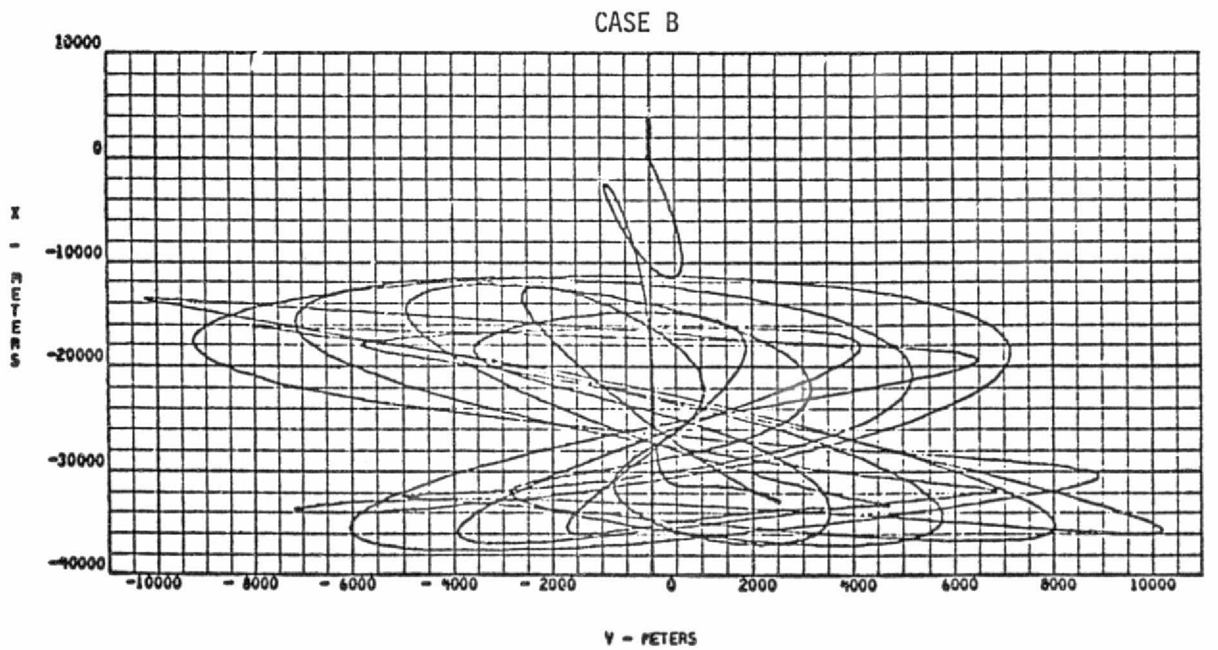
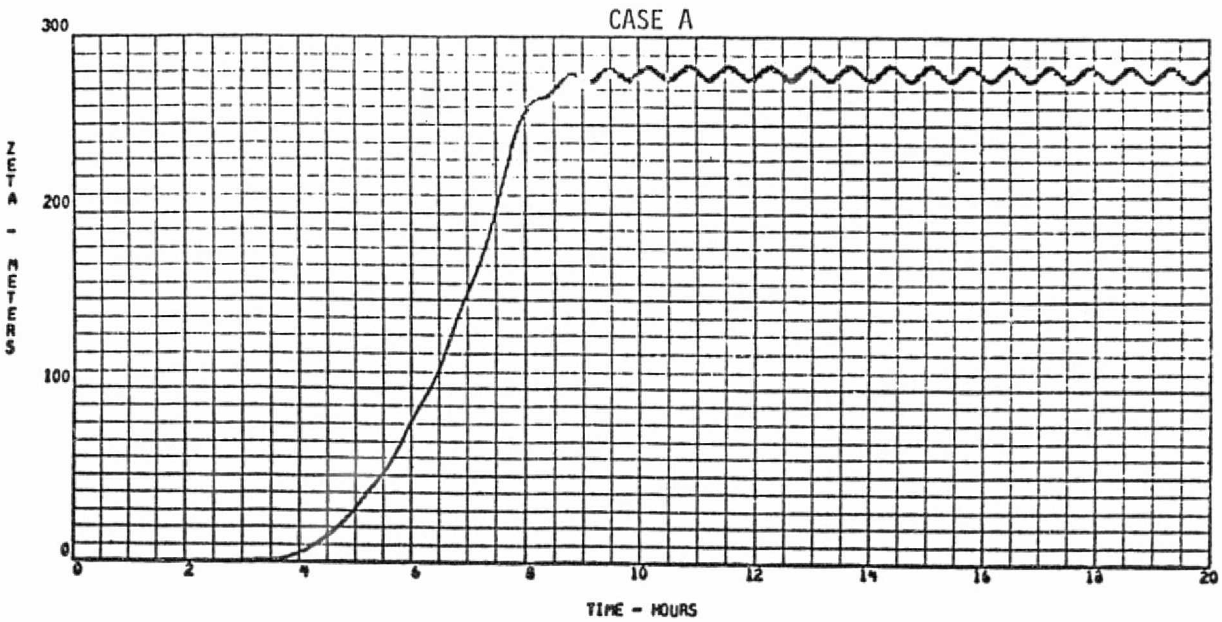


Figure D-12. Equatorial motion (X-Y plot), polar orbit, 90 km tether.



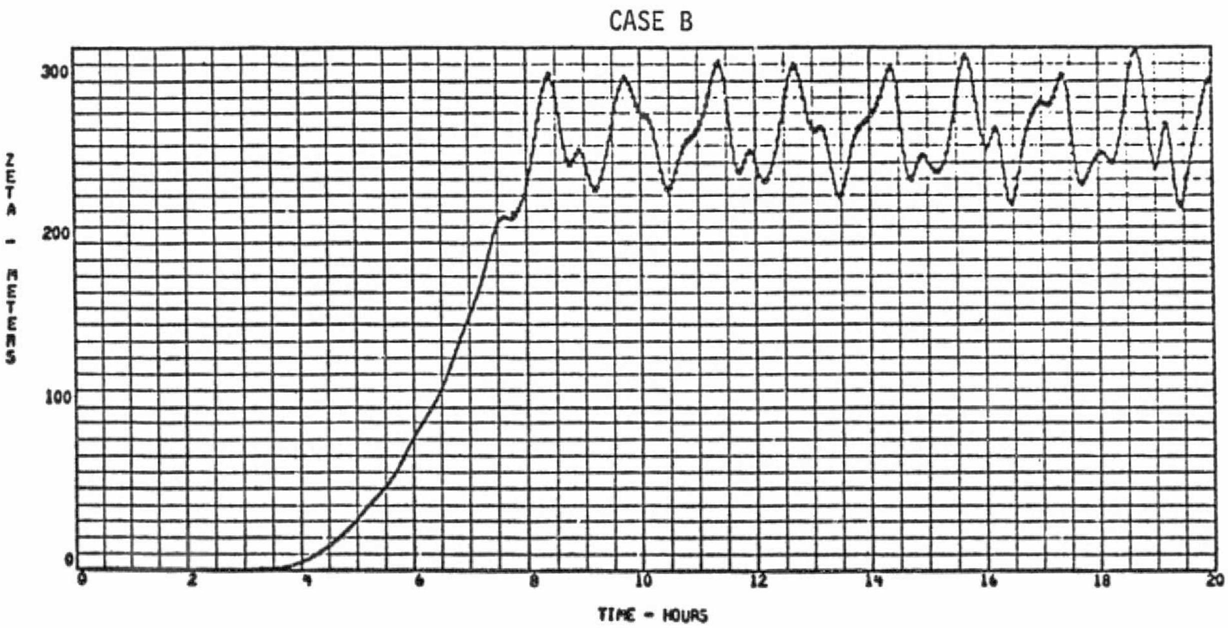


Figure D-13. Tether elongation versus time, polar orbit, 90 km tether.

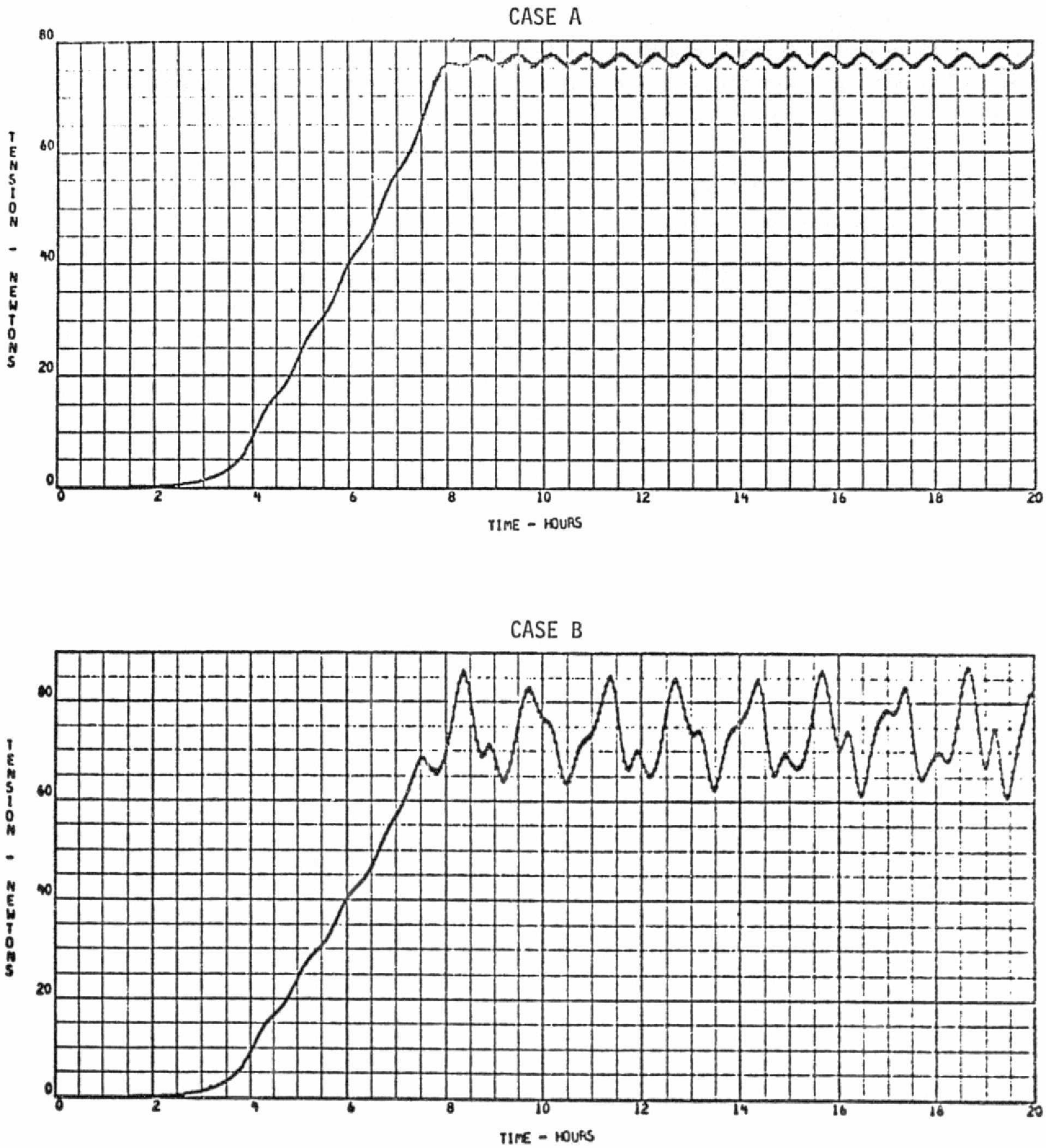


Figure D-14. Tether tension versus time, polar orbit, 90 km tether.

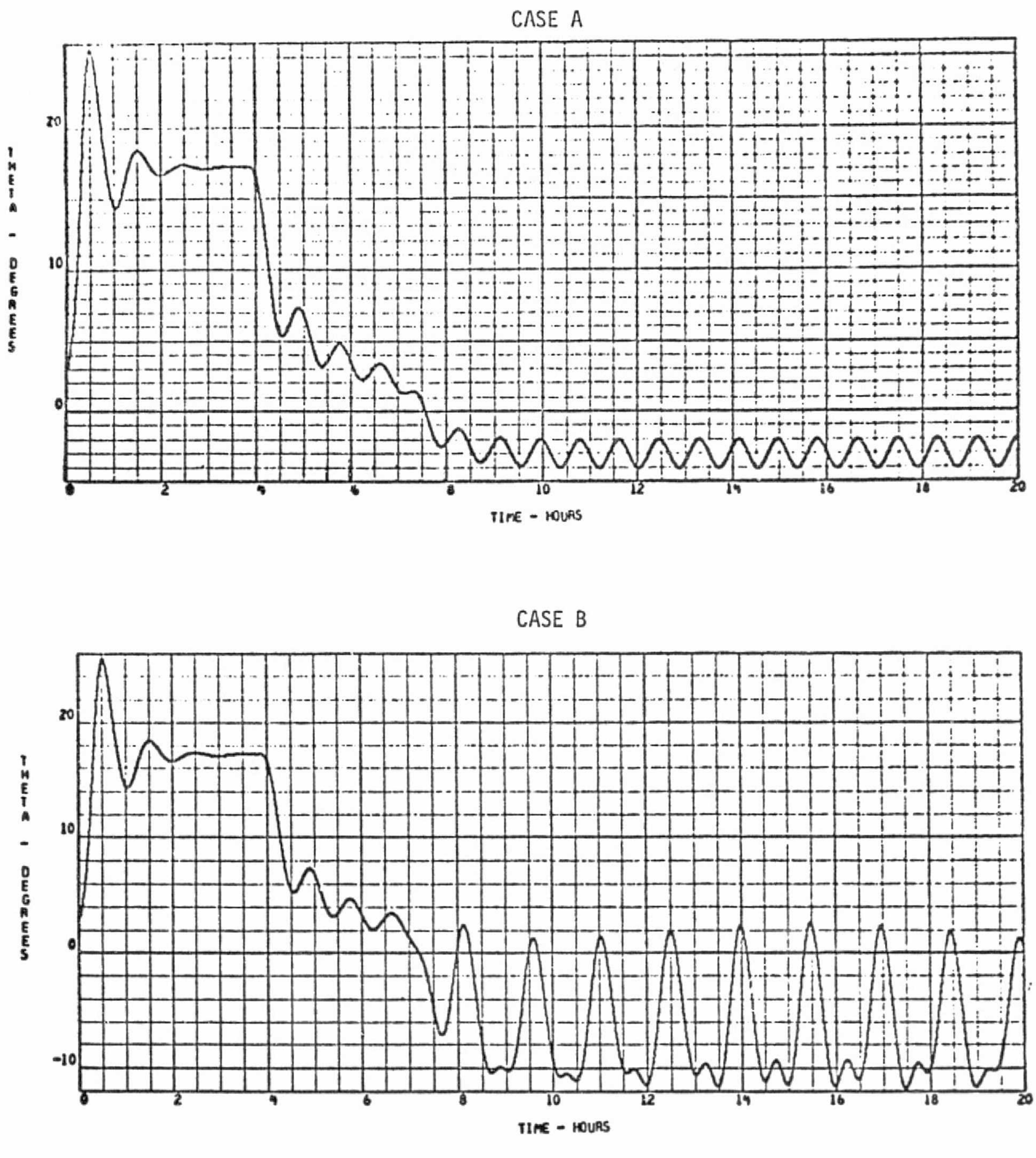
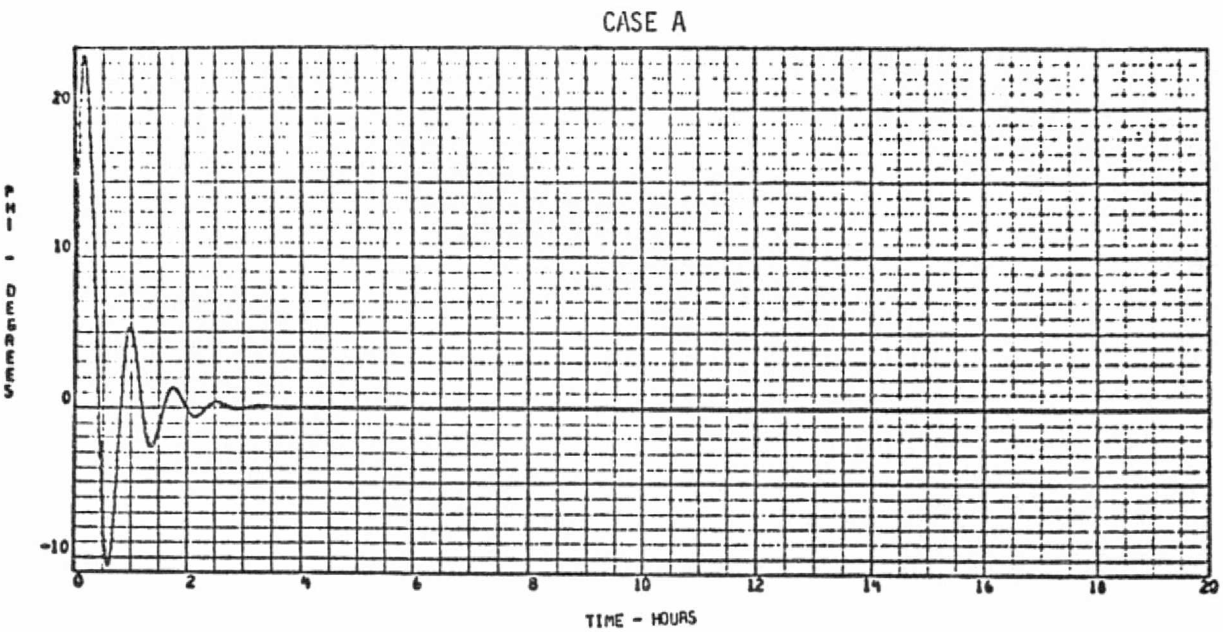


Figure D-15. Pitch angle versus time, equatorial orbit, 80 km tether.



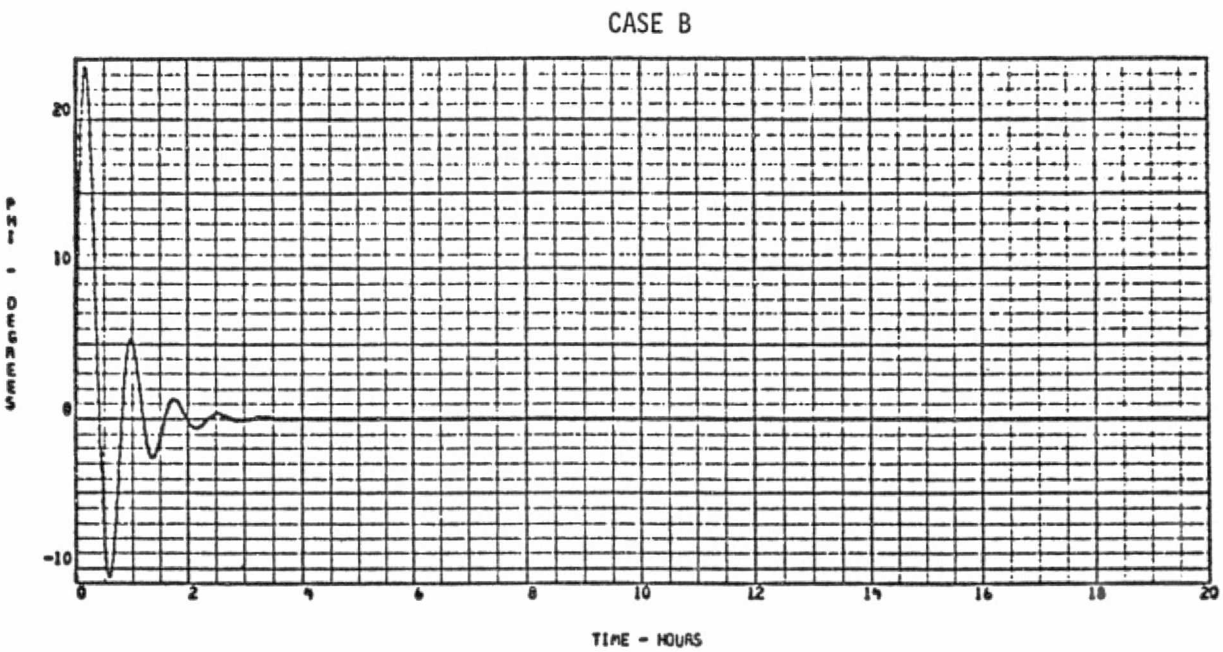
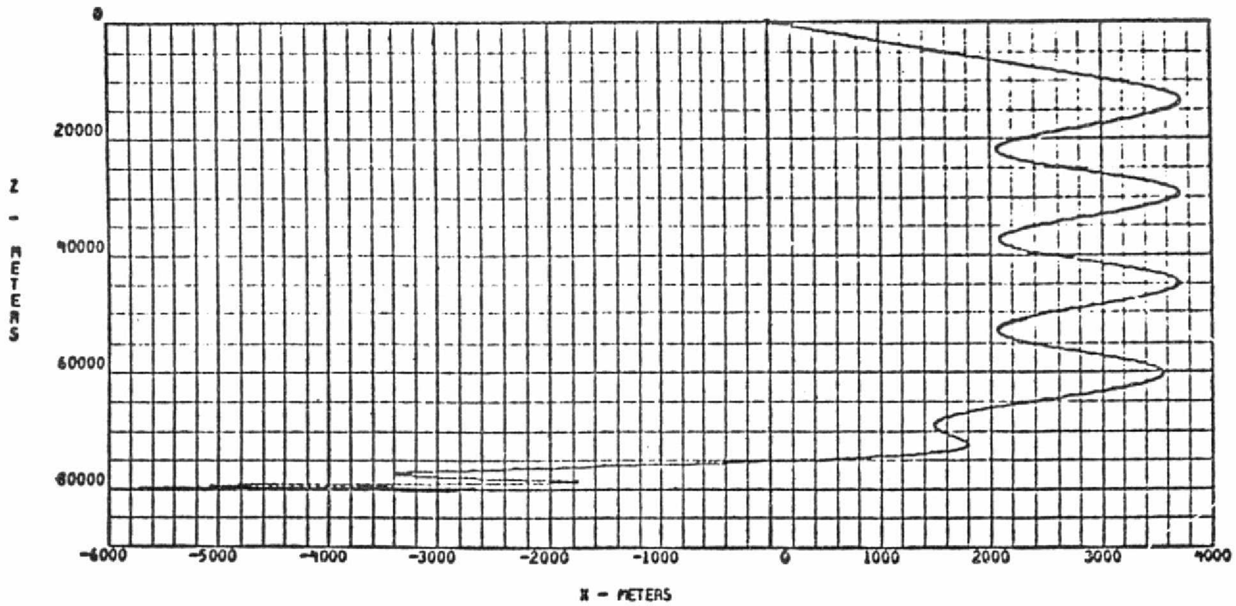


Figure D-16. Roll angle versus time, equatorial orbit, 80 km tether.

CASE A



CASE B

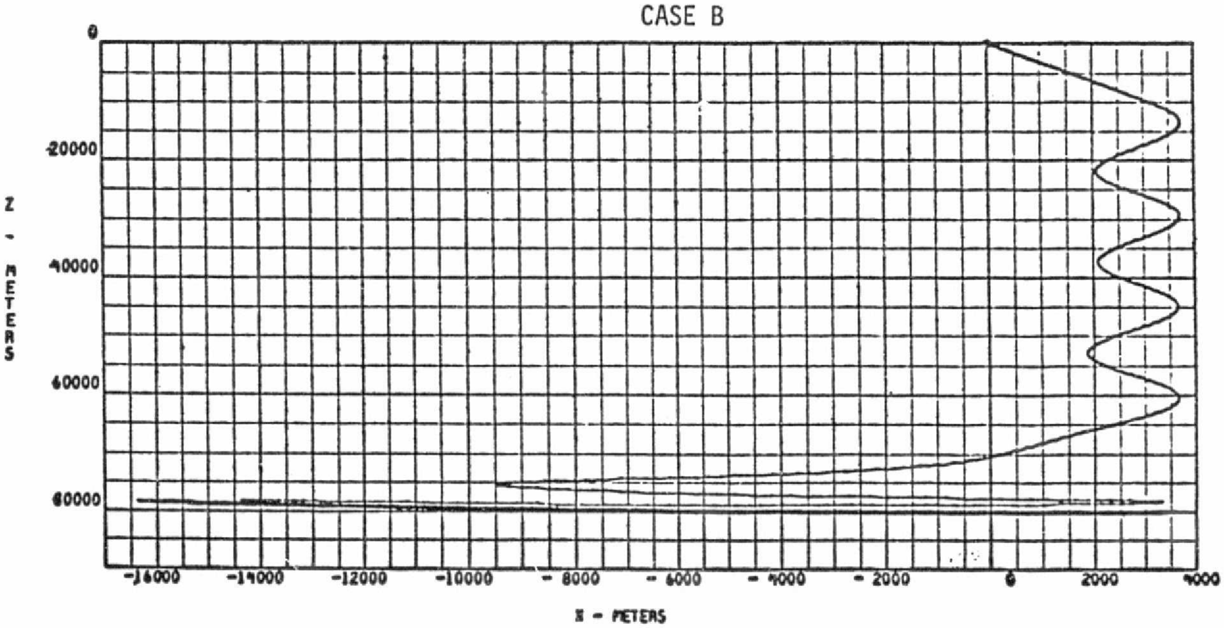
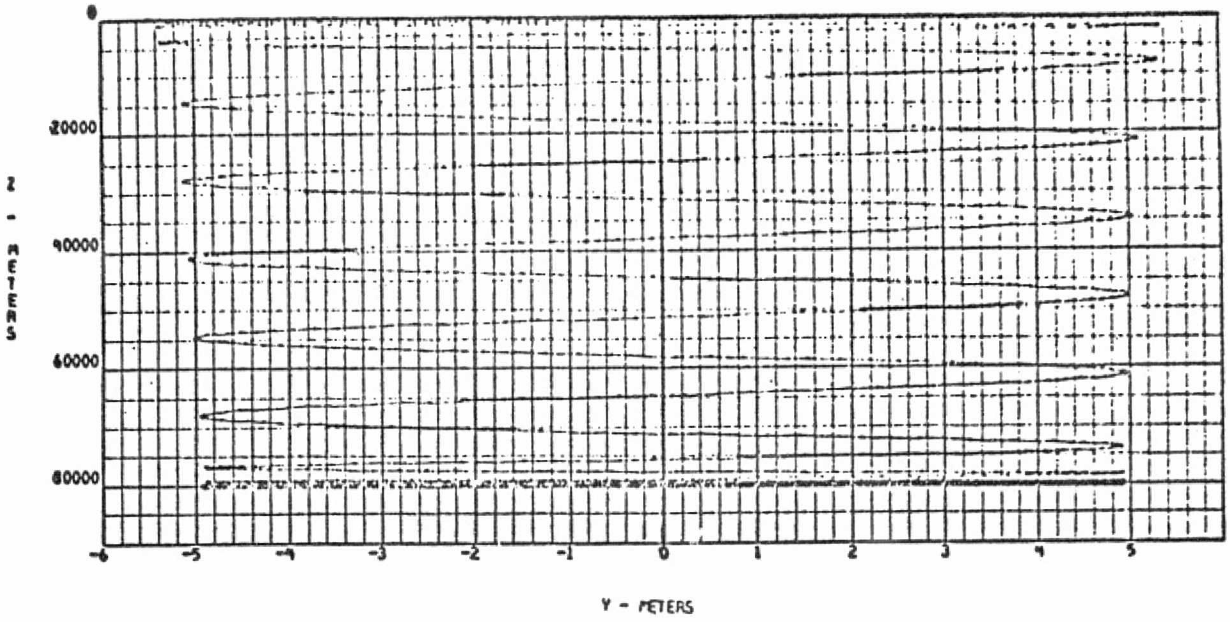


Figure D-17. In-plane motion (X-Z plot), equatorial orbit, 80 km tether.

CASE A



CASE B

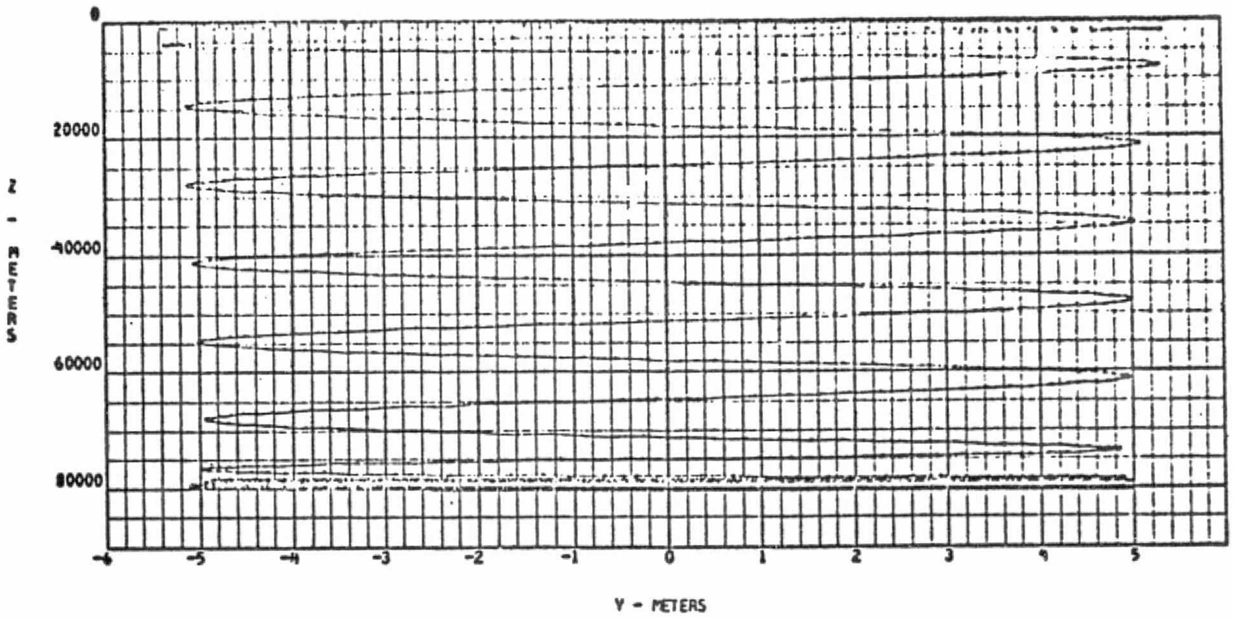
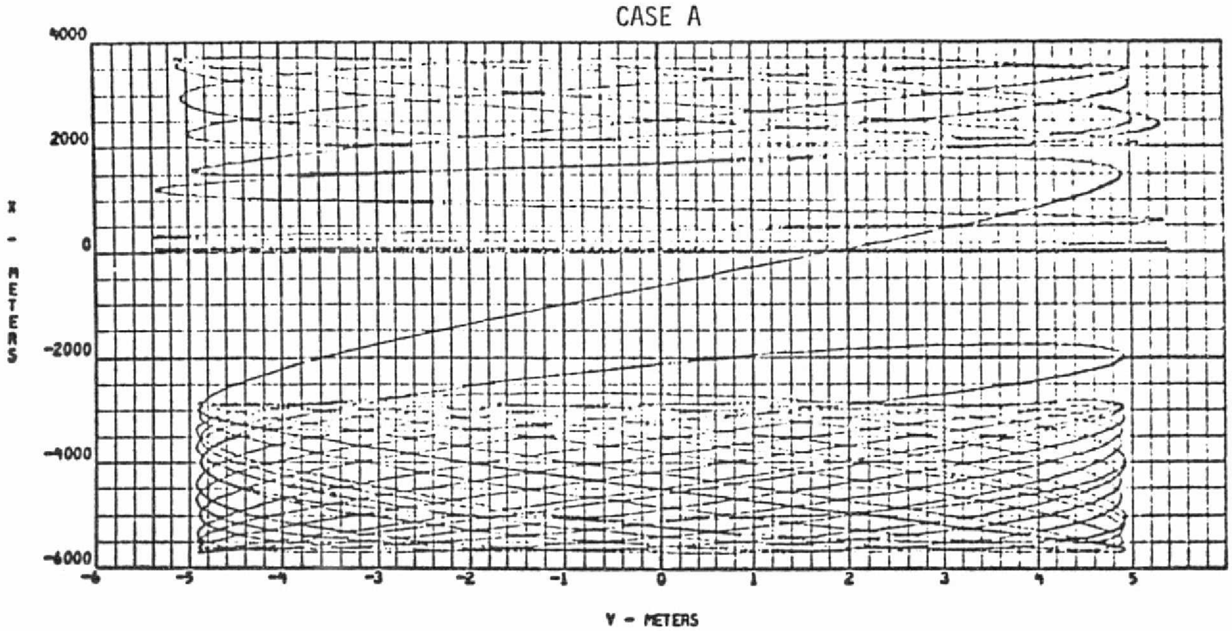


Figure D-18. Out-of-plane motion (Y-Z plot), equatorial orbit, 80 km tether.



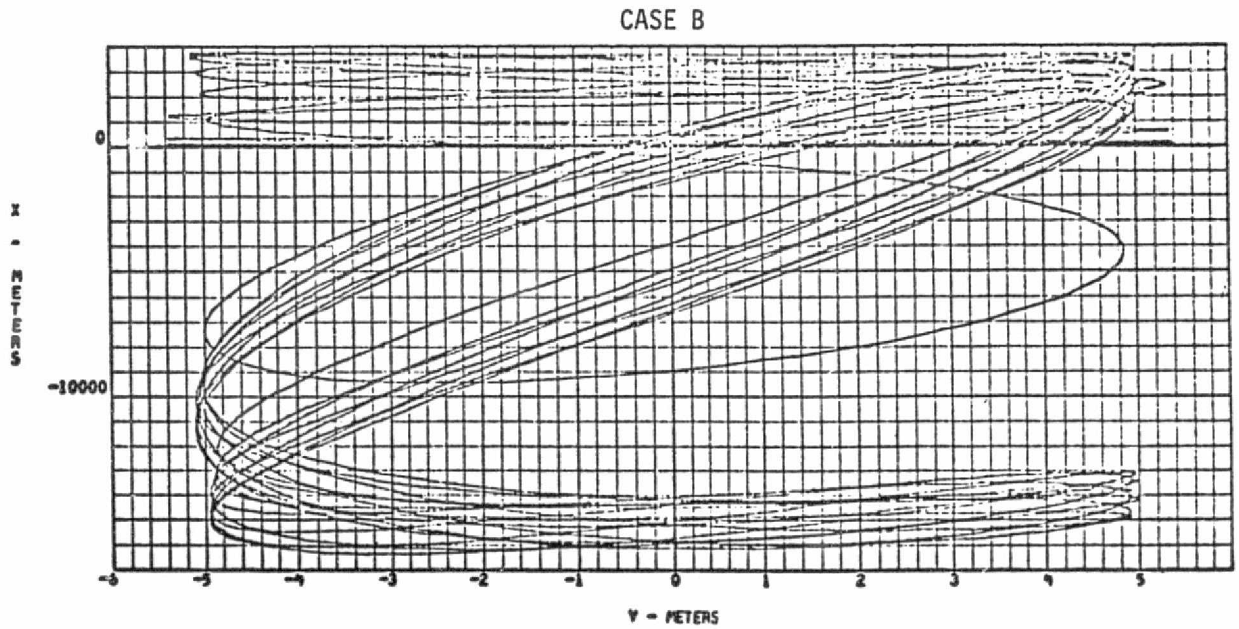
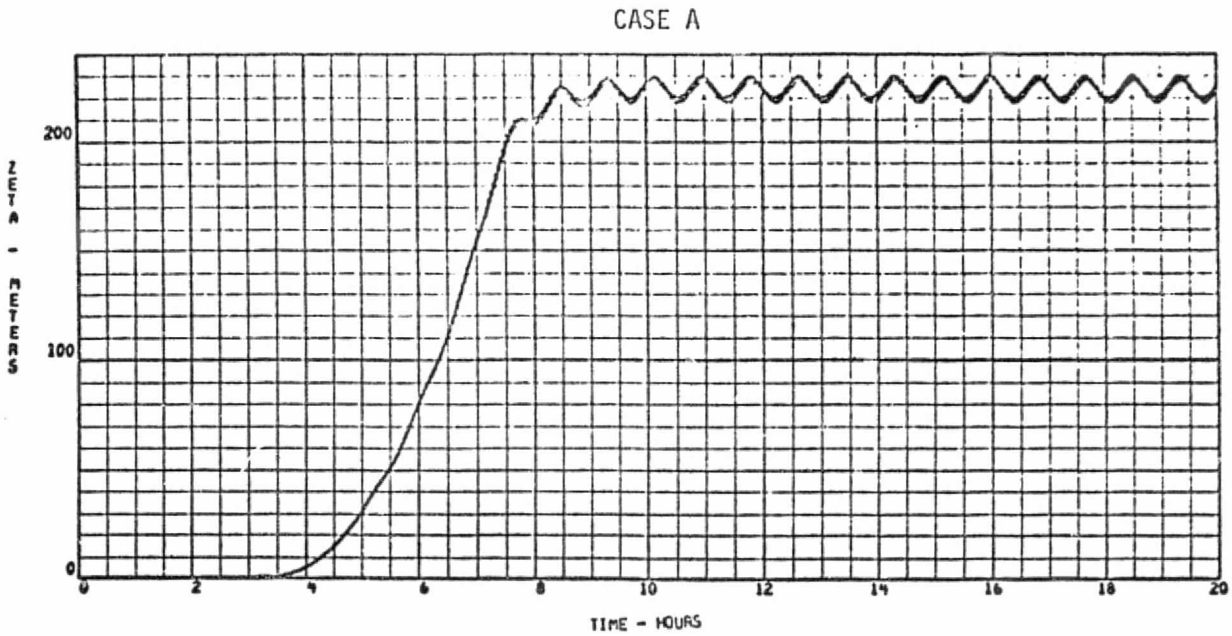


Figure D-19. Equatorial motion (X-Y plot), equatorial orbit, 80 km tether.



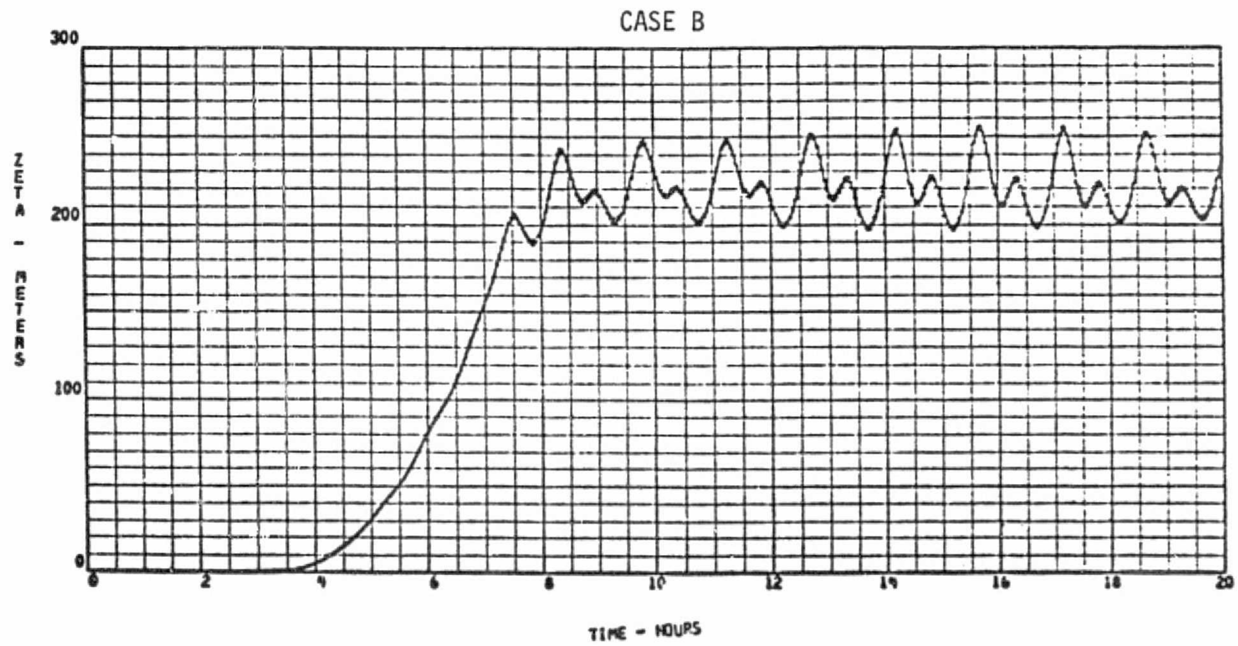
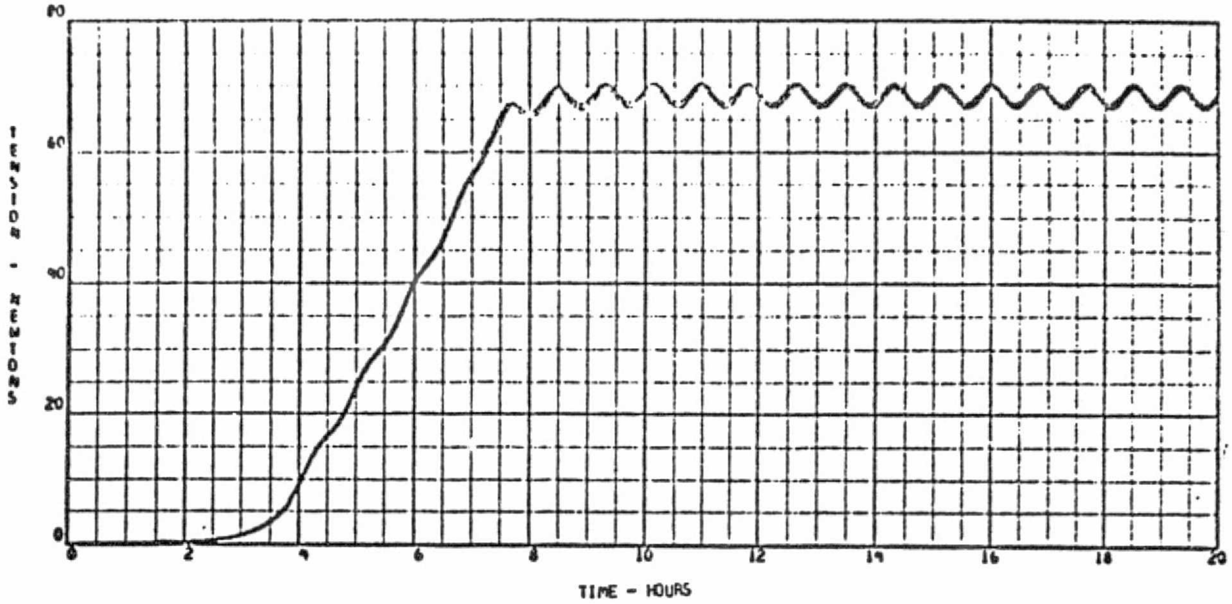


Figure D-20. Tether elongation versus time, equatorial orbit, 80 km tether.

CASE A



CASE B

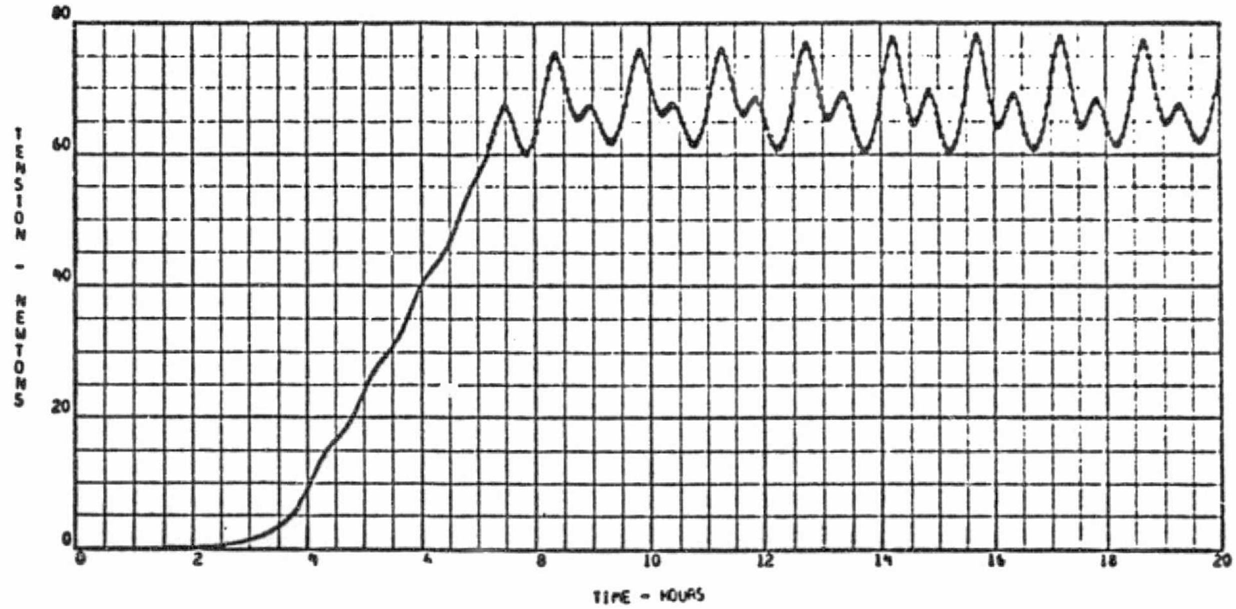
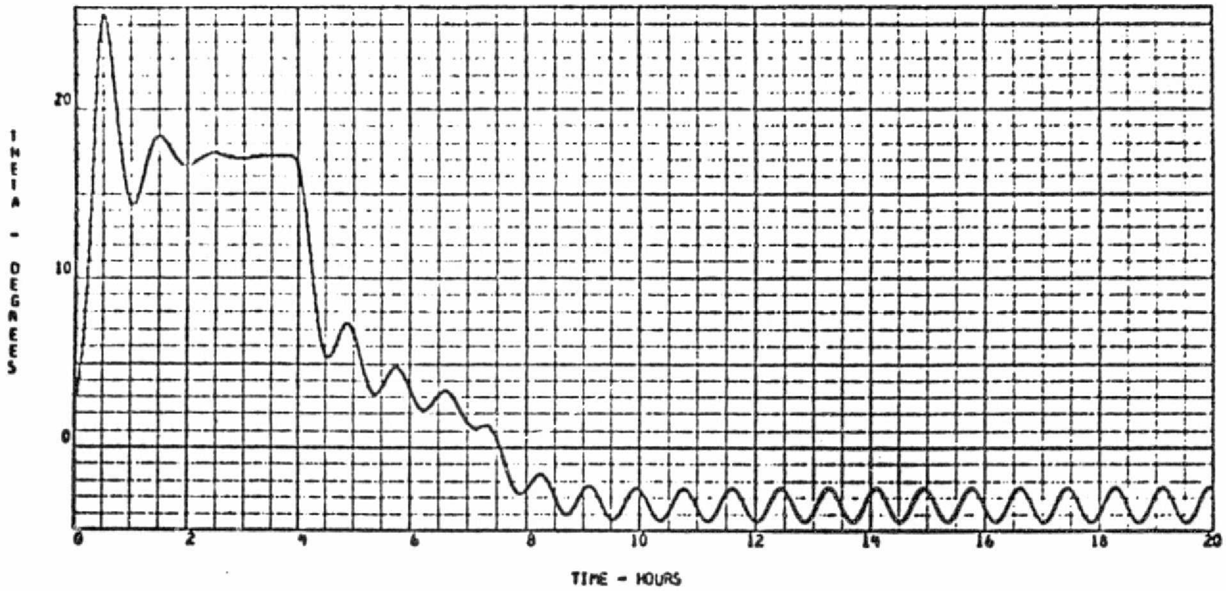


Figure D-21. Tether tension versus time, equatorial orbit, 80 km tether.

CASE A



CASE B

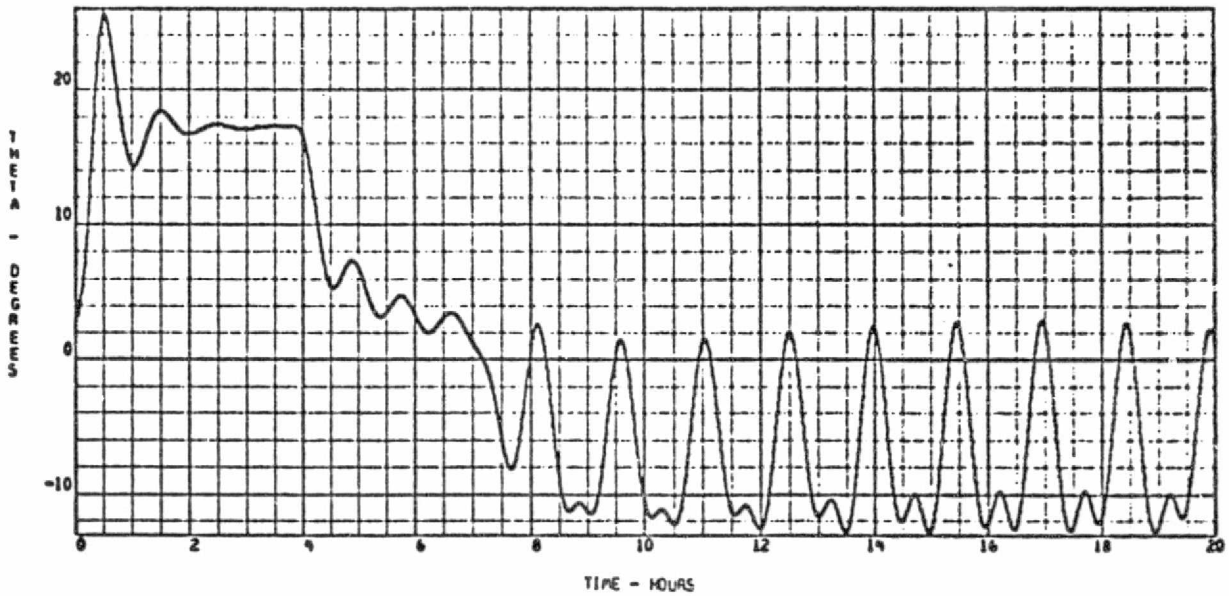


Figure D-22. Pitch angle versus time, polar orbit, 80 km tether.

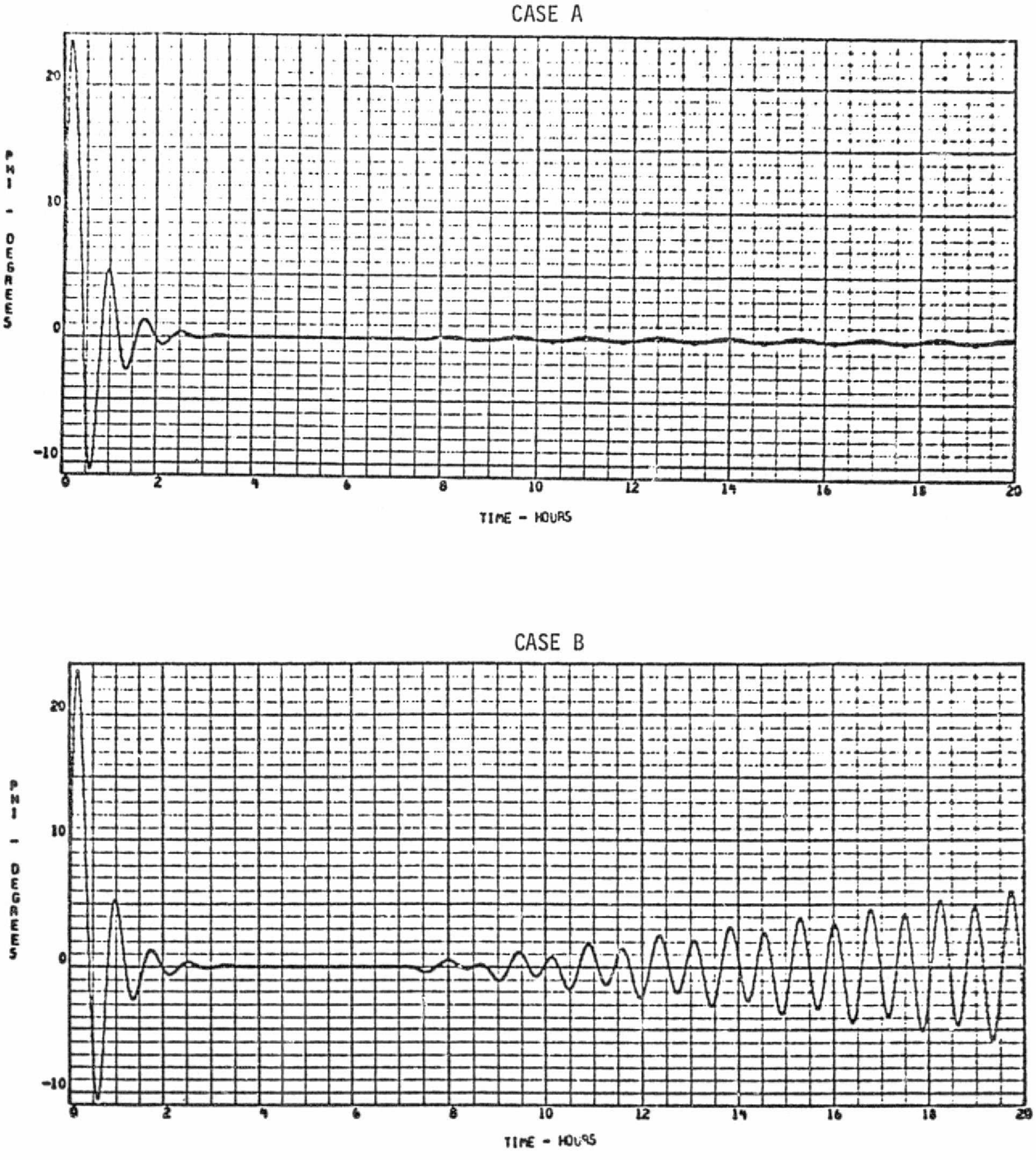
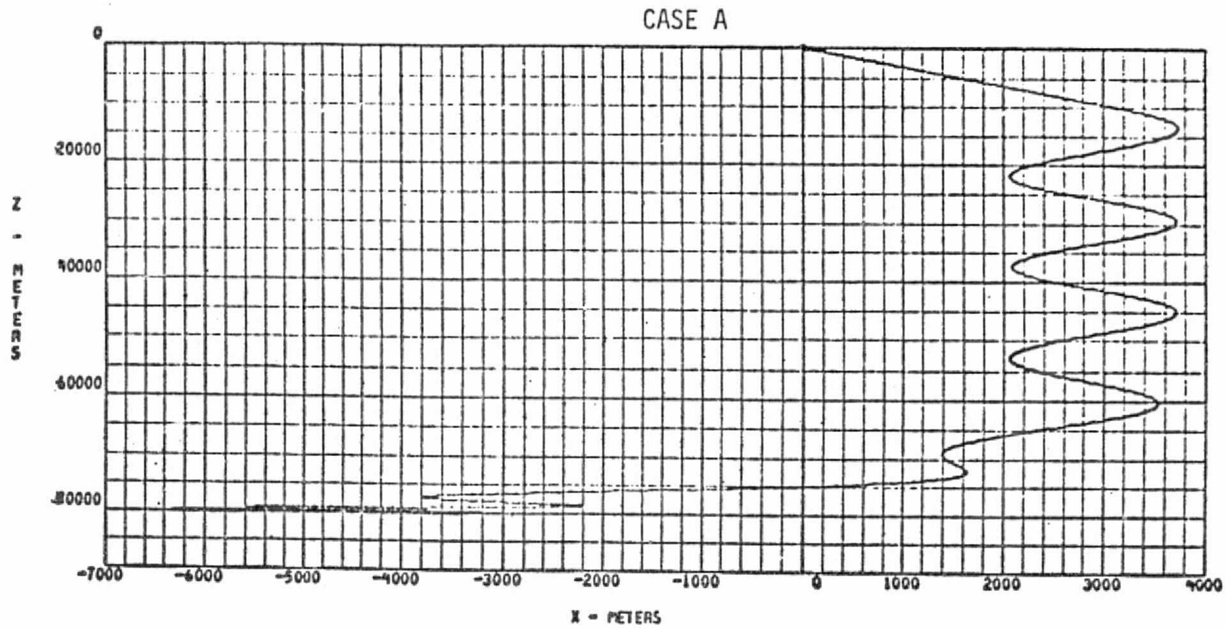


Figure D-23. Roll angle versus time, polar orbit, 80 km tether.



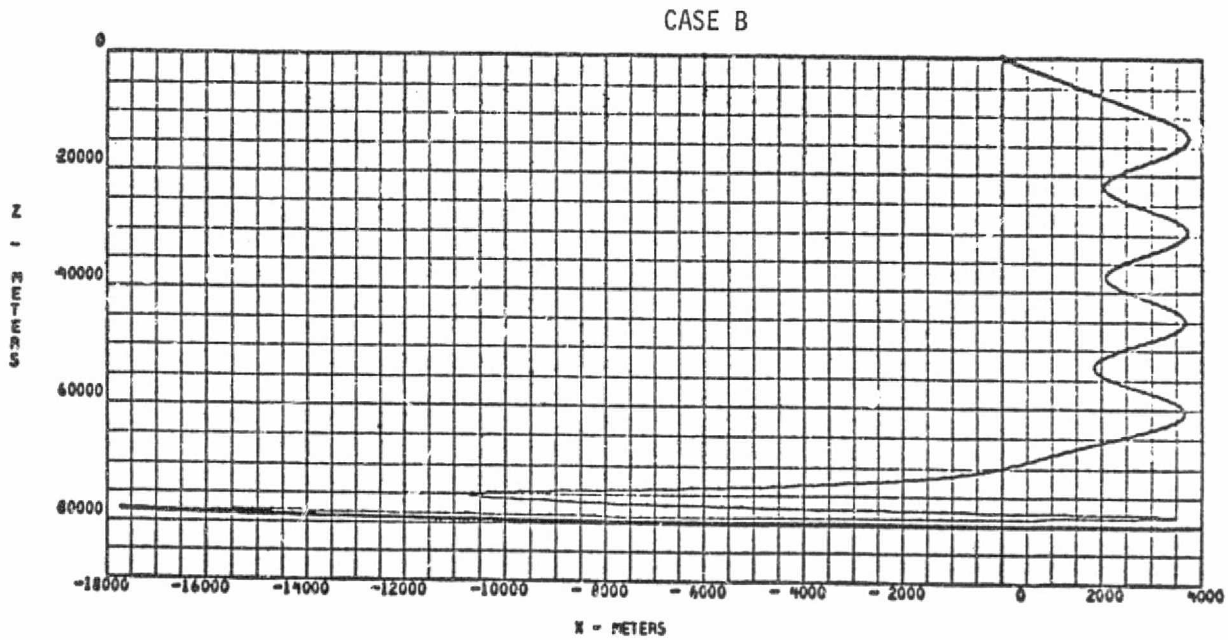
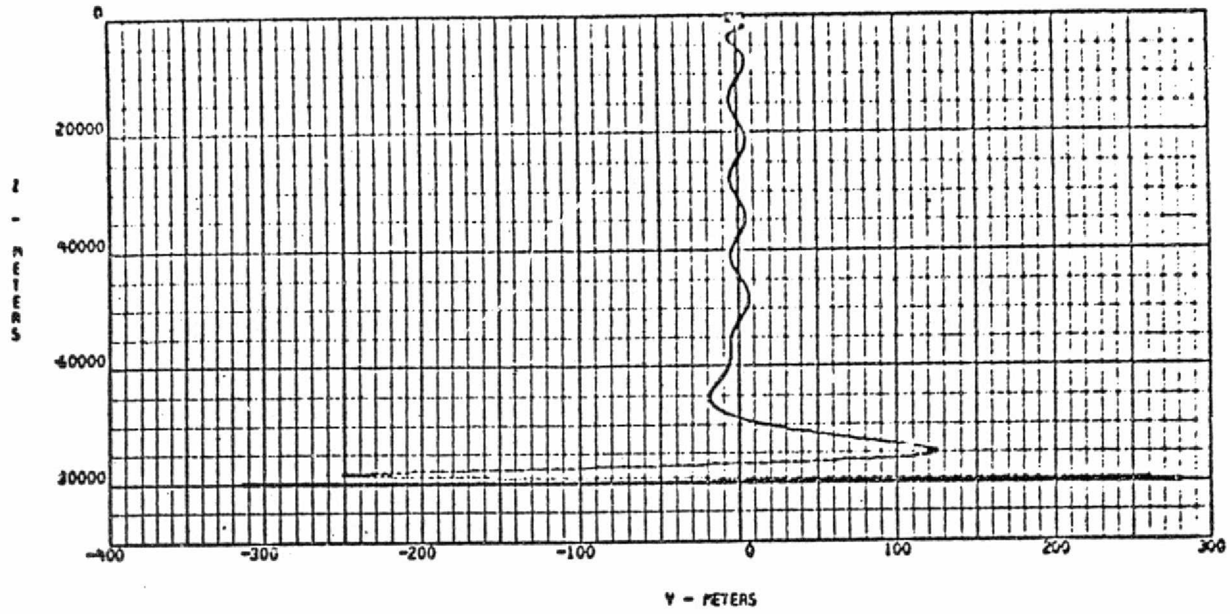


Figure D-24. In-plane motion (X-Z plot), polar orbit, 80 km tether.

CASE A



CASE B

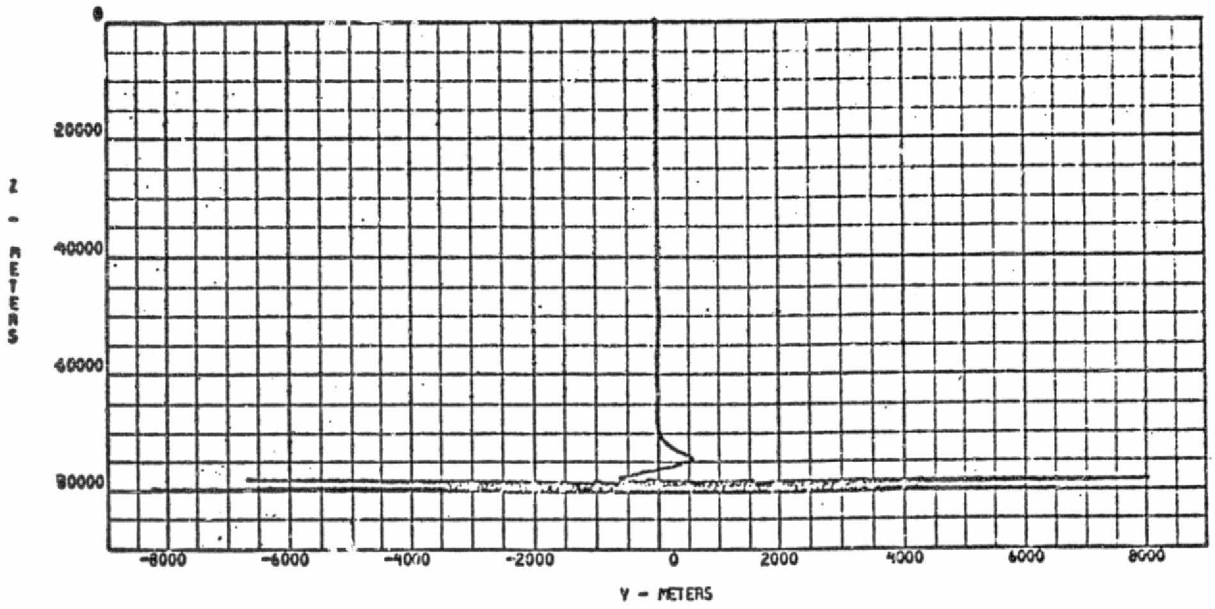
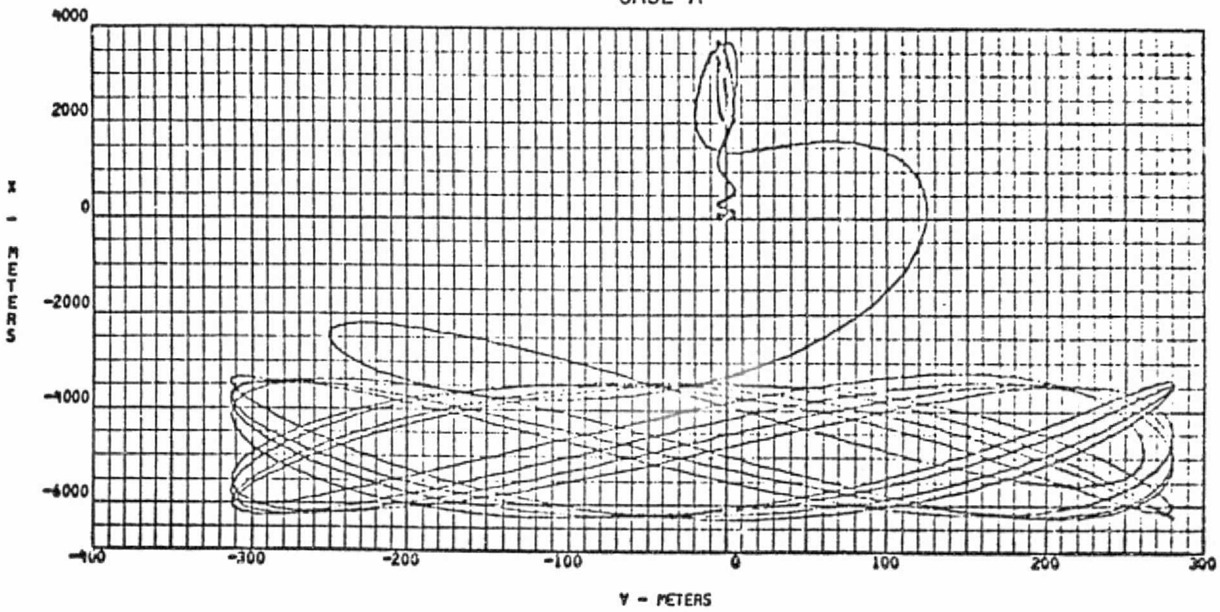


Figure D-25. Out-of-plane motion (Y-Z plot), polar orbit, 80 km tether.



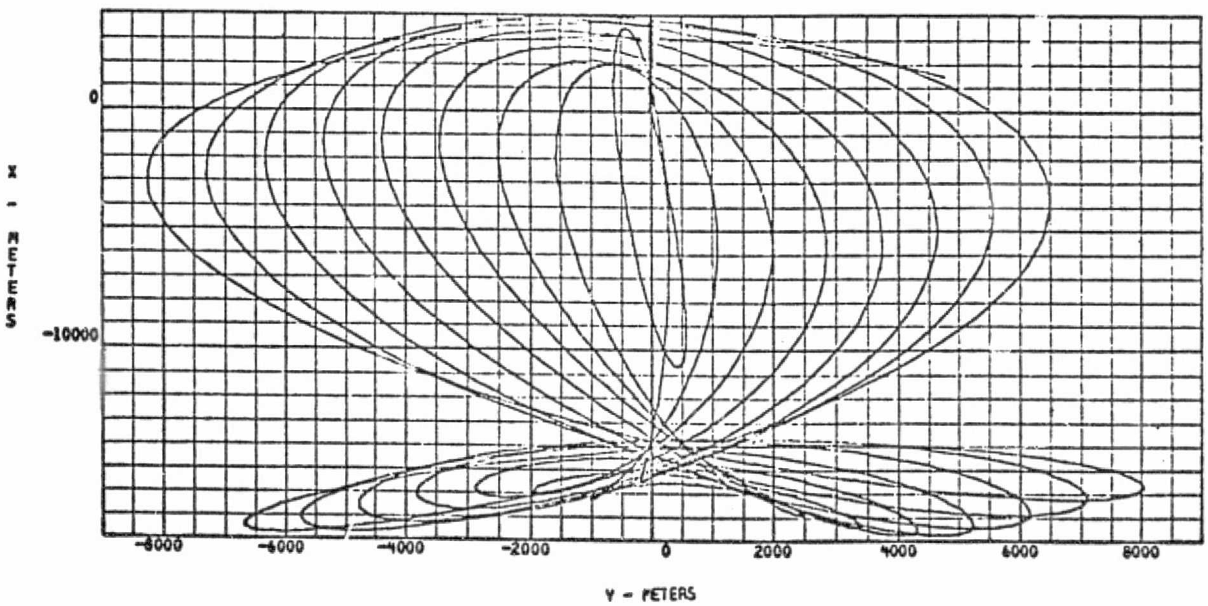
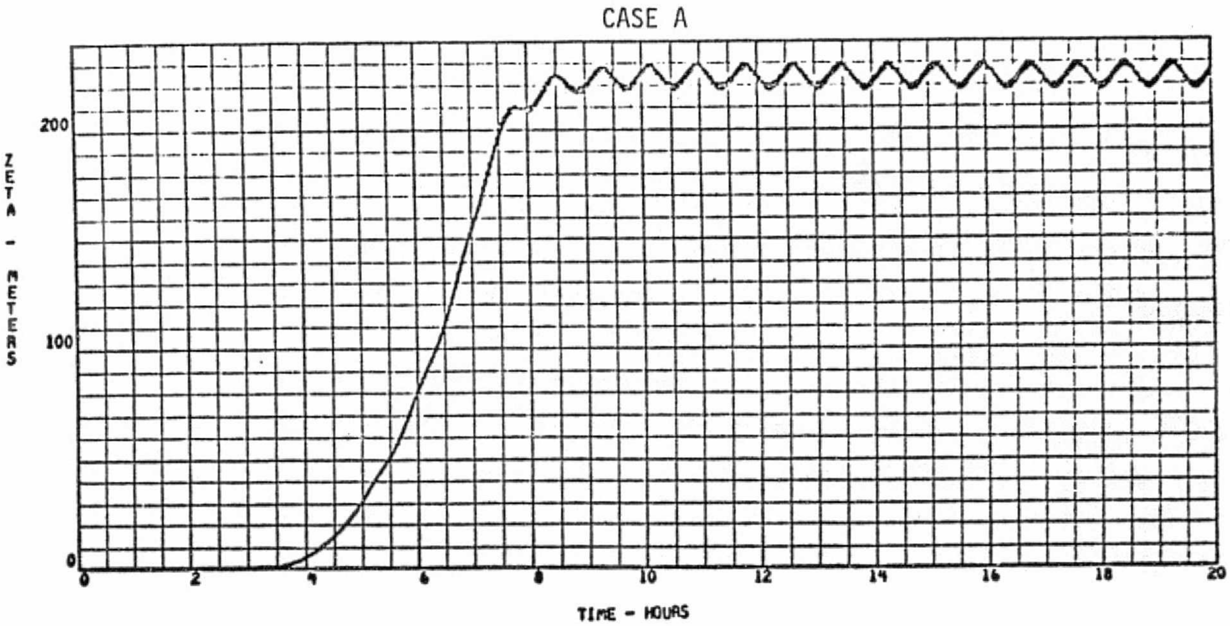


Figure D-26. Equatorial motion (X-Y plot), polar orbit, 80 km tether.



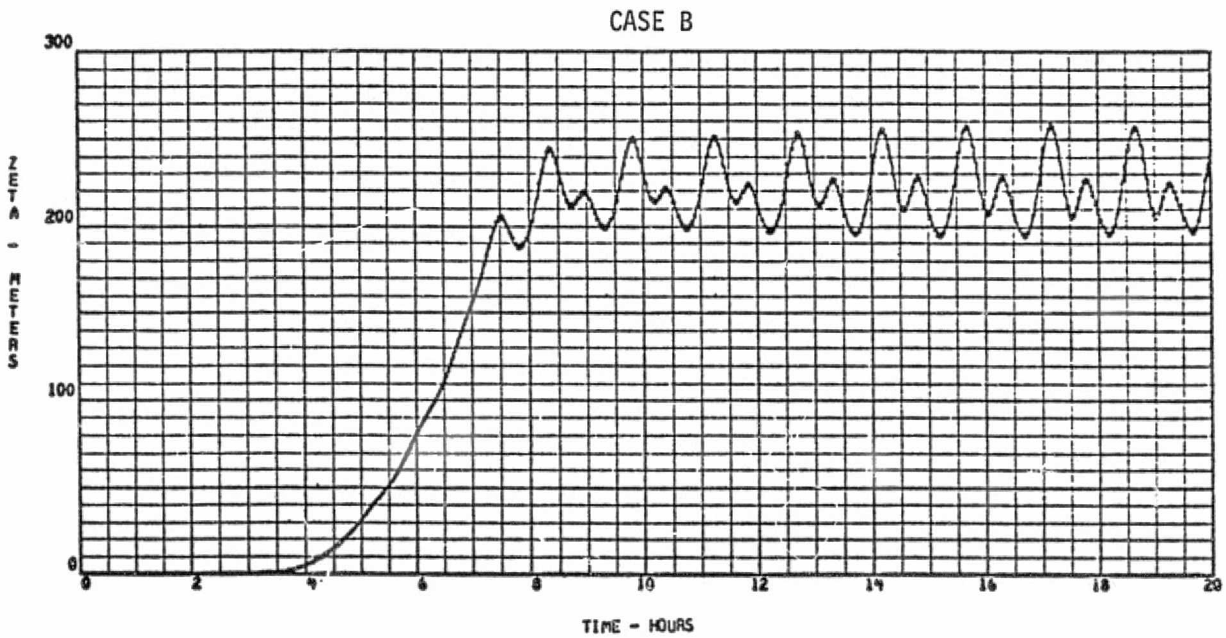


Figure D-27. Tether elongation versus time, polar orbit, 80 km tether.

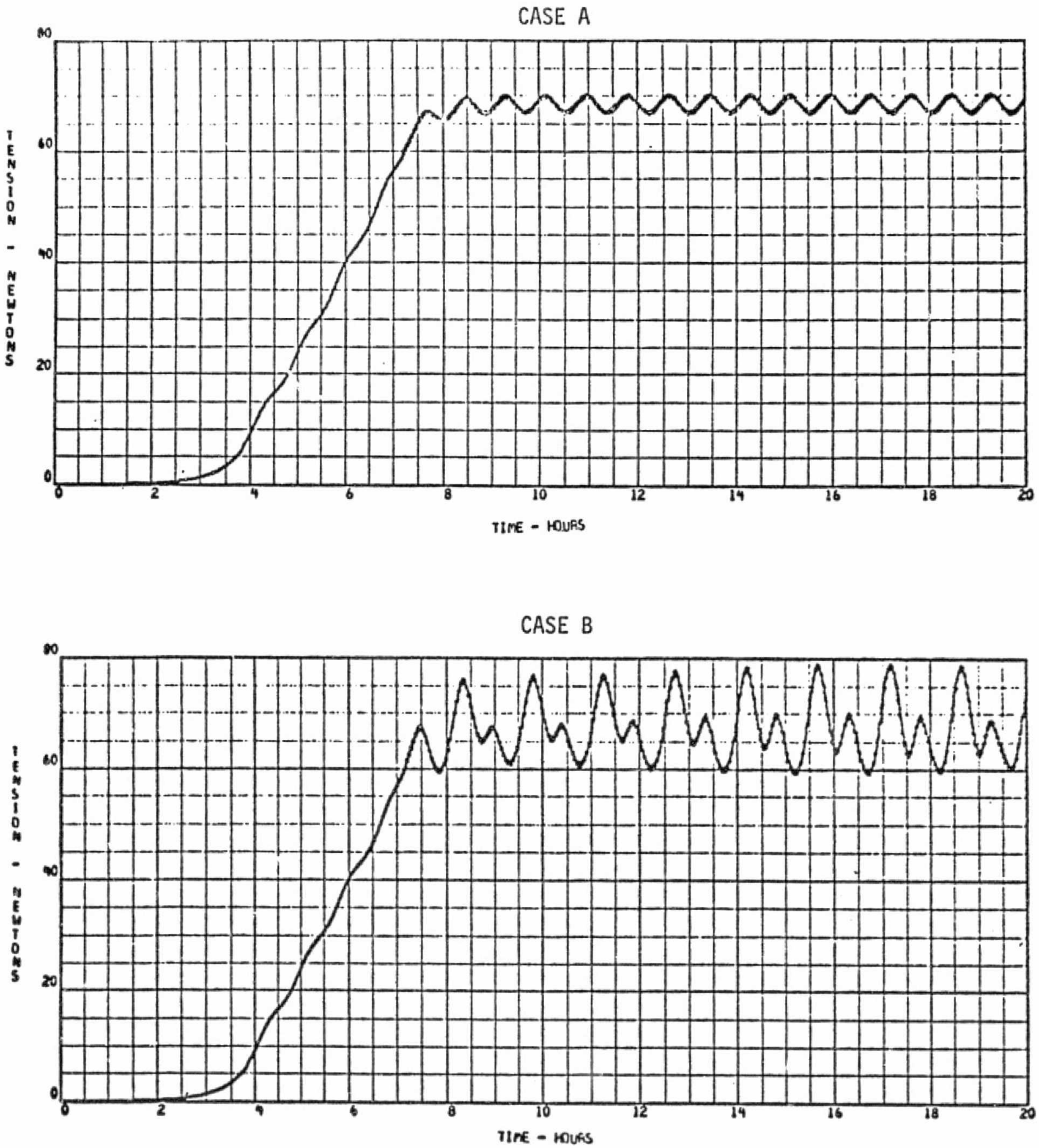


Figure D-28. Tether tension versus time, polar orbit, 80 km tether.

APPENDIX E

STATEMENT OF WORK
MICROMETEOROID IMPACT TESTS ON WIRES AND ROPES

Introduction

The use of load bearing wires (or ropes) in space has been proposed for a number of applications, e.g., tethered satellites, antennas, and light structures. The effect of micrometeoroid impacts upon such wires is not known and cannot be calculated from the known tests on flat plates. It is therefore desired to perform impact tests on wires using a hypervelocity gas gun with nylon and glass projectiles. The objective is to determine the criteria for failure of 1 mm diameter wire as proposed for the tethered satellite system. The test program will require:

a. The manufacture of suitable specimen holders and targets

b. The testing of the specimens

c. The evaluation of the tests.

Specimens and Targets

The hypervelocity gun is capable of firing 1.5 mm diameter nylon projectiles or using a sabot, 0.100 to 0.200 mm diameter glass spheres. The initial tests will be performed using the 1.5 mm nylon projectiles on wires appropriately scaled up in diameter. Later tests will be run using the 1 mm diameter wire and the small glass projectiles.

For the initial tests the following target materials will be used:

Stainless steel wire, bare

Stainless steel wire, sheathed with insulator

Stainless steel rope, bare

Stainless steel rope, sheathed with insulator

Kevlar monofilament

Kevlar woven rope.

Each of these target materials will be provided in three diameters: 6 mm (1/4 in.), 4-1/2 mm (3/16 in.), and 3 mm (1/8 in.).

The samples will be provided to the tester in any desired length for direct mounting in the test chamber.

For the latter tests using the small glass projectiles, test specimens made from the same materials will be provided. The targets will be made from tensioned 1 mm diameter specimens disposed in a way such as to ensure an impact on each successful shot; the target area so designed will be approximately 1 cm^2, as seen in the direction of the projectile flight. The individual wires will be tensioned to a maximum of approximately 40 kg load by appropriate mechanical devices in a reusable target holder.

Testing

For the initial 1.5 mm diameter nylon projectile tests, it is desired to impact each kind of specimen so that the trajectory of the projectile crosses the centerline of the specimen; in further tests "grazing" impacts will be attempted.

The specimens will be tested in at least two attitudes: one where the centerline of the specimen is normal to the trajectory of the projectile and the other where that angle is 30°. Four test conditions are needed for each specimen. Two or three tests should be performed in each configuration for confirmation of the results.

Similar test procedures will be used for subsequent tests on the multi-specimen target of 1 mm diameter specimens using the glass projectiles.

Evaluation of the Tests

The tests will first be evaluated qualitatively by inspection of the specimens. Tensile tests will then be performed to determine the strength of the impacted specimens. Scaling factors will be applied to determine the likelihood of failure in a space environment. Scaling allowances will be made for the velocity of the projectiles, for their density and for the unidirectionality of the test conditions.

APPENDIX F

DEPLOYED BY ORBITER TETHERED SATELLITES (DOTS)

The object of the DOTS flight is to study the dynamics of light and cable connected structures, and to gather data on the survivability of cable supported structures and tethered subsatellites when exposed to the micrometeorite environment in Earth orbit.

The DOTS experiment will consist of one or more completely passive packages to be deployed by the Orbiter. Each package will contain two aluminized plastic inflatable balloons of differing ballistic coefficients connected by a 10 km length of woven synthetic cable approximately 1 mm in diameter. The cable will be deployed from a spool similar to a spinning reel as the differential drag causes the satellites to separate. Ground tracking and visual observations will be used to determine if, and when, the balloons separate more than the cable length indicating that the cable has been severed.

The information derived from the flight will be used to verify calculations of catastrophic micrometeorite impact probabilities. These calculations involved extrapolating empirical data on micrometeorite flux density and laboratory test data on damage to flat plates. The two extrapolations combine to reduce the confidence in the calculation results to the point where further flight testing is required. It is expected that orbital lifetime of the balloon experiments should be at least 6 months and deployment of at least three packages would be required to be of statistical significance. The balloons would use echo technology with very little fabrication required for the spools to house the cable. An alternate packaging approach to the experiment would be a series of balloons of gradually increasing ballistic coefficient strung in a string. This string would appear as a series of dots in the early evening sky.

APPROVAL

SHUTTLE/TETHERED SATELLITE SYSTEM CONCEPTUAL DESIGN STUDY

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

Charles R. Darwin

CHARLES R. DARWIN
Director, Preliminary Design Office

James T. Murphy

JAMES T. MURPHY
Director, Program Development

☆U S. GOVERNMENT PRINTING OFFICE 1977–740-049/164 REGION NO. 4