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STRUCTURES FOR SMALL SCIENTIFIC SATELLITES

BY

JESSE M. MADEY
ROBERT C. BAUMANN

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NASA
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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Goddard Space Flight Center
National Aeronautics and Space Administration

May 1965

GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland

FROM THE AUTHORS

We, the authors, realize that this publication is not as complete and comprehensive as it should be. We are in the process of preparing the first revision which we plan to make more complete especially in the areas of torsional vibration, double hinged appendages, method for determining solar paddle arm angles, materials and vibration isolation.

We solicit the comments, questions, and constructive criticism on the content and method of presentation.

Jesse M. Madey
Robert C. Baumann

Code 670
Goddard Space Flight Center
Greenbelt, Maryland

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ABSTRACT

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Satellites whose primary mission is space science research are called "scientific satellites."

Small satellites are defined as that class of satellites whose mission can be accomplished on a Delta or Scout launch vehicle.

The primary function of the satellite structure is to provide the scientific experiments with a suitable housing compatible in every respect to the experimentation which it is to contain.

The structural engineer has many factors to consider in the evolution of the satellite structure.

Among the design considerations are such items as schedule preparation, determining the basic shape of the satellite, launch vehicle - satellite compatibility, accessibility, materials, thermal design factors, R.F. design factors, structural loads, fabrication, integration, test, shipping and handling, and field operations.

The structural engineer's task is not completed even after launch for he must perform post-flight analysis in his areas of responsibility.

author

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STRUCTURES FOR SMALL SCIENTIFIC SATELLITES

INTRODUCTION

The present day field of satellites is so broad that it becomes necessary to confine and define the area of the field that one discusses.

My presentation will be confined to small scientific satellites. Small, for purposes of this presentation, will be that class of satellites whose mission can be accomplished on a Delta or Scout launch vehicle. Scientific satellites are defined as those satellites whose primary mission is space science research.

The advantages of small satellites far outweigh the advantages of large satellites. The advantages of the smaller satellites are: (1) easier to tailor the spacecraft to the needs of the experiment, i.e., minimize electronic or magnetic interference; (2) with less experiments the chances are that the experimenters will obtain the orbit requested; (3) also the orientation of the spacecraft should satisfy most of the experimenters; (4) easier to integrate and test a less complex system; and (5) the most important advantage is that it is possible to launch a smaller satellite with a much shorter lead time. This enables the experimenter to fly the latest rather than obsolete experiments, therefore accelerating the state-of-the-art and gathering more sophisticated and meaningful data.

Generally speaking, the physical dimensions of a satellite during the launch configuration are much smaller than the orbital configuration. During launch, all booms, appendages and antennas are either folded or retracted. Before separation from the launch vehicle these booms, appendages or antennas are erected unfolded or released according to a well-planned operational sequence. Dimensionally, the satellite, once in orbit, could be extremely large in the orbital configuration. This case is illustrated by Figure 1 (130-foot dimension from tip to tip on galactic noise antenna).

The primary function of the satellite structure is to provide the scientific experiments with a suitable housing compatible in every respect to the experimentation which it is to contain.

The structure must be (1) efficiently designed in order to insure that the maximum weight is available for the scientific payload and supporting subsystems; (2) rugged enough to withstand the environmental conditions it will

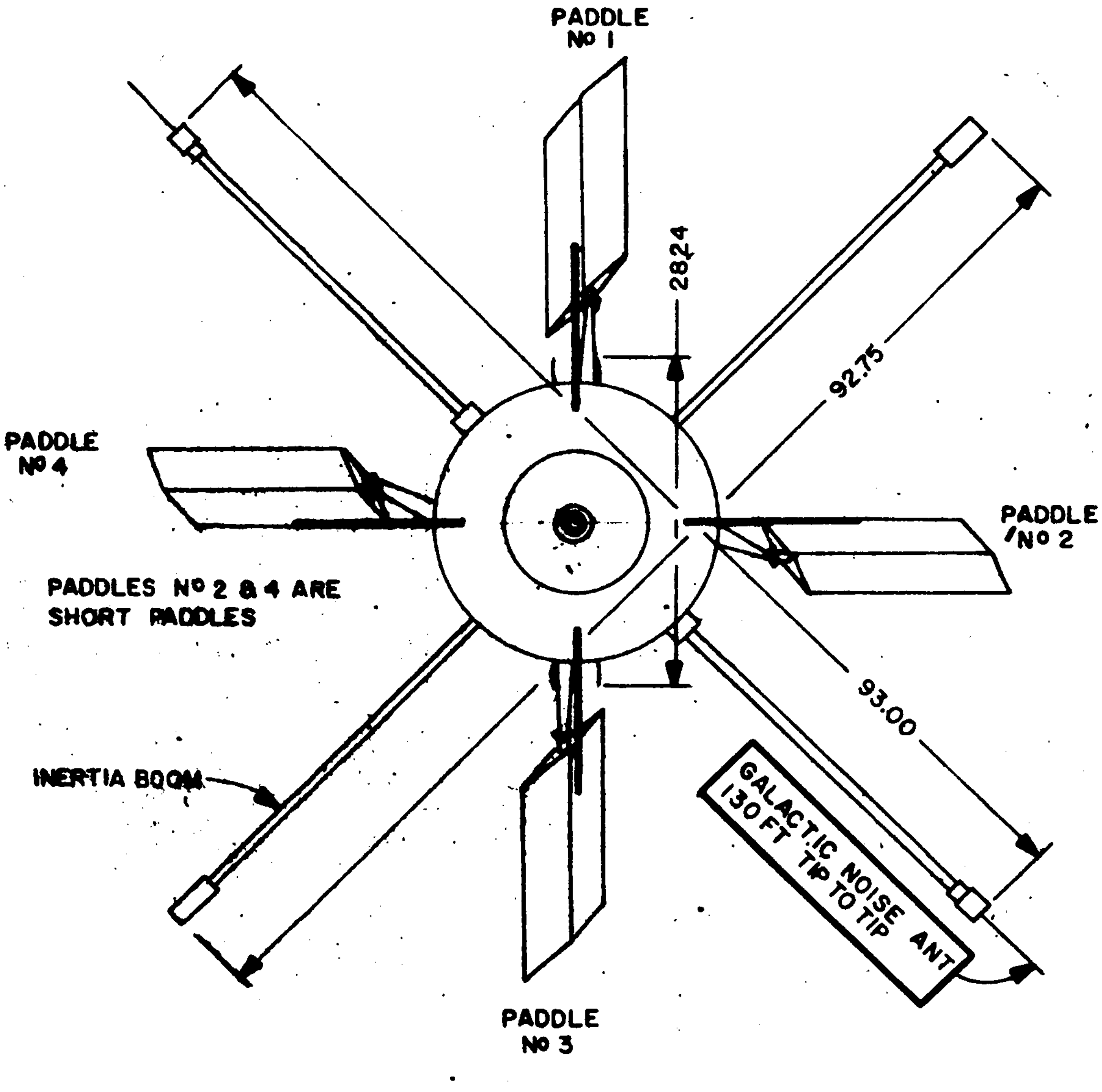


Figure 1—Orbital Configuration (S-52)

encounter such as ground handling, test, launch and space environmental conditions; and (3) constructed in such a manner that the experimentation and the various sub-systems may be readily installed, removed, inspected, and tested.

A satellite structure is defined as a series or combination of members, beams or plates held together by screws, rivets or similar fasteners as illustrated by Figure 2a. Hopefully, the structure has a high strength to weight ratio, is easy to design, is inexpensive but can sustain all the rigors of powered flight as well as orbital environment. Since all of this is not that simple but rather complex, an attempt will be made to separate the principle structure from the secondary structure.

The principle structure is one that carries the major portion of the weight or dynamic loads. (See Figure 2b). This is normally the main body of the satellite, whether it be a sphere, cylinder or quasi-spheroid. The secondary structure is composed mostly of brackets, hinges, arms, booms or similar hardware whose function is to attach some mass, whether it be an experiment or mechanism, to the main body of the satellite.

If the designer has only a static structure to provide with no continuously moving mechanisms, his task is a relatively straightforward one. Most of the small scientific satellites are of this static type. Some however have tape recorders, scan devices, stepping devices, and other mechanisms. These mechanisms and their design, fabrication and test are excluded from this paper; but, when they are to be included in a satellite system, the structural designer must consider them carefully. For a small spinning satellite that requires a certain spin-axis-sun relationship neglecting the effects of moving parts in the satellite analysis could be disastrous.

Also, one must not neglect the aerodynamic drag, magnetic damping, solar pressure, and other such forces in the analysis of the orbital attitude and spin decay. The analysis of the effects (Reference 1) of these types of forces is a subject unto itself and will not be dealt with in this paper.

There are numerous factors that go to make up the items generalized above. This paper will attempt to treat these many factors in sufficient depth that the apprentice designer of small scientific satellites will be able to use this paper as a guide that points to the milestones along the way from conception of the mission through launch and orbit, i.e., from "cradle to grave."

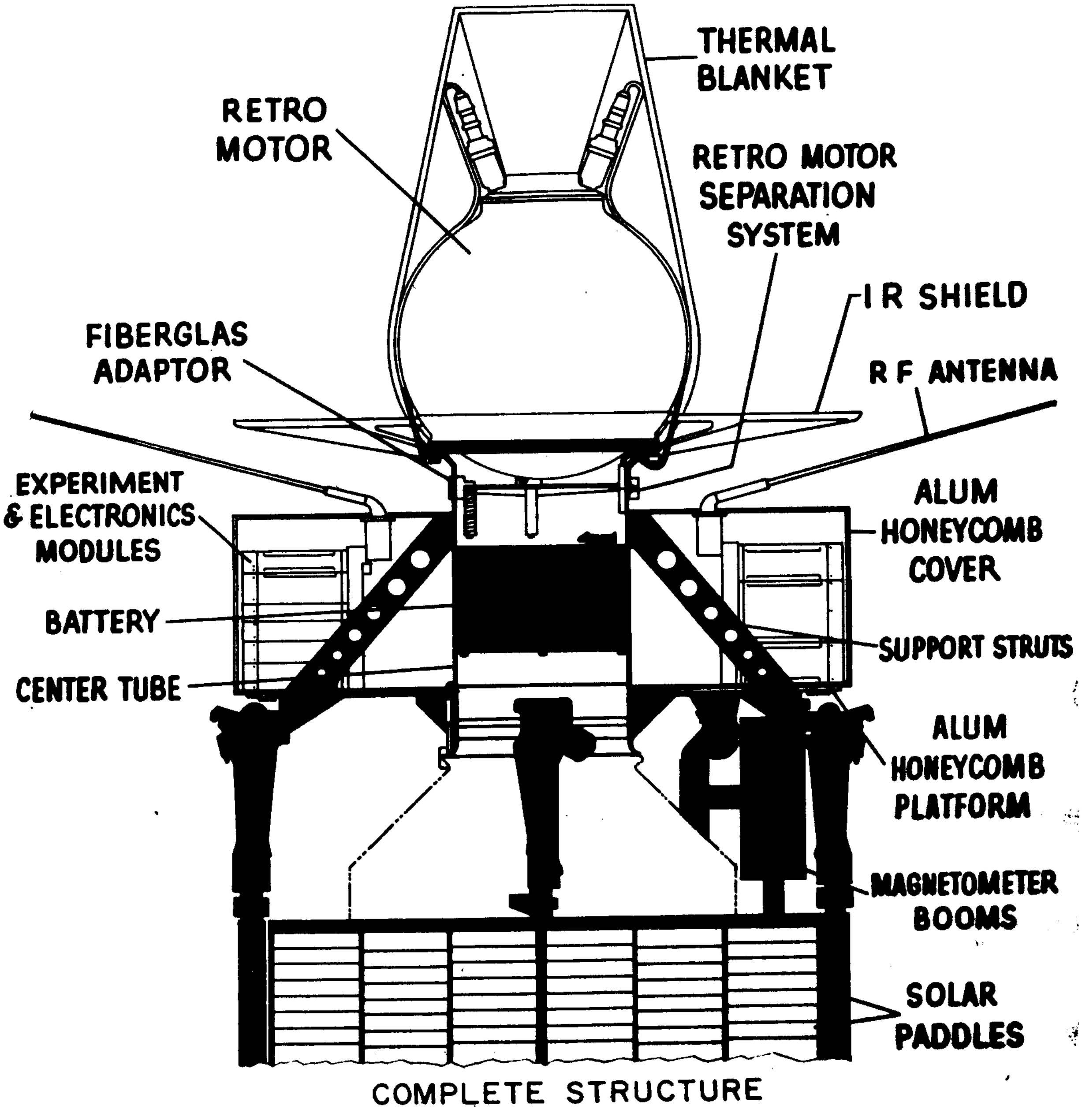


Figure 2a–Complete Structure (A-IMP Cross-Section)

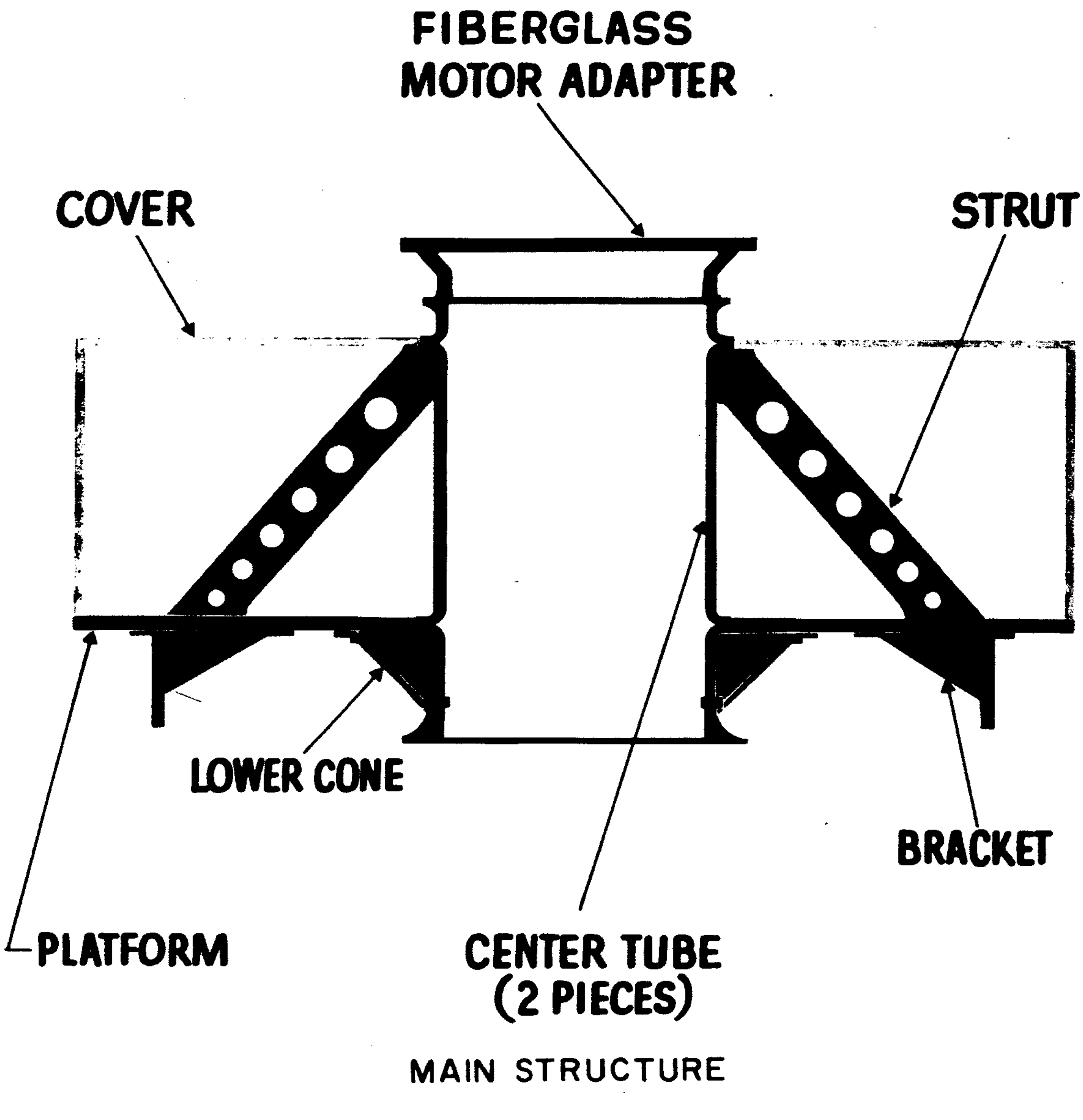


Figure 2b–Main Structure (A-IMP)

GENERAL DESIGN EVOLUTION (Figure 3)

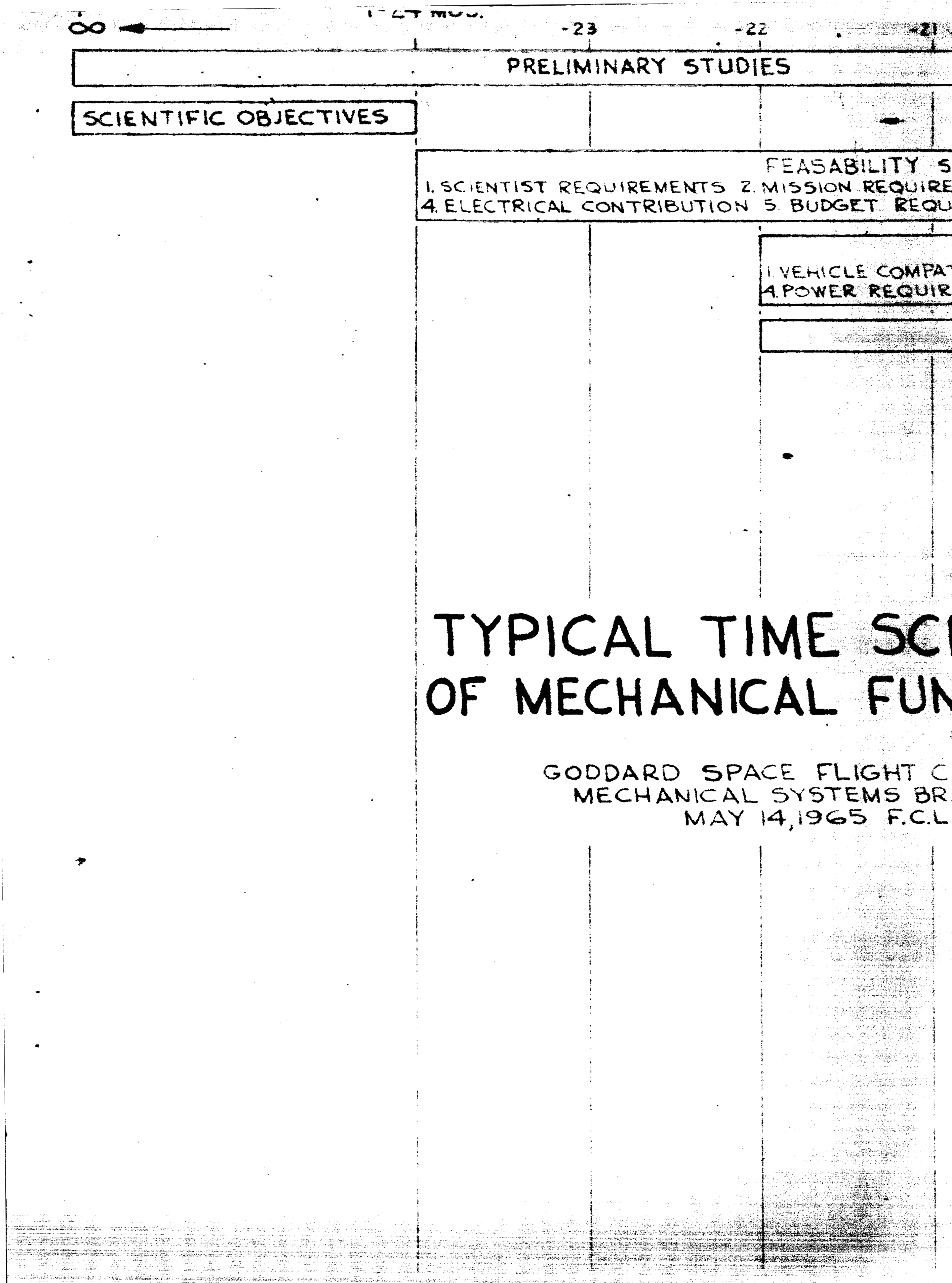
Long before the structural designer becomes engaged in the feasibility, preliminary, and final design phases of a scientific satellite structure decisions are being made by responsible scientists. These decisions result in the combining of scientifically compatible experiments into a single satellite. The experiment complement is normally chosen for the purpose of investigating closely inter-related space phenomena.

Once the experiment complement is determined it becomes the job of the project manager to investigate the practical feasibility of combining all of the experiments with their various requirements into a single integrated systems design. Here is where the satellite structural designer begins his extremely important role in the evolution and development of the satellite.

Feasibility design and layouts, taking into account all the numerous and various restraints, are made. These feasibility designs and studies usually result in pin-pointing, at a very early stage of development, incompatibilities between experimenters' requirements and what can actually be accomplished within the various constraints. Compromise is usually the answer to the dilemma the designer is faced with. Tradeoffs are proposed and negotiated until an acceptable spacecraft design can be realized.

The structural designer then starts what we call the preliminary design phase. In this phase detailed analysis in the many areas for which the structural designer has responsibility is undertaken. Again during this phase of the satellite evolution it is often necessary to negotiate further trade-offs with the experimenters, the other subsystem designers, and others involved in the fulfillment of the mission. During this phase, models, an engineering test unit, various structural and mechanical subsystems, are built and evaluated. Long lead-time parts and materials are ordered. Ground support equipment is designed and its acquisition is undertaken. Liaison to define in detail all spacecraft interfaces is completed. By the end of this phase the satellite design is usually "frozen."

Sometimes the prototype and flight unit satellite structures are manufactured somewhat in parallel during the last stages of structural development. This is only feasible when the basic design is reasonably firm and the risk of major changes is small.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PRELIMINARY STUDIES
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FEASABILITY S
1. SCIENTIST REQUIREMENTS 2. MISSION REQUIRE
4. ELECTRICAL CONTRIBUTION 5. BUDGET REQU
1. VEHICLE COMPAT
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TUDIES
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DESIGN, FABR

ENGINEERING CONSIDERATIONS
IBILITY 2. SIZES, SHAPES & WEIGHT 3. EXPERIMENTERS REQUIREMENTS
EMENTS 5. HANDLING REQUIREMENTS 6. MISSION REQUIREMENTS

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7. MOMENT OF INERTIA CALCU
EXPERIMENTS 10. HANDLING
EXPERIMENTS AND SUB-SY

1. MATERIALS
LEAD TIME

1. STRUCTURAL

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DESIGN, FABRICATE & ASSEMB
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1. TRANSPORTATION EQU
2. LIFTING DEVICES

MECHANICAL TEST FIXTURES
1. SOLAR ARRAY 2. SOLAR ARR
4. CENTER OF GRAVITY 5 MO
7. ZERO-GRAVITY SPIN SIMUL
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WIRING HARNESS

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ANTENNA MOCKUP

CATION & MECHANICAL ASSEMBLY OR INTEGRATION

PROTO

DESIGN PHASE

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ERIMENT WEIGHT DISTRIBUTION 5. EXPERIMENT VIEW ANGLES 6. PRELIMINA
ATIONS 8. PRELIMINARY BALANCE CALCULATIONS 9. ENCAPSULATION
EQUIPMENT 11. PROTECTIVE SPACECRAFT & EXPERIMENT COVERS 12.
TEMS

PROCUREMENT

2. FASTENERS 3. SPECIAL EQUIPMENT 4. SPECIAL TOOLS 5. LONG
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ETU ASSEMBLY

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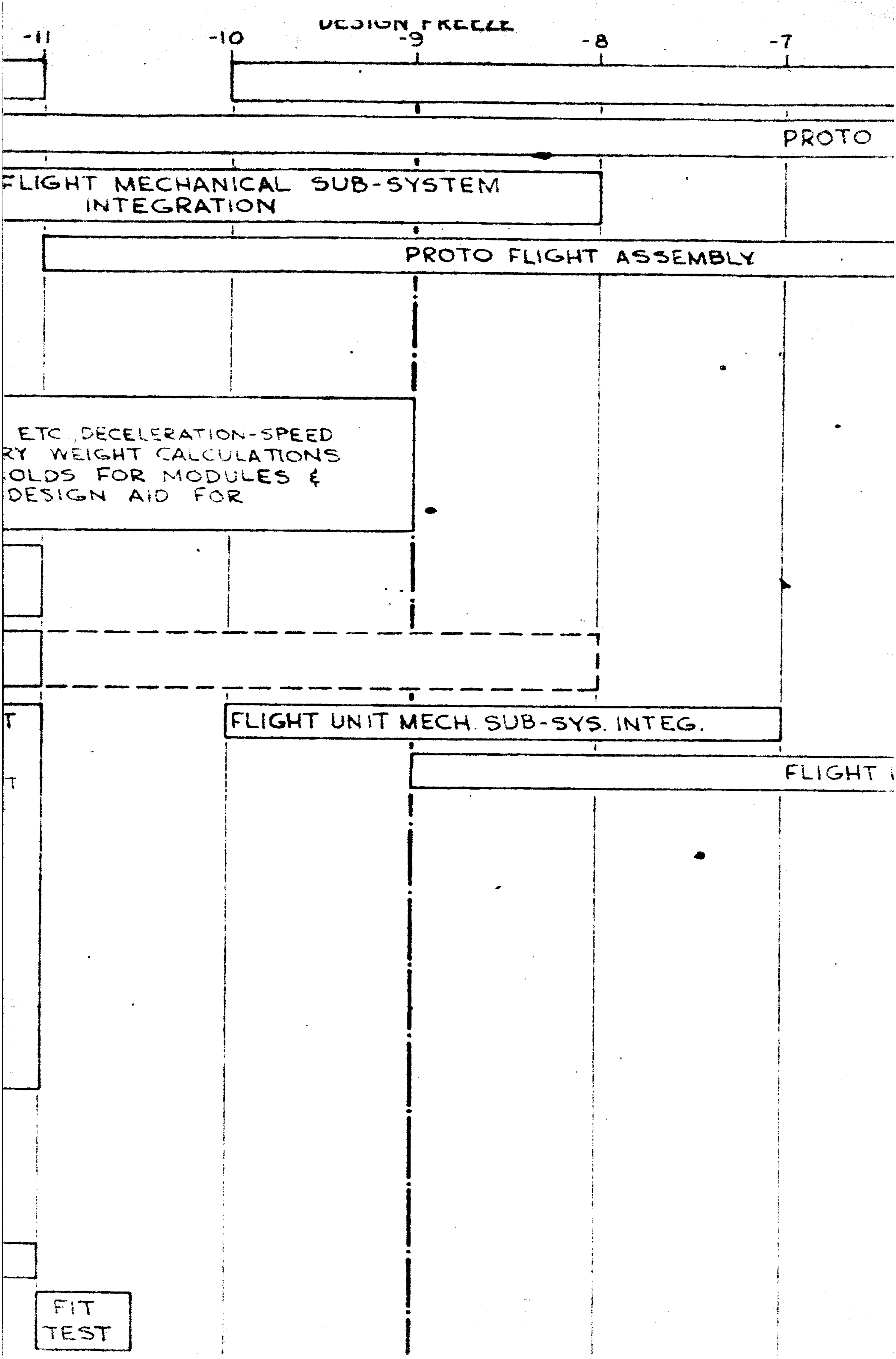
MODEL 10. ELECTRICAL INTEGRATION

SHIPPING CONTAINER



DESIGN FREEZE
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FLIGHT UNIT

FLIGHT UNIT

PROTO FLIGHT QUALIFICATION
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3. SOLAR ARRAY CENTER OF GRAVITY
4. STATIC & DYNAMIC BALANCE
5. PRELIMINARY VIBRATION -1.5x FLIGHT LEVELS 6. THERMAL CALIBRATION
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UNIT ASSEMBLY

FLIGHT UNIT QUALIFICATION
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3. SOLAR ARRAY CENT GRAVITY 4. STATIC &
BALANCE - PRELIMINA VIBRATION - FLIGHT L
5. THERMAL CALIBRA
6. THERMAL VACUUM
7. VIBRATION - RANDOM LEVEL ONLY 8. FINAL BALANCE 9. MOMENT INERTIA

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PRE-LAUNCH PREPARATIONS
1. PREPARE FOR FINAL PRE-FLIGHT CHECK-OUT
2. ASSIST IN ANTENNA CHECK - PROTO & FLIGHT
3. FINAL MECHANICAL ASSEMBLY PER CHECK-OFF SHEET
4. R F CHECKS - PROTO - GANTRY
5. FINAL FIT CHECK
6. SPIN BALANCE
7. PREPARE GANTRY AREA FOR INTEGRATION CHECK
8. FINAL MECHANICAL CHECK, GANTRY

LAUNCH

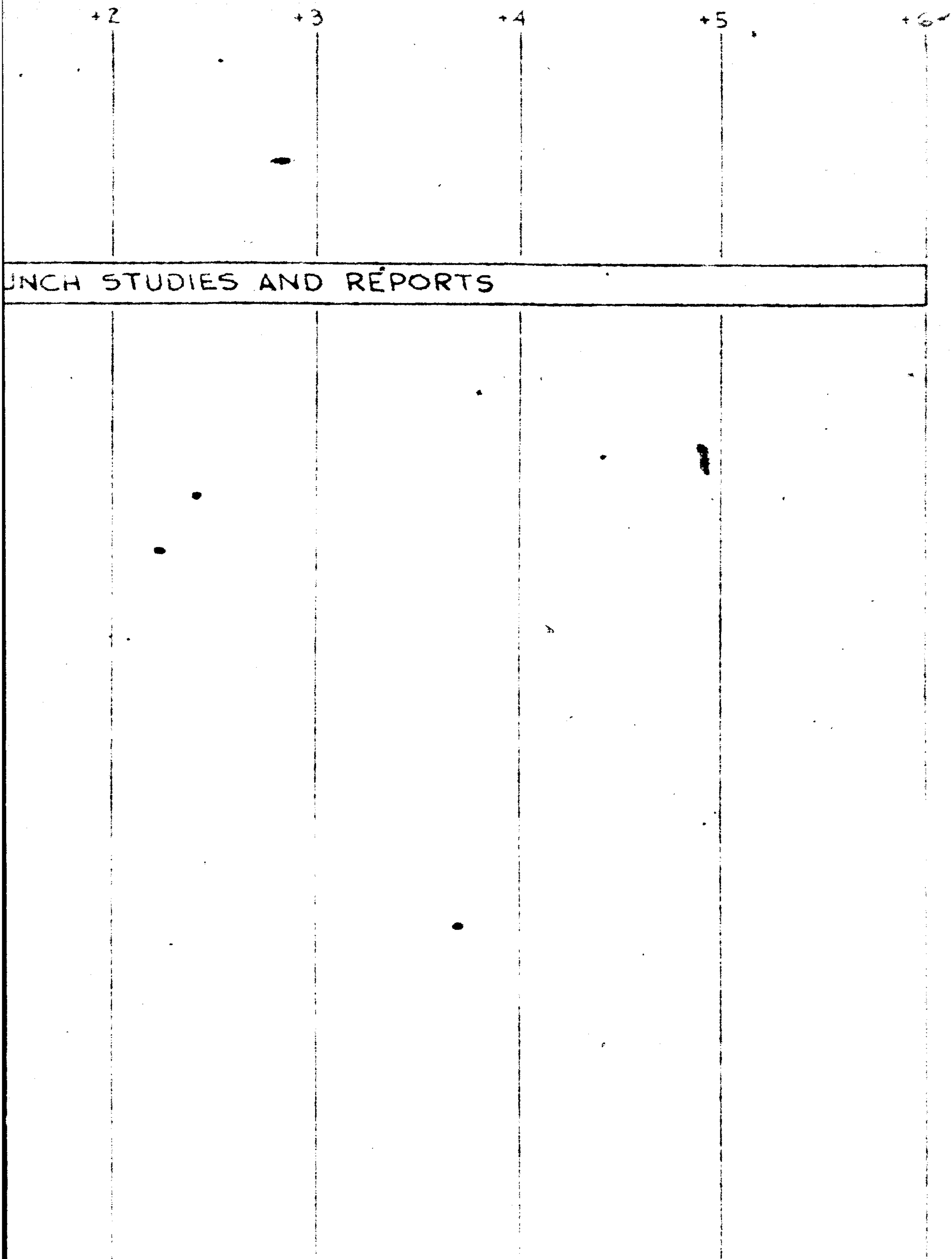


Figure 3 [illegible] Time Schedule of Mechanical Functions

DESIGN PRACTICES

The small scientific satellite designer must consider all the various factors that any good designer considers. He has one significant consideration that most other designers do not. There are no opportunities to repair, maintain, or adjust the satellite once the launch vehicle leaves the ground.

Simplicity in design generally means reliability of a system. Keep in mind what the structure or mechanism must do and design using the simplest approach which will accomplish the desired end result.

If a well-known technique for accomplishing a certain function is in existence, use it for it has generally been perfected. The same applies to the use of well-known materials. Why use a new exotic material that has not been qualified for space use, when there are many materials that have been subjected to the launch and orbital environment?

One cannot design to cover all possible failure modes. One can, however, design a system in such a way as to minimize the risk of failures. For example, by designing independent redundancy into a system and/or subsystem one can increase his confidence by a tremendous factor. One has to take the attitude that if anything can go wrong, it will, and make design trade-offs accordingly.

For example: Let us assume a satellite with 4 appendages folded alongside the last stage motor. These appendages are held in place by a Dacron or Nylon cord. To release these appendages the cord must be cut. For independent redundancy use two separate timers, two separate power sources, wired separately into two guillotine cutters. One may further assume that this redundant system may also fail and as a result try to design the appendages so that at vehicle/satellite separation the appendages will slip from underneath the cord without any interference from the vehicle or cord.

All stress calculations are based on maximum dynamic forces expected or calculated as a result of subjecting the spacecraft to a specification which has been generated for a specific vehicle. The criteria used at the Goddard Space Flight Center is to dynamically test the prototype structure to 1.5 times the flight levels. This approach insures a safety factor of at least 1.5.

For reliability's sake, order all fasteners, mechanisms and materials to an accepted and approved specification. Avoid using uncoated aluminum screws with tapped holes in aluminum. Mating aluminum to aluminum has a tendency to gall, making it difficult and sometimes impossible to remove these screws. Heli-coils are recommended for use in soft materials like fiberglass, epoxies, magnesium and at times in aluminum.

All structural hardware should be inspected for drawing conformity. It is not unusual to receive hardware that has not been fabricated per drawings and/ or specifications. Some of the discrepancies noted are: Out of tolerance, wrong finish, deep scratches, tool marks (gouges), sharp corners, wrong materials, wrong heat treatment, etc., etc.

Sharp corners should be avoided like a plague. Most failures in satellite structures occurred because a generous radius was not provided in some critical highly stressed area, such as a sharp corner.

Avoid designing a structure with flat head screws because most structural sections are too thin to properly accommodate a flat head screw and the screw head is usually overstressed when the tapped hole in one component and the countersunk hole in the other component are not concentric. The latter results in the inner tapered surface of the screw making contact with one side of the countersunk hole causing the screw head to bend and frequently break.

Provide a table (Attachment I) of recommended torques for every screw used on the satellite in order to be consistent in assembling the spacecraft. This should insure maximum reliability of joints in the event disassembly in the field is necessary.

DESIGN CONSIDERATIONS

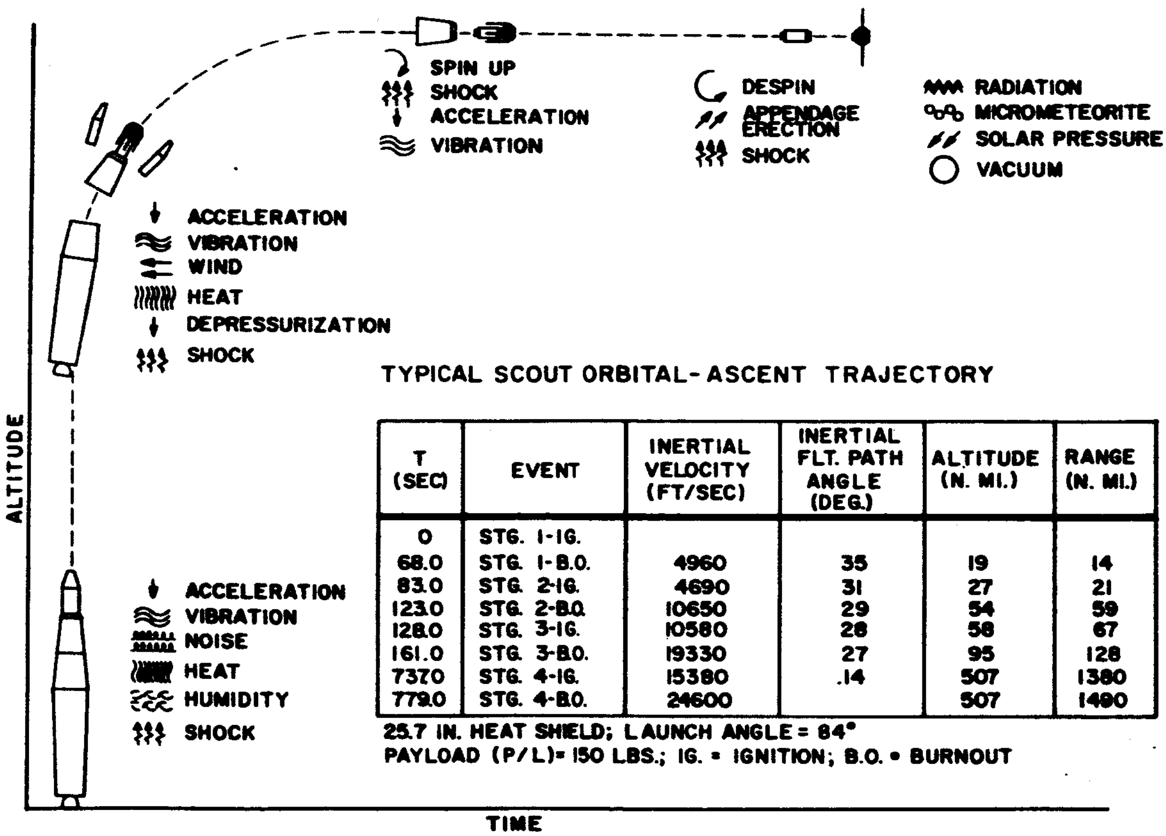
General

One of the biggest mistakes made by machinists and metal fabricators is the lack of attention given to fabrication techniques. Although most of the engineering analysis and design is done on paper the fabrication shops could possibly provide satellite parts that are machined, welded or riveted per design drawings, but yet these parts could be inferior or weaker than the calculated figures. This is normally attributed to high localized stresses within the hardware that were created by excessively deep cuts or cold working of the materials, improper adherence to drawing tolerances or the use of incorrect welding rods for welding these materials. These are only a few but important areas that should be considered. It is important to check on fabrication facilities, and any technique that is questionable should be improved upon and the importance and the reasons for the changes should be explained. Educating fabrication personnel is as important as educating yourself.

The ground handling environment, which includes shock and vibration during transportation, humidity during assembly and transportation, and corrosion, oxidation or contamination of mechanisms and/or satellite, has not been as severe as the launch environment. However, this is mainly due to an expended effort on the part of the cognizant spacecraft personnel to control this environment. This does not mean that the ground handling environment should be ignored for the occasion may someday arise when the ground handling environment may have to be considered in the design of satellites.

The loads imposed on the satellite structure are mostly a result of the launch phase environment as illustrated by Figure 4. This environment will expose the satellite to shock, vibration, acceleration, angular acceleration, noise, centrifugal forces, and possible aerodynamic heating.

After injection into orbit the structure/satellite is exposed to extremely severe vacuums, and temperatures, possible radiation exposure and micrometeroid damage. Since most of the small satellites are designed for one (1) year life, all of these parameters must be investigated to insure little or no degradation of materials or mechanisms within the satellite for its intended lifetime.



TYPICAL SCOUT ORBITAL-ASCENT TRAJECTORY

T (SEC)	EVENT	INERTIAL VELOCITY (FT/SEC)	INERTIAL FLT. PATH ANGLE (DEG.)	ALTITUDE (N. MI.)	RANGE (N. MI.)
0	STG. 1-IG.				
68.0	STG. 1-B.O.	4960	35	19	14
83.0	STG. 2-IG.	4690	31	27	21
123.0	STG. 2-B.O.	10650	29	54	59
128.0	STG. 3-IG.	10580	28	58	67
161.0	STG. 3-B.O.	19330	27	95	128
737.0	STG. 4-IG.	15380	.14	507	1380
779.0	STG. 4-B.O.	24600		507	1490

25.7 IN. HEAT SHIELD; LAUNCH ANGLE = 84°
PAYLOAD (P/L) = 150 LBS.; IG. = IGNITION; B.O. = BURNOUT

LAUNCH AND ORBITAL ENVIRONMENT

Figure 4—Launch and Orbital Environment

The basic design considerations discussed above provide a general outline of the primary areas that the structural engineer has to consider. Initially the engineer must formulate an approach and establish in his mind how the task will be undertaken. He must visualize the shape or configuration that will eventually involve as the design of the various subsystems becomes finalized the difficult task begins.

The one question that has been raised by many structural engineers and some of you may also ask this question; is the satellite designed to withstand an environment or is it designed to the test specification? To be safe, it is prudent to design to whichever is most severe and this generally is the dynamic test specification. The dynamic test specification is generally more severe than the launch environment but it must be remembered that some level of confidence should be established. This is generally done by using whatever data is available and flight test levels established at what might be considered the worst possible expected condition. These levels are then increased by a safety factor of 1.5 for both the Engineering Test Unit and prototype. If both units pass the dynamic tests it is almost certain that the flight unit will pass the lower levels, but the environmental test does not guarantee the satellite a successful launch and long operating life. It discloses defects and establishes the readiness for flight of the flight unit. Since the test specification does not provide for testing the satellite for more than two (2) weeks in thermal-vacuum it is the structural engineer's responsibility to select his materials so that there will be no appreciable degradation during the satellite's design life.

What are the actual design considerations? There are seven areas that must be considered, these are listed below:

1. Scientific experiments
2. Orbital environment
3. Launch environment
4. Pre-launch environment
5. Materials
6. Fabrication techniques
7. Testing

The scientific experiments will generally establish most of the orbital requirements. Some of the requirements that are generated by the experimenters are as shown on the following page.

1. Either rotating or non-rotating satellite
2. If rotating, spin rate will be given
3. Experiment orientation within the satellite
4. Orbit requirements (Apogee and Perigee and inclination)
5. Aspect of satellite at launch
6. Use of non-magnetic materials (Magnetometer type satellites)
7. Whether hermetically sealed or not
8. Maximum permissable coning angle (dynamic unbalance)

Since most of the scientific satellites designed by GSFC were spin stabilized, I will only discuss the approach for designing this type of spacecraft and present the step by step approach by which the satellite is conceived, designed, assembled, tested, shipped to the launch facility and finally launched.

The information that is generally needed to properly begin an initial design layout is as follows:

1. The number of experiments, plus associated electronics
2. Scientific objective, e.g., measure energetic particles in the Van Allen Belts or ionosphere research, etc.
3. Maximum allowable weight
4. Spin rate (For stabilization and/or experimenters' requirements)
5. Total electrical power required
6. Orientation of sensors in relation to spin axis
7. Look angles (per Figure 5)
8. Estimated weight of sensor plus associated electronics
9. Physical dimensions of sensor plus associated electronics
10. Orbit aspect
11. The need for a tape recorder or other specialized equipment
12. Physical size and quantity of batteries
13. Special requirements, i.e., some experiment on a boom with a specified minimum distance from center of gravity, a requirement for a kick or retro motor, nutation dampers or an attitude control system.
    NOTE: For a more complete set of information requirements see Attachment 2 - Mechanical Interface Requirements.

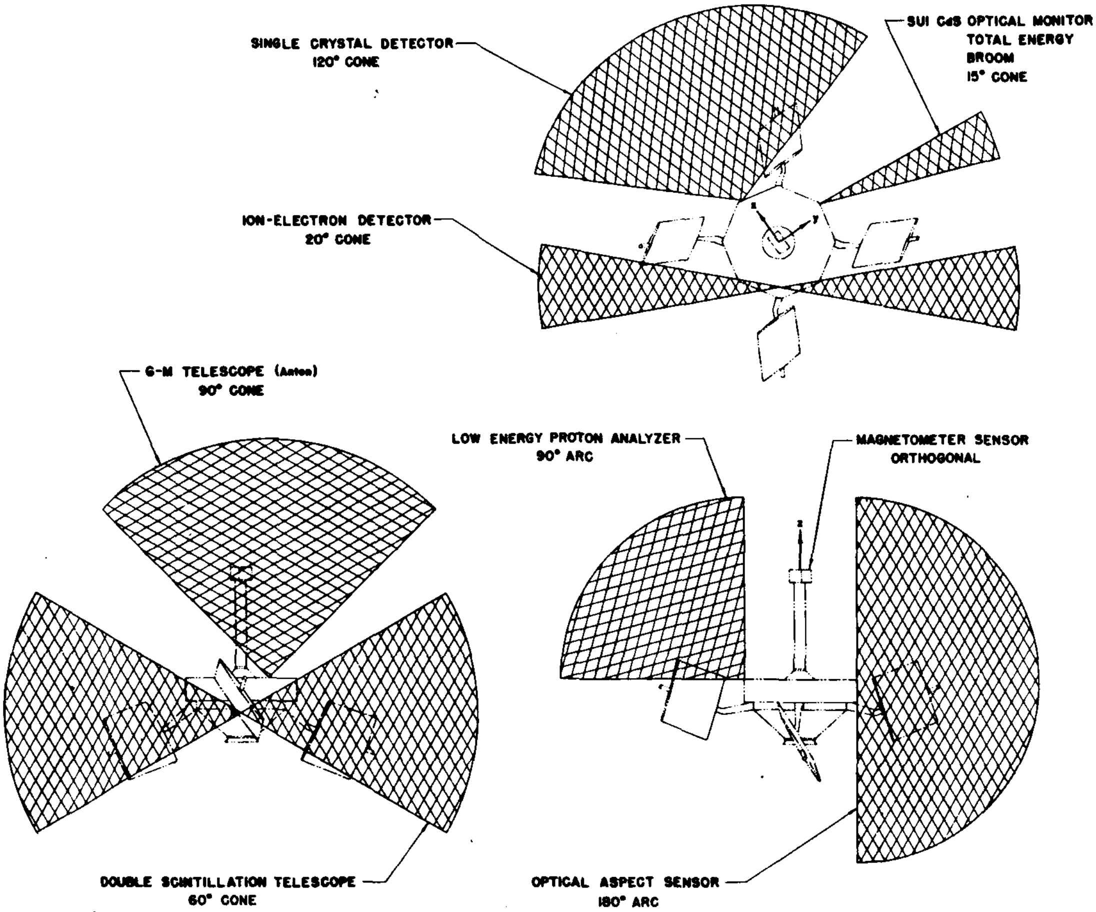


Figure 5–Sensor Look Angles (Explorer XII)

Schedule Preparation

One of the first and most important steps to be taken is the preparation of a realistic schedule (Figure 6). Realistic in that a reasonable time should be estimated to perform all the necessary functions. The schedule should cover all the important milestones, from design conception to shipment to the field. Included in this schedule should be all the various items for which the structural engineer is responsible. By including all of the various items one can tell at a glance how the flight unit is effected if the prototype's milestones begin to slip, etc. The schedule may also be used as an excellent reminder for ordering long lead-time items, for planning manpower requirements, and for showing how the structural engineer must interface in time and function with the other portions of the satellite team.

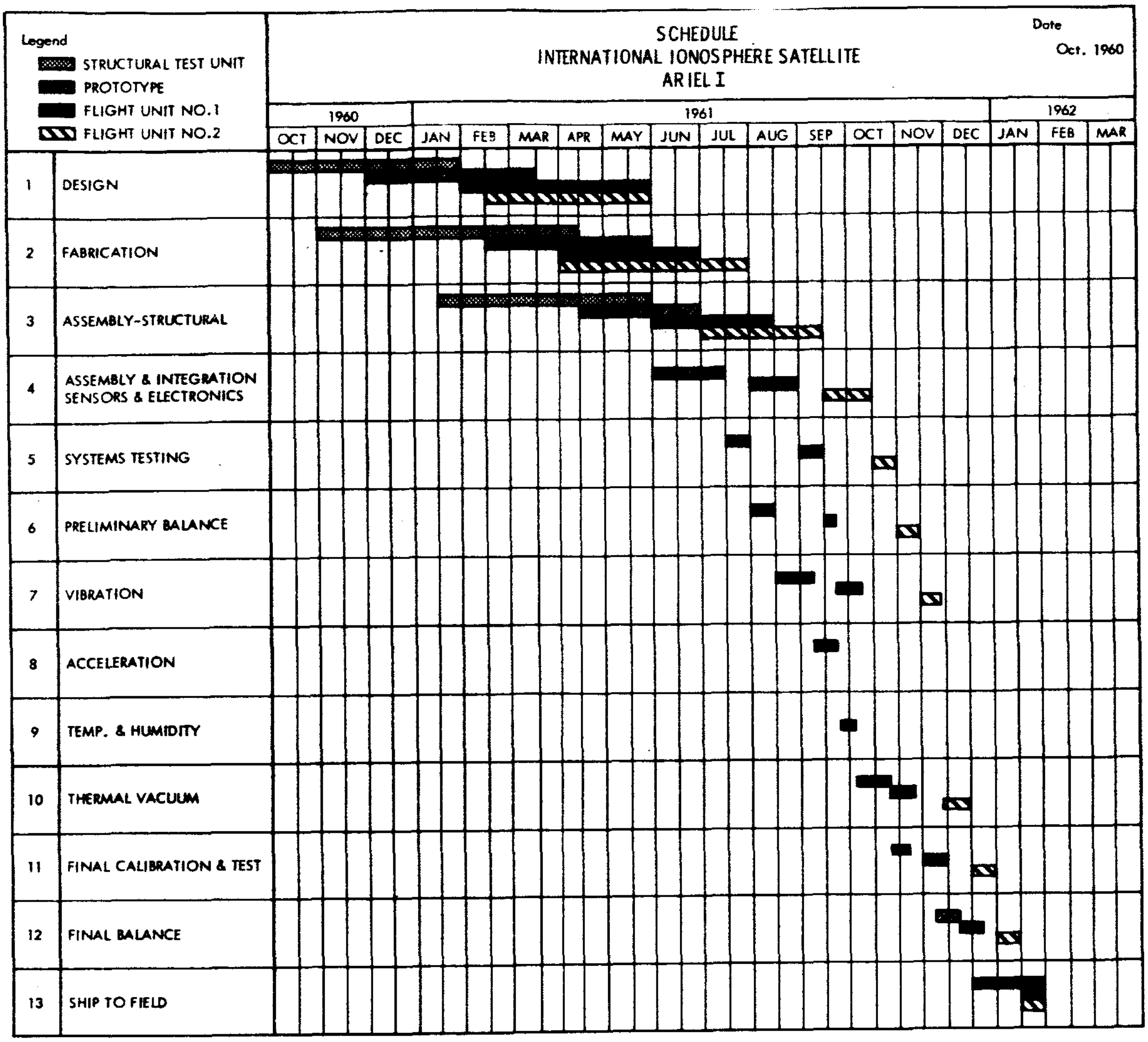


Figure 6–Planned Program Schedule (Ariel I)

Determining the Basic Shape

From the power and orbital requirements, the structural engineer can determine if the satellite configuration has adequate area for attachment of solar cells or if solar paddles must be employed. Generally if the subsystems and sensors have been designed with the use of microelectronic components the satellite will undoubtedly have sufficient experiments aboard to require more power than could be obtained from a satellite whose surface is covered by solar cells. This means solar celled paddles. One advantage of solar paddles is that generally the paddle area can be increased without requiring a major redesign, but a surface-covered

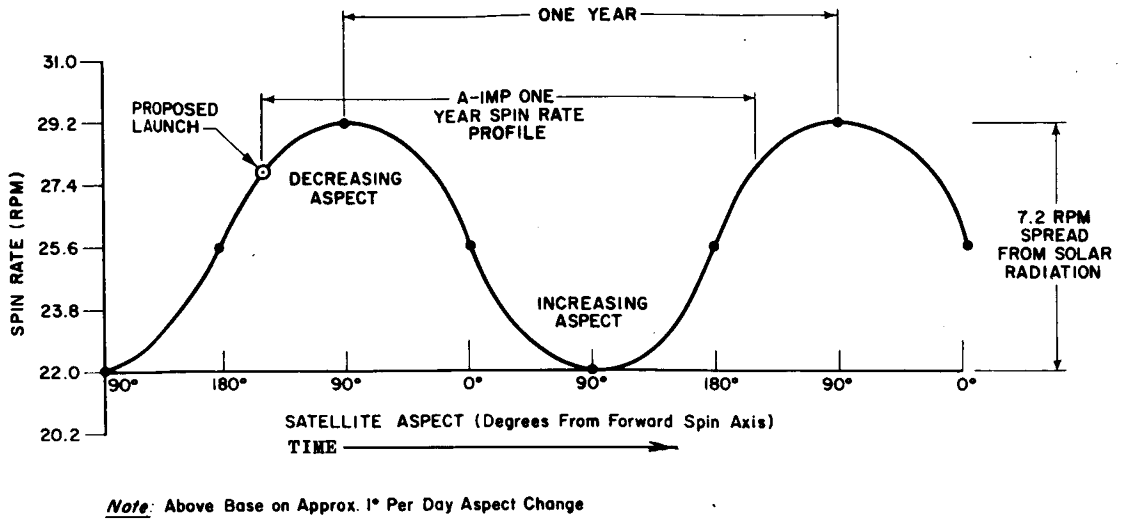


Figure 7–Spin Rate vs Time (Result of Solar Pressure)

satellite would require a complete redesign or elimination of some sensor but if paddles are employed remember to calculate the effect on spin as a result of solar pressure (Figure 7 and Reference 1). It is always wise to design a satellite with this thought in mind, i.e., the power requirements will be increased. One bit of information that may influence your decision for selecting a particular shape is that a sphere is the easiest configuration for calculating thermal coatings and temperature gradients.

Initial Layout and Interfaces

This initial layout determines the feasibility of meeting all the requirements of the experimenters. If all the requirements cannot be met because of some fixed restraints, the experimenter is asked to compromise. This process of compromise and trade-off may repeat itself several times until an acceptable spacecraft design has been established. At this time the structural engineer should prepare a mechanical interface document (Attachment 2) whose purpose is not only to gather additional information but also to clarify all the mechanical aspects (materials and hardware to be used and how they anticipate fastening their components to the structure - etc.) The reason for this is that generally the experimenters and scientists may not have as good a mechanical background as the structural engineer. Also, and this is very important if you can control and minimize the mechanical interface between the experiment and the satellite you will eliminate most of the problems associated with compatibility and interchangeability of packages. This may sound exaggerated but experience gained on past programs has shown that a lack of compatibility and interchangeability of packages is time consuming and costly. It was not unusual to find the structural engineer expending a large effort trying to rework most of the spare experiments and electronic components to fit within the flight spacecraft. If you carefully check Attachment 1, you will notice that this document goes one step further, in that it requires the experimenters and subsystems designers to fit their experiments and circuits within a frame of a fixed configuration. The only variable is the height. Also, the structural designer did not only have the frames designed but also fabricated. This approach may mean more preliminary work for the structural engineer but by having full control of every item that is attached to the satellite, the time and manpower saved far outweighs the effort required to supply this hardware. A Mechanical Integration document should be distributed to every person who is directly associated with a given program and a deadline should be set for providing all the information that will be needed to begin finalizing the spacecraft design.

The initial layout is compared with the new data, if the changes are insignificant (and they normally are) the design engineer initiates a mathematical analysis to determine the section modulus of all the structural components. This is also the time to order all the long lead-time items whether they be special screws, fasteners or materials.

Moment of Inertia Considerations

When the satellite configuration has been established the structural engineer undertakes the task of computing the mass Moments of Inertia (MOI) about three mutually perpendicular axes. Of these axes, the spin axis is designated Z axis and the two lateral axes as the X axis and the Y axis. The reason for the computations is to make certain that the satellite is designed and assembled with its large MOI about its spin (Z) axis and thereby insure the inherent stability of the spinning spacecraft. It has been our practice at Goddard Space Flight Center to design spin stabilized spacecraft with the spin MOI a minimum of 5% greater than the principal lateral axis MOI. The above 5% figure serves only as a guide and the structural engineer may be required to deviate from it in order to meet more demanding scientific requirements.

In determining an acceptable minimum difference between the spin MOI and principal lateral MOI the following areas which could affect stability should be investigated:

Appendages and Projections—Solar paddles and other appendages could, as a result of solar pressure and/or aerodynamic drag, produce torques (Reference 1) and cause the satellite to perturbate. Nutation dampers could be added to the spacecraft to eliminate or reduce the undesirable motions.

Spin Rate–The lower the spin rate the greater the chance of the satellite becoming unstable; therefore a minimum spin rate should be selected which will insure stability for the life of the satellite.

Accuracy of Measurements–The method used for measuring the MOI should have an error less than 2%, and a minimum of three reasonably-spaced transverse MOI should be measured and plotted to insure that the largest transverse MOI has been located.

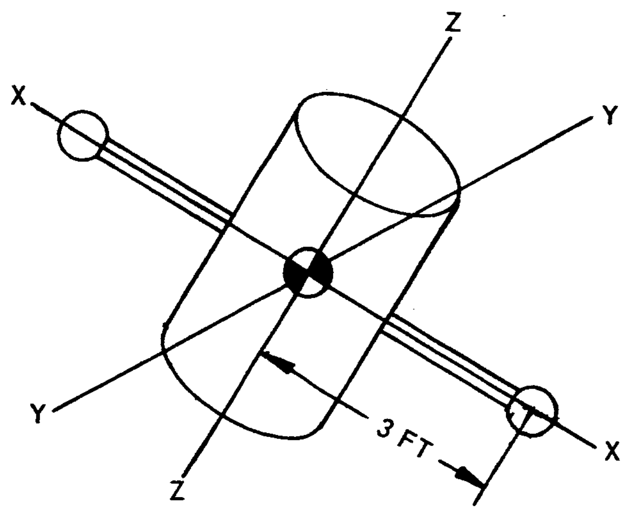
Component Replacement–If last-minute replacements in the field are required careful consideration should be given and a math analysis made so that the recommended 5% difference in MOIs is maintained.

Environmental Effects–Consideration must be given to short-term and long-term environmental effects in orbit to insure that the 5% figure is maintained. For example release of stored gas, sublimation, mechanism deployment, etc., must all be planned for.

For a satellite that is flat and large in diameter with 3 or more equally-spaced appendages, stability is no problem. This configuration approaches a toroid or a flat disk whose spin axis to lateral axis ratio ($I_{spin}/I_{lateral}$) approaches (2) two but, for satellites that are spheres or cylinders without appendages the ratio could be less than (1) one. This is why it is very important to calculate and recalculate the MOI every time some changes are made. If the experimenters' requirements are such that the MOI ratio is less than (1) one the solution would be either to compromise with the experimenter by locating the experiment closer to the C.G. or attach weighted booms in the plane through the C.G. and perpendicular to the spin axis. Note of caution: Use three or more booms. Reason: two booms would not only increase the spin MOI but also the lateral MOI by the same amount or possibly more. The lateral axis would be increased more if the booms were placed below the C.G. plane rather than through it.

Example:

Let us assume a cylindrical-shaped satellite with a 5 slug-ft^2 MOI through the spin axis. The lateral X-X and Y-Y is equal to 5.1 slug-ft^2, or 0.1 slug-ft^2 greater. Let us add (2) two weighted booms to axis X-X. These booms weigh 2 lb each and their C.G. is (3) three feet from the spin-axis. MOI of the booms is equal to Mr^2 or

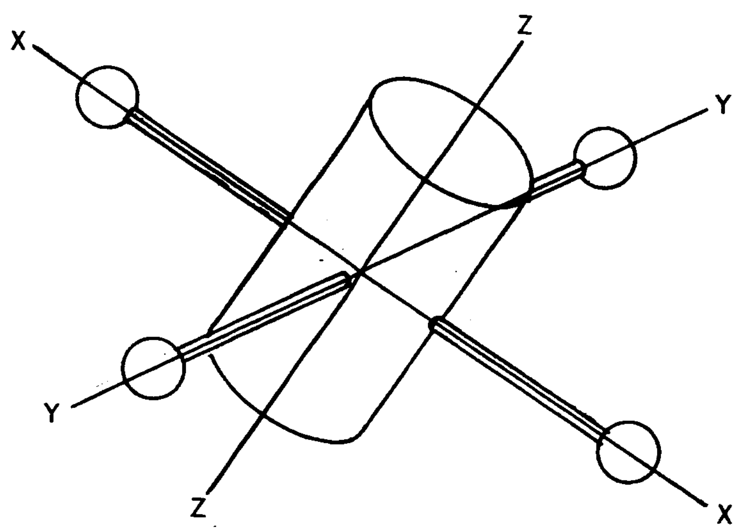


$$MOI_{Z-Z} = \frac{4}{32} \cdot 3^2 \qquad MOI_{Y-Y} = \frac{4}{32} \cdot 3^2$$

$$= \frac{9}{8} \qquad = \frac{9}{8}$$

$$MOI_{Z-Z} = 1.125 \text{ slug-ft}^2 \qquad MOI_{Y-Y} = 1.125 \text{ slug-ft}^2$$

Looking at the illustration you will notice that the same increase applies to both the Z-Z axis and the Y-Y axis. Therefore, the MOI ratio is still less than (1) one. But let us simplify the above case and assume we add (2) two more identical booms to axis Y-Y.



Example:

MOI of booms only about Z-Z axis.

1. $$MOI_{B\text{-}Z} = \frac{8}{32} \cdot 3^2$$

$$= \frac{9}{4}$$

$$= 2.25 \text{ slug-ft}^2$$

MOI of booms only about Y-Y axis

2. $$MOI_{Y\text{-}Y} = \frac{4}{32} \cdot 3^2$$

$$= \frac{9}{8}$$

$$= 1.125$$

MOI of booms only about X-X axis

3. $$MOI_{X\text{-}X} = \frac{4}{32} \cdot 3^2$$

$$= \frac{9}{8}$$

$$= 1.125$$

4. $$MOI_{\text{Corrected } Z} = MOI_{\text{Initial } Z} + MOI_{\text{Booms}}$$

$$MOI_{C.Z.} = 5 + 2.25$$

$$MOI_{C.Z.} = 7.25 \text{ slug-ft}^2 \text{ total Z}$$

5. $$MOI_{\text{Corrected lateral}} = MOI_{\text{Initial lateral}} + MOI_{\text{Booms}}$$

$$MOI_{C.1.} = 5 + 1.125$$

$$MOI = 6.125 \text{ slug-ft}^2 \text{ total lateral}$$

6. $$\frac{7.25 - 6.125}{6.125} = .186 \text{ or } 18.6\%$$

By adding (4) four booms the spin MOI is 18.6% higher than the lateral axes. Notice the spin MOI was increased by 2.25 slug-ft^2 but the lateral axis by half as much. If (3) three booms are used the spin MOI will increase about 1.75 times the lateral MOI increase.

GSFC has used the torsion pendulum method (Figure 8) to measure the actual MOI of all GSFC-built satellites; although there are other methods for measuring the mass moments, the accuracy and simplicity of the torsion rod pendulum is slightly better than the other systems.

The system is simple, effective and accurate to less than (2%) two percent.

The method employed is to design a torsion rod that will provide for a period of anywhere from 10 to 15 seconds. This of course is for the spin axis in orbital configuration (paddles, booms, etc., extended). The period will be much faster for other configurations and axes. The reason for this time approach is to minimize outside disturbances (torques due to sudden air movement) as much

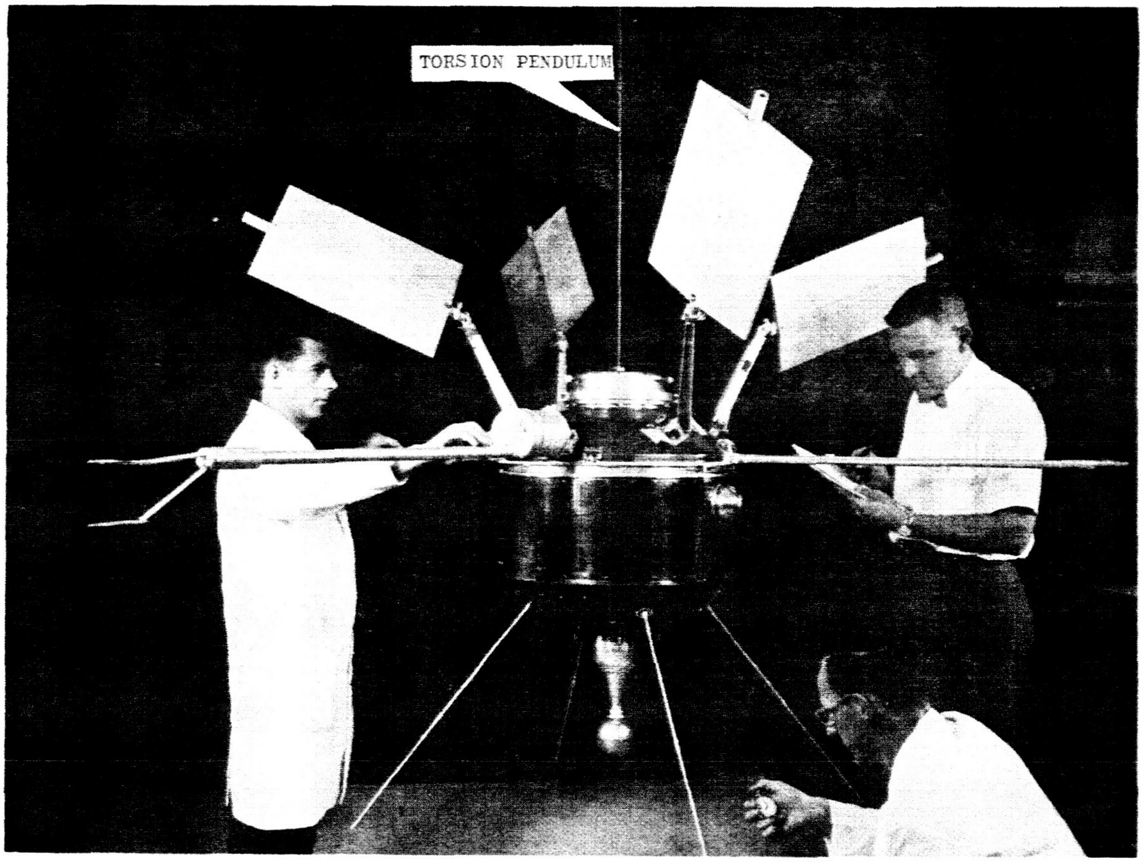


Figure 8—Moments of Inertia Determination (Ariel I)

as possible. A torsion rod with a relatively long period generally possesses very little restoring torque. This torque cannot cope with outside forces as readily as a larger diameter rod. Also a larger diameter rod has more strength and rigidity; it can also be used repeatedly without fear of failing from fatigue. To reduce or eliminate other disturbances always attach the fixture to rigid surface preferably to part of the building, i.e., steel I-beam in ceiling or some similar steel plate that is cemented in reenforced concrete. One word of caution; do not attach to a ceiling or building that is subjected to vibrations, i.e., large machinery in building or close by. Vibrations will introduce additional error in measurements.

The rod should be tested to at least (4) four time the satellite's weight. This can be done by either a tensile tester or suspended weight.

The equations for the design of the Torsion Rod are:

a. $T = 2\pi \sqrt{1/k}$

b. $k = \dfrac{JG}{L}$

c. $J = \dfrac{\pi d^4}{32}$

d. $M = \theta k$

e. $G = .4E$

Where:

T = time of period, seconds

I = Moment of Inertia of Mass, lb-in.-sec^2

L = Length of Rod, inches

G = Modulus of Rigidity, psi

J = Polar Moment of Inertia of Cross Section of Rod, $(inches)^4$

E = Modulus of Elasticity, psi

θ = Twist of Rod, radians

disk and satellite by dividing the number of oscillations into the total time for these oscillations. The MOI of the satellite is derived by substituting the measured and calculated values in the equation below.

$$\frac{I_{sat.}}{T^2_{sat.}} = \frac{I_{disk}}{T^2_{disk}}$$

Satellite-Launch Vehicle Compatibility

In designing a spacecraft it is necessary to make the satellite compatible with the vehicle. (See Figure 9) Most vehicle/payload restraint documents provide enough information to enable the structural engineer to design a spacecraft that will mate properly with the launch vehicle. Most of the small satellites designed by the Goddard Space Flight Center had several appendages that were folded parallel or along the side of the last stage of the launch vehicle. These appendages have to be designed so as not to interfere with vehicle functions, i.e., should not prevent or hinder proper separation of the preceding stage from the last stage; also they should fit within the fairing envelope.

The satellite is affixed to the last stage of the launch vehicle by means of a Marmon-type clamp. (See Figure 10). This clamp is designed with two distinct purposes in mind. One is to rigidly affix the spacecraft to the vehicle and the other is to enable a clean and quick separation from the last stage, at some preset time.

Special explosive bolts or bolts with bolt cutters are used to properly torque the payload to the last stage and provide separation. A clean separation is obtained by physically separating both halves of the clamp with large, flat springs. (See Figure 10). Relative velocity between the spacecraft and last stage is obtained by means of a separation spring. This separation spring is located between the payload and the last stage, and stays with the last stage.

Accessibility

The biggest time saver that enables a satellite program to proceed with some degree of efficiency is the accessibility of components or subsystems within the spacecraft. It is not unusual for an instrument or an experiment to be removed from the satellite at least (100) one hundred times from the time the satellite is first assembled to the time it is placed in orbit. In planning for this requirement, it is prudent to design the satellite so that all the subsystems can be easily and quickly removed with little or no degradation of all mating components. To make

M = Twisting Moment on Rod, in.-lb

K = Spring Constant of Rod, $\frac{\text{in.-lb}}{\text{rad.}}$

d = Rod diameter, inches

Example:

Assumed Rod Length = 30 inches

Diameter = .218 inches

I = 9.3 slug-ft^2 or 111.6 lb-in.-sec^2

Material = Stainless Steel E = 29×10^6 psi

Angular Rotation = 15° = .2618 radians

$$M = \theta K = \theta \frac{JG}{L} = \theta \frac{\pi d^4 \ (.4E)}{32\,L}$$

$$M = .2618\,\pi \frac{(.218)^4 \ (.4(29 \times 10^6)}{32 \times 30}$$

$$M = 22.5 \text{ in.-lb}$$

$$T = 2\pi\sqrt{I/k} = 2\pi\sqrt{IL/GJ}$$

$$T = 2\pi\left(\frac{32\,IL}{.4\pi\,Ed^4}\right)^{\frac{1}{2}} = \frac{2\pi}{d^2}\left(\frac{80IL}{\pi E}\right)^{\frac{1}{2}}$$

$$T = \frac{2\,\pi}{(.218)^2}\left(\frac{80(111.6)30}{\pi(.4)\ (29 \times 19^6)}\right) = 11.5 \text{ sec period}$$

It is a simple matter to measure the MOI of a satellite with a torsion rod. All that is needed is a cylindrical solid disk. Knowing the weight and the radius of this disk it is a simple matter to calculate its MOI (Attachment 3 goes into a more detailed measurement), or MOI = 1/2 times its weight divided (÷) by g times radius squared. MOI = 1/2 Mr^2.

This MOI is stamped on the disk and recorded in the log book for ready reference.

This disk is suspended on the rod, oscillated through angles less than 20° and the period recorded for about 10 to 20 oscillations; the satellite is also subjected to the same procedure. The average period is then obtained for both the

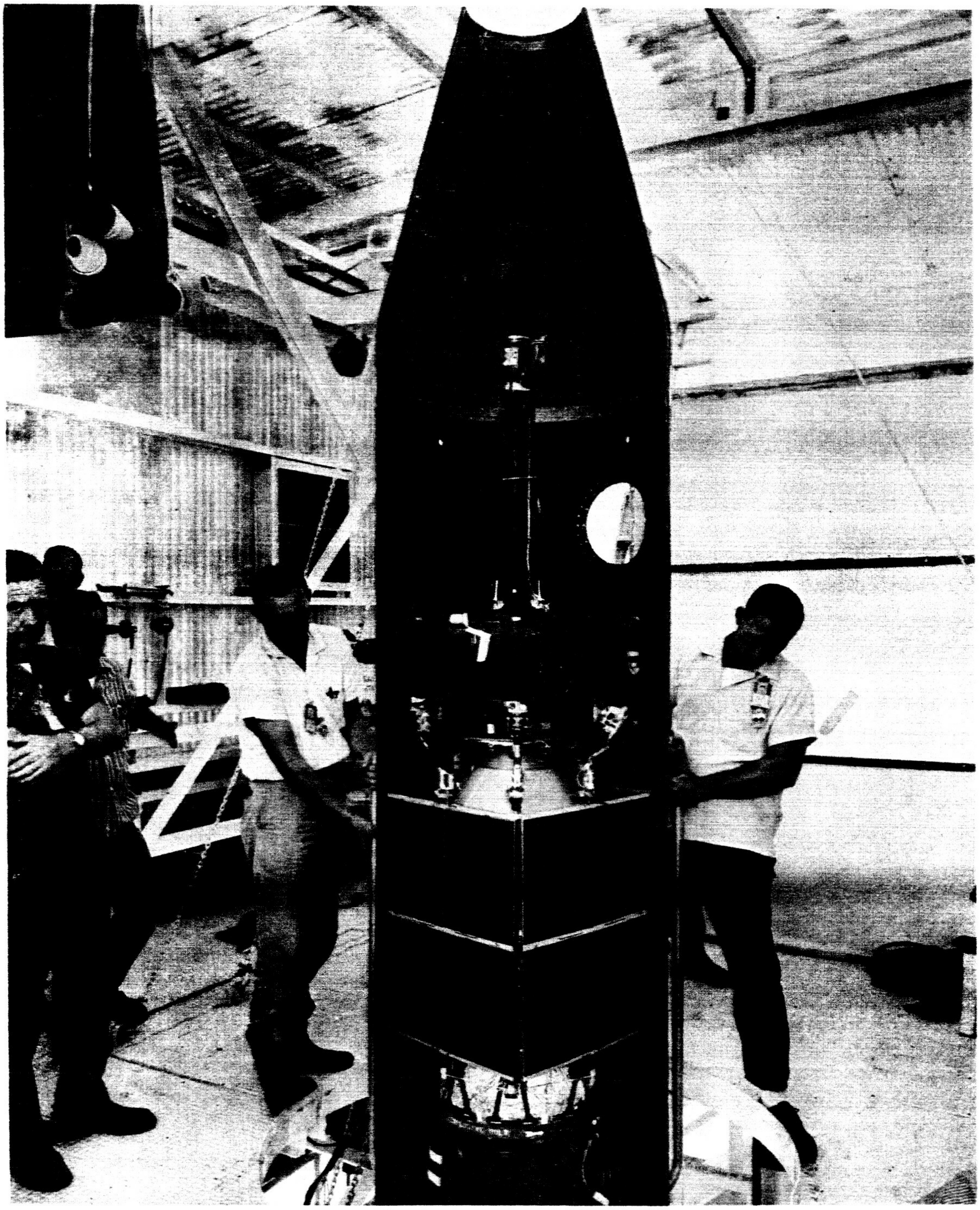


Figure 9—Vehicle Satellite Compatibility

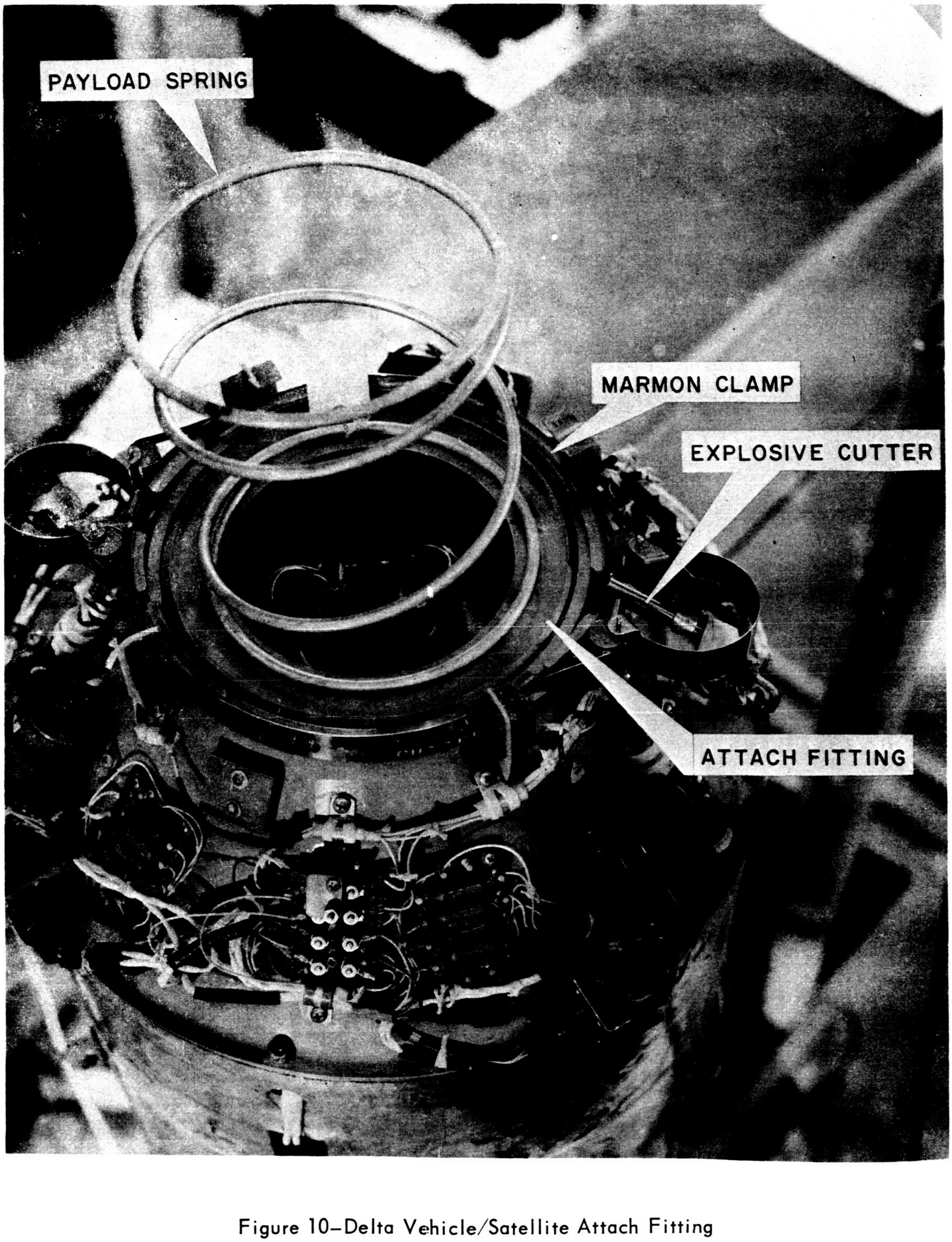


Figure 10–Delta Vehicle/Satellite Attach Fitting

a satellite accessible, it must be kept in mind that the satellite be designed and constructed with the least possible number of pieces. Also, if a subassembly, cover or a mechanism can be held in place with two (2) screws why fasten it with more, even though it may look safer and stronger.

Materials

The question some of you may ask is: What kind of materials do I use for a structure? The answer to this would be: Materials that are easy to obtain; easily machined; homogeneous; consistent from one lot to another; a good thermal conductor; has a very low vapor pressure; and most of all has a high strength to weight ratio.

It is very important that all the materials selected be compatible with not only the structure but with the subsystems and electronic components. Long-term problems could develop that might cause failures. These problems could be in the form of chemical reaction or redeposition of metals and/or organic materials on precision instruments and electronic circuitry.

Outgassing of certain materials and metals could cause failure of an experiment and/or subsystem. Materials with a high vapor pressure will coat optics thereby causing either a malfunction or erroneous data. Some metals that have a high vapor pressure have been known to form metal whiskers on electrical terminals thereby causing an electrical short in the system. One of these metals is cadmium. Avoid using cadmium-plated materials.

The materials most commonly used by the Goddard Space Flight Center on its small satellites are aluminum, magnesium and fiberglass.

Use of materials, such as titanium and beryllium, should be confined to special applications. If you decide to use such materials for space applications, it is recommended that you contact the manufacturers and other users for data.

Thermal Design Considerations

In designing a structure, consideration should be given to a very important area that is often overlooked. This is the area of thermal conductivity between mating surfaces. The structural engineer should make every effort in close conjunction with the people responsible for the passive or active thermal control to determine the total power dissipation of each and every subsystem. The subsystems with the highest power dissipation should be given priority when bolting to a good heat-sink. If calculations indicate that a particular subsystem has insufficient thermal paths, plan on providing additional paths whether they be screws, rivets or metal straps, made out of a good thermal conductor like aluminum. Subsystem suppliers of battery packs, electromechanical timers, as well as other heat generating subsystems, are highly dependent on the structural engineers to provide assistance in designing containers and locating these subsystems within the structure. The structural designer must provide a good thermal connection to the structure. In outer space, convection cooling is not available for dissipation of heat produced by subsystems. The structural engineer must have a good knowledge of thermal properties of materials used for outer space.

It is sometimes required to use beryllium oxide or boron nitride washers in order to prevent certain electronic components from failing due to excessive heating. These washers provide both excellent electrical insulation and excellent thermal conductivity.

Normally, thermal radiation within the satellite can be improved by painting everything inside the satellite with heat absorbing paint. Either black or certain white plastic paints are used. This approach reduces the temperature gradient between the hottest and coldest subsystems inside the satellite by several degrees.

RF Design Considerations

When designing the exterior covers of the satellite keep in mind a common source of trouble, i.e., RF leakage. RF has a tendency to work its way inside the satellite through wires, openings and loose fitting covers. To prevent this from happening take extra care in designing all the covers. Eliminate all unnecessary openings and shield the openings that are mandatory. Shielding, as well as the covers, to be effective should be either metal or have a metal coating.

Electrical continuity between mating structural parts has not been a design or assembly problem in the past. The use of aluminum structural components and aluminum fasteners that are anodized may cause electrical difficulties during electrical integration or testing. To alleviate this possibility, remove all anodizing from mating surfaces and, after assembly, test for continuity between all mating surfaces.

Structural Design Loads and Calculations

General–The section modulus of each component is calculated on the basis of exposure to the maximum dynamic forces. These forces are generated by both the pre-launch and launch environment. Of these two, the launch environment is the most severe. Therefore, all calculations are based on a test specification that is usually generated for a specific vehicle. It is also the Goddard

Space Flight Center's policy to test the Engineering Test Unit and prototype units to levels that are 1.5 times higher than flight levels. This means that the satellite should be designed to pass the prototype levels of shock, vibration, acceleration, noise and appendage erection loads. This approach will provide a 1.5 safety factor for the flight unit.

<u>Vehicle Vibration Environment</u>—Listed below is the Delta launch vehicle vibration specifications that will dictate the loads to which the spacecraft will be designed and tested.

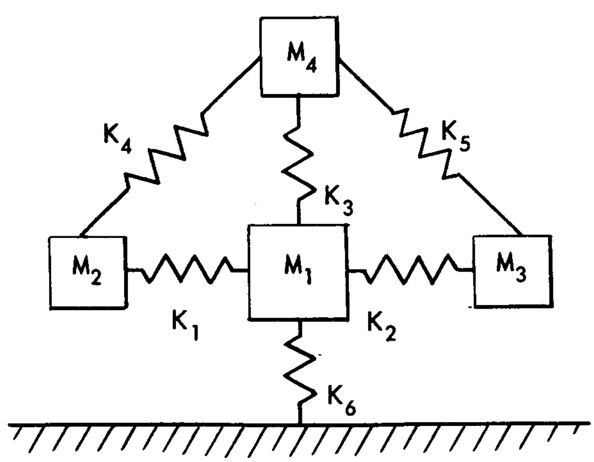
SPACECRAFT DESIGN QUALIFICATION
(For Delta (DSV-3C and 3D)
Sinusoidal Vibration*

Axis	Frequency (cps)	Duration (min)	Acceleration (g. 0-to peak)	Sweep Rate
Thrust (Z-Z-Axis)	10-50 50-500 500-2000	1.17 1.66 1.00	3.8* 7.5 21.0	2 octaves per minute
Lateral X-X Axis and Lateral Y-Y Axis	8-500 500-2000	2.99 1.00	2.3* 3.8	2 octaves per minute
	Grand Total: 11.81 minutes			
When the specified accelerations cannot be attained due to armature displacement limitations, the input may be a constant displacement not less than 0.5 inch double amplitude.				

RANDOM VIBRATION				
Axis	Range (cps)	PSD Level (g 2/cps)	Acceleration (g-rms)	Duration
Thrust (Z-Z) Lateral X-X Lateral Y-Y	20-2000	0.07	11.8	4 minutes each axis
Grand Total: 12 minutes				

Vibration Amplification—The specification levels are not the true criteria for determining the stresses that will be created by sinusoidal vibration, rather amplification within the satellite at resonant frequencies are the predominant loads. It is not unusual to record an amplification or "Q" level of twenty (20) at the resonant frequency of some structural member. Several years ago the structrual engineer would design a structure with the assumption that the amplification could create loads as high as 100 g's in the thrust axis and as much as 50 g's in the lateral axis. Of course these assumptions applied to an old Delta specification whose vibration levels were much higher in the upper frequencies, but as more satellite experience is gained and a better knowledge and understanding of the vehicle's dynamic responses is known the structural engineer can equate this valuable information and design his spacecraft to be compatible with the vehicle. A good example is the dynamic responses of the Delta vehicle. Flight vibration data from one of the earlier Delta flights, (Reference 2), recorded several distinct vibration frequencies. Most of these were transients of less than one (1) second duration but there were two (2) frequencies (one in thrust and the other lateral) that could be detrimental to a satellite if a satellite had resonant frequencies equal to the measured values. One of these was a 26 cycles-per-second thrust frequency of a six (6) second duration and the other was lateral 9.4 cps for approximately one (1) second. Based on this data it would be prudent to design a spacecraft well above these frequencies. The structure can be compared to a multi-spring-mass system as per the illustration. A system that most of the time is too complex to analyze because of its complexity. Therefore reliance is placed on past experience.

The multi-spring-mass illustration is analogous to a complex structure. A system where all the masses (M) and spring constants (K) are different. The



MULTI-SPRING-MASS SYSTEM

satellite structure has a bit more complexity than depicted by this illustration. There are several more masses than illustrated and the spring constants are normally undefinable. Transmissibility ("Q") is impossible to calculate; therefore reliance is placed on obtaining this information by sinusoidal vibration testing.

Resonant Frequencies of the Basic Structure of Some of the Successful Satellites

Satellite	Weight (lb)	Shape	Thrust	Lateral
1. Ariel I (Joint U.K.-U.S.)	136	Cylindrical with 4 paddles and booms	90-110	35
2. Explorer XVII (Atmospheric Structures)	410	Sphere Heremetically Sealed	50-100	9
3. Explorer XVIII (Interplanetary Monitoring Platform)	137	Octagon with 4 paddles and 2 booms	75	Clearly Undefined
4. Explorer XXVI (Energetic Particles Explorer)	101	Octagon with 4 paddles	110	45-55

Small Scientific Satellites that were designed by the Goddard Space Flight Center have had a thrust axis resonant frequency of between 50 cps to 110 cps. In the lateral axes the resonant frequencies have been between 9 cps to 55 cps (see chart).

It must be remembered that at resonance the "Q" level (or amplification) in certain structural parts could build up to several orders of magnitude above the input. Therefore some thought must be given to a means of minimizing this amplification. This is generally done by using dissimilar materials, friction devices, or special rubber compounds. I will list some of the points that may help to keep the "Q" level from going over 7.

These points are:

1. Use honeycomb wherever possible (Figure 11).
2. Use Fiberglass internally where possible (one drawback; Fiberglass is difficult to thermally coat).

3. Design load-carrying members and covers to provide some relative movement (friction) under high loads.
4. Use special vibration isolator rubber compounds.
5. Avoid (1) one piece structures (the more pieces to a structure the better the chance for obtaining relative movement and therefore dissipating energy by means of friction).

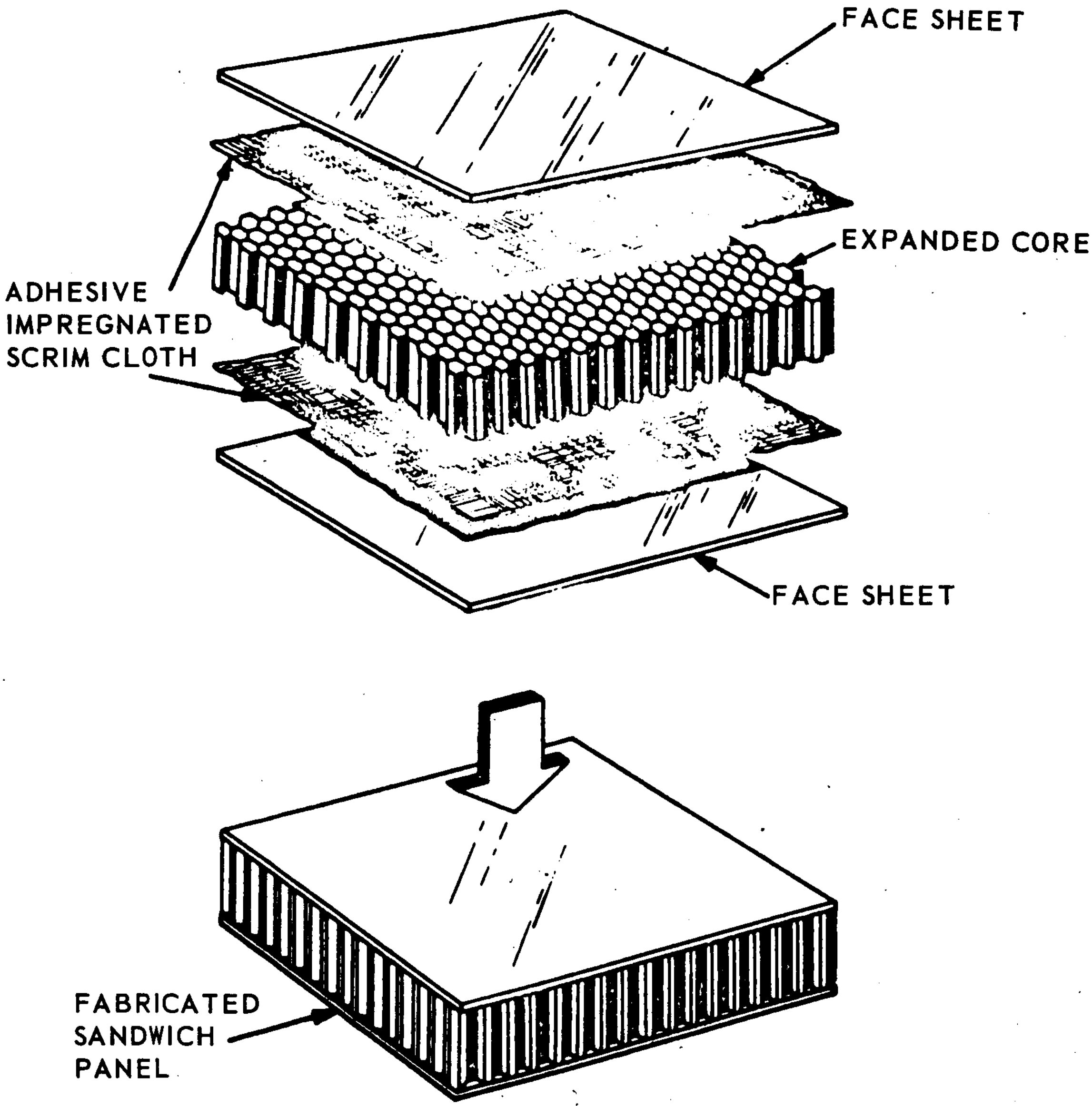


Figure 11–Honeycomb Material

6. However, do not build a loose structure. (This will creating banging and create thermal problems).

7. Encapsulate all subsystems.

Calculation of Structural Loads—

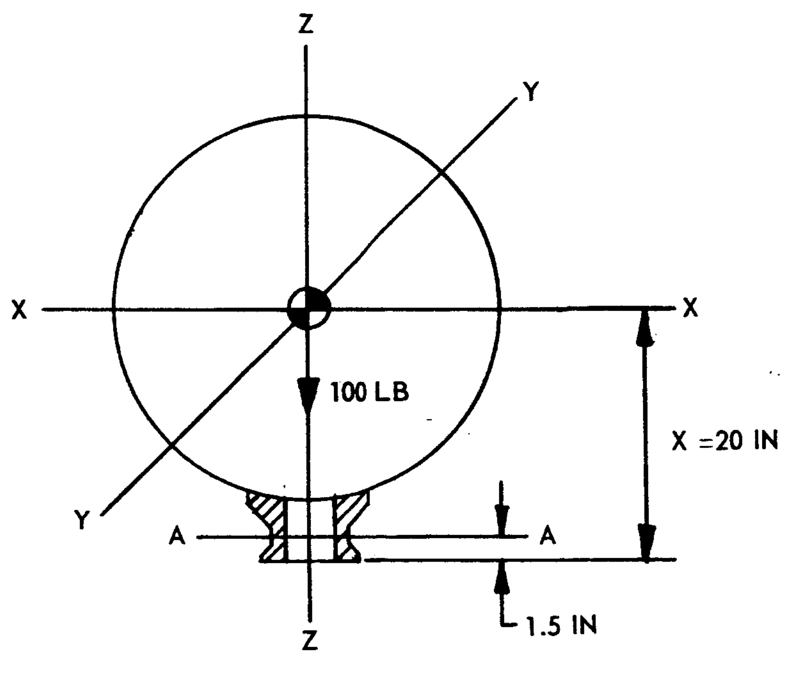
a. Vibration

One can proceed to calculate the structural loads. Looking at the Delta specification you will notice that the highest "g" level in the thrust axis for frequency between 50 to 500 is 7.5 "g's." You multiply this times a "Q" of 7 (A "Q" of 7 is based upon the assumption that energy is dissipated as a result of utilizing some of the suggestions listed above.), and this is the level you use to calculate thrust loads. The same approach is used in designing the structure for lateral loads. As a simple example, assume a 100 lb payload that will be launched by the Delta vehicle and therefore designed to pass the sinusoidal vibration specification.

Thrust (Z-Z Axis)

1. (a) $F = Q \cdot "g" \cdot W$
$F = 7 \cdot 7.5 \cdot 100$
$F = 5250$ lb

Where:
$Q = 7$ (estimated value)
$g = 7.5$ (from Spec. Thrust 50-500 cps)
$W = 100$ lb (Total weight of Satellite)



(b) $A = \frac{F}{S}$

$A = \frac{5250}{35,000}$

$A = .150$ sq in. (Area through Section A-A required to pass thrust vibration levels)

F = (Tensile or compressive load due to sinusoidal vibration)
A = (Area of lower cylinder at Section A-A)
S = (Yield stress of material, assume aluminum 35,000 psi)

Lateral or bending mode (X-X or Y-Y Axis)

1. $M = Q \cdot \text{"g"} \cdot W \cdot X$
   $M = 7 \cdot 2.3 \cdot 100 \cdot 20$
   $M = 32{,}200$ in.-lb

2. $S = \frac{M}{Z}$

   $35{,}000 = \frac{32{,}200}{Z}$

M = Moment (in.-lb)
g = 2.3 (from Specification lateral 8-500 cps)
Q = 7 (estimated value)
W = 100 lb
X = 20 in. (distance of C.G. to base of satellite)
Z = Section Modulus (in.^{3})
S = Stress (psi)

$Z = .89$ in.^{3} (Section Modulus required at base)

3. Stress through Section A-A (18.5 in. below C.G.)

$S = \frac{M}{Z}$

$S = 29{,}800$ psi (stress at Sect. A-A)

Where: $M = Q \cdot \text{"g"} \cdot W \cdot X_{A-A}$

$M = 7 \cdot 2.3 \cdot 100 \cdot 18.5$

The example illustrates the method for determining the area and section modulus in the lower cylindrical section; also the stresses are calculated through Section A-A. The attached chart is a simple case, but a spacecraft may have booms, paddles, etc.

b. Appendages and Yo-Yo

It is then necessary to treat the appendages as independent pieces of the structure when calculating their stresses and natural frequencies. Also, it must be remembered that appendage erection loads have to be calculated and compared to the vibration loads and the hardware designed accordingly.

Example:

Assume: (1) (4) Four appendages equally spaced

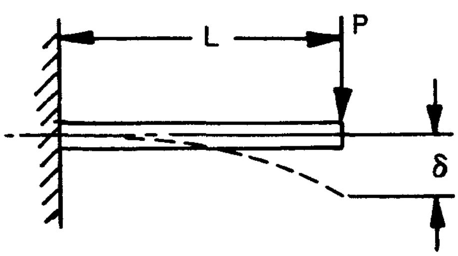
(2) Parallel to spin axis before release

(3) Perpendicular to spin axis after release

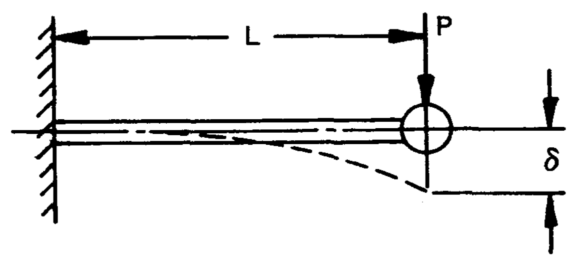
When the appendages approach the fully erected position (assumed perpendicular to spin axis), they possess Kinetic Energy equal to the difference between Kinetic Energy before (initial) and after (final) erection.

Kinetic Energy in all four appendages = $1/2\ (I_i\ \omega_i^2 - I_f\ \omega_f^2)$ in ft-lb. Since there are (4) four appendages each appendage will possess 1/4 the total energy calculated.

Load must now be equated to an equivalent static load so that its effect can be compared to the stresses created by the vibration loads, and the structural member to which the appendage is fastened should be designed for the condition that creates the greatest stresses. Assuming the appendage is a simple cantilever, the strain energy (σ) equation can be used to determine an equivalent static load (P).



Simple Cantilever



Cantilever Appendage

Equation 1. $\sigma = \dfrac{P^2\ L^3}{6EI}$

Equation 2. $\delta = \dfrac{P\ L^3}{3EI}$

Where: σ = Strain Energy
P = Load (concentrated)
L = Distance to Load
I = Moment of Inertia
δ = Deflection
E = Modulus of Elastivity

The only unknowns in the above equations are P and δ.

To calculate P, let σ = the potential energy of one of the appendages.

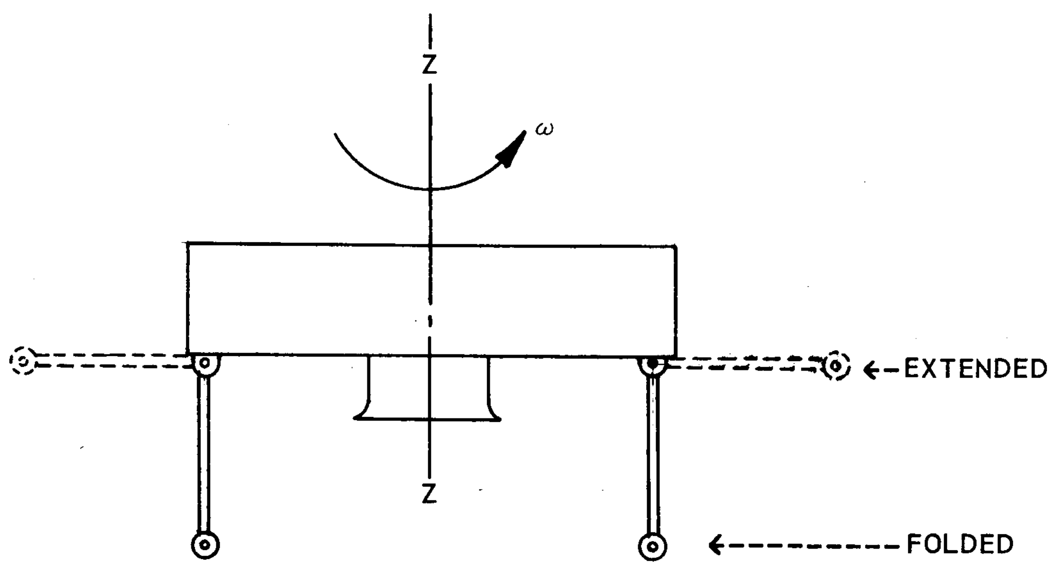
The energy that an appendage possesses is equal to the difference in energy between the initial condition and final condition or the energy the system had before paddle erection and after paddle erection.

Example:

Assume the following:

$$I_{\text{paddles folded}} = 5 \text{ slug-ft}^2 \begin{pmatrix}\text{Initial}\\ \text{MOI}\end{pmatrix} \quad \omega_{\text{paddles folded}} = \text{16 radians/sec (initial spin rate)}$$

$$I_{\text{erected}} = 16.8 \text{ slug-ft}^2 \begin{pmatrix}\text{Final}\\ \text{MOI}\end{pmatrix}$$



Satellite With Four Equally-Spaced Appendages

Using the equation for conservation of angular momentum:

1. $I_i \omega_i = I_F \omega_F$ or

   $5\ (16) = 16.8\ \omega_F$

   $\omega_F = 4.75$ rad/sec (final spin rate)

Where: $I_{p.f.} = I_i$

$I_{p.e.} = I_F$

$\omega_{p.f.} = \omega_i$

$\omega_{p.e.} = \omega_F$

2. Solving for the difference in Kinetic Energy:

$$E_K = 1/2\ (I_i\ \omega_i^2\ -\ I_F\ \omega_F^2) \qquad \text{or}$$

$$E_K = 1/2\ (5 \cdot 16^2\ -\ 16.8 \cdot 4.75^2)$$

$$E_K = 1/2\ (1280\ -\ 380)$$

E_K = 450 ft/lb of Kinetic Energy in (4) four appendages or

E_K = 112.5 ft/lb in one (1) appendage.

Changing the units to 1350 in.-lb you would then proceed to substitute this number for σ in Equation 1 solving for P. Once P is obtained it can be used in Equation 2 to solve for (δ) deflection. The assumption up to this point is that the length (L) has been established by design requirements and that the section modulus was also computed.

To evaluate the appendage for adequacy of design, obtain (1) one appendage and attach to the ETU structure. The experimental (K) constant can be obtained by the recording deflection versus load. Let us designate deflection as X_d and load as F_L so that we can differentiate these experimental values from calculated values and symbols. Now with the experimental K we can establish whether the appendage can withstand the 1350 in.-lb of Kinetic Energy computed above in the preceding examples:

Example:

Let us assume that we placed a 60 lb load at the C.G. of the appendage and the deflection was .8 in. therefore:

1. $K = \dfrac{60}{.8}$ or

K = 75 lb/in.

The above mentioned 1350 in.-lb of Kinetic Energy will be transferred to Potential Energy (deflecting the boom X distance) by the equation of P.E. = $1/2\ KX^2$

Therefore:

2. $1350 = 1/2 \cdot 75 \cdot X^2$

$X^2 = 36$

or $X = 6$ in. deflection
The force to deflect X distance is:

3. $$K = \frac{F}{X}$$

$$75 = \frac{F}{6}$$

$$F = 75 \cdot 6 \quad \text{or}$$

$$F = 450 \text{ lb.}$$

With this information, load the C.G. of the appendage with a 450 lb static load. This load will realistically test the appendage to duplicate the 1350 in.-lb Kinetic Energy due to appendage erection. This would also be an excellent opportunity to place strain gauges on specific critical areas of the appendage for the purpose of comparing calculations vs test data as well as for locating what may look like possible high stress areas.

The appendage is not a simple cantilever; therefore this equation is an approximation, but if used it will enable the structural engineer to design an appendage to a realistic load which is the appendage erection load. Energy dissipation in the above example is assumed to be equal to zero.

c. Yo-Yo

The Scout and Delta vehicles depend on spin of the last stage for stability of the satellite and last stage assembly. Depending on the moment of inertia and configuration of the satellite the vehicle contractor will spin-up the assembly from 80 rpm to 180 rpm with a tolerance of ±10%. Most of the mission requirements of the Goddard Space Flight Center small scientific satellites dictated de-spinning the satellite to some lower spin rate. To accomplish this requirement, the satellites were de-spun by incorporating a Yo-Yo mechanism.

The rigid Yo-Yo de-spin mechanism is essentially two (2) small identical weights attached to two (2) separate but equal length wires. (Figure 12). These weighted wires are symmetrically wrapped around the satellite (opposite direction to spin) and the weights held in place by some electromechanical actuated device. At some preselected time (normally after last stage burnout or after separation) both release mechanisms are simultaneously activated by small electric pyrotechnics. As a result, both weights unravel in the same direction as the satellite's rotation. Release of both wires is simultaneously at a time when the wires are perpendicular to the satellite's spin axis. The de-spin

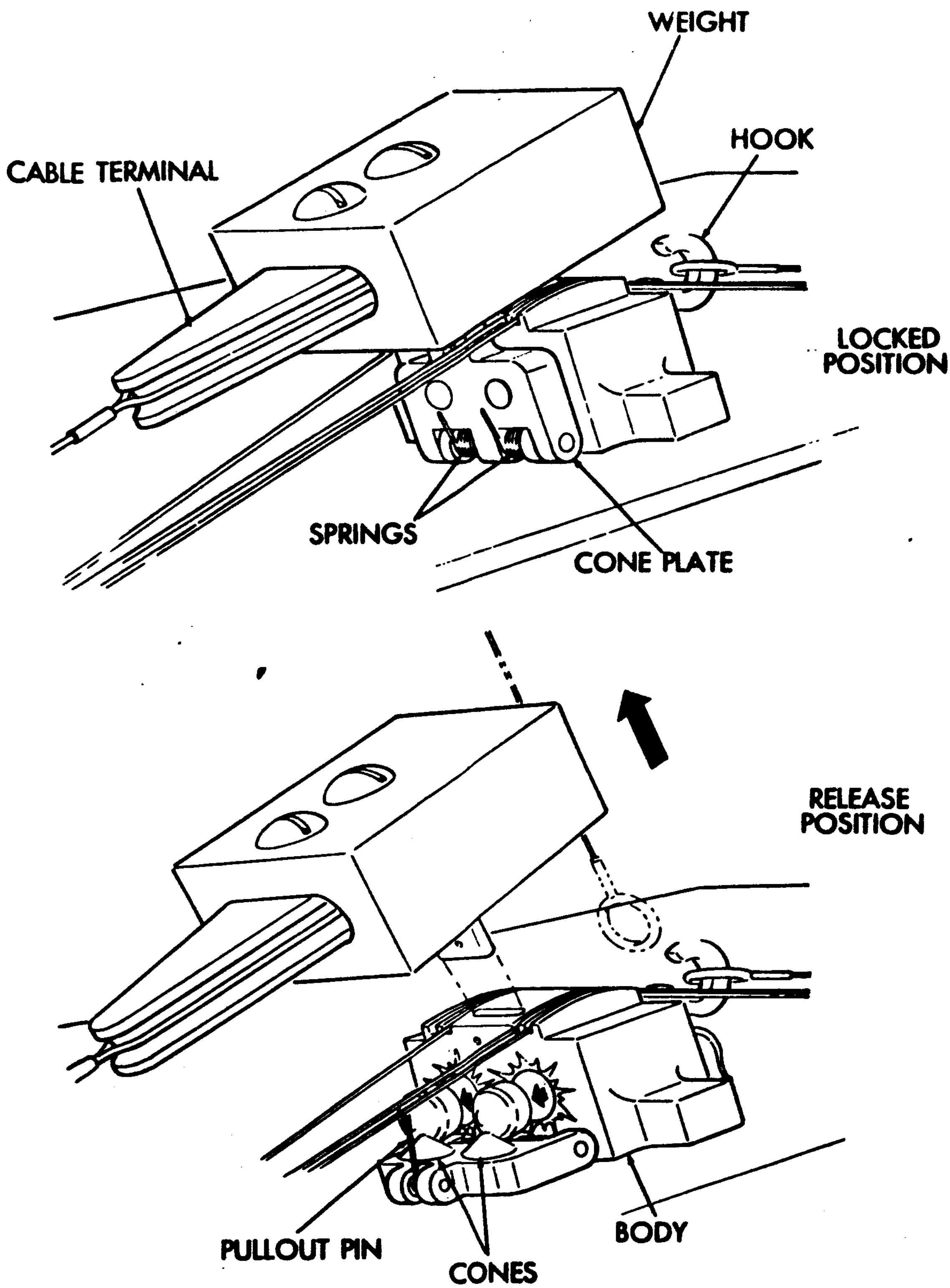


Figure 12–De-Spin Cable Release Device

function is accomplished by the transfer of some or all of the satellite's angular momentum into Kinetic Energy of the Yo-Yo weights and wires.

The Yo-Yo system can be designed to de-spin a satellite to a zero rpm condition and if necessary to spin up the satellite in the opposite direction. Spin-up in the opposite direction would require a somewhat complex wire release device.

The final spin obtained would be dependent on error (less than 1-1/2%) in calculations and the tolerance on vehicle spin-up. The vehicle spin-up tolerance is ±10%. This same tolerance applies when calculating the final de-spin rpm. If you plan to de-spin a satellite to 10 rpm the tolerance is ± 1 rpm. Notice, that as you design for an rpm close to zero the tolerance approaches zero.

If the mission requirements are such that a ±10% tolerance cannot be tolerated, it is recommended to design a de-spin system that could at times provide a ±1% tolerance of the final spin rate. This system is the stretch Yo-Yo. It is similar to the system discussed above, in operation. The only exception, the wire is replaced either entirely or partially with a spring. The spring compensates by either elongating or retracting depending on whether the spin rate is higher or lower than expected. This device senses the spin rate and corrects accordingly.

For a complete dynamic analysis and theory of the Yo-Yo de-spin system consult References 3, 4, and 5.

The total weight of the de-spin Yo-Yo system is mostly dependent on both the wire length and satellite radius. If the system is designed with two (2) complete wraps or turns of wire the total weight should be less than 1% of the spacecraft weight. The Yo-Yo system is capable of inducing spacecraft coning if the system is designed haphazardly. This coning is induced by an unbalance of forces or torques which are attributed to the following factors:

1. Yo-Yo mechanism not spaced diametrically opposite each other (180° apart).

2. One weight slightly heavier.

3. Weights and wires not released simultaneously.

4. One wire slightly heavier and/or longer.

The further the Yo-Yo is located above or below the C.G. the greater the induced coning. Remembering these factors, design the complete system as accurately as feasible. Design of the de-spin system is accomplished by completing the equations on "Yo-Yo De-Spin Calculation Sheet (Radial Release)." (See Figures 13 a and 13 b).

Some other important considerations that must be considered in de-spin design are:

1. If the satellite is to remain attached to the final rocket stage during de-spin be sure to take into account the inertias of all parts of the system.

2. Proper account for de-spin caused by inertia changes which can be due to appendage erection, gas depletion, and other such factors

d. Linear Acceleration

Acceleration due to rocket thrust has not been a problem in the past but as new propellants are utilized the thrust is being increased to such high levels that it is becoming the dictating factor in designing satellites below 125 lb. In addition the latest launch vehicles do not have the same dynamic response as the older vehicles. As a comparison, the specification for the new Delta DSV-3E and DSV-3F vehicle lists 3.0 "g" thrust axis level for a 25 cps to 250 cps frequency. Which means that the acceleration due to thrust would very likely be the criteria for determining structure design loads, for small satellites in the thrust axis. Since this presentation is centered around experience gained on small satellites, launched by the old Delta DSV-3C and DSV-3D vehicle, we will base all of the examples on this vehicle.

Example:

A comparison of loads as a result of increased vehicle thrust. Several years ago the prototype thrust level for a 125 lb satellite on the X248 solid stage motor was:

1. $\text{Accel.} = \frac{1.5 \times 3000}{125 + 77}$ "g's"

   Accel. = 22.3 "g's"

2. now on the X258 solid stage motor acceleration is:

   $\text{Accel.} = \frac{1.5 \times 6700}{125 + 77}$

   Accel. = 50 "g's"

Where: 1.5 = safety factor (prototype level)

125 lb = Satellite weight

3000 lb = Thrust of X248

77 lb = Expended weight of X248

6700 lb = Thrust of X258

Yo-Yo De-Spin Calculation Sheet (Radial Release)

DEFINITION OF SYMBOLS AND UNITS:

I - moment of inertia about spin axis (slug ft^2)
a - radius of satellite (ft)
ℓ - length of one yo-yo wire (ft)
m - total mass of both spin weights + 1/3 mass of both wires (slugs)
F_{max} - maximum tension in wire (lb)
ω_0 - initial spin rate (rad/sec)
ω_f - final spin rate (rad/sec)
r - final spin rate divided by initial spin rate
g - acceleration of gravity (ft/sec^2)

TO CALCULATE THE TOTAL MASS (WEIGHT) OF SPIN WEIGHTS AND WIRE (m):
Record

I = __________ slug-ft^2 $\quad$ ω_0 = __________ rad/sec
a = __________ ft $\quad$ ℓ = __________ ft. $\quad$ ω_f = __________ rad/sec

Calculate

$$r = \frac{\omega_f}{\omega_0} = \_\_\_\_\_\_ = \_\_\_\_\_\_.$$

With this value of r, read the value of $I/m(\ell + a)^2$ from the design curve; call this value B. Then calculate the following:

$$w = mg = \frac{Ig}{B(\ell + a)^2} = \frac{(\quad)\,32.2}{(\quad)(\quad)^2} = \_\_\_\_\_\_ = \_\_\_\_\_\_ \text{ lbs.}$$

TO CALCULATE MAXIMUM TENSION IN ONE WIRE: Calculate λ by

$$\lambda^2 = \frac{I}{m} + a^2 = \_\_\_\_\_\_ = \_\_\_\_\_\_.$$

or

$$\lambda = \_\_\_\_\_\_ \text{ ft.}$$

Also

$$\omega_0^2 = \_\_\_\_\_\_ /\text{sec}^2;$$

$$F_{max} = 1.3\frac{m}{2}\omega_0^2\lambda = 1.3\,\frac{(\qquad)}{2}(\qquad)^2(\qquad) = \_\_\_\_\_\_ \text{ lbs.}$$

CHECK OF UNDERLYING ASSUMPTION OF THE EQUATIONS: Calculate G as follows:

$$G = \frac{(1 - r)I}{ma^2} = \_\_\_\_\_\_ = \_\_\_\_\_\_.$$

If $G \geq 100$ and $\ell/a > 2\pi$, the answers are accurate to about 1-1/2 percent of the theoretically correct value.

Figure 13a–Yo-Yo De-Spin Calculation Sheet

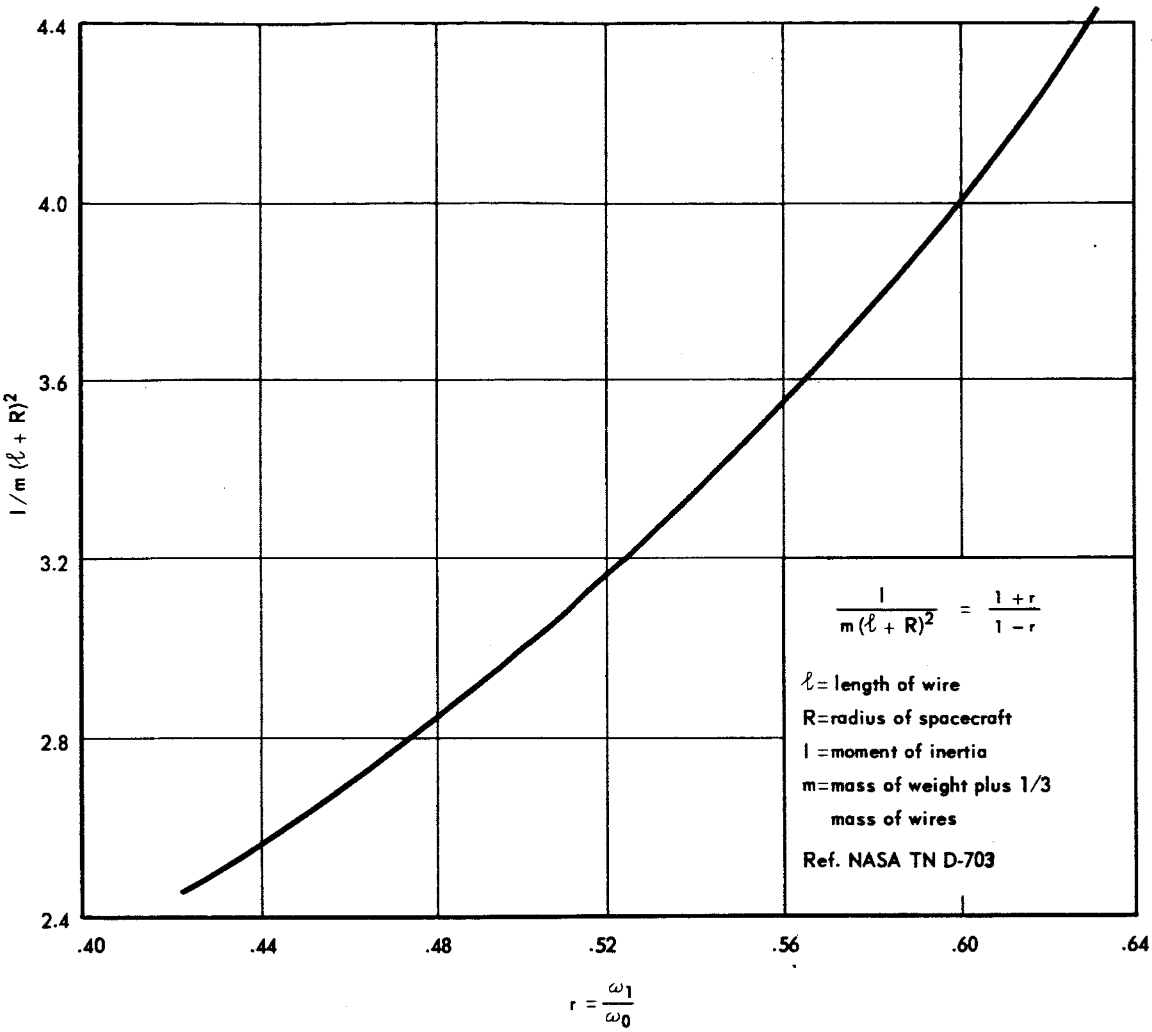


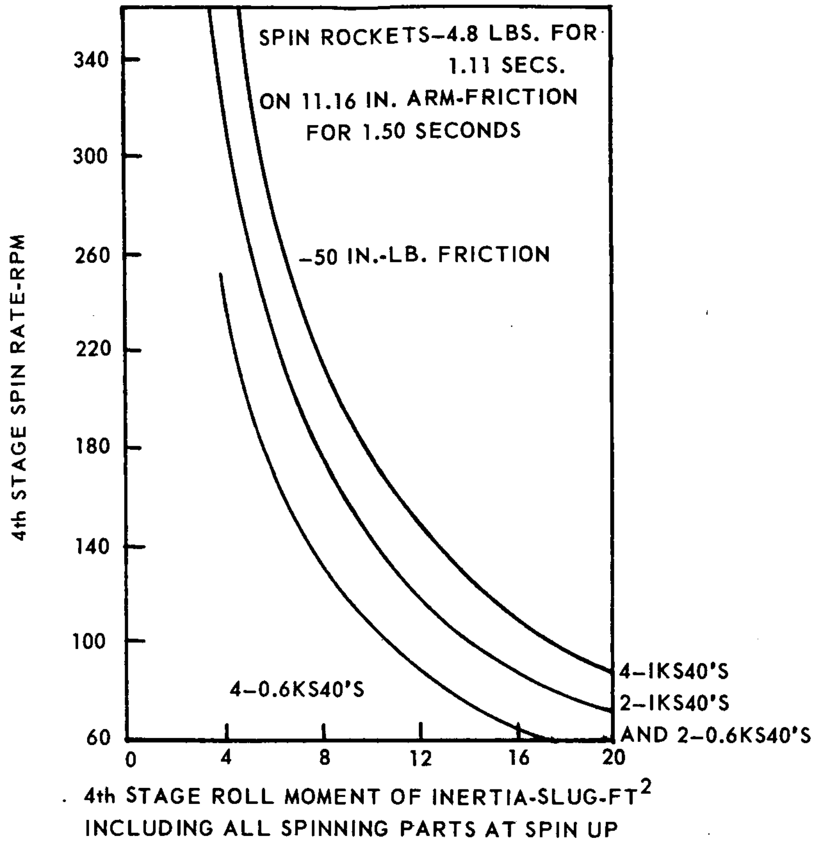
Figure 13b—Yo-Yo De-Spin Curve

Notice that this 50 "g" level is very close to the computed vibration design level. Computed by multiplying a "Q" of 7 times a "g" level of 7-1/2.

e. Angular Acceleration

Angular Acceleration due to spin rockets has not been a problem but it should not be ignored. Angular acceleration figures are normally in a Vehicle Restraints Manual, (see below) which provide curves showing angular acceleration for spin rates vs Moments of Inertia.

The Scout payload design parameters—Spin environment. The spin environment for various fourth stage total moments of inertia is shown in the graph



Spin Rate Vs. 4th Stage Moment of Inertia Vacuum

above for vehicles incorporating the cold (spring ejection) separation system. The spin rates shown in the graph above are those which will occur at fourth stage ignition.

As a result of the internal gas dynamics, the spin rate at fourth stage burnout will be approximately 11% greater than those at ignition.

<u>Static & Dynamic Balance & Alignment</u>–A parallel effort should also be undertaken at the outset of stress calculations. This is in the area of Static and Dynamic balance (See Reference 6).

Static mass unbalance (δ) is the shift of the principal axis parallel to the geometric axis. Dynamic mass unbalance is the tilt (α) of the principal axis to the geometric axis (Attachment 4).

1. Equation for Static Unbalance

$S = W\delta$

Where: S = static unbalance
W = payload weight
δ = axis shift

2. Equation for Dynamic Unbalance

$D = g\,(I_z - I_x)\tan\alpha$ (For small angles let $\tan\alpha = \alpha$)

Therefore:

$D = g\,\alpha\,(I_z - I_x)$

Where: D = dynamic unbalance
g = gravitational constant
α = principal axis tilt
I_x = Moment of Inertia (MOI) lateral
I_z = MOI spin axis

Figure 14 illustrates the axis shift and tilt with the appropriate equations to solve the static and dynamic unbalance.

To facilitate proper alignment of the Flight Satellite's spin axis with the vehicle's spin axis, a machined surface should be provided as far above the separation plane as possible. The run-out (Total Indicator Reading) of this surface should be obtained during the balancing operation and the high spot (maximum reading) either marked on the satellite or recorded in the log book. This information will be required by the vehicle personnel during the field operation prior to launch.

The vehicle usually has a requirement for what is considered as an acceptable maximum static and dynamic unbalance. The purpose of this requirement, by the vehicle, is to insure a good alignment of the thrust vector with the spin axis. On most of the Goddard Space Flight Center built small scientific satellites the dynamic unbalance requirements as dictated by the experimenters were more stringent than the vehicle requirements. Therefore, based on the most critical requirement, the structural engineer has the added responsibility to try to mathematically balance the spacecraft by shifting or interchanging electronic components and subsystems. It must be pointed out that the weights of all experiments and subsystems are mostly approximations. Even though every effort is made to balance a satellite on paper, this cannot and will not be accomplished except by pure luck or by continued balance computations that continue until the design freeze. This may sound like an unnecessary exercise for testing the engineers' mathematical capabilities, but it is not, for this is one area where it behooves the engineer to save

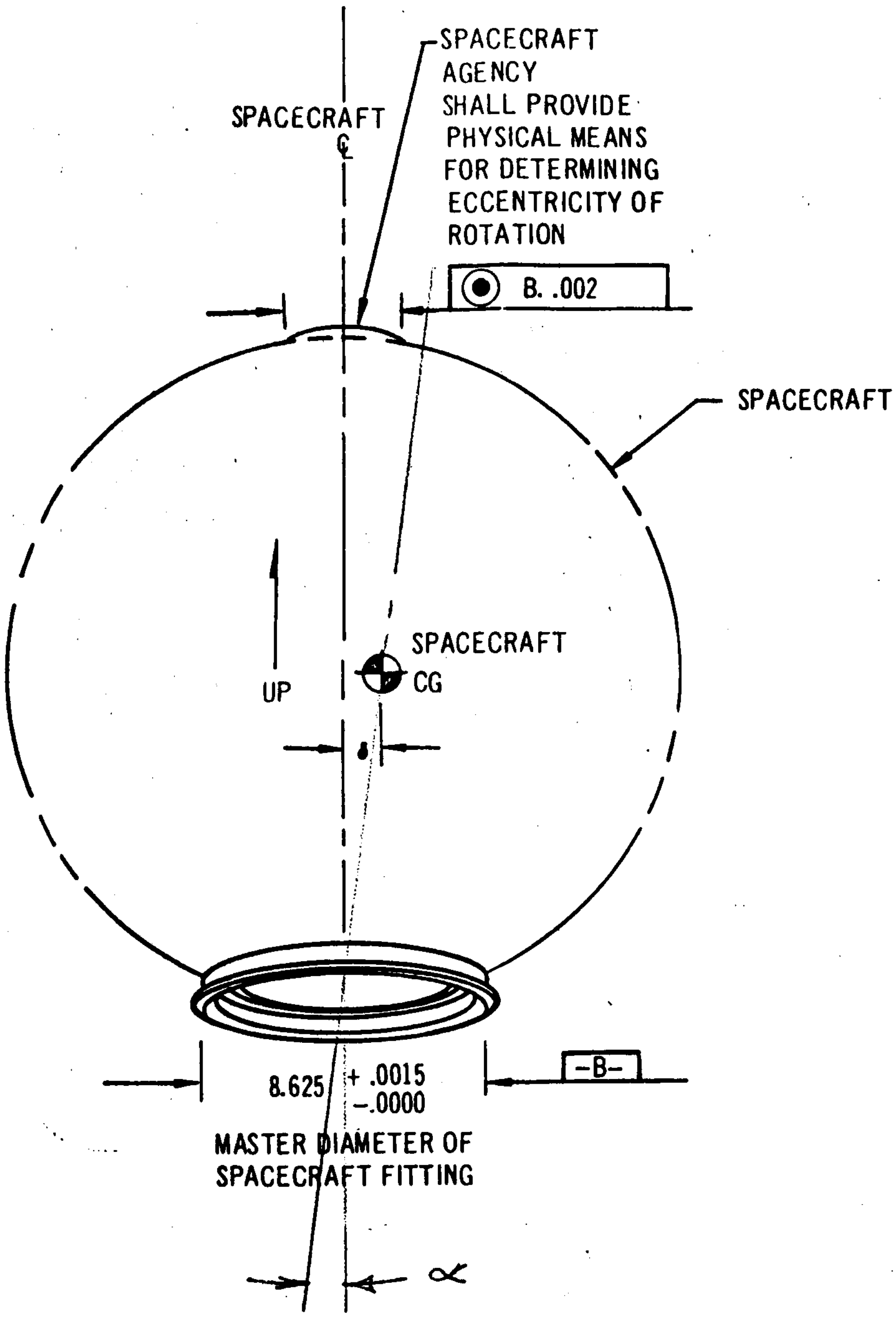


Figure 14—Balancing Considerations

that which is most critical, weight. Also, flying a large needless weight or weights (generally lead), whose only value is to balance the spacecraft could possibly be detrimental to some experiment, in that it could act as a radiation shield for an experiment whose function is to measure radiation.

Consideration should be given to provide for the placement of balance weights on the satellite. The weight should be located as far from the spin axis as possible and in balance planes located as far as practical above and below the center of gravity. Accessibility to the weights should be considered for it will be necessary to install, remove and relocate the weights a number of times during the balance operations. Therefore, locate the weights in such areas that it won't be necessary to remove parts of the satellite during the balance operation.

Handling—It takes approximately one (1) year to launch a satellite from the time that it is first assembled. Some consideration must be given to the handling problems that will be encountered from its birth till the time it is launched. Whether the satellite is moved from one room to another room or shipped to a launch facility, plans should be made to provide a handling cart and shipping container that will assure complete safety during its movement.

Considerable thought should also be given to providing an assembly area that is dust free, uncluttered, adequately illuminated and most of all not overcrowded.

Every effort must be made to inspect the satellite at every opportunity for damages, quality of workmanship, loose hardware and dust. Errors should be corrected immediately and loose hardware removed or tightened and the satellite covered with a protective cover during periods of idleness.

The protective cover should be designed and fabricated from material that does not possess a tendency to build-up an electrostatic charge. It is not unusual for a vinyl cover to build a potential of several thousand volts between the cover and the satellite. Upon discharge, this energy could very easily cause damage to sensitive circuitry within the satellite or ignite an electric squib, cutter or dimple motor. These electric actuators are used to perform a special task, e.g., Yo-Yo release. The same precautions should be undertaken with working clothes plus assembly areas.

Antenna Pattern Mock-up

One additional small but critical item that should be fabricated is an antenna pattern mock-up (Figure 15). The purpose of this mock-up is to determine the effect of the satellite's configuration on the antenna pattern. The exterior shape or configuration and the location of these antennae on the mock-up determine the

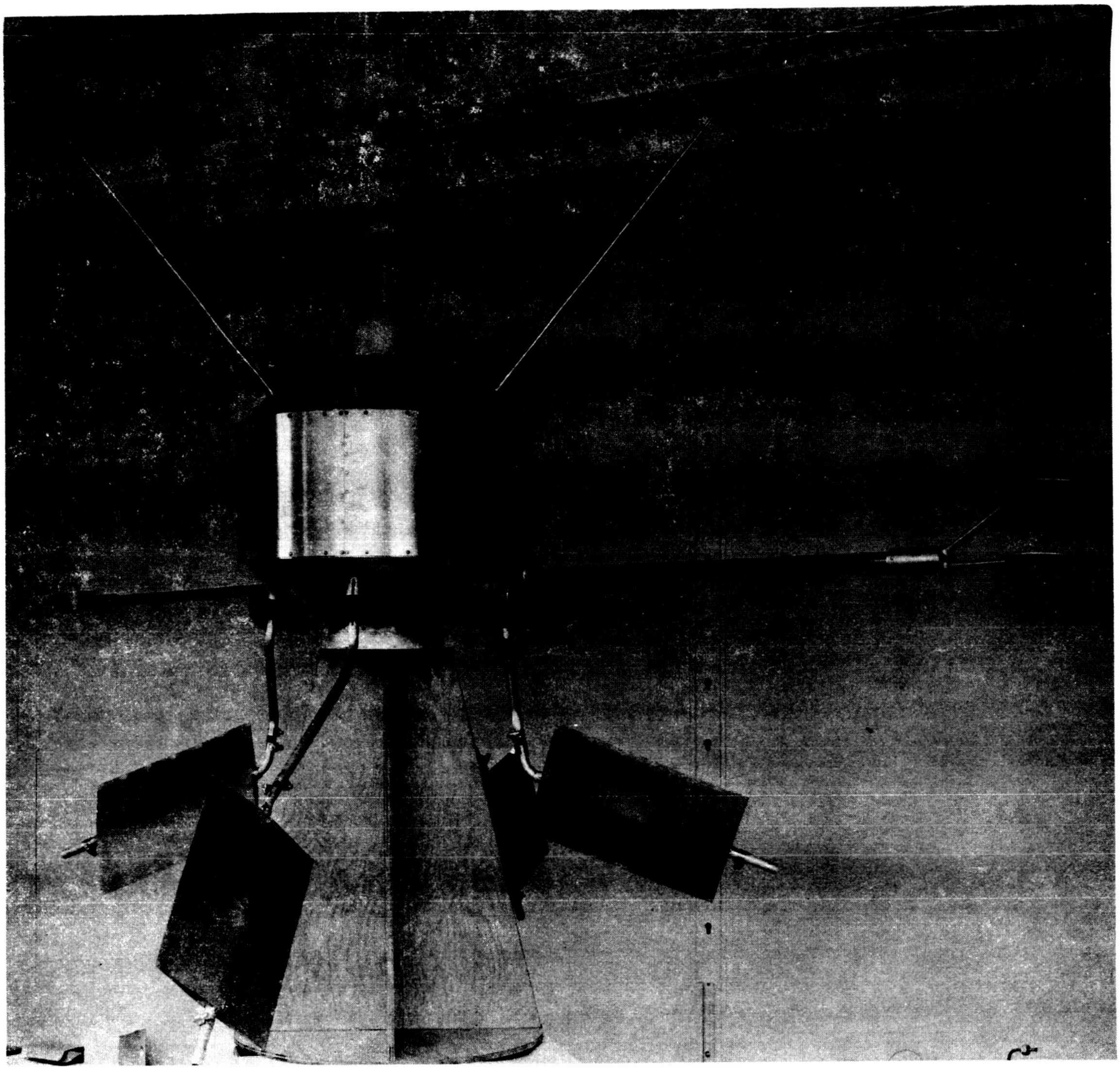


Figure 15—Antenna Mock-Up (Ariel I)

pattern. Therefore, the mock-up can be welded or riveted out of sheet aluminum. It does not have to be a precise piece of fabrication but it should resemble the final expected shape, i.e., solar paddles and all appendages be simulated properly.

Engineering Test Unit

Upon completing all the computations and design drawings the next step is to plan for fabricating and assembling an Engineering Test Unit (ETU) (Figure 16). The primary purpose of this unit is to thoroughly test the satellite's structural integrity before ordering hardware for the prototype and flight units.

Fabrication—To properly evaluate the ETU, it must be characteristic of the flight unit, i.e., it must be weighted and the weights distributed to at least the expected flight unit weight or possibly 5% more. This can easily be accomplished by bolting weighted wooden blocks in place of all the subsystems. Better yet, it would be desirable to utilize dummy weighted cards with accelerometers located within. The structural engineer could go one step further and select all or most of the critical structural members and instrument these with strain gauges. An ETU with all these transducers, will provide recorded data that will be useful in analyzing the dynamic stresses and responses during dynamic testing. It can also

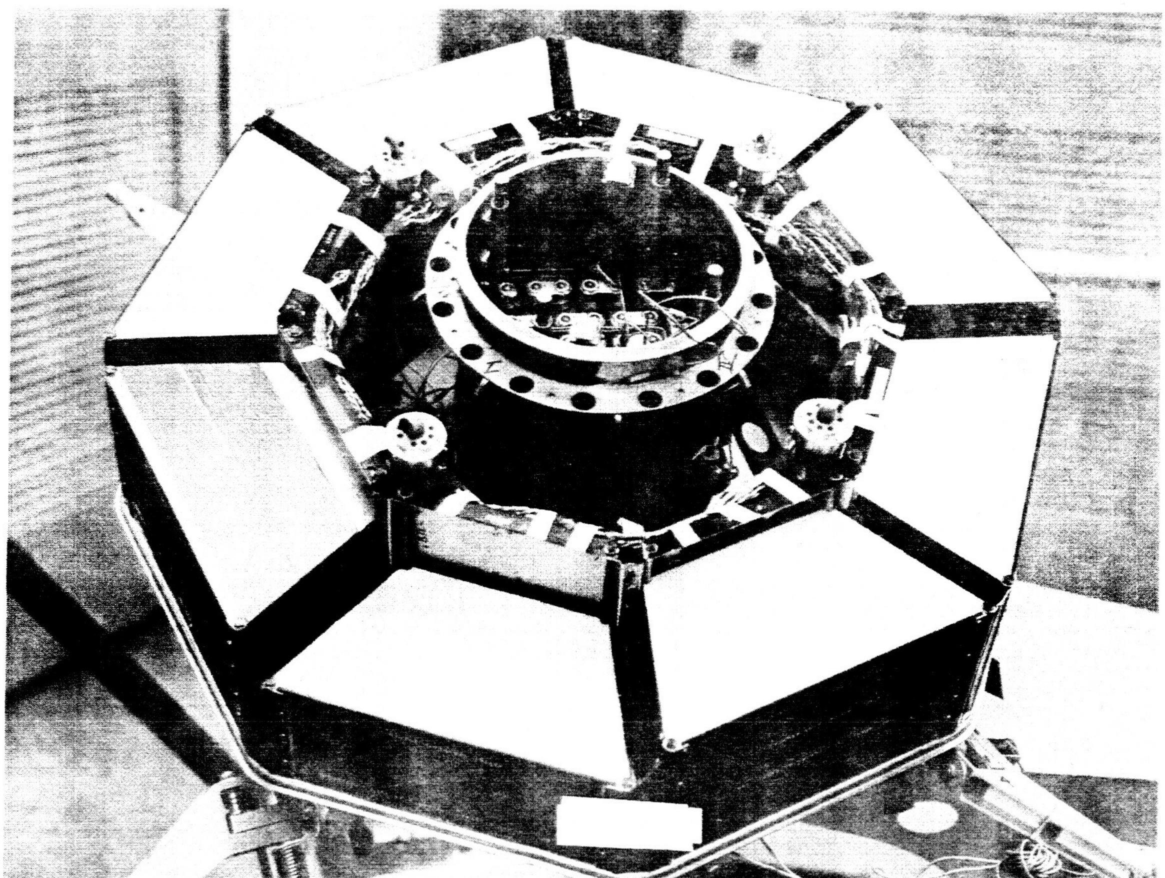


Figure 16—Engineering Test Unit (A-IMP)

be used as a thermal model by employing resistors and thermistors in each card to determine the thermal temperature gradients.

Testing–

a. Vibration

Assume the ETU is assembled and ready for testing. The first test is the prototype thrust vibration test per table on page 29. If any failures are detected during or upon completion of this test, the satellite should be removed from the vibration shaker and the failure inspected, analyzed and redesigned before proceeding further with the test. If no failures occur, then proceed to the lateral vibration test. If failures occur during this test use the same approach as above, i.e., correct the fault and resume testing. Random vibration generally follows the sine sweep combined into a spectral density envelope to meet the vehicle specification. This test normally should pose no difficulty with the possible exception of a few loose screws.

b. Acceleration

The next test is the acceleration test. If the acceleration levels were not the critical design factors for the structure this test should pose no problems but if they were this could be a source of trouble.

c. Spin

The spin test which follows the acceleration test has never given the structural designers any trouble at the Goddard Space Flight Center. This is due mainly to the fact that the spin rate for the Scout and the Delta vehicles has been less than 180 rpm, and the center of gravity of the individual subsystem packages is no more than (12) twelve inches from the spin axis. In checking the "g" level that these packages are undergoing it is found that the level is less than the 15 "g" lateral design vibration level that was discussed in the paragraph concerning design considerations. To check the "g" forces on the subsystems:

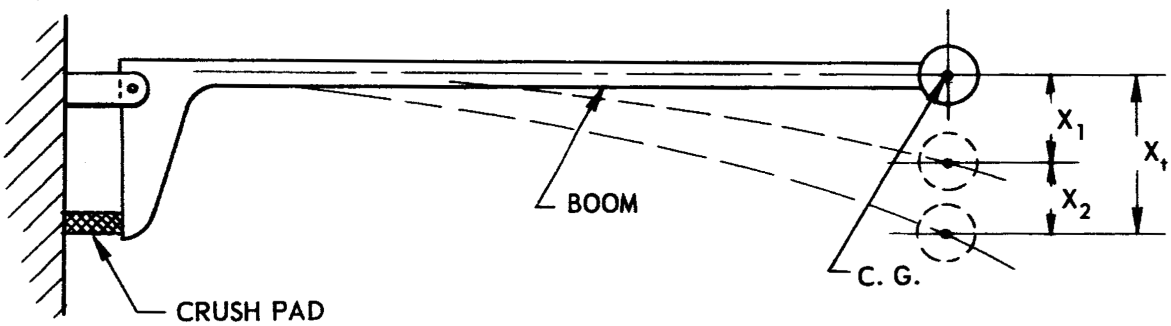
$$\text{Force in "g's"} = \frac{r\omega^2}{g} \qquad \text{Where: } r = 1 \text{ foot}, \quad \omega = 18.9 \text{ rad/sec}$$

$$\text{Therefore: } \frac{1}{32.2}(18.9)^2 = \underline{\underline{11.16}} \text{ "g's"}$$

In the future this force may become a problem, when satellite diameters become much larger. Also if the satellite is designed with appendages, it would be wise to calculate all the forces that are tending to unfold or unseat these appendages from their fixture supports. These calculations should be the basis of designing a reliable tie-down system to hold these appendages secure during the applicable dynamic tests. The last test is to perform appendage erection tests if applicable. The theory used at the Goddard Space Flight Center is to assume a de-spin failure and design to survive the maximum anticipated spin rate. This is done by designing the structure to withstand the erection loads of Kinetic Energy encountered during a normal de-spin sequence. Now, if the satellite does not de-spin, the additional Kinetic Energy that the appendages possess is dissipated by the use of a shock absorber or some other similar mechanical device. The common and simple energy dissipator is the crushing or yielding of materials.

Example:

A simple appendage with the mass concentrated at the extreme end, and the system possesses more Kinetic Energy than the structure could withstand without a crush pad. The crush pad in this particular case is used to dissipate this additional energy.



X_1 = The deflection of the boom as a result of Kinetic Energy

X_2 = The additional distance that the mass travels in the process of doing work or yielding the crush pad.

X_t = Total distance the boom must deflect to store energy and travel to dissipate the remaining energy.

Assume a de-spin failure. As a result of this failure the appendage will erect at a higher spin rate. Since the appendate boom was designed to pass the lower spin rate, the additional Kinetic Energy in the appendage would cause structural failure (if the crush pad was not included in this system).

Assume:

1. The Kinetic Energy that the appendage possesses is equal to (E_K) 1800 in.-lb.

2. The spring constant (K) is equal to 75 lb/in. (From previous example on design and test of boom).

3. Structural damage occurs at 1400 in.-lb of energy.

4. A safety margin by crushing the pad at an energy level less than the 1400 in.-lb figure or at a (E_K) 1000 in.-lb level.

Solving for the deflection (X_1) of the boom:

$$E_K = 1/2KX_1^2$$

$$1000 = 1/2 \cdot 75\ X_1^2$$

$$X_1^2 = 26.7$$

$$X_1 = 5.16 \text{ in.}$$

Up to this point we have stored (Potential Energy) 1000 in.-lb of energy. We must dissipate the remaining 800 in.-lb of energy by allowing the boom to travel a distance of X_2. In order to solve for X_2 we must calculate the peak force of the 1000 in.-lb stored energy, or

$$E_K = \frac{FX_1}{2}$$

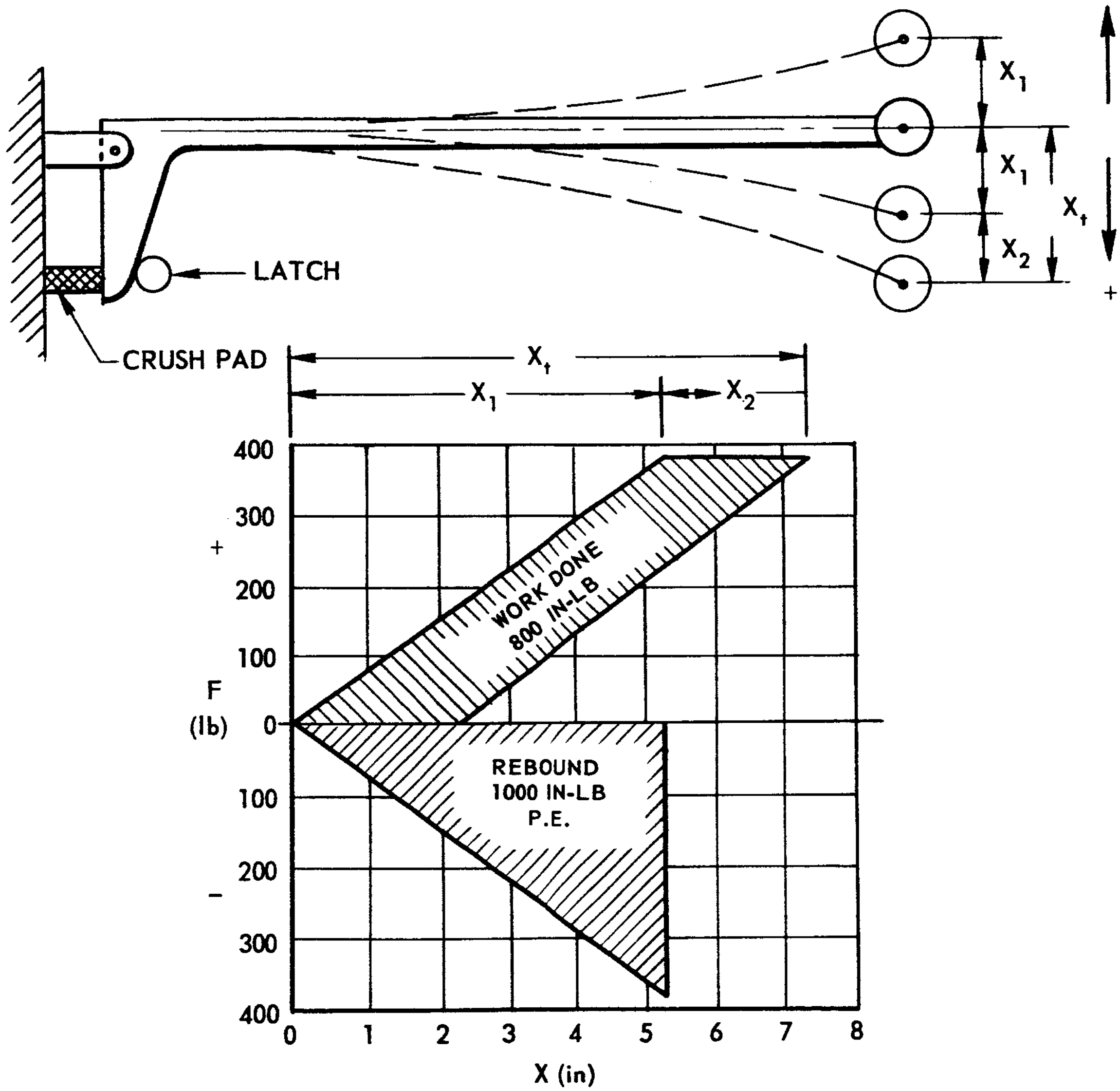
$$F = \frac{2E_K}{X_1}$$

$$F = \frac{2000}{5.16}$$

or F = 388 lb. This is the peak force. At this point the pad begins to yield. The distance X_2 it must yield is equal to the energy remaining divided (÷) by the peak force.

$$\text{or } X_2 = \frac{800 \text{ in.-lb}}{388 \text{ lb}} \qquad X_2 = 2.06 \text{ in.}$$

This can be shown graphically as follows:



The area under the upper enclosure is equal to the energy dissipated. The area under the lower enclosure is the rebound potential energy. Energy dissipation due to Friction and Boom Flexure is assumed to be zero.

Therefore, the residual or potential energy in the boom is still 1000 in.-lb. This energy will rebound the appendage in the opposite direction as shown by the graph.

In orbit these appendages will oscillate back and forth for some period of time dependent on the damping factor or friction or possibly an additional energy dissipator. One point to remember when testing on earth, the residual energy plus the earth's gravity, may cause failure on the down-swing. The reason for this condition is that on the down-swing the boom possesses most of the Kinetic Energy the boom had on the up-swing plus the energy generated by the earth's gravitational pull. High speed photography will normally establish if failure occurs on the up-swing or down-swing. Since this condition is peculiar to tests conducted on earth using a rigid fixture, the solution here is to either test by free-falling the simulated satellite or if the pictures definitely establish a failure on the down-swing, determine the increase of energy due to gravity and if it is appreciable, test the appendage under an equivalent static load as determined by the use of above-mentioned equations. Realistic tests can be simulated by using a rigid fixture providing gravity is accounted for by overspin. (See Reference 4 and Attachment 5.)

The above calculations are approximations and only hold true for small deflection angles. Also, the boom is considered as weightless with all the weight concentrated some distance from the hinge point. In addition, the diagram indicates a straight line for the K of the boom and a constant force to crush the pad. This is an ideal case but the calculations are valid and the error is relatively small.

d. De-Spin

Following the appendage erection tests the ETU can be utilized for Yo-Yo or De-Spin tests or the ETU could be used as a thermal model by the thermal engineers for monitoring spacecraft temperature gradients when exposed to solar simulation, or by utilizing the resistors in each card as previously mentioned.

As an alternative to using the ETU for de-spin and appendage erection test, one can design and use a flat circular disk with adjustable weights for varying the moments of inertia. To this disk attach a shell or cover that is similar to the flight satellite cover with a duplicate flight expected de-spin system.

Perform de-spin and appendage erection tests in a large vacuum chamber under a free-fall condition. Remember, this is the ideal method. If the free-fall cannot be utilized the error will be very small, usually less than 1% for the de-spin tests but the error could be much larger for appendage erection tests run under atmospheric conditions. The aerodynamic drag could also be appreciable if the de-spin weights are physically large and the cable long.

As a precaution, the de-spin wires and mechanisms plus the structure to which the mechanism is attached should be exposed to a pull test that is 1.5 times the force or tension calculated per equation listed on Figure 13a, under design of Yo-Yo.

Prototype Unit

At the completion of the appendage and de-spin tests, the prototype structural hardware should be ordered. This unit will be an excellent indicator as to the final outcome and configuration of the flight unit. For this will be the first time that all the experiments and subsystems will be mechanically integrated with the prototype structure. Upon receiving the structure hardware it is carefully inspected, cleaned, marked with serial numbers and assembled in a dust free room with the humidity controlled to less than 40%. Also a log book is assigned to the prototype unit to record all the components that make up the structure with their respective serial numbers. The book will record all events on a day by day basis so that an accurate record is kept as to who worked on this unit, what was done to it, and what problems were encountered and how they were resolved and by whom.

The first step is to begin assembly of the main structure. This usually is a joint effort by the structural personnel and the electronic integration team. The reason is that the wiring harness must be installed rather early or difficulty will be encountered in trying to force-fit this harness. The subsystems and electronic components are then installed one by one and carefully inspected and examined for: (1) hole alignment, (2) proper connector mating, (3) mechanical interference between components and structure, and (4) proper seating. When the spacecraft has been completely assembled, it is released to the electronic integration team for a thorough electronic check-out. During this stage, it is not unusual to assist the integration team in removing some subsystem several times a day. This is done for a period of about eight weeks until all the problems have been resolved and the prototype unit is operating flawlessly. At this point the unit is attached to a balance machine and a preliminary or rough balance performed. The reason for this exercise is, the vibration shaker and/or spacecraft could be damaged by a force (couple) created by an excessive C.G. shift from the geometric axis.

The prototype will be exposed to the higher vibration levels. Since this is the first time the electronics are exposed to dynamic testing, the problems are generally in this area. Very seldom will problems develop in the structure. Acceleration which follows, does not usually pose any difficulties in any part of the spacecraft.

Temperature and humidity is a one-week test that uncovers defects and weaknesses in electronic circuitry. The structure should pass this test without any difficulty.

Thermal vacuum is normally a three-week test that may extend to more than three weeks depending on the difficulty encountered in the electronic system. Upon completion of this test, all effort is directed to the flight unit.

Final balance is generally a one to three day operation depending on the complexity of the satellite. The personnel performing the balance operation should be informed as to the eventual location of the balance weights and once the size and weight of these balance weights are established they should be fastened to the satellite by the structural assemblers. At this point, the prototype will be set aside until such time when it will be shipped to the launch facility with the flight unit.

Flight Unit

The ideal time to begin assembly of the flight unit is after completion of environmental testing of the prototype. But this is very seldom the case. The troubles and problems that develop during prototype integration are numerous. These problems are determined by the complexity of the spacecraft. As the state-of-the-art is improved, the physical size of sensors and support electronics keeps reducing, so that more sensors and circuitry can be packaged in smaller volumes. This micro-miniaturization adds more complexity to existing problems and the result is a longer time needed for correcting these problems. This results in a slip in schedule and one way to make up this loss is to begin assembly of the flight unit before the completion of prototype tests.

A flight unit log book is also assigned to this unit and the same information is recorded in this book as in the prototype log book.

Most of the design problems should have been corrected as a result of prototype testing. The only problems that are normally disclosed by environmental testing the flight unit are generally in the area of workmanship, i.e., poor solder joints, faulty connectors, defective fasteners, etc.

One area that has not been discussed so far is the area of thermal coating. Thermal patterns have not been finalized until the completion of flight unit thermal vacuum test. Based on these results, the patterns could be corrected or changed as late as two weeks before launch. The reason for this is the short time left between the end of thermal vacuum test and field operations. The joint effort by the structural engineer and the thermal engineer to complete this phase before launch is accomplished with very little difficulty.

ENCAPSULATION AND CONFORMAL COATING

Sometime before the satellite's design freeze date, the task of designing and fabrication of encapsulation molds for the experiments and subsystems should be undertaken. The function of these molds is to restrain (prevent buckling) the experiments' and subsystems' frame or container during encapsulation.

Encapsulation and conformal coating of electronic components and other components are usually done during and after completion of electronic integration. Reason: During electronic integration the cards and experiments have to be removed many times for modification or repairs before all the problems have been solved. Once the prototype is functioning properly, the cards and experiments are removed, conformal coated, encapsulated and re-assembled into the prototype to see if the encapsulant changed the characteristics of the experiments and subsystems.

Conformal coating is the coating of a electronic circuit board with a protective coating of semi-rigid epoxy approximately 2 mils thick. This is accomplished by either spraying, brushing or dipping the complete card.

Encapsulation is the filling of all voids within a frame, card, or experiment container by the use of a low density material. This material is a closed cell polyurethane foam or something similar. The densities can be varied from two (2) pounds per cubic foot to over twenty (20) pounds per cubic foot. The polyurethane foams used by the Goddard Space Flight Center are the Ecco foam FP and Ecco foam FPH. (Reference No. 4). Ecco foam FP is recommended for use below 66°C and FPH above 66°C.

In areas where corona may be a problem, careful attention should be given in selecting an insulation compound with excellent insulating properties, good resiliency and capable of long exposure to high vacuums. A note of caution: The reason for recommending semi-rigid or resilient epoxies and insulation compounds is that the coefficient of expansion of these coatings is different than the coefficient of expansion of the electronic components coated.

During thermal cycling the rigid type of coating would stress and fracture some of the electronic components whether they be glass diodes or small precision resistors.

INTEGRATION AND TEST

The mechanical and electronic integration of the subsystems into the structure will begin to show discrepancies in the spacecraft. Discrepancies like: interference of one experiment or its electronic circuit with another experiment. R-F leakage into some other circuitry or an additional requirement or change as a result of above-mentioned troubles. These may require some structural changes. Most of these changes are minor but there are times when some subsystem has to be re-designed which effects a design change in the structure. This is why it is best to hold up production on the flight unit until the prototype has undergone complete environmental testing. This again is an ideal schedule and most schedules are tight and slightly unrealistic; therefore, it may be necessary to order long lead-time flight structural components at the same time as prototype hardware. Also, if the schedule slips drastically, it may be necessary to begin assembly and mechanical and electronic integration of the flight unit before the prototype has completed environmental testing. This is a gamble but if the structural engineer plans his structure to be somewhat flexible he will be able to accommodate these changes without any difficulty.

Dynamic tests, on the prototype, very seldom cause any structural failures. The failures that do occur are fatigue failures which are a result of repetitive testing. This is natural and should be expected but to prevent this type of failure it is recommended that the prototype not be tested repeatedly if at all possible. It is recommended that the ETU be further utilized to qualify alternate flight subsystems rather than using the prototype and thereby further fatiguing of the prototype structural members.

The problems associated with the flight unit are strictly oriented to subsystem defects. From here on out the structural engineer concentrates mainly on excellence of workmanship and making sure that the flight unit is assembled with only one thought in mind, that upon completion of the environmental testing the unit will be shipped to the launch facility and finally launched. To insure flight readiness, the structural engineer should have prepared a field check-off list for the flight unit. The purpose of this list is to keep a running log of everything that has been removed and re-installed from the time the satellite is made flight ready to the time it is launched. The flight unit is made flight ready at the onset of environmental testing. If the unit passes these tests without any malfunctions then it can be shipped to the field without having to undergo final assembly. Flight readiness means every screw, bolt, pin or fastener locked for flight, either locked by the use of a chemical compound or special screws with locking features. A sample field check-off list is attached as a ready reference. (See Attachment 6, titled: Field Operations Check-Off List (IMP-B).

SHIPPING

The design and fabrication of the shipping containers should have been initiated before the onset of the ETU dynamic tests. It is recommended that these shipping containers be fabricated from either plywood or sheet aluminum. Aluminum would be preferred in that it can be designed as an excellent water-tight container. The container should be strong, not too heavy and easy to handle. A spacecraft interface/mounting fixture should be fastened to the interior of this container by means of vibration isolators at a frequency, different from the thrust resonant frequency of the spacecraft. A good rule to follow is to design to a frequency between 10 and 25 cycles per second. At this frequency with a 3 "g" input the amplitude is not large enough to cause interference within the container and at the same time will dampen out most of the dynamic forces that are created by handling and transporting. To prevent lateral movement, the same technique could be incorporated, i.e., the other end should be braced or supported with specially designed dampening materials. In packing a satellite for shipment remember to include desiccant inside the shipping container. Desiccant for those who may not have had the occasion to use this material is used for absorbing moisture. This is where a good tight container is beneficial. On an extremely humid day there is very little chance that the interior will be drastically affected. But keep this in mind, satellites are normally packed indoors where the temperature is 70°F or more and the humidity could be as high as 50% but once you expose this container to a cold outdoor temperature you could have condensation within the container. This is why you should use desiccant at all times and in sufficient quantities to prevent any accumulation of moisture. The amount of desiccant used is determined by container volume and the manufacturer's instructions with some additional safety factors. Something that I feel is worthwhile to mention is exterior container size. The question you should ask yourself is, how am I going to ship this satellite to the launch facility? By train, boat, truck or airplane, or maybe by a combination of these modes of transportation. But keep one thing in mind, if you design the container too large you may not be able to get this satellite aboard an airplane, (Figure 17), so it is best to obtain dimensions of cargo space and the physical size of the opening to the cargo space. If you find that a passenger airline cannot get this package through their doorway you may have to hire a special cargo plane for the sole purpose of shipping this spacecraft. One approach that is feasible most of the time, is to disassemble the spacecraft and ship it in two separate containers. If it is a solar paddle and boom-type satellite, these could also be shipped in separate containers. From the time the satellite arrives at the airport to the time it is launched the personnel working with the structural engineer should have full handling control of the satellite at all times. This includes standing nearby whenever the vehicle people may be working near the spacecraft. The reason for this close scrutiny is to witness any damage that may have been done to the spacecraft so that it can be immediately analyzed and corrected if it is decided that it could cause a failure.

FIELD OPERATIONS

The smaller scientific satellites are normally shipped to the field from three to five weeks before launch. The last two weeks of this time are devoted to

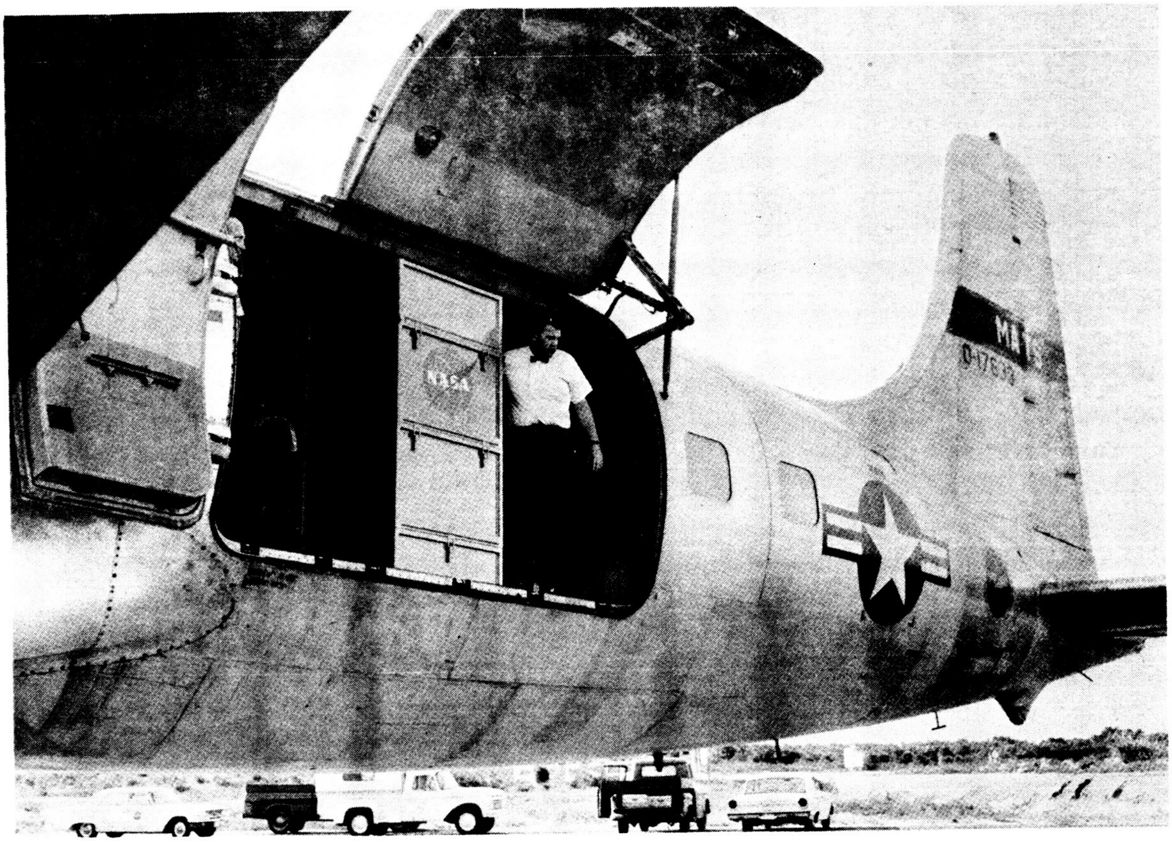


Figure 17–IMP Satellite in Cargo Plane

vehicle/satellite operations. Prior to this, the satellite undergoes operational checks and calibration. If the satellite is very complex then approximately three weeks would be required for operational checks and calibration, but if it is simple one week may be sufficient. Since both the prototype and flight unit are shipped to the launch facility the same week, it is necessary to have enough cognizant personnel to simultaneously perform all the scheduled tasks on both units.

The first few days are generally spent in operating both the prototype and flight unit. All voltages and currents are carefully checked and compared to previous records. Sensors are exposed to calibration sources and data analyzed with prior data. When it has been established that the prototype and flight unit are both operating properly the units are prepared for further field checks.

Both units are transported to an antenna range to test RF transmission. This is generally about a 1-day test.

The following day the flight unit may undergo further check-out and calibration and the prototype would be taken to the spin facility for determining last stage and satellite compatibility; also the vehicle personnel may attach supports to the last stage for appendages.

The next day the prototype and last stage would be taken to the gantry and the complete assembly attached to the lower stages. This is done to check for spacecraft/vehicle interference and blockhouse interface (umbilical connections).

One final test to determine if any RF interference exists on the gantry is to turn the spacecraft on, transmit a signal and receive and analyze this signal about a mile distant. This brings us to about 10 days before launch.

The flight unit is shipped to the alignment and spin facility area where the satellite is attached to the last stage. The complete satellite/last stage assembly is first aligned to insure proper alignment of both the satellite and last stage axis. The reason for this operation is that the complete assembly must be dynamically balanced. Therefore, misalignment would require more weight to balance the assembly. The balance operation follows: A note of caution: If the satellite has solar paddles, then balance the assembly using dummy-weighted paddles. The weight and C.G. should be identical with the flight paddles. The reason for balancing the assembly with dummy paddles is that the vehicle personnel must attach and remove lead weights in the general area of the solar paddles. These paddles are therefore susceptible to damage and it is not worth the risk of canceling the flight when it is just as easy and accurate to balance the assembly with weighted paddles. The active solar paddles could be attached a day or two before launch.

Upon completion of the balance operation the assembly is installed on the lower stages followed by additional spacecraft checks prior to fairing installation. Once the fairing is installed, the only functions that remain to be done are installation of the turn-on plug and removing of a cord to release the antennas, so that they may rest on the inner surface of the fairing.

Once all of the above items have been completed the satellite is ready for launch. As an illustration of what the field operations consist, Attachment No. 6 is included as ready reference, Titled: Field Operations Check-Off List.

POST FLIGHT ANALYSIS

Achieving a successful orbit does not mean the end of the structural engineer's problems. Assuming that all dynamic functions were achieved per some well-planned operational sequence, there still is the problem of checking periodically on the housekeeping data to determine if the satellite is experiencing some unexplained perturbations and temperature excursions. If the satellite experiences some subsystem or sensor failure, assistance is generally provided by the structural engineer to determine the probable cause. If the satellite ceases transmission, additional failure analysis is conducted to determine the probable causes.

Generally all the past launches were not perfect. The Ariel I encountered some difficulty due to premature paddle and boom deployment. The Explorer XV experienced Yo-Yo de-spin failure and Explorers XVIII and XXI fell short of the expected apogee.

Investigations following each of the four programs resulted in several design changes. On the Ariel I the hypothesis was that the motor case temperature exceeded the maximum temperature design limit of the tie-down system resulting in premature failure. To safeguard against this problem, improved tie-down cord was designed that could withstand much higher temperatures for a longer period of time.

The reason for de-spin failure on Explorer XV was never really established. The complete de-spin system was tested to try to duplicate this failure, but without success. The end result was to redundant wire the electrical system and improve on the de-spin weight release mechanism.

The low apogee of Explorer XVIII and Explorer XXI was attributed to sub-performance of the third stage of the Delta vehicle.

These are examples of some problems that needed investigation and for testing and redesign to improve future launches. It is the structural engineer's responsibility to analyze all flights, i.e., obtain as much data as possible, including launches of satellites designed and assembled by other government agencies and contractors carefully analyze any deviations of operational sequence and determine if this data can be used to improve future satellite designs.

SUMMARY

The engineering, design, assembly and mechanical integration through launch of small scientific satellites is a complex function that requires skill and/or experience.

Experience is said to be the best teacher.

We have presented information based on the experience we have gained from several of Goddard Space Flight Center's designed and built small scientific satellites. We recognize that the information presented is not complete but we trust that it will provide valuable assistance to satellite structural engineers.

We have outlined a step by step approach which has been used successfully on small scientific satellite structures from inception to launch and orbit.

We have included such topics as shape determination, design loads for structural members and/or appendages, design techniques and materials, dynamic stability criteria for spin stabilized satellites, mechanical tests and integration, the type of units to be fabricated and their functions, handling and shipping of the flight units, field operations and Post Flight analysis.

We have also included sample calculations to aid the engineer in designing and testing of appendages, Moment of Inertia Fixtures, Yo-Yo de-spin systems, dynamic loads and Section Modulus.

We have recommended procedures for handling and shipment of the satellite to the launch facility.

ACKNOWLEDGMENTS

The authors gratefully acknowledge the assistance of W. S. Logan for contributing the torsional pendulum rod equations; T. W. Flatley for contributing the strain energy equations; and X. W. Moyer for contributing the typical time schedule of mechanical functions illustration (page 7).

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FASTENER TORQUE VALUE REFERENCE

ATTACHMENT 1

AIMP MECHANICAL MEMORANDUM

No. 3

FASTENER TORQUE VALUE REFERENCE

E. W. Travis
D. K. McCarthy
D. L. Miller
April 15, 1965

AIMP MECHANICAL MEMORANDUM NO. 3

SUBJECT: Torque Values for AIMP Spacecraft

Torque values for AIMP spacecraft as taken from the following source, except for magnesium:

Torque Manual
Fourth Edition – 1963
P. A. Sturtevant Co.
Addison, Illinois

These torque values will be used on all AIMP hardware to properly equalize the loads throughout the spacecraft, unless exception is taken by the cognizant engineer.

D. K. McCarthy
D. Miller
E. W. Travis

Enclosure

In-lb.

Bolt Size	18-8 and 300 Series S St.	Brass	Phosphor Bronze	Aluminum 2024-T4	Magnesium ZK60-T5	Tolerance
2-56	2.0	1.5	1.8	0.9	0.5	±0.5
4-40	4.7	3.8	4.3	2.4	1.4	±0.5
5-40	7	5	6	3	2	±1
6-32	8	7	8	4	3	±1
8-32	18	14	16	9	5	±2
10-24	21	16	19	12	7	±2
10-32	30	24	27	17	11	±2
1/4-20	70	55	60	40	25	±5
1/4-28	90	70	80	50	30	±5
5/16-18	120	100	110	70	45	±10
5/16-24	130	105	120	75	45	±10
3/8-16	210	170	200	120	75	±20
3/8-24	240	190	220	130	85	±20

How to Use: Choose the smaller torque value for any combination of bolt and insert/fastener. For threaded inserts (helicoils, etc.), compare screw and insert materials.

Examples: #4-40 Al. screw in Phos. Bronze helicoil = 2.4 in-lb.
#4-40 screw (18-8 SST) in tapped Magnesium = 1.4 in-lb.
#4-40 screw (18-8 SST) in Phs. Bronze Helicoil in Magnesium = 4.3 in-lb.

MECHANICAL INTERFACE REQUIREMENTS

ATTACHMENT 2

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GODDARD SPACE FLIGHT CENTER

GREENBELT, MARYLAND

Interplanetary Monitoring Platform (IMP F and G)

Code 673.1 Mechanical Interface Requirements

Contacts for Mechanical Interface Information:

Elmer W. Travis - Project Engineer, Ext. 4481
Bill Moyer - Assistant Project Engineer, Ext. 5096
Dennis McCarthy - Research Engineer, Ext. 5096

Size

Body Mounted Electronics and Experiments

Dimensions are given as length, depth and height respectively using the sample body illustration as shown by Figure 1.

The basic electronics module and experiment package mounted in the main body (on octagon platform) of the spacecraft shall be a trapezoidal shaped card as shown by Figure 2. The height shall not be less than .9375 in. and not more than 9.000 in. Any height above the .9375 must be approved by NASA/GSFC. If an experiment is of such dimensions that it cannot fit within the dimensions shown on Figure 2, the experimenter should plan to extend the package through the back of the card as shown by Figure 3. For those requiring a smaller volume than that shown by Figure 2, a half card is available as shown by Figure 2A.

Connector Location

All main harness connectors shall be located horizontally as illustrated by Figures 1, 2, 2A, 3 and 5.

All test connectors must come out the front of each package and be provided with plastic dust covers. (Connector must be flush with front face of module card.)

Weight

NASA/GSFC shall have the responsibility for the control of the weight budget for all IMP electronics and experimental packages.

Windows and Experiment Look Angles

Body Mounted Experiments

Window: Can be located on top and bottom of the octagon.

All windows (openings) located on the facets of the octagon shall be located perpendicular to the spin axis as shown by Figure 1. Look Angles for the body mounted experiment should be submitted to GSFC/NASA for approval. The experiment window shall not extend more than .031 inches beyond the outer frame of the trapezoidal module as shown by Figure 4.

Materials

The use of magnetic materials will be avoided. In order to minimize or prevent a build-up of magnetic fluxes the materials used in the construction of the experiments should be either aluminum or fiberglass. Certain brasses and magnesiums and other non-ferrous metals exhibit some magnetic properties. Prior approval from GSFC/NASA should be obtained for use of brass or magnesium.

The magnetic restrictions of each subsystem or experiment will meet the following:

a. After a 25 gauss exposure have residual magnetism of 32 gamma at 18 inches.

b. After a 50 gauss deperm have residual magnetism of 2 gamma at 18 inches.

c. Have a stray magnetism of 4 gamma at 18 inches.

Hardware

Screws, Nuts, Fasteners, Washers, Etc.

In order to eliminate possible failure when the spacecraft is subjected to environmental testing, it is mandatory that all designs incorporate screws made by the Long-Lok Corporation of Los Angeles, California. The screws should be anodized aluminum with the Kel-F insert. All other hardware should be anodized aluminum.

Cannon Connectors

Only gold plated "D" series cannon connectors shall be used. Use of a 37-pin connector must be approved by GSFC/NASA.

Encapsulating (Potting)

All electronic components, circuit boards and solder joints will be potted with eccofoam having a density of 6 to 8 lb/ft^3. All potting and encapsulating will be accomplished at GSFC with the assistance and concurrence of all designers.

Heat Sinks

All high heat-liberating components will be attached to the trapezoidal frames either directly or indirectly through a BeO insulator. GSFC/NASA should be notified as to the location of all hotspots.

Approval of Mechanical Interface

NASA/GSFC shall have the responsibility for the control of all IMP mechanical interface areas mentioned in the preceding paragraphs. All mechanical interface information and deviations to the above-listed requirements shall be submitted to NASA/GSFC for approval through the IMP F & G project office.

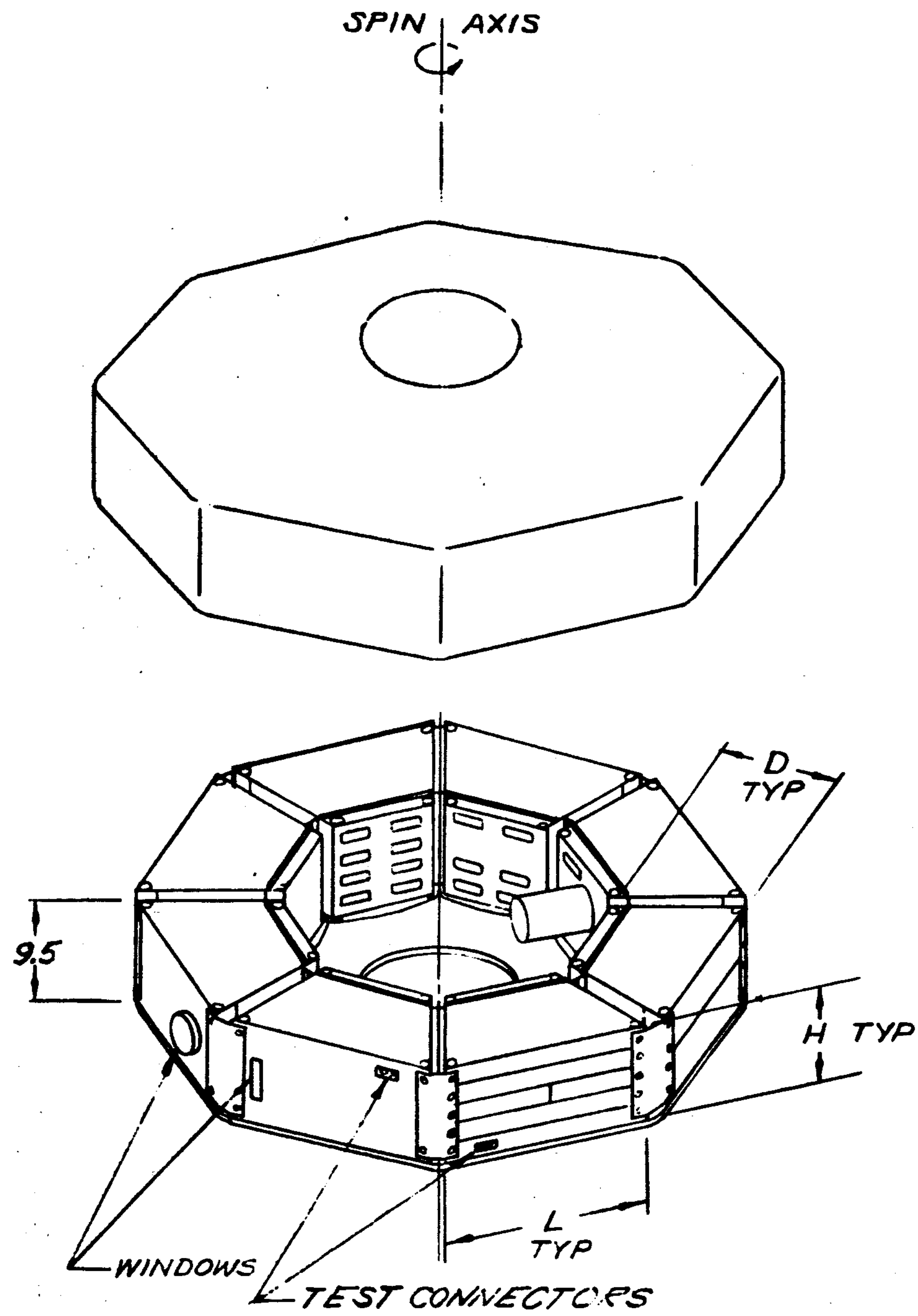


Figure 2-1—Interplanetary Monitoring Platform IMP F & G

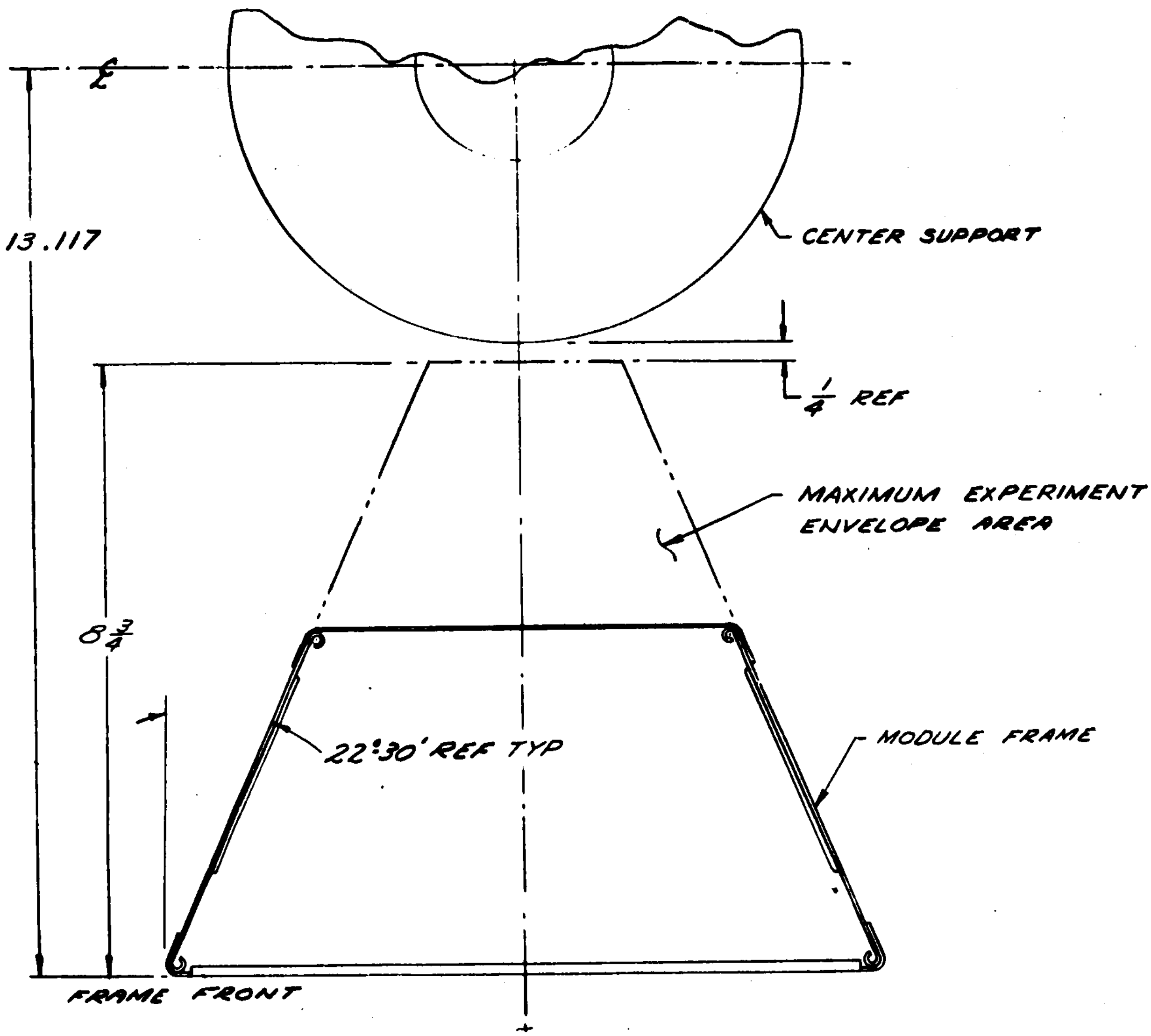


Figure 2-1a—Interplanetary Monitoring Platform IMP F & G

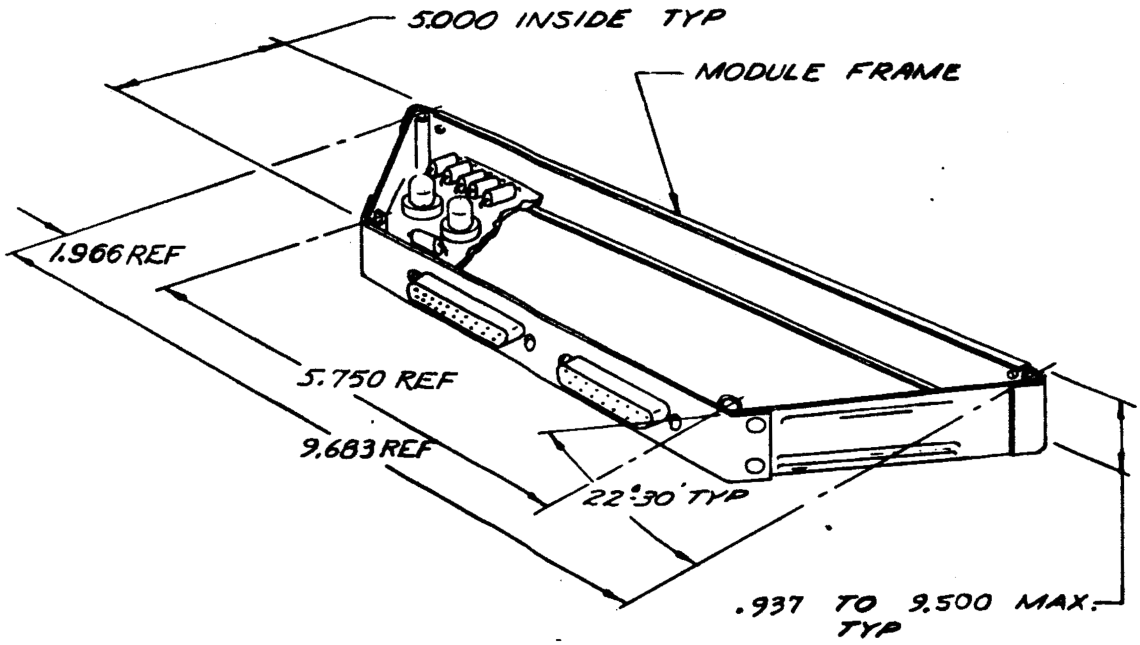


Figure 2-2–Interplanetary Monitoring Platform IMP F & G

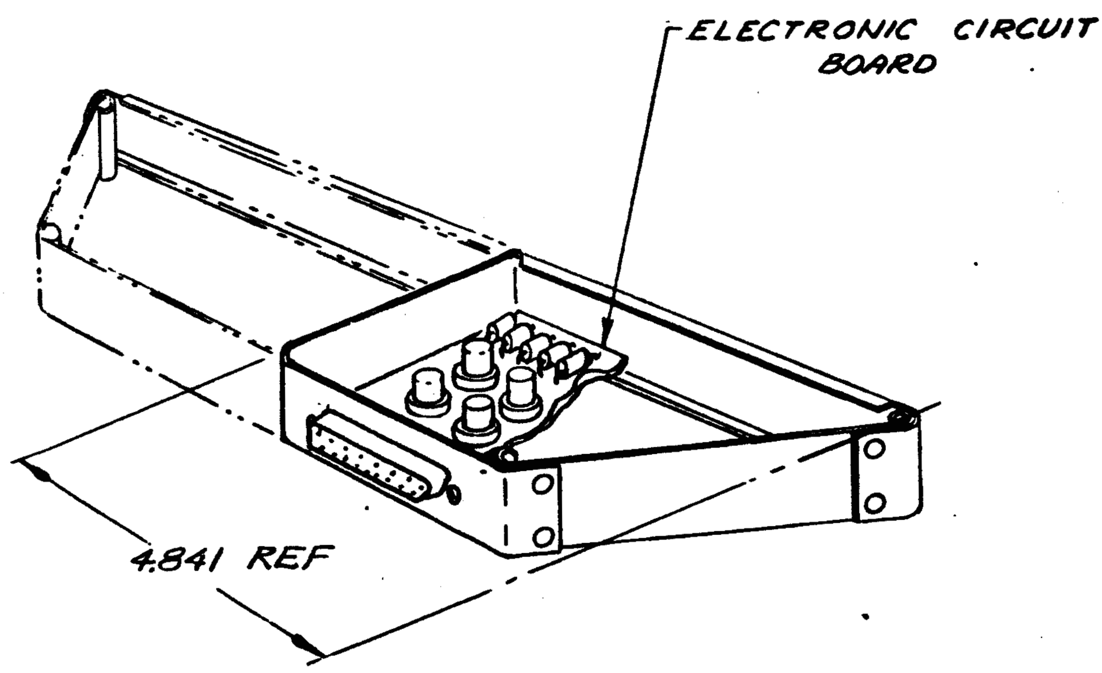


Figure 2-2a–Interplanetary Monitoring Platform IMP F & G

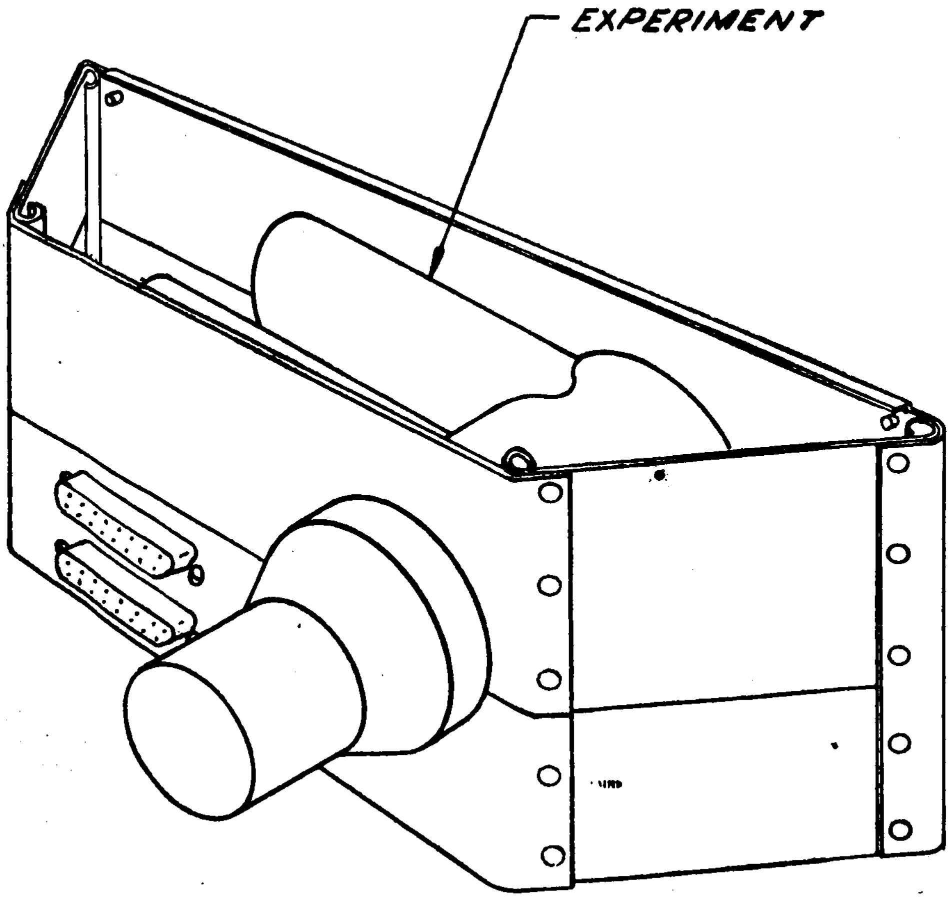


Figure 2-3—Interplanetary Monitoring Platform IMP F & G

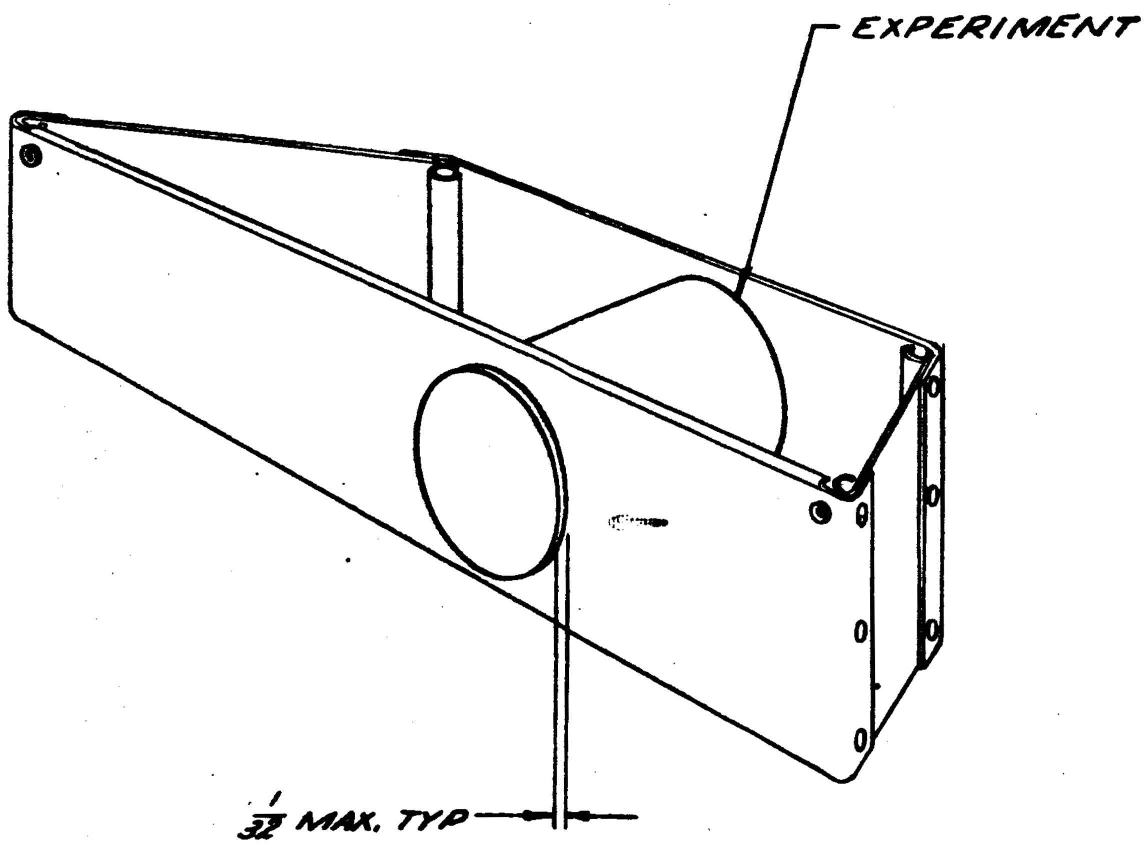


Figure 2-4—Interplanetary Monitoring Platform IMP F & G

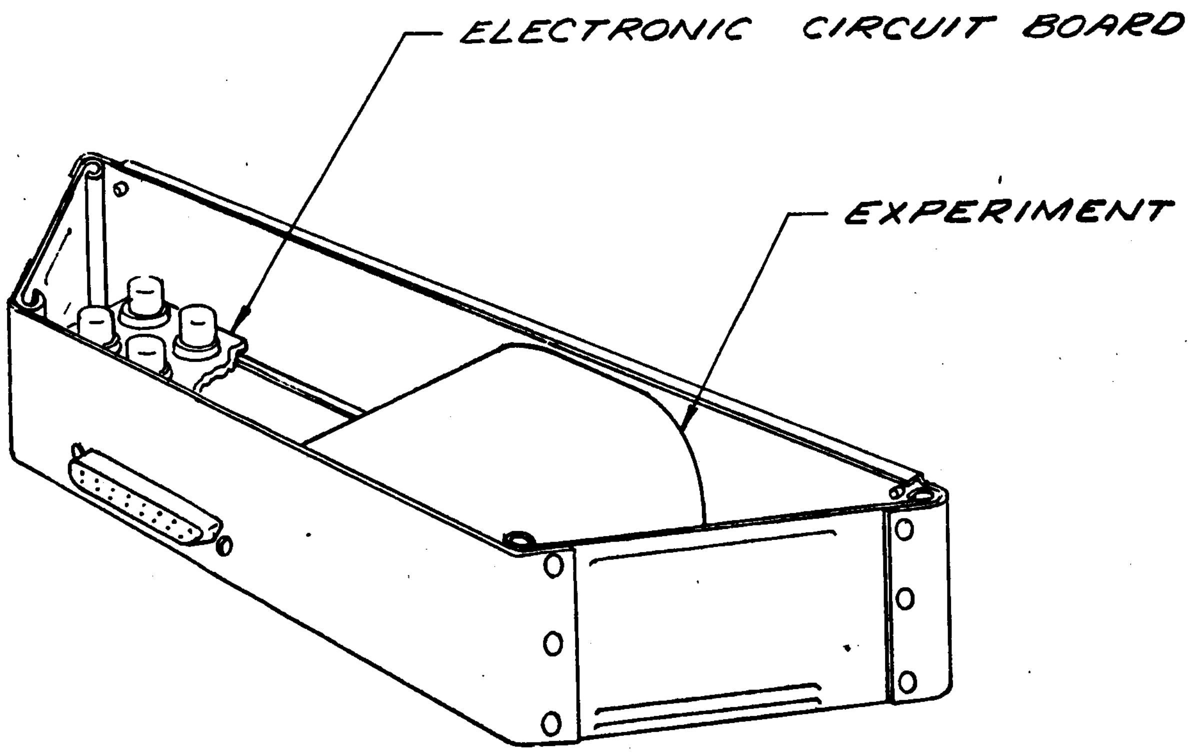


Figure 2-5–Interplanetary Monitoring Platform IMP F & G

NOTES:

1. IDENTIFICATION CODE NO. (1-LINE BOTTOM LEFT).
2. NOMENCLATURE (COMPONENT NAME) 2-LINES TOP RIGHT.
3. ALL OTHER MARKINGS TO BE LOCATED IN BOTTOM RIGHT HAND CORNER AS SHOWN.
4. ALL LETTERING TO BE 1/8 HIGH CAPITAL LETTERS .005 DEEP.

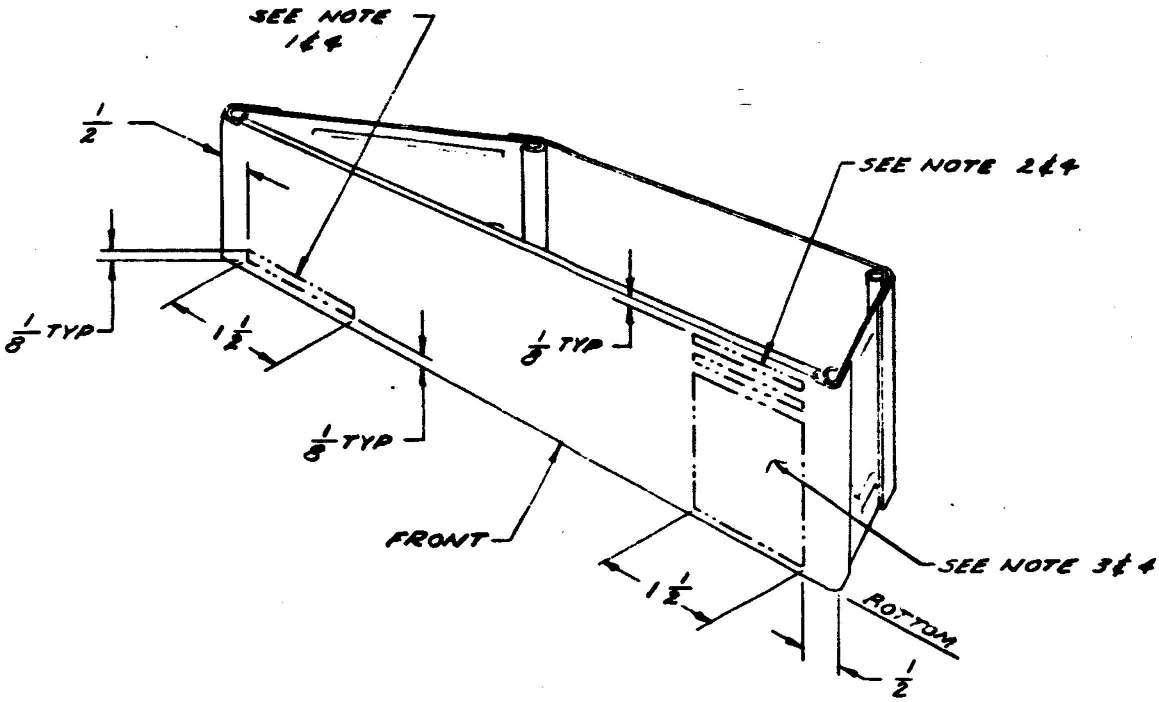
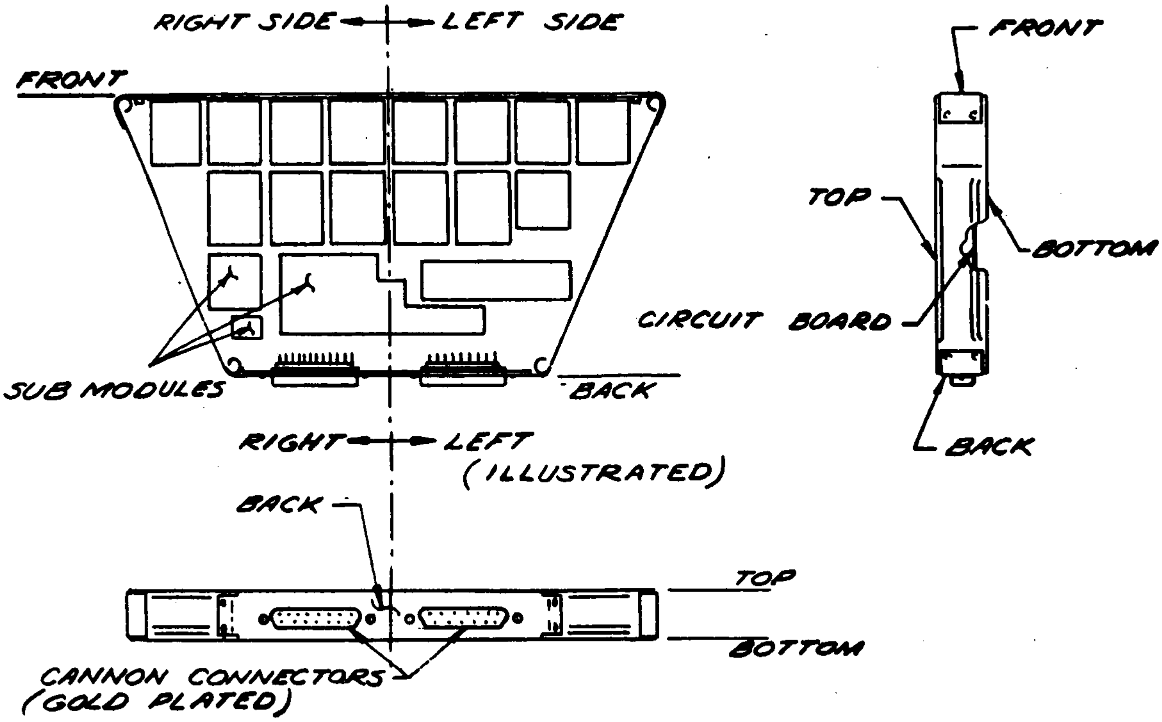


Figure 2-6—Interplanetary Monitoring Platform IMP F & G



NOTE:
PLUGS PLACED IN MODULES AS ILLUSTRATED WITH WIDE SIDE OF PLUG TOWARD TOP.

Figure 2-7–Interplanetary Monitoring Probe

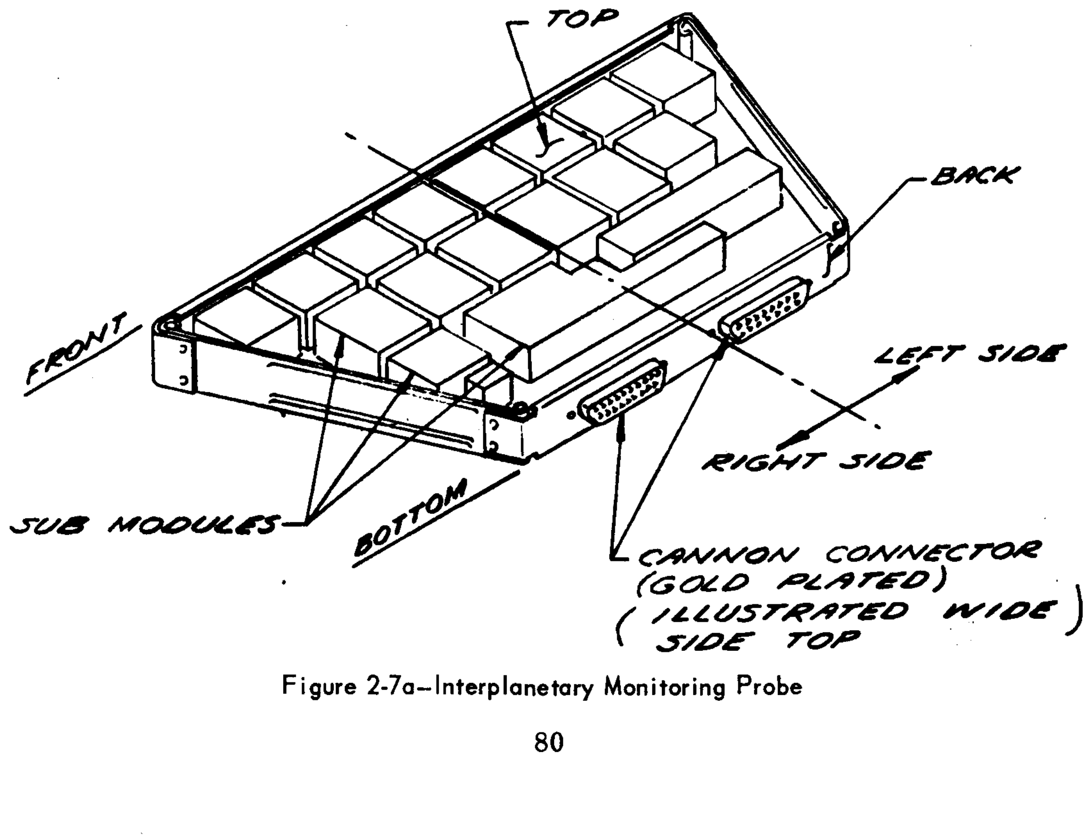


Figure 2-7a–Interplanetary Monitoring Probe

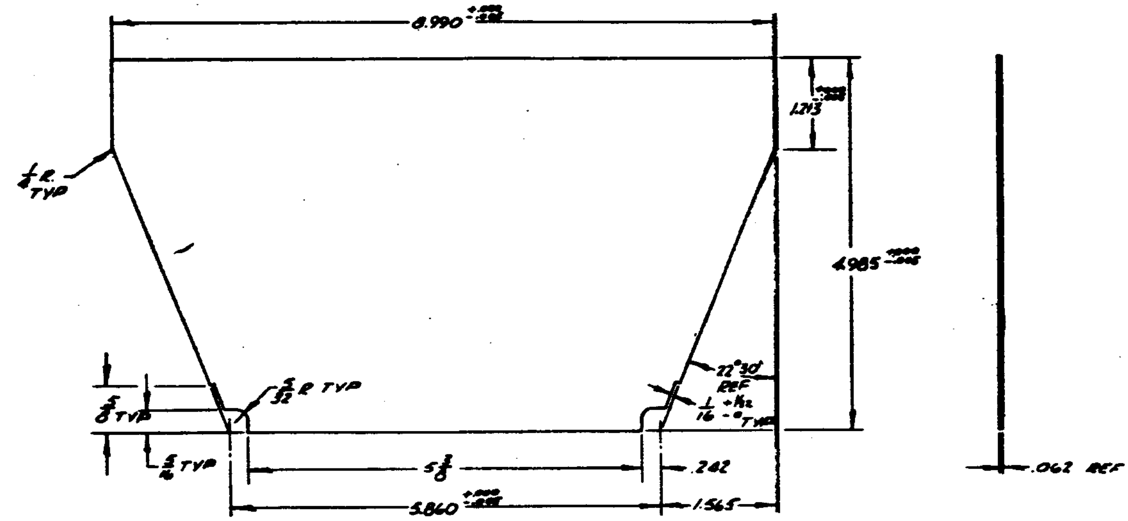


Figure 2-8—Circuit Board

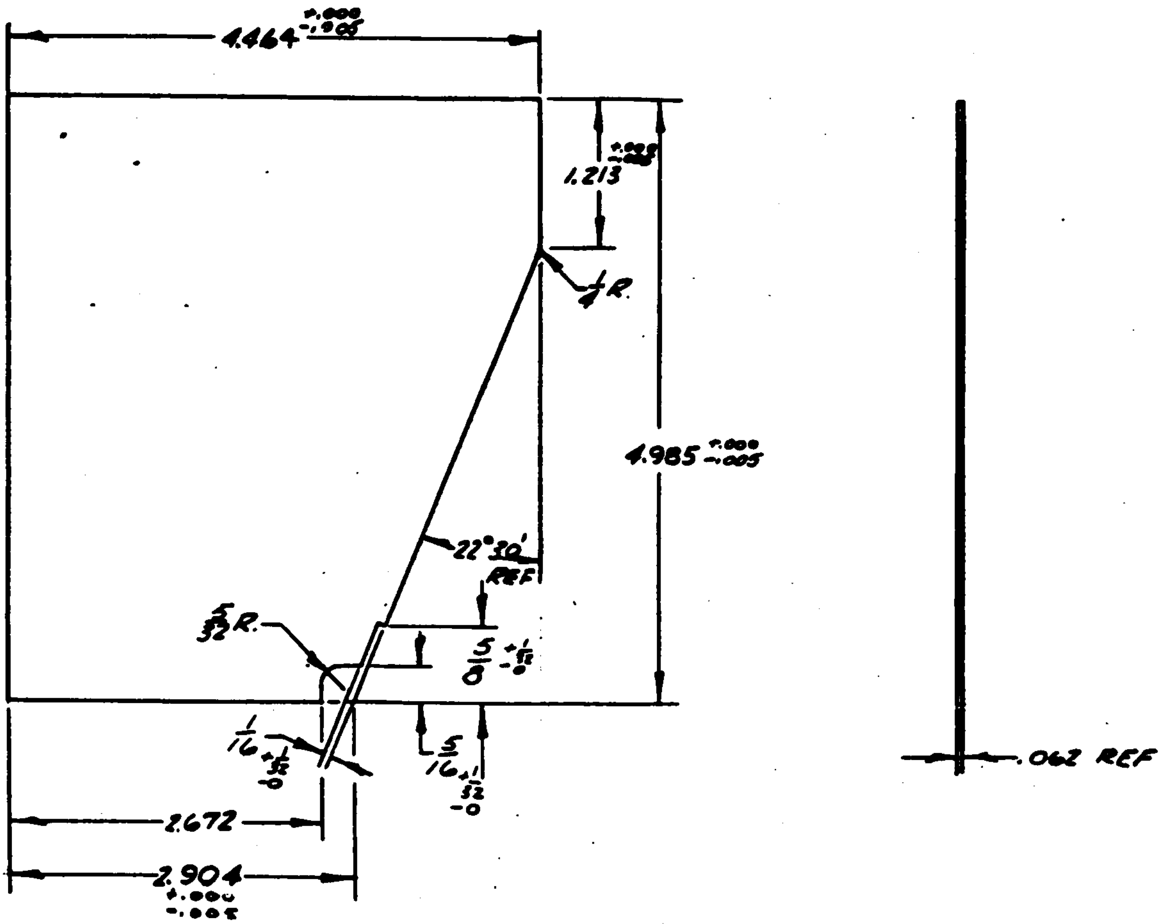


Figure 2-9–Circuit Board

MOMENT OF INERTIA

ATTACHMENT 3

AIMP MECHANICAL MEMORANDUM

No. 7

MOMENT OF INERTIA MEASUREMENTS

E. W. Travis
D. K. McCarthy
D. Miller

AIMP MOMENT OF INERTIA MEASUREMENTS

This document outlines the procedure to be followed in conducting the AIMP moment of inertia measurements. The inertia measurements will be made utilizing the torsion rod principle.

1.0 Standard Disc

1.1 Standard Disc Weight - 50.156 lb

1.2 Standard Disc Inertia

The standard disc's moment of inertia about the Z-Z axis (see Figure 1) is:

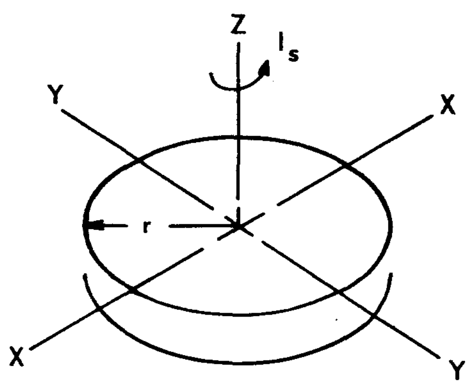


Figure 1

$$I_s = 1/2\ mr^2$$

$$= \frac{(50.156\ \text{lb})}{64.4\ \text{ft/sec}^2}\ (.5885\ \text{ft}^2) =$$

$$I_s = \underline{\quad .265 \quad}\ \text{slug ft}^2$$

1.3 Standard Disc Period (T_{s_r}) for I_{roll}

The disc is mounted to the torsion rod, a picture taken, torqued approximately ±10°, released, and the free oscillation timed.

The torsion rod is employed with an extension rod for clearance.

Test	Oscillations	Time (Sec.)	Period (Sec), T_{s_r}
1	50		
2	50		
3	50		

1.4 Measuring Unknown Inertias in Roll Plane

The unknown spacecraft inertias will now be obtained by handing the spacecraft from the torsion rod in various orientations, obtaining the period, (T_p) of each configuration, and calculating the moment of inertia from:

$$I_T = \frac{I_s}{T^2_{s_r}} T_T^2 = k_{s_r} (T_T)^2$$

where $k_{s_r} = \dfrac{I_s}{T^2_{s_r}} = \dfrac{.265}{(\quad)^2} = \_\_\_\_\_\_$

1.5 Inertia of Roll Attachments (I_{ra})

A. The attachment plate, adapter, 2 stainless steel marmon clamps, 2 eyebolts are attached to the standard disc, a picture taken and the combination period, (T_c), obtained.

Test	Oscillations	Time (Sec)	Period (Sec), T_c
1	50		
2	50		
3	50		

B. The attachment plate, adapter, two (2) stainless steel marmon clamps, two (2) eyebolts and the folded hardware are attached to the standard disc, a picture taken and the combination period, (T_{cf}), obtained

Test	Oscillations	Time (Sec)	Period (Sec), T_{cf}
1	50		
2	50		
3	50		

The combination inertia (I_c), is therefore

a. $I_c = \dfrac{I_s}{T^2_{s_r}} T_c^2 = k_{s_r} (\quad)^2 = (\quad)(\quad)^2 = \_\_\_\_\_\_$ slug ft^2

b. $I_{c_f} = \frac{I_s}{T^2_{s_r}} T^2_{c_f} = k_{s_r} (\quad) = (\quad)(\quad)^2 =$ ______ slug ft²

The inertia of the roll attachments are:

a. $I_{ra} = I_c - I_s =$ ______ $-$ $\underline{.265}$ $=$ ______ slug ft²

b. $I_{raf} = I_{cf} - I_s =$ ______ $-$ $\underline{.265}$ $=$ ______ slug ft²

1.6 Spacecraft Roll Inertias

Utilizing the attachments and marmon clamps, the spacecraft is hung from the torsion rod with its roll axis colinear with the torsion rod central axis. Safety lines are tied to the payload and pictures taken of each configuration. The various configurations are torqued and the spacecraft inertias obtained by calculating the combination inertia of the payload and roll attachments (either a or b), I_T, and subtracting the roll attachments, I_{ra} (either a or b).

$I_T = k_{s_r} T_T^2 = (\quad) T_T^2$

DATA SHEET

I_{roll}

Configuration	Run Oscillations	Time (Sec)	Period (Sec), T_T
1. Paddles folded, F/G booms folded, 4th stage motor hdw.			
2. Paddles extended, F/G booms folded, 4th stage motor hdw.			
3. Paddles extended, F/G booms extended, 4th stage motor hdw.			
4. *Paddles extended, F/G booms extended with empty 4th stage motor	-	-	-
5. Paddles extended, F/G booms extended without 4th stage motor			

*Determined Analytically

Calculation Sheet for I_{roll}

$I_T - I_{ra} + I_{motor} = I_{roll\ total}$

Configuration 1

$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 =$ ________ slug-ft^2

$I_T - I_{raf} + I_{motor} + I_{roll\ total}$

$(\quad) - (\quad) + (\quad) =$ ________ slug-ft^2

Configuration 2

$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 =$ ________ slug-ft^2

$I_T - I_{ra} + I_{motor} = I_{roll\ total}$

$(\quad) - (\quad) + (\quad) =$ ________ slug-ft^2

Configuration 3

$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 =$

$I_T - I_{ra} + I_{motor} = I_{roll\ total}$

$(\quad) - (\quad) + (\quad) =$ ________ slug-ft^2

Configuration 4. (Determined Analytically with Data from Configuration 3)

$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 =$ ________ slug-ft^2

$I_T - I_{ra} + I_{motor\text{, }empty} = I_{roll\ total}$

$(\quad) - (\quad) + (\quad) =$ ________ slug-ft^2

Configuration 5

$$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 = \underline{\qquad\qquad} \text{ slug-ft}^2$$

$$I_T - I_{ra} = I_{roll\ total}$$

$$(\quad) - (\quad) = \underline{\qquad\qquad} \text{ slug-ft}^2$$

3.0 Phase III - Spacecraft Transverse Axes

The fixture (with stainless steel marmon clamp attached) used to hold the spacecraft during the transverse measurement is shown in Figure 2, Drawing No. GE-IMP(D)-2347 (Sheet 1 & 2).

3.1 Standard Disc Period (T_{s_p}) for I_{pitch}

The disc is mounted to the Torsion Rod, a picture taken, torqued approximately ±10°, released, and the free oscillations timed.

The Torsion Rod is employed <u>without</u> an extension for I_{pitch}

Test	Oscillations	Time (Sec)	Period (Sec), T_{s_p}
1	50		
2	50		
3	50		

$$\therefore I_p = \frac{I_s}{T_{s_p}^2} T_p^2 = k_{s_p} (T_p)^2 = \frac{(.265)}{(\quad)^2} T_p^2 = (\quad) T_p^2$$

3.2 Measuring Unknown Inertias in Transverse Plane

The unknown spacecraft inertias will now be obtained by hanging the spacecraft from the fixture, hanging from the Torsion Rod. To provide sufficient data for the calculation of the maximum and minimum moments of inertia for each particular configuration, it is necessary to measure the inertia about any three separate axes in the transverse plane. These three measurements, I_x, I_m, I_y, will be made 45° apart as marked on Figure 3, Drawing No. GE-IMP(D)-2422. Each inertia of the spacecraft alone, $I_{s/c}$, is obtained by measuring the

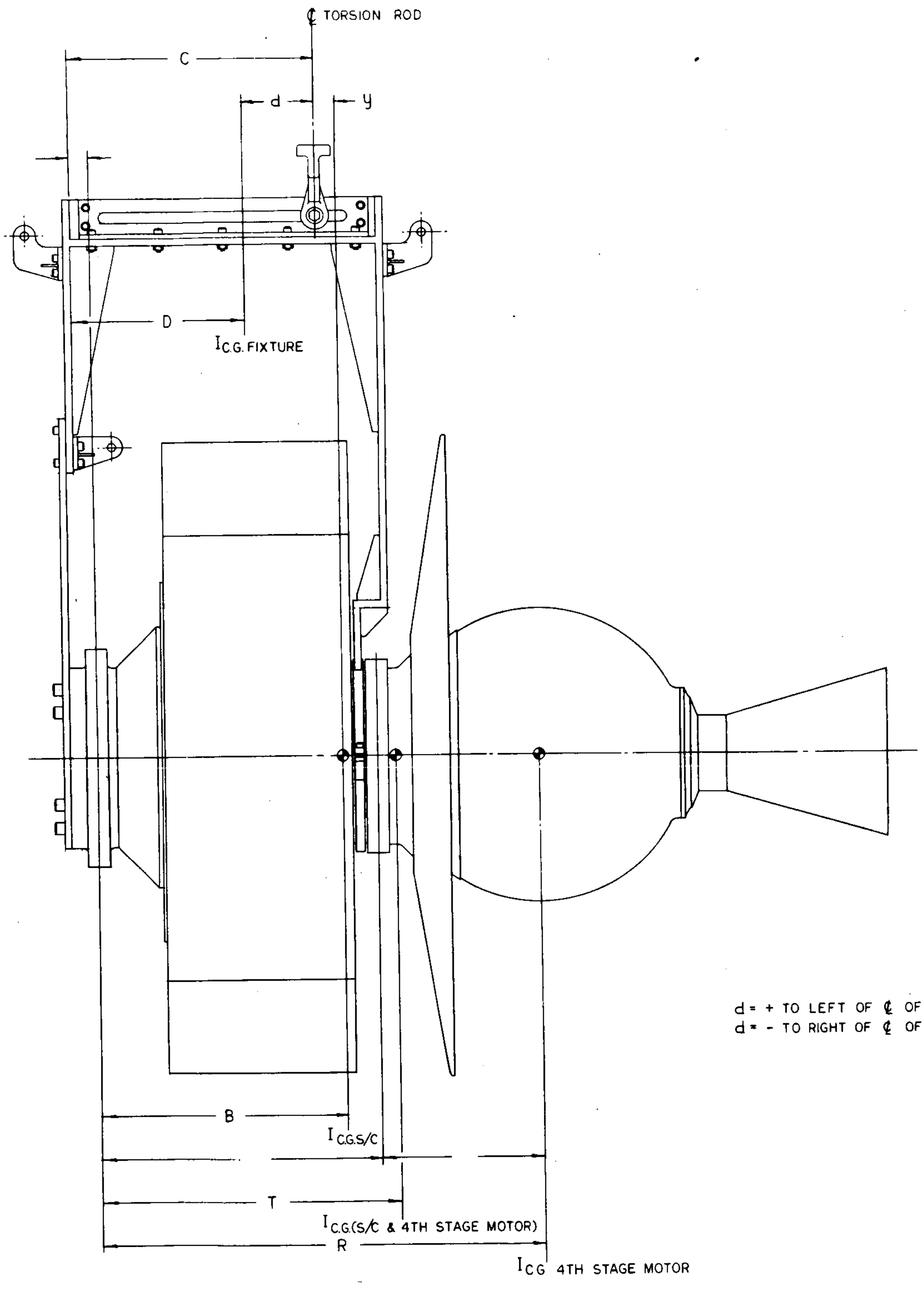


Figure 2 (Sheet 1)

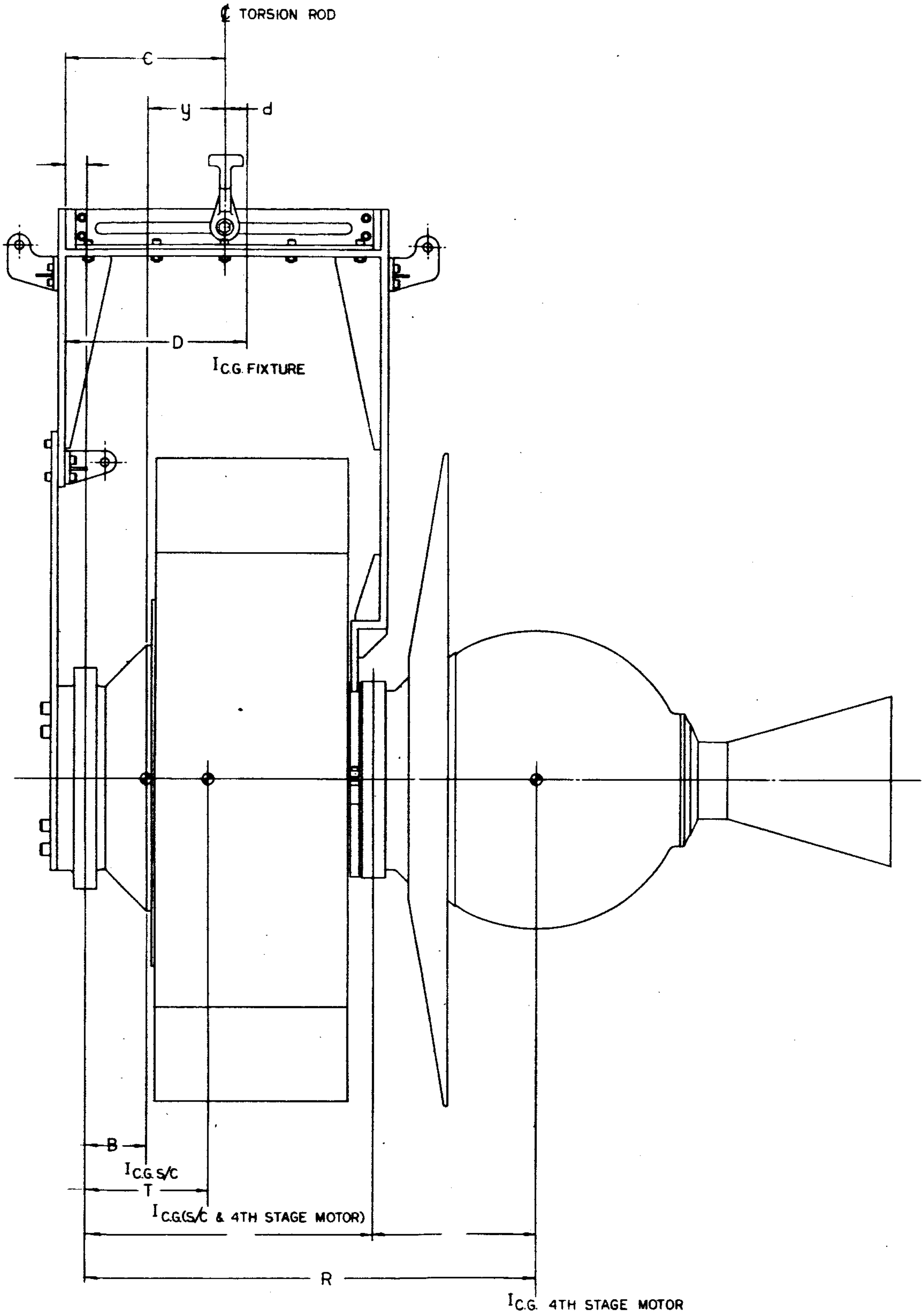


Figure 2 (Sheet 2)

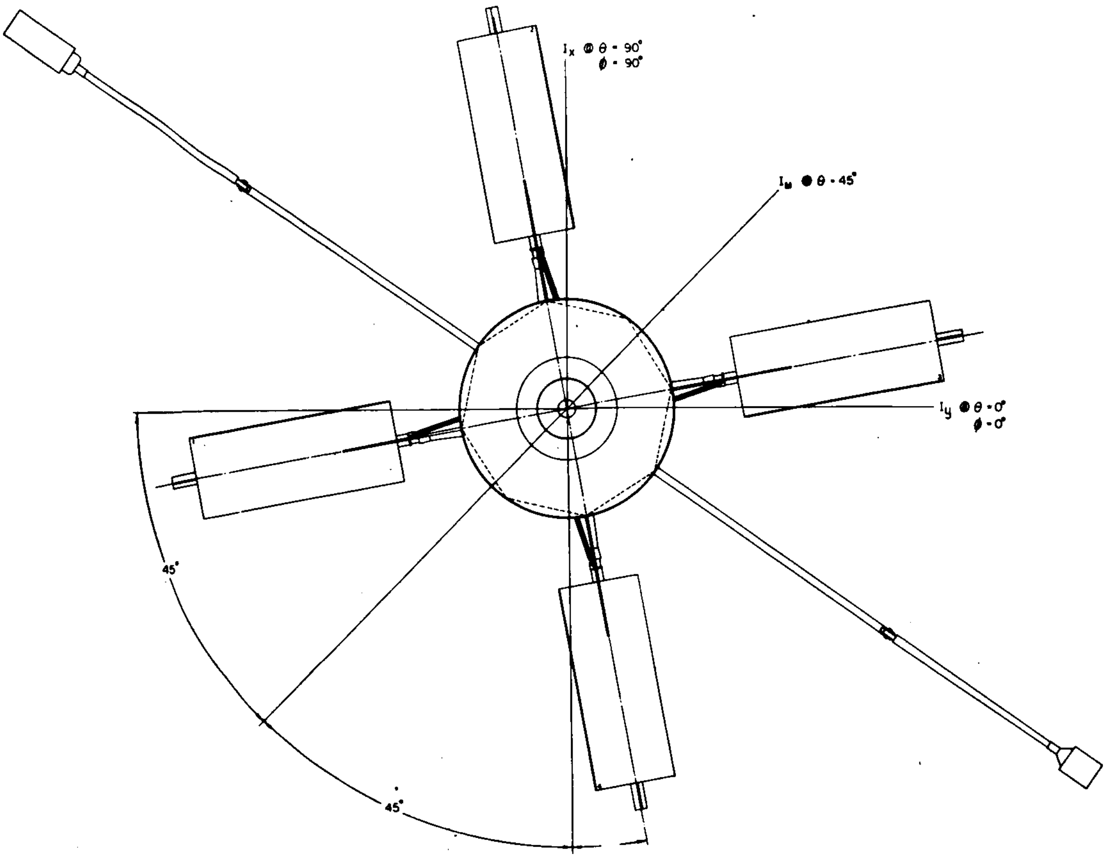


Figure 3

total inertia, I_{total}, in each orientation of the combination payload and transverse inertia fixture, I_{total}, and subtracting the corresponding inertia of the transverse inertia fixture, $I_{c.g._{fixture}}$ and the (M) $(x)^2$'S.

The center of gravity of the payload must also be obtained in each configuration, because the inertias will be obtained without the 4th stage retro motor, because it is not feasible to employ the live 4th stage.

Knowing the c.g. and $I_{c.g.\ 4th\ stage}$ of the 4th stage and determining the c.g. and $I_{c.g.\ s/c}$, the total transverse inertia of the various configurations can be obtained in the following manner:

Assume a configuration (Reference Figure 2, GE-IMP(D)-2347).

Determine the distance d and c, and D as a cross check.

ΣM @ Rod $= W_{fixture}\ (d) - W_{s/c}\ (y) = 0$ (When fixture is leveled)

d = + to left of of Torsion Rod

d = - to right of of Torsion Rod

$$\therefore y = \frac{(W_{fixture})\ (d)}{W_{s/c}}$$

$$\therefore c.g._{s/c},\quad B = C - A \pm y$$

$$I_{s/c} = I_{total} - I_{c.g.\ fixture} - M_{fixture}\ (d)^2 - M_{s/c}\ (y)^2$$

The data on the fixture (with stainless steel marmon clamp attached) used to hold the spacecraft during the transverse measurement is shown in Figure 2 (GE-IMP(D)-2347).

a. Fixture without folded attachment hardware

Weight =

Oscillations =

Time, sec =

Period, sec (T_F)

D =

b. Fixture with folded attachment hardware

Weight =

Oscillations =

Time, sec =

Period, sec (T_{FF}) =

D =

To Determine the Maximum and Minimum

Transverse Moments of Inertia

Ref. Figure 3 (Drawing No. GE-IMP(D)-2422)

Ref. "Principles of Mechanics," by Synge and Griffith

$$I_M = \frac{I_x + I_y}{2} + \frac{I_x - I_y}{2} \cos 2\theta - I_{xy \sin 2\theta}$$

in this case $\theta = 45°$ (Angle between I_x, I_y, I_M)

Solve for I_{xy}

$$I_{xy} = \frac{\frac{I_x + I_y}{2} + \frac{I_x - I_y}{2} \cos 2\theta - I_M}{\sin 2\theta}$$

$$\text{Tan } 2\phi = \frac{2 I_{xy}}{I_x - I_y}$$

The angle, ϕ determines how much to rotate the original axes (x,M,y) to find the Axes which contain the principal transverse moments of inertia.

$$I_{x_{max}} = \frac{I_x \cos^2 \phi - I_y \sin^2 \phi}{\cos 2\phi}$$

$$= \frac{I_x (1/2 + 1/2 \cos 2\phi) - I_y (1/2 - 1/2 \cos 2\phi)}{\cos 2\theta}$$

$$I_{y_{min}} = \frac{I_y \cos^2 \phi - I_x \sin^2 \phi}{\cos 2 \phi}$$

$$= \frac{I_y (1/2 + 1/2 \cos 2 \phi) - I_x (1/2 - 1/2 \cos 2 \phi)}{\cos 2 \phi}$$

Data Sheet

$I_{transverse}$

Configuration	Axis	Run	Oscillations	Time (sec)	T_{total} Period (sec)
I Paddle Folded		1			
	x-x	2			
F/G Booms Folded		3			
		1			
	M-M	2			
with 4th stage Hdw.		3			
		1			
	y-y	2			
		3			
		1			
	x-x	2			
II Paddle Extended		3			
		1			
	M-M	2			
F/G Booms Folded		3			
		1			
	y-y	2			
with 4th stage Hdw.		3			
		1			
	x-x	2			
III Paddles Extended		3			
		1			
	M-M	2			
F/G Booms Extended		3			
		1			
	y-y	2			
with 4th stage Hdw.		3			
		1			
	x-x	2			
IV Paddles Extended		3			
		1			
	M-M	2			
F/G Booms Extended		3			
		1			
	y-y	2			
without 4th stage Hdw.		3			

Calculation Sheet for $I_{transverse}$

Configuration (   )

$d =$

$c =$

$D =$ (Measure as a check of C + d)

$$y = \frac{(W_{fixture})\ (d)}{W_{s/c}} = \frac{(\quad)\ (\quad)}{(\quad)} = \underline{\qquad\qquad} \text{ in.}$$

$$c.g._{s/c},\ B = C - A \pm y = (\quad) - (\quad) \pm (\quad) = \underline{\qquad\qquad} \text{ in.}$$

$$c.g._{s/c + 4th},\ T = \frac{(W_{s/c})\ (B) + (W_{4th})\ (R)}{W_{s/c} + W_{4th}}$$

$$I_{c.g.\ fixture} = k_{sp}\ (T_p)^2 = (\quad)\ (\quad)^2 = \underline{\qquad\qquad} \text{ slug-ft}^2$$

$$M_{fixture}\ (d)^2 =$$

$$M_{s/c}\ (y)^2 =$$

Axis (x-x)

$$I_{total} = k_{sp}\ (T_{total})^2 = (\quad)\ (\quad)^2 = \underline{\qquad\qquad} \text{ slug-ft}^2$$

$$I_{c.g.\ s/c} = I_{total} - I_{c.g.\ fixture} - M_{fixture}\ (d)^2 - M_{s/c}\ (y)^2$$

$$= (\quad) - (\quad) - (\quad)\ (\quad)^2 - (\quad)\ (\quad)^2 + \text{slug-ft}^2$$

$$I_{s/c + 4th} = I_{c.g.\ s/c} + M_{s/c}\ (T\text{-}B)^2 + I_{4th} + M_{4th}\ (R\text{-}T)^2$$

$$= (\quad) + (\quad)\ (\quad)^2 + (\quad) + (\quad)\ (\quad)^2 =$$

$$= \underline{\qquad\qquad} = \text{slug-ft}^2$$

Axis (y-y)

$$I_{total} = k_{sp} \, (T_{total})^2 = (\quad)(\quad)^2 = \underline{\hspace{3cm}} \text{ slug-ft}^2$$

$$I_{s/c} = I_{total} - I_{c.g.\ fixture} - M_{fixture} \, (d)^2 - M_{s/c} \, (y)^2$$

$$= (\quad) - (\quad) - (\quad) - (\quad) = \underline{\hspace{3cm}} \text{ slug-ft}^2$$

$$I_{s/c + 4th} + I_{c.g.\ s/c} + M_{s/c} \, (T-B)^2 + I_{4th} + M_{4th} \, (R-T)^2$$

$$+ (\quad) + (\quad)(\quad)^2 + (\quad) + (\quad)(\quad)^2 =$$

$$= \underline{\hspace{3cm}} = \text{slug-ft}^2$$

Axis (M-M)

$$I_{total} = k_{sp} \, (T_{total})^2 = (\quad)(\quad)^2 \underline{\hspace{3cm}} \text{ slug-ft}^2$$

$$I_{s/c} = I_{total} - I_{c.g.\ fixture} - M_{fixture} \, (d)^2 - M_{s/c} \, (y)^2$$

$$= (\quad) - (\quad) - (\quad) - (\quad)$$

$$I_{s/c + 4th} = I_{c.g.\ s/c} + M_{s/c} \, (T-B)^2 + I_{4th} + M_{4th} \, (R-T)^2$$

$$= (\quad) + (\quad)(\quad)^2 + (\quad) + (\quad)(\quad)^2 =$$

$$= \underline{\hspace{3cm}} \text{ slug-ft}^2$$

$$I_{xy} = \frac{I_x + I_y}{2} + \frac{I_x - I_y}{2} \cos 2\theta - I_M$$

$$= \frac{(\quad) + (\quad)}{2} + \frac{(\quad) - (\quad)}{2} (\quad) - (\quad) =$$

$$\text{Tan } 2\phi = \frac{2 I_{xy}}{I_x - I_y} = \frac{2 (\quad)}{(\quad) - (\quad)}$$

$$\phi =$$

$$I_{x_{max}} = \frac{I_x \, (1/2 + 1/2 \cos 2\phi) - I_y \, (1/2 - 1/2 \cos 2\phi)}{\cos 2\phi}$$

$$= \frac{(\qquad)\,[.5 + .5\,(\cos \qquad)] - (\qquad)\,[.5 - .5\,(\cos \qquad)]}{(\qquad\qquad)} =$$

$$I_{y_{min}} = \frac{I_y \, (1/2 + 1/2 \cos 2\phi) - I_x \, (1/2 - 1/2 \cos 2\phi)}{\cos 2\phi}$$

$$= \frac{(\qquad)\,[.5 + .5\,(\cos \qquad)] - (\qquad)\,[.5 - .5\,(\cos \qquad)]}{(\qquad\qquad)} =$$

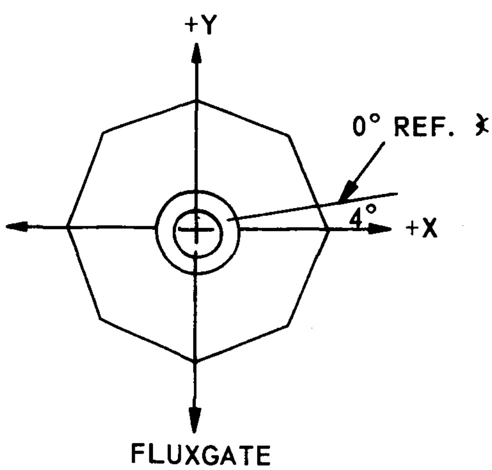
PHYSICAL DATA

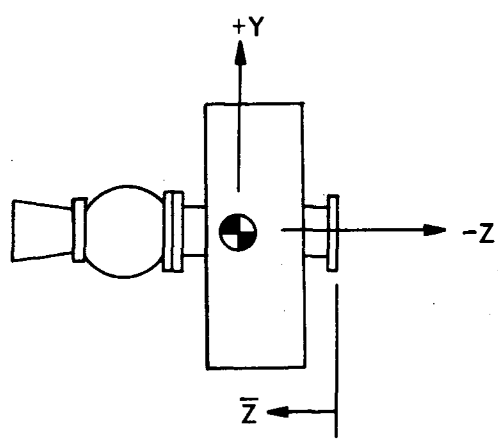
Spacecraft: ____________________ Date: ____________________

Estimated or Actual: ____________________ Prepared By: ____________________

Configuration	Weight (lb)	$\bar{x}$ (in.)	$\bar{y}$ (in.)	$\bar{z}$ (in.)	I_{xx} (Slug-ft^2)	I_{yy} (Slug-ft^2)	I_{zz} (Slug-ft^2)	Spin Rate (RPM)
Launch (all appendages folded)								
Yo-Yo Deployed								
Paddles Erected								
Booms Erected								
Post Retro-Fire								
Post Retro-Separation								

Remarks: 1. X258 Motor Inertias are NOT included in the above I_{zz} for X258 is 0.73 slug-ft^2.





DYNAMIC BALANCE VS
MASS MOMENTS OF INERTIA

ATTACHMENT 4

AIMP MECHANICAL MEMORANDUM

No. 2

DYNAMIC BALANCE VS MASS MOMENTS OF INERTIA

E. W. Travis
D. K. McCarthy
D. L. Miller

April 15, 1965

AIMP MECHANICAL MEMORANDUM NO. 2

SUBJECT: Dynamic Unbalance/Moments of Inertia

The equation employed to determine dynamic unbalance versus the moments of inertia is:

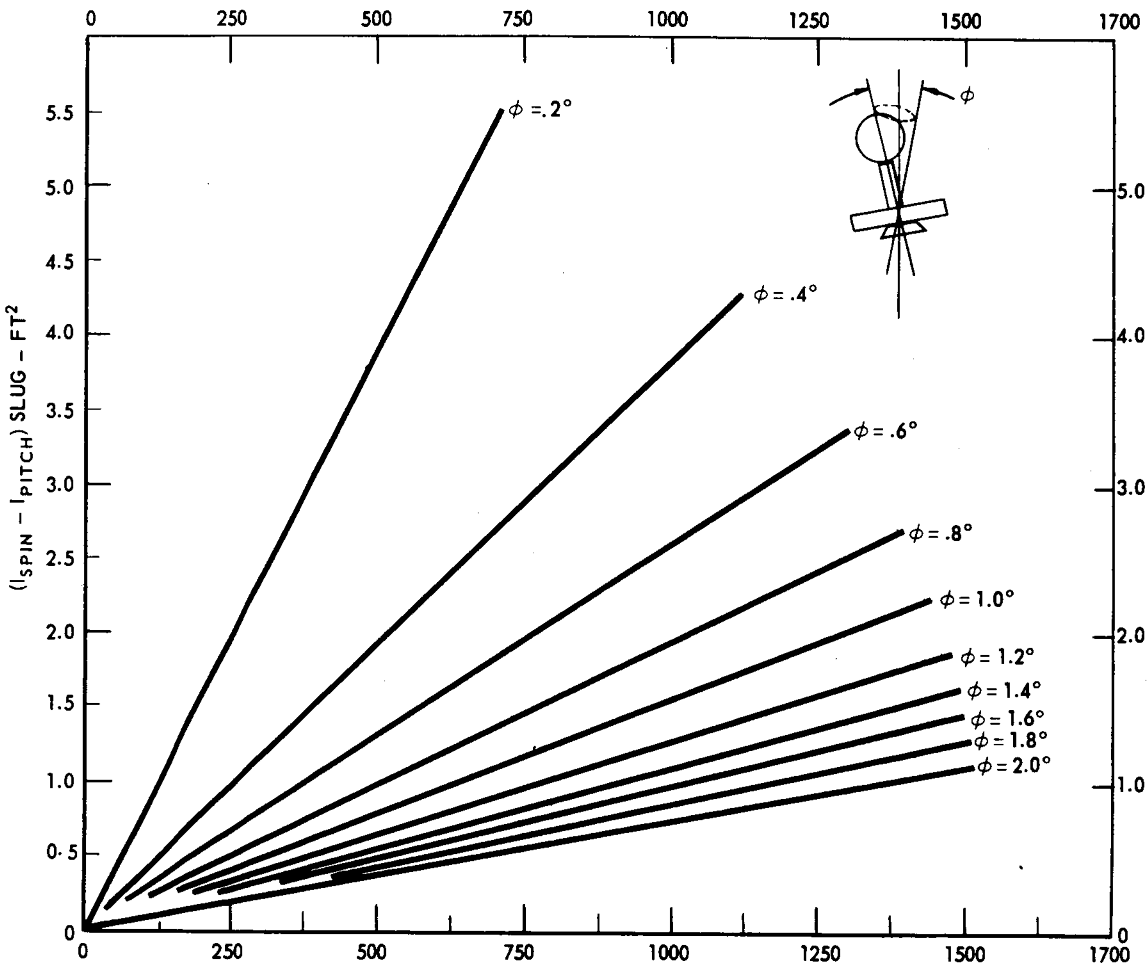
$$\text{TAN } 1/2\ \phi = \frac{\text{Dynamic Unbalance}}{(I_{\text{roll}} - I_{\text{transverse}})}\ \text{'g'}$$

where ϕ is the total coning angle (Ref. NASA TN D-1446).

The enclosed graph illustrates the relationship between dynamic unbalance versus the moments of inertia.

D. K. McCarthy
D. Miller
E. W. Travis

Enclosure



0
250
500
750
1000
1250
1500
1700
$(I_{SPIN} - I_{PITCH})$ SLUG - FT^2
5.5
5.0
4.5
4.0
3.5
3.0
2.5
2.0
1.5
1.0
0.5
0
ϕ =.2°
ϕ = .4°
ϕ = .6°
ϕ = .8°
ϕ = 1.0°
ϕ = 1.2°
ϕ = 1.4°
ϕ = 1.6°
ϕ = 1.8°
ϕ = 2.0°
ϕ

Graph 2

DERIVATION OF THE OVERSPIN EQUATIONS TO COMPENSATE FOR GRAVITY AND INERTIAL VARIATIONS DURING APPENDAGE ERECTION

ATTACHMENT 5

AIMP MECHANICAL MEMORANDUM

No. 1

DERIVATION OF THE OVERSPIN EQUATIONS TO COMPENSATE FOR GRAVITY AND INERTIAL VARIATIONS DURING APPENDAGE ERECTION

E. W. Travis
D. K. McCarthy
D. L. Miller

April 2, 1965

The energy absorbed by spacecraft appendages during erection in a zero gravity field is the difference of kinetic energy between the folded and erected states.

$$\underline{\text{Folded:}} \quad KE_1 = \frac{1}{2} I_1 \omega_1^2 \tag{1}$$

$$\underline{\text{Erected:}} \quad KE_2 = \frac{1}{2} I_2 \omega_2^2 \tag{2}$$

$$\Delta E_S = \frac{1}{2} (I_1 \omega_1^2 - I_2 \omega_2^2) \tag{3}$$

$$\Delta E_S = \frac{1}{2} I_1 \omega_1^2 (1-R) \text{ where } R = \frac{I_1}{I_2} \tag{4}$$

During erection testing in the influence of a 1g gravity field, the energy absorbed by an appendage is decreased by the potential energy imparted to the arm.

$$\underline{\text{Folded:}} \quad KE_1 = \frac{1}{2} I_1 \omega_{1T}^2; \quad P.E. = 0 \tag{5}$$

$$\underline{\text{Erected:}} \quad KE_2 = \frac{1}{2} I_2 \omega_{2T}^2; \quad P.E. = \Sigma\, wh \tag{6}$$

$$\Delta E_G = \Delta KE + \Delta PE \tag{7}$$

$$\Delta E_G = \frac{1}{2} (I_1 \omega_{1T}^2 - I_2 \omega_{2T}^2) - \Sigma\, wh \tag{8}$$

$$\Delta E_G = \frac{1}{2} I_1 \omega_{1T}^2 (1-R) - \Sigma\, w, \text{ where } R = \frac{I_1}{I_2} \tag{9}$$

The purpose of the ground test shall be to subject the appendage to the energy experienced in a space erection, therefore:

$$\text{Make } \Delta E_G = \Delta E_S, \text{ and assume } I_{1T} = I_{1S} \tag{10}$$

$$\frac{1}{2} I_1 \omega_{1T}^2 \ (1 - R_S) - \Sigma wh = \frac{1}{2} I_1 \omega_{1S}^2 \ (1 - R_S) \tag{11}$$

$$\boxed{\omega_{1T} = \sqrt{\omega_{1S}^2 + \frac{2 \Sigma wh}{I_1 \ (1 - R_S)}}} \tag{12}$$

NOTE: This derivation assumes that test inertias and flight inertias are identical.

ω_{1T} represents the increased initial spin rate which must be used if the appendage is to experience the proper energy input at erection.

In some test situations, it is either impossible or inconvenient to achieve an initial folded configuration inertia (I_1) equal to the initial flight inertia. An example of this has occurred on the AIMP. In flight, the initial inertia is the total of the spacecraft launch configuration inertia plus the inertia of the empty X258 motor. To match this inertia in test requires that the spin table, motor mock-up etc. be inertially identical to the empty X258. Sometimes this is not a reasonably attained goal, however, quite fortunately it is possible to produce the proper erection energy input to the appendages despite this inertial difference. The following is a derivation of the initial test spin rate necessary to compensate for both gravity effects and inertial variances.

a. Assume $I_{2T} - I_{1T} = I_{2S} - I_{1S}$ (13)

b. The object of the test is to make $E_T = E_S$

c. Using equation (9)

$$\frac{1}{2} I_{1T} \omega_{1T}^2 \ (1 - R_T) - \Sigma wh = E_S \tag{14}$$

$$\boxed{\omega_{1T} = \sqrt{\frac{2 \ (E_S + \Sigma wh)}{I_{1T} \ (1 - R_T)}}} \tag{15}$$

where $E_S = \frac{1}{2} I_{1S} \omega_{1S}^2 \ (1 - R_S)$ (4)

d. In the case where $R_S = R_T$ and $I_{1T} = I_{1S}$, equation (15) reduces to equation (12).

NOTE: Both the preceeding derivations neglect to show the erection spring energy. This was intentional since this energy is present in identical amounts for both flight and test conditions.

Terms

ΔE = Energy absorbed by appendages, ft-lb
h = height which the appendage center of mass is raised during erection, ft
I = Mass moment of inertia about the spin axis, slug-ft^2
w = weight of each appendage, lb
ω = spin rate, seconds^{-1}.

Subscripts:
1 = initial position, appendages folded
2 = final position, appendages erected
S = in space, zero gravity
T = on ground test, 1g
A = nominal DAC spin-up with successful yo-yo
B = 10% DAC overspin with successful yo-yo
C = 10% DAC overspin with yo-yo failure

Appendix

In the case of the AIMP ETU paddle erection tests, it will be necessary to alter the initial spin rates to compensate for both gravity and the inertial differences mentioned in the latter derivation of this memorandum.

A. Comparison of Initial Spin Axis Inertias (I.)

Item		Flight			Test
Spacecraft*	=	3.422 slug-ft^2			3.422 slug-ft^2
X258	=	.730 slug-ft^2			1.046**
$\therefore I_{1S}$	=	4.152 slug-ft^2	$\therefore I_{1T}$	=	4.468 slug-ft^2

*This value is for a launch configuration, appendages folded and loaded motor.

**This value includes X258 dynamic mock-up, DAC attach fitting, marmon clamp + 6 appendage cradles.

B. Adjustment

Section A reveals an inertial difference of 0.316 slug-ft^2. Since the condition of the kick motor was not important to this test, an empty motor was substituted for the full in an effort to reduce the inertial difference.

	4.468	slug-ft^2, Test Inertia
−	.315	slug-ft^2, loaded kick motor
	4.153	slug-ft^2,
+	.070	slug-ft^2, empty kick motor
	4.223	slug-ft^2, ETU for test

C. Flight Despin Sequence

$$\left.\begin{aligned} I_{3S} &= 16.163 \text{ slug-ft}^2 \\ I_{2S} &= 10.853 \text{ slug-ft}^2 \\ I_{1S} &= 4.152 \text{ slug-ft}^2 \end{aligned}\right\} \text{These include 0.730 slug-ft}^2 \text{ for the X258}$$

$$\omega_3 = 27.5 \text{ RPM}$$

$$\omega_0 = 150.0 \text{ RPM}$$

a. Boom erection spin rate $\omega_2 = \frac{I_3}{I_2} \omega_3 = \frac{16.163}{10.853}$ (27.5)

$\underline{\omega_2 = 41 \text{ RMP}}$

b. Paddle erection spin rate $\omega_1 = \frac{I_2}{I_1} \omega_2 = \frac{10.853}{4.152}$ (41)

$\underline{\omega_1 = 107.5 \text{ RPM};}$

D. Parameter Spin Rates:

a. Nominal = Nominal DAC spin up and successful yo-yo

$\underline{\omega_{1A} = 107.5 \text{ RPM}}$

b. Overspin = 10% DAC spin up and successful yo-yo

$\underline{\omega_{2A} = 118 \text{ RPM}}$

c. Yo-yo failure = 10% DAC spin up and yo-yo failure

$\underline{\omega_{3A} = 165 \text{ RPM}}$

E. Flight Energies (using Equation 4)

$$E_S = \frac{1}{2} I_{1S} \omega_{1SA}^2 \ (1-R_S),\ R_S = \frac{4.15}{10.85} = .383$$

$$\therefore E_S = \frac{1}{2} (4.15)\ (\omega_{1SA})^2\ (1-.383)$$

$$\underline{\therefore E_S = 1.28\ \omega_{1SA}^2}$$

$$E_{SA} = 1.28 \left(\frac{107.5}{9.55}\right)^2 = 162 \text{ ft-lb}$$

$$E_{SB} = 1.28 \left(\frac{118}{9.55}\right)^2 = 195 \text{ ft-lb}$$

$$E_{SC} = 1.28 \left(\frac{165}{9.55}\right)^2 = 380 \text{ ft-lb}$$

F. Test Spin Rates (using Equation 15)

$$\omega_{1T} = \frac{\sqrt{2(E_S + \Sigma wh)}}{I_{1T} \ (1-R_T)}$$

W = 6.15 lb/paddle, h = 2 ft, 4 paddles

$$R_T = \frac{4.22}{4.22 + (10.85 - 4.15)} = .386$$

$$\omega_{1TA} = \sqrt{\frac{2[162 + 4 \ (6.15) \ (2)]}{4.22 \ (1-.386)}} \approx 12.8 \text{ sec}^{-1}$$

$$\therefore \omega_{1TA} \approx 122 \text{ RPM}$$

Similarly:

$$\omega_{1TB} = 13.7 \text{ sec}^{-1} \approx 131 \text{ RPM}$$

$$\omega_{1TC} = 18.2 \text{ sec}^{-1} \approx 174 \text{ RPM}$$

G. Summarizing:

Case	Flight	Test
A = Nominal	107.5	122
B = Overspin	118	131
C = yo-yo failure	165	174

SUMMARY SHEET

DATE:____________________ CALCULATIONS BY:____________________

SPACECRAFT: ______________________________

CASE		Total Kinetic Energy Into Appendage (ft-lb)	Flight Spin Rate (RPM)	Test Spin Rate (RPM)
A	Nominal Spinup with Yo-Yo.			
B	10% Overspin with Yo-Yo.			
C	10% Overspin with Yo-Yo Failure.			

The data above was calculated using the following inputs:

I_{it} = ____________________ slug-ft^2

I_{is} = ____________________ slug-ft^2

Nominal Delta spin rate = ____________________ RPM

Nominal Orbital spin rate = ____________________ RPM

Paddle Weight = ____________________ lb/paddle

Height which Paddle is Raised = ____________________ feet.

DATA SHEET FOR TEST SPIN RATES

1. INERTIA DATA:

	Flight	Test
a. Spacecraft	________	________
b. X258	________	________
c. Spin Table	N/A	________
d. Others	________	________
Total:	________	________

Remarks:

I_{is} = ________ slug-ft^2

I_{it} = ________ slug-ft^2

2. DESPIN SEQUENCE: (use flight inertias only)

I_{3s} = ________ slug-ft^2, Paddles & booms erected + full motor.

I_{2s} = ________ slug-ft^2, Booms folded, paddles erected + full motor.

I_{1s} = ________ slug-ft^2, Booms and paddles folded + full motor.

ω_3 = ________ RPM, orbital spin rate

ω_0 = ________ RPM, Delta spin-up rate.

$$R_s = \frac{I_{1s}}{I_{2s}} = \_\_\_\_\_\_ , \quad R_t = \frac{I_{it}}{I_{it} + (I_{2s} - I_{1s})} = \_\_\_\_\_\_$$

a. $\omega_2 = \frac{I_{3s}}{I_{2s}} \omega_3$

$\omega_2 =$ ______________

$\omega_2 =$ ______________ RPM (boom erection)

b. $\omega_1 = \frac{I_{2s}}{I_{1s}} \omega_2$

$\omega_1 =$ ______________

$\omega_1 =$ ______________ Rpm (paddle erection)

3. SPIN RATES TO BE USED FOR ENERGY INPUT:

A. Nominal spin-up + nominal Yo-Yo, $\omega_{1A} = \omega_1$	$\omega_{1A} =$ ________ RPM
B. 10% overspin + nominal Yo-Yo, $\omega_{1B} = 1.1\omega_1$	$\omega_{1B}{}^2 =$ ________ RPM
C. 10% overspin + yo-yo failure, $\omega_{1c} = 1.1\omega_0$	$\omega_{1C} =$ ________ RPM

4. ENERGIES IMPARTED TO PADDLES FOR ZERO O 'g' ERECTION

a. $E_s = \frac{1}{2} I_1 (1 - R_s) \omega_1{}^2$

calculate k_0, where $k_0 = \frac{1}{2} I_1 (1 - R_s)$

$k_0 =$ __________ ft-lb-sec ^{2}.

b. Calculate energy inputs for each ω

using $E_s = k_0 \omega_1^2$

$E_{sA} = k_0 \omega_{1A}^2$ = ______________ ft-lb.

$E_{sB} = k_0 \omega_{1B}^2$ = ______________ ft-lb.

$E_{sC} = k_0 \omega_{1C}^2$ = ______________ ft-lb.

5. <u>TEST SPIN RATES TO COMPENSATE FOR GRAVITY AND INERTIA VARIANCES:</u>

a. Weight per appendage, W = ________ lb.

Height which appendage cg is raised, h = ________ ft.

ΣWh = ____________

b. Calculate spin rates using:

$$\omega_{1t} = k_1 \sqrt{\frac{2\,(E_s + \Sigma Wh)}{I_{it}\,(1 - R_t)}}$$

where k_1 = 9.55 RPM/SEC.

1. $$\omega_{1tA} = 9.55 \sqrt{\frac{2\,(E_{sA} + \Sigma Wh)}{I_{it}\,(1 - R_t)}}$$

ω_{1tA} = ____________ RPM

2. $$\omega_{1tB} = 9.55 \sqrt{\frac{2\,(E_{sB} + \Sigma Wh)}{I_{1t}\,(1 - R_t)}}$$

$$\boxed{\omega_{1tB} = \_\_\_\_\_\_\_\_\_\_\_\_ \text{ RPM}}$$

3. $$\omega_{1tC} = 9.55 \sqrt{\frac{2\,(E_{sc} + \Sigma\,Wh)}{I_{1t}\,(1 - R_t)}}$$

$$\boxed{\omega_{1tC} = \_\_\_\_\_\_\_\_\_\_\_\_ \text{ RPM}}$$

FIELD CHECK-OFF DOCUMENT

ATTACHMENT 6