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DESIGN TECHNIQUES FOR SMALL SCIENTIFIC SATELLITE STRUCTURES

by J. M. Madey and R. C. Baumann

Goddard Space Flight Center
Greenbelt, Md.

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ABSTRACT

This paper describes the design, development, integration, and testing of small scientific satellite structures in sufficient detail to serve as a design guide for the apprentice designer. It describes the many considerations and procedures involved from the initial concept phase through launch and post-flight analysis. To relate the general considerations and procedures to actual programs, data in the form of illustrations, specifications, checkoff lists, and other reference documents are provided for specific satellites.

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INTRODUCTION

This report concerns small scientific satellites, those 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, over large ones, are as follows.

1. The spacecraft can be tailored more easily to the needs of the experiment.
2. Electromagnetic interference can be minimized.
3. Because of fewer onboard experiments, the experimenters are more likely to obtain the orbit requested.
4. The orientation of the spacecraft should satisfy most of the experimenters.
5. The inherently less complex system is easier to integrate and test.
6. A much shorter lead time is required to launch a small satellite, thus enabling the experimenter to fly the latest experiments and gather more sophisticated, meaningful data.

Generally speaking, the physical dimensions of a satellite during launch are much smaller than in orbit. During launch, all booms, appendages, and antennas are either folded or retracted. Before separation from the launch vehicle, they are erected, unfolded, or released according to a well-planned operational sequence. Orbiting satellites can have extremely large dimensions as shown by the 130-foot galactic noise antenna in Figure 1.

The primary function of the satellite structure is to provide the scientific experiments with a suitable housing compatible in every respect to the experimentation it contains. The structure must be designed efficiently to assure that maximum weight is available for the scientific payload and supporting subsystems, yet rugged enough to withstand the environmental conditions it will encounter such as ground handling, test, launch, and space environmental conditions. Also the structure must be constructed in such a manner that the experimentation and the various subsystems may be readily installed, removed, inspected, and tested.

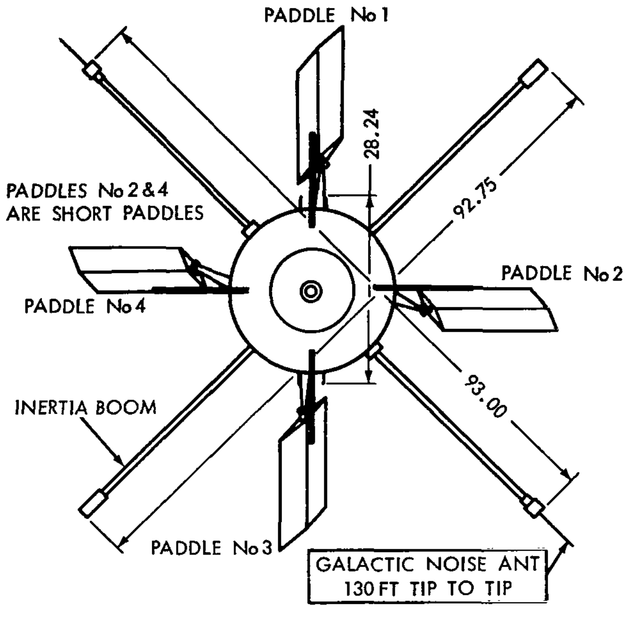

Figure 1—Top view of satellite orbital configuration (S-52).

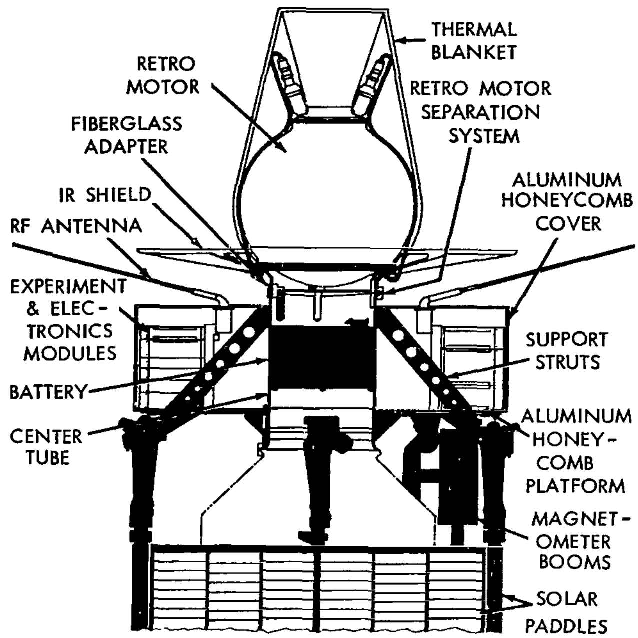

Figure 2—Complete spacecraft structure (AIMP cross-section).

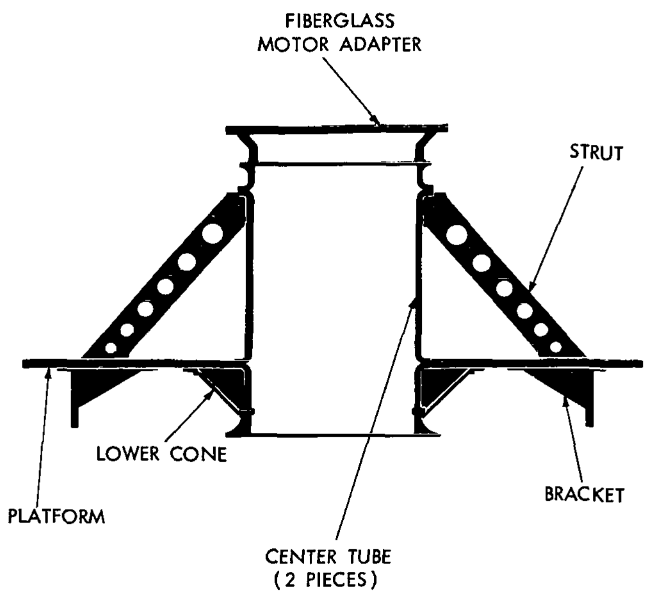

Figure 3—Principal satellite structure (AIMP).

A satellite structure is defined as a combination of members, beams, or plates held together by screws, rivets, or similar fasteners as illustrated by Figure 2. Hopefully, the structure has a high strength-to-weight ratio, is easy to design, and is inexpensive but can sustain all the rigors of powered flight as well as orbital environment. Since obtaining all of these features is rather complex, an attempt will be made to separate the principal structure from the secondary structure.

The principal structure (Figure 3) is one that supports the major portion of the weight or dynamic loads. This structure normally is 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 used to attach 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 relatively straightforward. Most small scientific satellites are of this type; however, some have tape recorders, scanners, stepping devices, and other mechanisms. These mechanisms and their design, fabrication, and test are excluded from this paper; however, when they are to be included in a satellite system, the structural designer must consider them carefully. In 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, the aerodynamic drag, magnetic damping, solar pressure, and other such forces must be considered in the analysis of the orbital attitude and spin decay. Analyzing the effects (Reference 1) of these types of forces is a separate subject and will not be considered in this paper.

Numerous factors comprise the foregoing generalizations. This document is intended to cover these factors in sufficient depth for the apprentice designer of small scientific satellites to be able to use the document as a design guide.

GENERAL DESIGN EVOLUTION

Long before the structural designer becomes engaged in even the feasibility design phase of a scientific satellite structure, decisions are being made by responsible scientists that result in the combining of scientifically compatible experiments into a single satellite (Figure 4). The experiment complement is normally chosen for the purpose of investigating closely interrelated space phenomena. When the experiment complement is determined, the project manager investigates the feasibility of combining all of the experiments into a single, integrated systems design. Here the satellite structural designer becomes involved in the development of the satellite.

First, feasibility design and layouts are made within the various constraints. These designs usually result in pinpointing, at a very early stage of development, incompatibilities between experimenter requirements and what can actually be accomplished within the various constraints. Compromise is usually the answer to the problems facing the designer. Tradeoffs are proposed and negotiated until an acceptable spacecraft design can be realized.

The structural designer then begins the preliminary design phase in which he performs detailed analyses in the many areas in which he has responsibility. Again during this phase, 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, and 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. Also, all spacecraft interfaces are defined. By the end of this phase, the satellite design is usually "frozen."

Occasionally, the manufacture of the prototype and the manufacture of the flight unit satellite structures overlap during the last stages of structural development. However, this is only feasible when the basic design is reasonably firm, and the risk of major changes is small.

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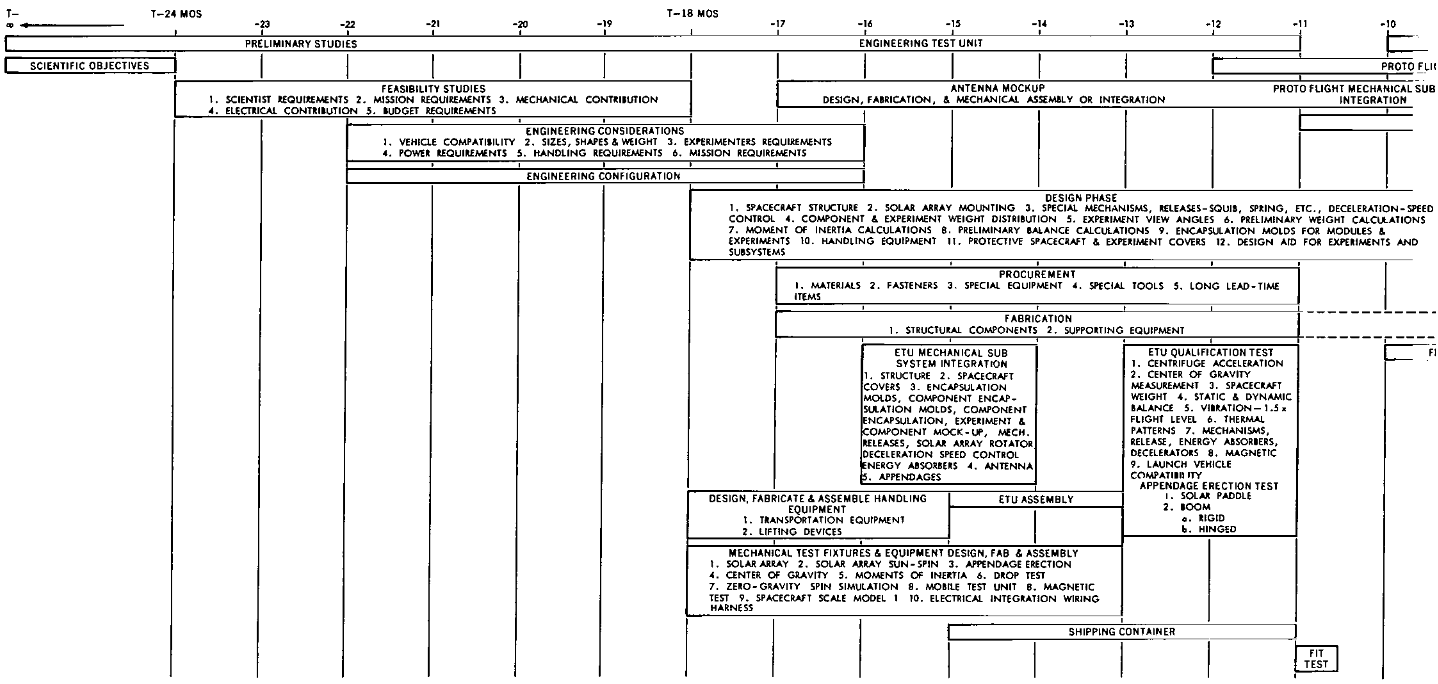

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SHIPPING CONTAINER
FIT TEST

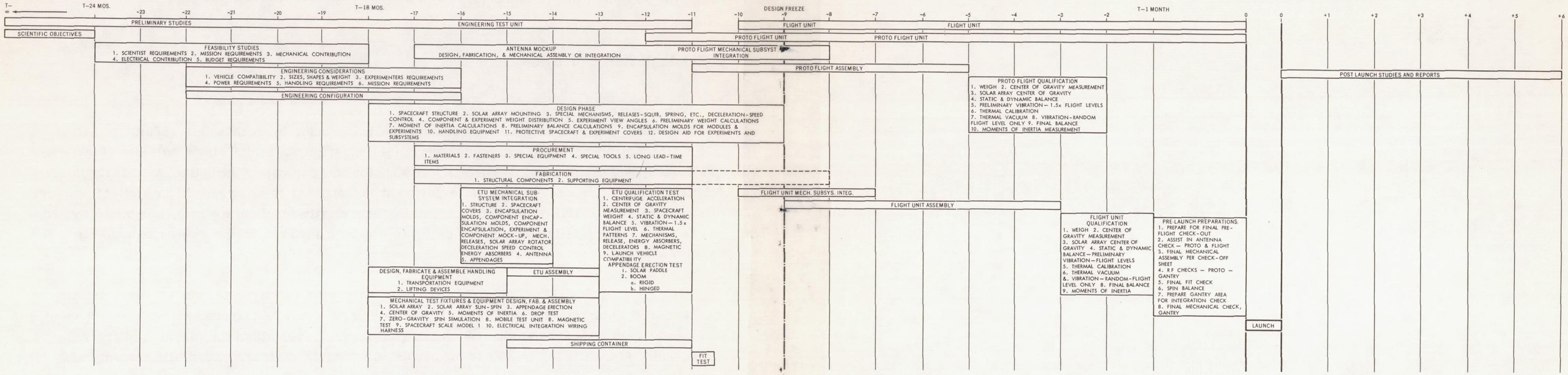

Figure 4—Typical time schedule of mechanical functions.

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DESIGN PRACTICES

The designer of small scientific satellites must consider all of the various factors that any good designer considers. In addition, he has one significant consideration that most other designers do not have, in that there are no opportunities to repair, maintain, or adjust the satellite after the launch vehicle leaves the ground. Thus, reliability becomes vitally important. Design simplicity generally assures system reliability. Remembering the mission of the structure or mechanism and using the simplest design approach will accomplish the desired end result.

Well-known design techniques should be used where possible since generally they have been perfected. The same rule 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; however, he can design a system in which failures can be minimized. For example, designing independent redundancy into a system increases reliability considerably. The designer must assume that if anything can go wrong, it will, and make design trade-offs accordingly.

For an example, assume a satellite with four appendages folded alongside the last stage motor, held in place by a Dacron or Nylon cord. To release these appendages, the cord must be cut. For independent redundancy, two separate timers and two separate power sources are wired separately into two guillotine cutters. Assume further that this redundant system may also fail and 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 test levels which have been specified for a specific vehicle. The criteria used at the Goddard Space Flight Center is to test dynamically the prototype structure at 1.5 times the flight levels. This approach assures a safety factor of at least 1.5.

To help increase reliability, order all fasteners, mechanisms, and materials in accordance with an accepted and approved specification. Avoid using uncoated aluminum screws in holes tapped in aluminum since 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 such as fiberglass, epoxies, magnesium, and, at times, aluminum.

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

Sharp corners should be avoided like a plague. Most failures in satellite structures have occurred because a generous radius was not provided in some critical, highly stressed area, such as a sharp corner. Also, the designer should avoid designing a structure with flat head screws because most structural sections are too thin to properly accommodate a flat head screw. Therefore, 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 condition causes the inner tapered surface of the screw to make contact with one side of the countersunk hole, causing the screw head to bend and, frequently, break. The designer should provide a table (Appendix A) of recommended torques for every screw used on the satellite in order to be consistent in assembling the spacecraft. This should assure maximum reliability of joints in the event that disassembly in the field should become necessary.

DESIGN CONSIDERATIONS

General

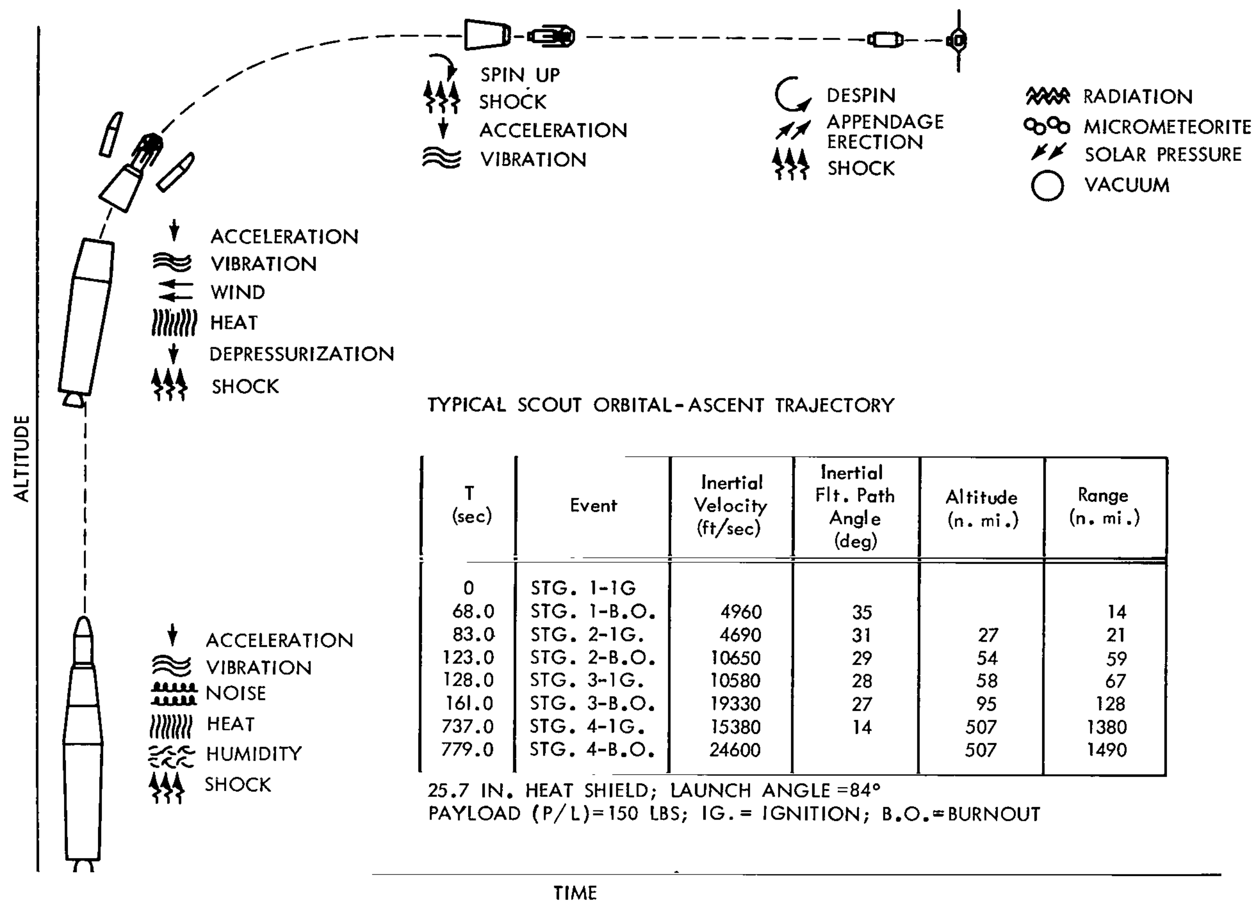
One of the biggest mistakes made by machinists and metal fabricators is the lack of attention given to fabrication techniques. Although most of the engineering analysis and design is done on paper, the fabrication shops can possibly provide satellite parts that are inferior or weaker than the calculated figures even though the points are machined, welded, or riveted in accordance with design drawings. This situation normally is 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. These are only a few—but important—areas that should be considered. It is important to check fabrication facilities; any technique that is questionable should be improved, and the importance and the reasons for the changes should be explained. Educating fabrication personnel is as important as educating the designer.

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

The loads imposed on the satellite structure primarily are caused by the launch phase environment as illustrated by Figure 5. This environment exposes the satellite to shock, vibration, acceleration, angular acceleration, noise, centrifugal forces, and possible aerodynamic heating. After injection into orbit, the structure and satellite are exposed to extremely severe vacuum, temperatures, possible radiation exposure, and micrometeroid damage. Since most of the small satellites are designed for 1-year life, all of these parameters must be investigated to assure little or no degradation of materials or mechanisms within the satellite for its intended lifetime.

The basic design considerations previously discussed provide a general outline of the primary areas which the structural engineer must consider. Initially, the engineer must formulate an approach and establish in his mind how the task will be undertaken. Then, he must visualize the configuration that will eventually evolve as the design of the various subsystems becomes finalized.

The question raised by many structural engineers is whether the satellite is designed to withstand environment or 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



T (sec)	Event	Inertial Velocity (ft/sec)	Inertial Flt. Path Angle (deg)	Altitude (n. mi.)	Range (n. mi.)
0	STG. 1-1G				
68.0	STG. 1-B.O.	4960	35		14
83.0	STG. 2-1G.	4690	31	27	21
123.0	STG. 2-B.O.	10650	29	54	59
128.0	STG. 3-1G.	10580	28	58	67
161.0	STG. 3-B.O.	19330	27	95	128
737.0	STG. 4-1G.	15380	14	507	1380
779.0	STG. 4-B.O.	24600		507	1490

Figure 5—Launch and orbital environment.

generally more severe than the launch environment, but some level of confidence must be established. This is generally done by using whatever data are available and establishing flight test levels at the worst possible expected condition. These levels are then increased by a safety factor of 1.5 for both the engineering test unit and the prototype. If both units pass the dynamic tests, it is almost certain that the flight unit will pass the lower levels; however the environmental test does not guarantee a successful launch and long operating life. It discloses defects and establishes the flight-readiness of the flight unit. Since the test specification does not provide for thermal-vacuum testing the satellite for more than 2 weeks, the structural engineer must select his materials so that there will be no appreciable degradation during the satellite design life. Altogether, there are seven areas that must be considered; these are listed below.

1. Scientific experiments
2. Orbital environment
3. Launch environment
4. Prelaunch environment

5. Materials
6. Fabrication techniques
7. Testing

The scientific experiments generally will establish most of the orbital requirements. These requirements include the following.

1. Either a rotating or nonrotating satellite (if rotating, spin rate will be given)
2. Experiment orientation within the satellite
3. Orbit requirements (apogee, perigee, and inclination)
4. Aspect of satellite at launch
5. Use of nonmagnetic materials (magnetometer-type satellites)
6. Whether hermetically sealed or not
7. Maximum permissible coning angle (dynamic unbalance)

Since most of the scientific satellites designed by GSFC have been spin-stabilized, only the approach for designing this type of spacecraft will be presented. Also, the step-by-step approach by which the satellite is conceived, designed, assembled, tested, shipped to the launch facility, and finally launched will be described.

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

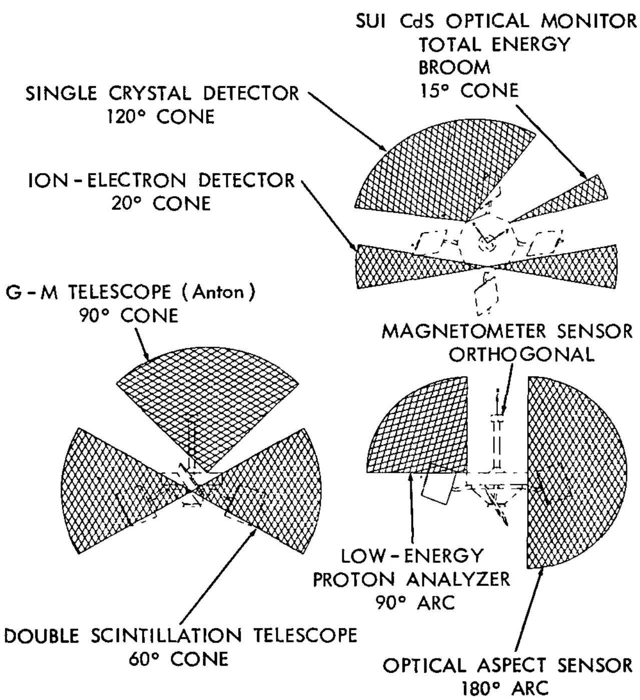


Figure 6—Sensor look angles (Explorer XII).

1. Number of experiments and plus associated electronics
2. Scientific objective; e.g., to measure energetic particles in the Van Allen Belts, or ionosphere research
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 (see Figure 6)
8. Estimated weight of sensor and associated electronics
9. Physical dimensions of sensor and associated electronics
10. Orbit aspect

11. Need for a tape recorder or other specialized equipment

12. Physical size and quantity of batteries

13. Special requirements (e.g., an experiment on a boom with a specified minimum distance from center of gravity, a requirement for a kick motor or retromotor, nutation dampers, or an attitude control system)

    Note: For a more complete set of information requirements see Appendix B, Mechanical Interface Requirements.

Schedule Preparation

One of the first and most important steps to be taken is the preparation of a realistic schedule (Figure 7). The schedule should cover all the important milestones from design conception to

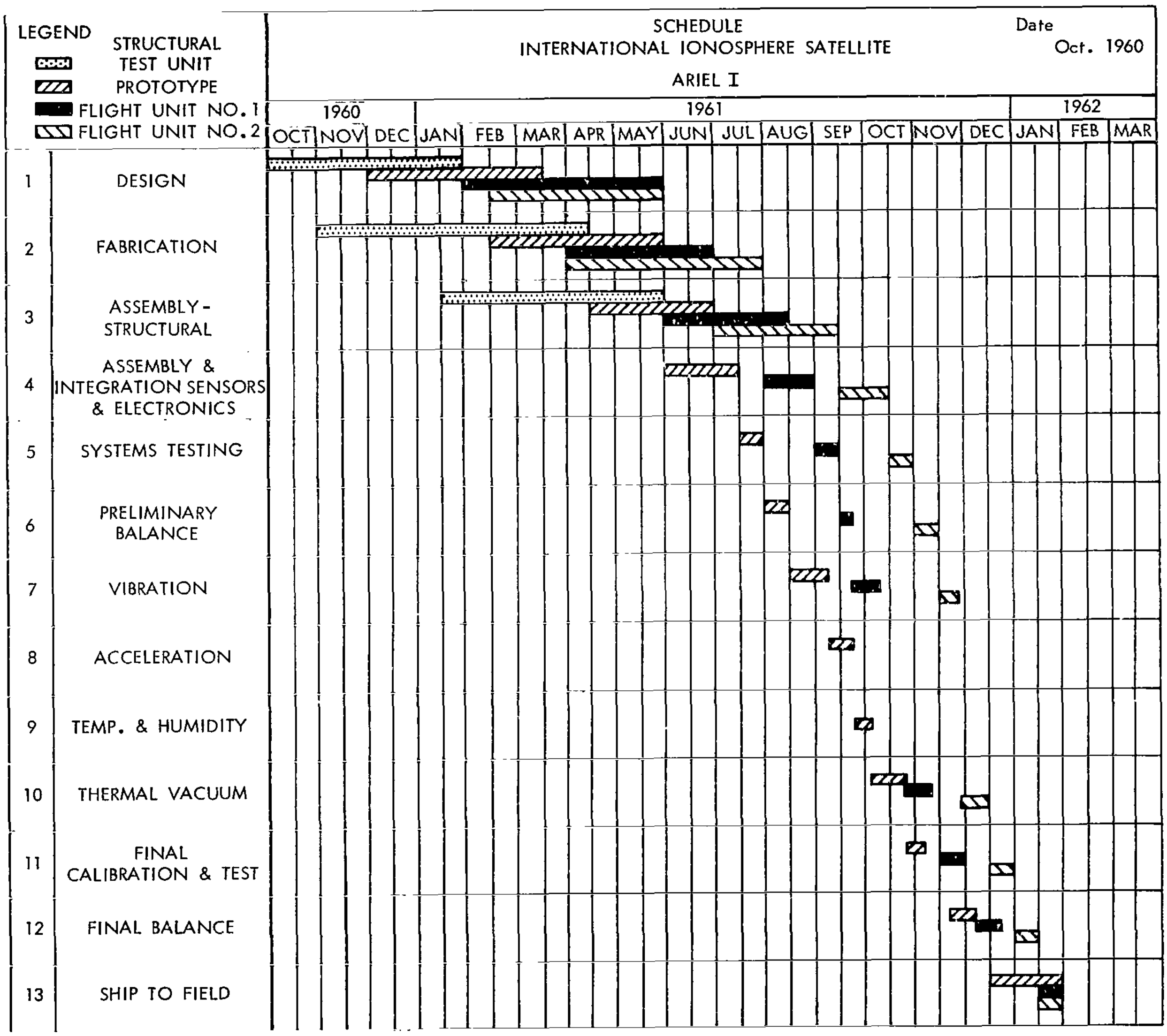
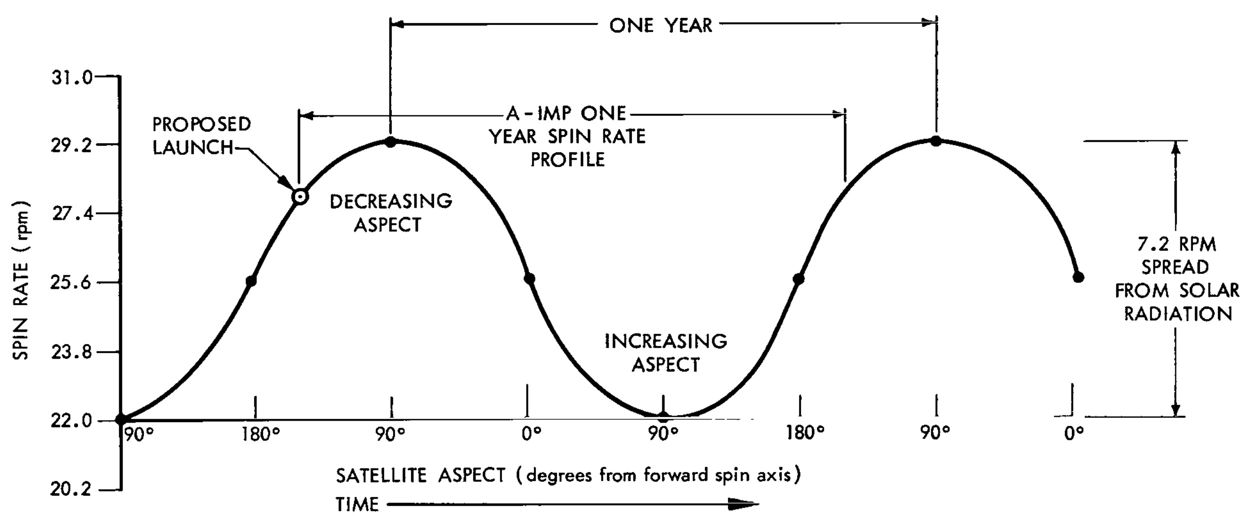


Figure 7—Planned program schedule (Ariel I).

shipment. Included should be all the items for which the structural engineer is responsible. By including all of these, one can tell at a glance how the flight unit is affected if the prototype schedule begins to slip. The schedule may also be an excellent reminder for ordering long lead-time items, for planning manpower requirements, and for showing how the structural engineer interfaces both in time and in function, with the rest of the satellite team.

Determining the Basic Shape

From the power and orbital requirements, the structural engineer can determine whether the satellite has adequate area for attaching solar cells or whether solar paddles must be employed. Generally, if the subsystems and sensors have been designed with the use of microelectronic components, the satellite will have sufficient experiments aboard to require more power than could be obtained from a satellite whose surface is covered by solar cells. This situation requires solar-cell paddles. One advantage of solar paddles is that generally the paddle area can be increased without requiring a major redesign; however, a surface-covered satellite would require a complete redesign or elimination of some sensor. If paddles are employed, remember to calculate the effect on spin as a result of solar pressure (Figure 8 and Reference 1). It is always wise to design a satellite with the thought in mind that power requirements will increase. One factor that may influence the decision for selecting a particular shape is that a sphere is the easiest configuration for calculating thermal coatings and temperature gradients.



NOTE: Above based on approx. 1° per day aspect change

Figure 8—Spin rate versus time (result of solar pressure) for IMP.

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 (Appendix B) 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 fastening components to the structure, etc.), is anticipated. The reason for this is that generally the experimenters and scientists may not have as good a mechanical background as the structural engineer. Also controlling and minimizing the mechanical interface between the experiment and the satellite 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. The structural engineer may expend a large effort in trying to rework most of the spare experiments and electronic components to fit within the flight spacecraft. Appendix A is a document that goes one step further, in that it requires the experimenters and subsystems designers to fit their experiments and circuits within 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 outweigh 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 computes the mass moments of inertia (MOI) about three mutually perpendicular axes. The spin axis is designated Z axis, and the two lateral axes as the X and Y axes. 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 the practice at Goddard Space Flight Center to design spin-stabilized spacecraft with the spin MOI a minimum of 5 percent greater than the principal lateral axis MOI. The 5-percent 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 is, the greater is the chance of the satellite becoming unstable; therefore a minimum spin rate should be selected which will assure stability for the life of the satellite.

Accuracy of measurements—The method used for measuring the MOI should have an error less than 2 percent and a minimum of three reasonably spaced, transverse MOI should be measured and plotted to assure 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 an analysis made so that the recommended 5 percent difference in MOIs is maintained.

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

For a satellite that is flat and large in diameter with three or more equallyspaced 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; however, for sperical or cylindrical satellites without appendages, the ratio could be less than unity. Therefore, it is very important to calculate and recalculate the MOI every time changes are made. If the experimenters' requirements are such that the MOI ratio is less than unity, the solution would be either to compromise with the experimenter by locating the experiment closer to the center of gravity (C.G.) or to attach weighted booms in the plane through the C.G. and perpendicular to the spin axis. Three or more booms should always be used since 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.

As an example, assume a cylindrical-shaped satellite (Figure 9) with a 5 slug-ft^2 MOI about 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 two weighted booms to axis X-X; these booms weigh 2 lb each, and their C.G. is 3 feet from the spin-axis. The MOI of the booms is equal to Mr^2 or

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

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

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

Examination of Figure 9 reveals that the same increase applies to both the Z-Z axis and the Y-Y axis. Therefore, the MOI ratio is still less than unity. However, consider the following

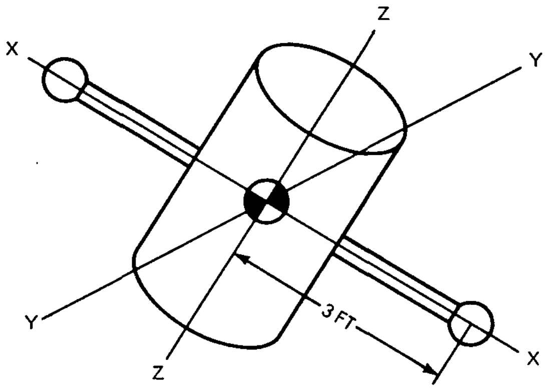


Figure 9—Model of cylindrical-shaped satellite with a 5.1-slug-ft^2 MOI about its spin axis.

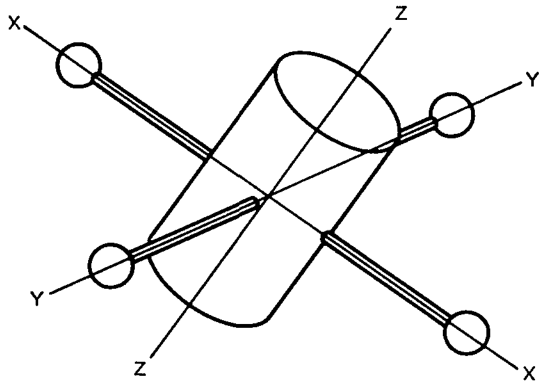


Figure 10—Same model as shown in Figure 9 with booms added to Y-Y axis.

examples when the foregoing case is simplified and two more identical booms are added to axis Y-Y (Figure 10):

1. MOI of booms only about Z-Z axis

$$
\begin{aligned}
MOI_{Z\text{-}Z} &= \frac{8}{32} \cdot 3^2 \\
&= \frac{9}{4} \\
&= 2.25 \text{ slug-ft}^2
\end{aligned}
$$

2. MOI of booms only about Y-Y axis

$$
\begin{aligned}
MOI_{Y\text{-}Y} &= \frac{4}{32} \cdot 3^2 \\
&= \frac{9}{8} \\
&= 1.125
\end{aligned}
$$

3. MOI of booms only about X-X axis

$$
\begin{aligned}
MOI_{X\text{-}X} &= \frac{4}{32} \cdot 3^2 \\
&= \frac{9}{8} \\
&= 1.125
\end{aligned}
$$

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

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

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

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

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

$MOI = 6.125$ slug-ft^2 total lateral

6. $\frac{7.25 - 6.125}{6.125} = 0.186$ or 18.6 percent

Adding four booms makes the spin MOI 18.6 percent higher than the lateral axes. Notice that the spin MOI was increased by 2.25 slug-ft^2, but the lateral axis by only one-half as much. If three booms are used, the spin MOI will increase approximately 1.75 times the lateral MOI increase.

GSFC has used the torsion pendulum method (Figure 11) 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 methods. The method, accurate to better than ±2 percent, uses a torsion rod designed for a period ranging from a 10 to 15 seconds, allowing for the spin axis in the orbital configuration (paddles, booms, etc., extended). The period is much faster for other configurations and axes. This time approach minimizes outside disturbances (torques caused by sudden air movement) as much 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 large 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 the fixture should always be attached to a rigid surface, preferably to part of the building; i.e., a

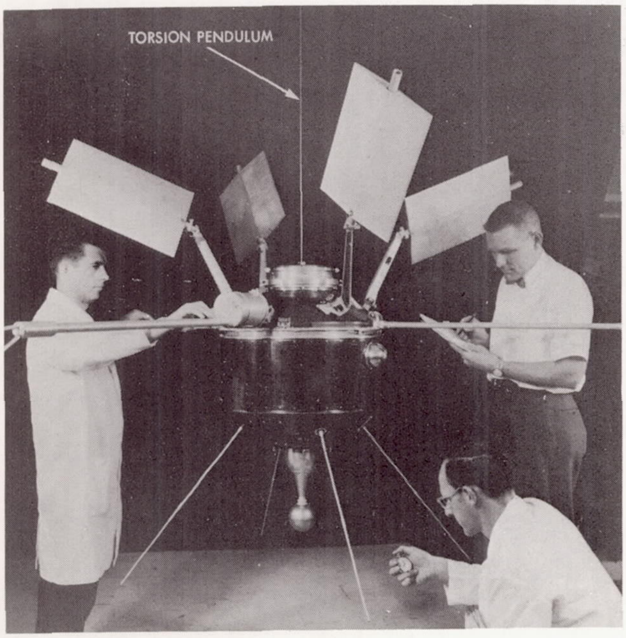


Figure 11—Moment of inertia determination, using the torsion pendulum method (Ariel I).

steel I-beam in the ceiling or some similar steel plate that is cemented in reenforced concrete. The fixture should not be attached to a ceiling or building that is subjected to vibrations; i.e., large machinery in the building or close by. Vibrations introduce additional measurement errors. The rod should be tested to at least four times the satellite weight. This can be done by either a tensile tester or suspended weight.

The equations for the design of the torsion rod are

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

$$k = \frac{JG}{L} \ ,$$

$$J = \frac{\pi d^4}{32} \ ,$$

$$M = \theta k \ ,$$

and

$$G = 0.4E \ ,$$

where

T = time of period in seconds,

I = moment of inertia of mass in lb-in.-sec ^{2},

L = length of rod in inches,

G = modulus of rigidity in psi,

J = polar moment of inertia of cross section of rod in (inches)^4,

E = modulus of elasticity in psi,

θ = twist of rod in radians,

M = twisting moment on rod in in.-lb,

K = spring constant of rod in in.-lb/rad., and

d = rod diameter in inches.

The following example illustrates the use of the equations:

assumed rod length = 30 inches

diameter = 0.218 inch

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

material = stainless steel $\quad E = 29 \times 10^6$ psi

θ(angular rotation) = 15° = 0.2618 radian

$$M = \theta K = \theta \frac{JG}{L} = \theta \frac{\pi d^4 (0.4E)}{32L}$$

$$M = 0.2618\pi \frac{(0.218)^4 (0.4)\left(29 \times 10^6\right)}{32 \times 30}$$

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

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

$$T = 2\pi \left(\frac{32IL}{0.4\pi E d^4}\right)^{1/2} = \frac{2\pi}{d^2} \left(\frac{80IL}{\pi E}\right)^{1/2}$$

$$T = \frac{2\pi}{(0.218)^2} \left(\frac{80(111.6)30}{\pi\left(29 \times 19^6\right)}\right)^{1/2} = 7.16 \text{ sec period.}$$

It is a simple matter to measure the MOI of a satellite with a torsion rod. All that is needed is a cylindrical homogeneous solid disk. If the weight and the radius of this disk are known, it is simple to calculate its MOI (Appendix C provides more detailed measurement), or MOI = 1/2 times its weight divided by g times the radius squared: MOI = 1/2 Mr^2. This MOI is stamped on the disk and recorded in the log book for ready reference.

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

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

Satellite-Launch Vehicle Compatibility

To provide satellite-launch vehicle compatibility, 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 have had several appendages that were folded parallel to, or along the side of, the last stage of the launch vehicle. These appendages must be designed to prevent interference with vehicle functions; i.e., they must not prevent or hinder proper separation of the preceding stage from the last stage. Also, they should fit within the fairing envelope.

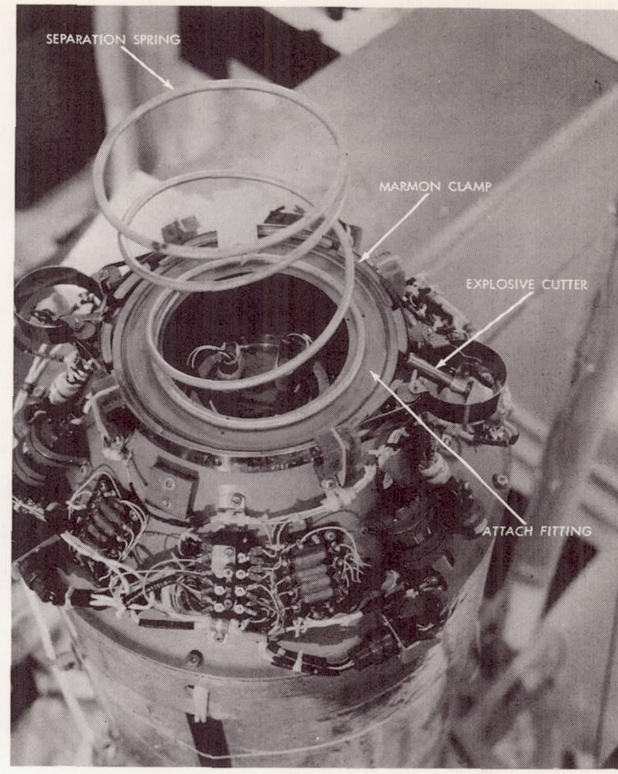


Figure 12—Delta vehicle/satellite attach fitting.

The satellite is affixed to the last stage of the launch vehicle by means of a Marmon-type clamp (Figure 12), which has two purposes. 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 torque properly the payload to the last stage and to provide separation. Clean separation is obtained by physically separating both halves of the clamp with large, flat springs (Figure 12). Relative velocity between the spacecraft and the last stage is obtained by means of a separation spring located between the payload and the last stage; it 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 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 removed easily and quickly with little or no degradation of all mating components. To make a satellite accessible, it must be designed and constructed with the least possible number of pieces. For example, if a subassembly, cover, or mechanism can be held in place with two screws, no more than two screws should be used to fasten it, even though it may not look safer and stronger.

Materials

Materials used for a structure should be easy to obtain, easily machined, homogeneous, consistent from one lot to another, a good thermal conductor; they should also have a very low vapor pressure and a high strength-to-weight ratio. It is very important that all the materials selected are compatible with, not only the structure, but also 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 or organic materials on precision instruments and electronic circuitry.

Outgassing of certain materials and metals also could cause failure of an experiment 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, hence, cadmium-plated materials should be avoided.

The materials most commonly used by the Goddard Space Flight Center on its small satellites are aluminum, magnesium, and fiberglass. Materials such as titanium and beryllium should be confined to special applications. If such materials are used for space applications, data pertaining to their use should be obtained from the manufacturers and other users.

Thermal Design Considerations

In designing a structure, consideration should be given to a very important area that is often overlooked—the area of thermal conductivity between mating surfaces. The structural engineer should determine, in close conjunction with the activity responsible for thermal control, the total power dissipation of each subsystem. The subsystems with the highest power dissipation should be given priority in bolting to a good heat sink. If calculations indicate that a particular subsystem has insufficient thermal paths, additional paths such as screws, rivets, or metal straps, made of a good thermal conductor such as aluminum, should be provided. Subsystem suppliers of battery packs, electromechanical timers, and other heat-generating subsystems are highly dependent on the structural engineer for assistance in designing containers and in 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; therefore, the structural engineer must have a good knowledge of the thermal properties of materials to be used in outer space.

Occasionally beryllium oxide or boron nitride washers are required to prevent certain electronic components from failing because of 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, a common source of trouble, RF leakage, must be considered since RF energy has a tendency to penetrate the satellite through wires, openings, and loose-fitting covers. To prevent this, all the covers must be designed carefully, eliminating all unnecessary openings and shielding the openings that are mandatory. Shielding, as well as the covers, 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. However, the use of aluminum structural components and anodized aluminum fasteners may cause electrical difficulties during electrical integration or testing. To alleviate this possibility, all anodizing should be removed from mating surfaces and, after assembly, a test for continuity should be made between all mating surfaces.

Structural Design Loads and Calculations

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 prelaunch 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 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.

Tables 1 through 5 contain the Delta and Scout launch vehicle vibration specifications that will dictate the loads to which the spacecraft will be designed and tested.

Table 1

Three-Stage Improved Delta (DSV-3E and 3F) Spacecraft Design Qualification, Sinusoidal Vibration.

Axis	Frequency (Hz)	Duration (min)	Level (g, 0-to-peak)	Sweep Rate
Thrust (Z-Z)	10-19	0.46	3.0*	2 octaves per minute
	19-25	0.20	4.5	
	25-250	1.66	3.0	
	250-400	0.34	4.5	
	400-2000	1.17	7.5	
		Total 3.83		
Lateral (X-X) and (Y-Y)	5-250	2.83	2.3*	2 octaves per minute
	250-400	0.35	3.0	
	400-2000	1.16	7.5	
		Total 4.34 (Each axis)		
Grand total: 12.51 min				

*When the specified accelerations cannot be attained because of armature displacement limitations, the input may be a constant displacement not less than 0.5 in. double amplitude.

Table 2

Three-Stage Improved Delta (DSV-3E and 3F) Spacecraft Design Qualification, Random Vibration.

Axis	Frequency (Hz)	PSD Level (g^2/Hz)	Acceleration (g-rms)	Duration
Thrust (Z-Z) Lateral (X-X) Lateral (Y-Y)	20-150 150-300 300-2000	0.023 0.045	9.23	4 minutes each axis
		Grand total: 12 minutes		

Table 3

Two-Stage Improved Delta (DSV-3G and 3H) Spacecraft Design Qualification, Sinusoidal Vibration.

Axis	Frequency (Hz)	Duration (min)	Level (g, 0-to-peak)	Sweep Rate
Thrust (Z-Z)	10-19	0.46	2.3*	2 octaves per minute
	19-25	0.20	3.8	
	25-150	1.30	2.3	
	150-500	0.87	0.923 in./sec constant velocity	
	500-2000	1.00	7.5	
		Total 3.83		
Lateral (X-X) and (Y-Y)	5-250	2.83	1.5*	2 octaves per minute
	250-400	0.35	3.0	
	400-2000	1.16	7.5	
		Total 4.34 (Each axis)		
		Grand total: 12.51 min		

*When the specified accelerations cannot be attained because of armature displacement limitations, the input may be a constant displacement not less than 0.5 in. double amplitude.

The specification levels are not the true criteria for determining the stresses that will be created by sinusoidal vibration; amplifications within the satellite at resonant frequencies are the predominant loads. It is not unusual to record an amplification or "Q" level of 20 at the resonant frequency of some structural member. Several years ago, the structural 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

Table 4

Scout FW-4 and X258 Launch Vehicles Spacecraft Design Qualification, Sinusoidal Vibration.

Axis	Frequency Range (Hz)	Level (0-to-peak)	Sweep Rate
Thrust Z-Z	10-53	12 in./sec constant velocity	2 octaves per minute
	53-100	±10.5 g	
	100-2000	±7.5 g	
Lateral X-X and Y-Y	5-150	±1.5 g	2 octaves per minute
	150-400	±3.0 g	
	400-2000	±7.5 g	

Table 5

Scout FW-4 and X258 Launch Vehicles Spacecraft Design Qualification, Random Vibration.

Axis	Frequency Range (Hz)	PSD Level (g^2/Hz)	Acceleration (g-rms)	Duration
Thrust Z-Z and Lateral X-X and Y-Y	20-2000	0.07	11.8	4 minutes each axis

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 1-second duration, but there were two 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-Hz thrust frequency lasting 6 seconds, and the other was a lateral 9.4-Hz frequency for approximately 1 second. Based on these data, it would be prudent to design a spacecraft so that its major resonances are not near these frequencies. The structure can be compared to a multi-spring-mass system (Figure 13). This system is usually too complex to analyze; therefore, reliance is placed on past experience.

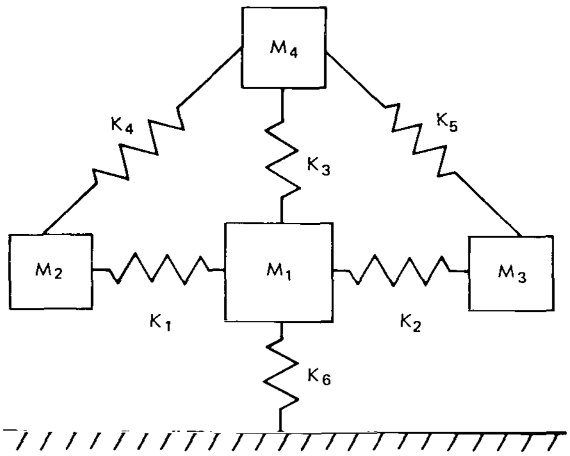


Figure 13—Multi-spring-mass system, schematic diagram.

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 satellite structure is more complex 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.

Small Scientific Satellites designed by the Goddard Space Flight Center have had a thrust axis resonant frequency varying between 50 to 110 Hz. In the lateral axes, the resonant frequencies have ranged between 9 and 55 Hz (see Table 6).

At resonance, the "Q" level, or amplification, in certain structural parts can build up to several orders of magnitude above the input; therefore this amplification must be minimized. This is

Table 6

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

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

*Undesirable; same as third vehicle mode frequency.

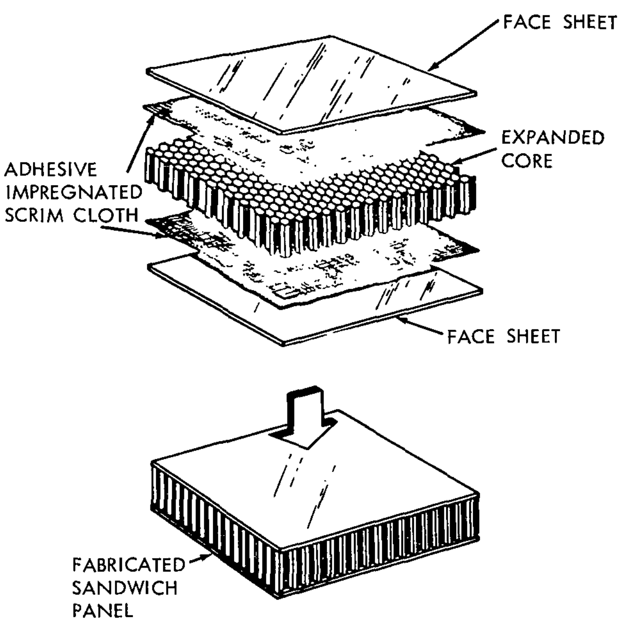


Figure 14—Honeycomb material.

generally done by using dissimilar materials, friction devices, or special rubber compounds. Some of the points that may help to keep the "Q" level from exceeding 7 include the following.

1. Use honeycomb material wherever possible (Figure 14).
2. Use fiberglass internally where possible (fiberglass is difficult to coat thermally).
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 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. However, a loose structure should be avoided since it will create banging and thermal problems).
6. Encapsulate all subsystems.
7. Design spacecraft so that the C.G. is as close as possible to spacecraft-vehicle interface.

Structural Resonant Frequency

Before any loads can be calculated, the resonant frequency of the spacecraft or structural component must be established, either by calculations or by testing. If the structure or component is too complex for calculating the resonant frequency, then a simple static load test for determining the spring constant (load versus deflection) would provide a simple method for determining the natural frequency; i.e., $F_n = 1/2\pi\sqrt{K/m}$. This equation is valid for both the thrust and lateral resonant frequencies as long as the following assumptions are made:

1. No appreciable damping
2. Yield strength of structure not exceeded
3. Slope of load vs deflection is nearly constant.

In the foregoing equation

m = mass supported by structure (W/g) (lb-sec ^{2}/in.).

K = lateral spring wt. (lb/in.).

F_n = Hz.

Calculation of Structural Loads

Vibration

To calculate the structural loads, assume that, as a result of a static loading test and the foregoing equation, F_n = 80 Hz. Examination of the Delta (DSV-3E and 3F) specification reveals that the highest g-level in the thrust axis for frequencies between 25 and 250 Hz is 3.0 g's. Multiplying this value 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 earlier) provides the level to be used for calculating thrust loads. The same approach is used in designing the structure for lateral loads. As a simple example, assume a 100-lb payload (Figure 15) that will be launched on the Delta vehicle and therefore must be designed to pass the sinusoidal vibration specification.

1. To calculate thrust in the Z-Z axis, let

$$\begin{aligned} F &= Q\,g\,W \\ &= 7(3.0)\,(100) \\ &= 2100 \text{ pounds}, \end{aligned}$$

where

Q = 7 (estimated value),

g = 3.0 (from thrust specification for frequencies of 25 to 250 Hz), and

W = 100 lb (total weight of satellite).

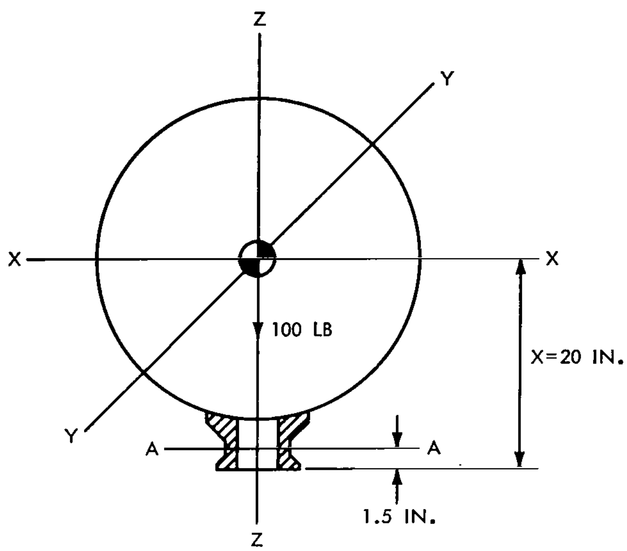


Figure 15—Model for calculating structural loads of 100-pound payload launched by Delta vehicle.

2. To calculate the area through section A-A (Figure 15) required to pass the thrust vibration levels, let

$$A = F/S = 2100/35{,}000 = 0.060 \text{ sq in.},$$

where

A = area of lower cylinder at section A-A, and

F = tensile or compressive load caused by sinusoidal vibration,

S = yield stress of material, assumed at 35,000 psi for aluminum.

3. To calculate the moment and the section modulus required at the base in the lateral or bending mode (X-X or Y-Y axis), assume a lateral natural frequency of 40 Hz and let

$$M = QgWX = 7(2.3)(100)(20) = 32{,}200 \text{ in.-lb},$$

where

M = moment in in.-lb,

Q = 7 (estimated value),

g = 2.3 (from lateral axis specification for 5 to 250 Hz),

W = 100 lb, and

X = 20 inches (distance from center of gravity to satellite base);

also

$$S = M/Z$$

$$35{,}000 = 32{,}200/Z$$

$$Z = 0.89 \text{ in}^3,$$

where

S = stress in psi and

Z = section modulus in in^3.

Knowing the area (A) and section modulus (2), it is possible to calculate the diameter of section A-A.

4. To calculate the stress through section A-A (18.5 in. below the center of gravity), let

$$S = M/Z$$

$$= 29,800 \text{ psi} ,$$

where

$$M = Q g W X_{A-A}$$

$$= 7(2.3)(100)(18.5) .$$

The foregoing example illustrates the method of determining the area and section modulus in the lower cylindrical section and the stresses through section A-A. However, this example represents a simple case; a spacecraft may contain booms, paddles, etc., which must be considered in the calculations, thus making the problem more complex.

Appendages and Yo-Yo

The appendages must be treated as independent pieces of the structure in calculating their stresses and natural frequencies. Also, appendage erection loads must be calculated and compared to the vibration loads; then the hardware must be designed accordingly. For example, assume that there are four appendages equally spaced, parallel to the spin axis before release, and perpendicular to the spin axis after release. When the appendages approach the fully erected position (assumed perpendicular to the spin axis), they possess kinetic energy equal to the difference between the kinetic energy before and after erection. Thus the kinetic energy in all four appendages = 1/2 $(I_i \omega_i^2 - I_f \omega_f^2)$ in ft-lb. Since there are four appendages, each appendage will possess one-fourth 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 that the appendage is a simple cantilever, the strain energy σ equation can be used to determine an equivalent static load P (Figure 16). Thus,

$$\sigma = \frac{P^2 L^3}{6EI} \qquad \text{also} \quad \sigma = 1/2\, PX^2$$

and

$$\delta = \frac{P L^3}{3EI} ,$$

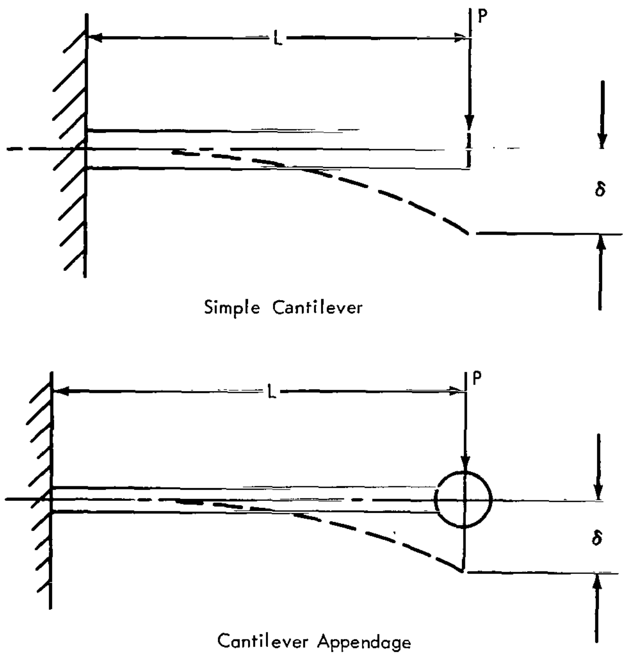


Figure 16—Equivalent cantilever static loads, schematic diagram.

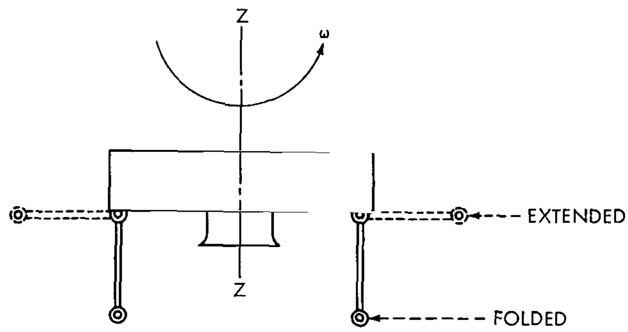


Figure 17—Satellite with four equally spaced appendages, schematic diagram.

where

σ = strain energy,

P = load (concentrated),

L = distance to load,

I = moment of inertia,

δ = deflection,

E = modulus of elasticity.

The only unknowns in the foregoing 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 that the system had before paddle erection and after paddle erection. As an example, using the schematic diagram in Figure 17, assume the following.

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

$$3.\ I_{\text{erected}} = 16.8 \text{ slug-ft}^2 \begin{pmatrix}\text{final}\\ \text{MOI}\end{pmatrix} . \tag{1}$$

Then, using the equation for conservation of angular momentum gives

$$I_i \omega_i = I_F \omega_F \quad \text{or}$$

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

$$\omega_F = 4.75 \text{ rad/sec (final spin rate) ,}$$

where

$$I_{p.f.} = I_i \ ,$$

$$I_{p.e.} = I_F \ ,$$

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

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

Solving for the difference in kinetic energy gives

$$E_K = 1/2 \left(I_i \omega_i^2 - I_F \omega_F^2 \right) \ , \quad \text{or}$$

$$E_K = 1/2 \left[5 \left(16^2 - 16.8 \right) 4.75^2 \right]$$

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

$$E_K = 450 \text{ ft-lb of kinetic energy in four appendages, or}$$

$$E_K = 112.5 \text{ ft/lb in one appendage.} \qquad (2)$$

Changing the units to 1350 in.-lb and substituting this number for σ in Equation 1 allows 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 also has been computed.

To evaluate the appendage for adequacy of design, obtain one appendage and attach it to the ETU structure. The experimental K constant can be obtained by the recording deflection versus the 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. With the experimental K, we can establish whether the appendage can withstand the 1350 in.-lb of kinetic energy computed in the preceding examples. To illustrate, consider a 60-lb load placed at the center of gravity of the appendage and a 0.8-in. deflection. Thus,

$$K = \frac{60}{0.8}$$

$$= 75 \text{ lb/in.}$$

The previously mentioned 1350 in.-lb kinetic energy will be transferred to potential energy (deflecting the boom X distance) by the equation P.E. = 1/2 KX^2. Therefore,

$$1350 = 1/2\,(75)\,(X^2)\ ,$$

$$X^2 = 36\ ,$$

and

$$X = 6 \text{ in. deflection}\ .$$

The force required to deflect X distance is given by

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

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

$$F = 75\,(6)$$

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

Thus, the appendage center of gravity should be loaded with a 450-lb static load. This load will test realistically the appendage to duplicate the 1350 in.-lb kinetic energy caused by appendage erection. This would also be an excellent apportunity to place strain gauges on specific critical areas of the appendage for the purpose of comparing calculations versus test data and locating possible high-stress areas.

The appendage is not a simple cantilever; therefore the foregoing equation is an approximation. However, if used it will enable the structural engineer to design an appendage in accordance with a realistic load, which is the appendage erection load. Energy dissipation in the foregoing example is assumed to be zero. When testing any appendage, the spring constant "K" must be the same as the flight "K" value.

Yo-Yo

The Scout and Delta vehicles depend on the spin of the last stage for the 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 percent. Most of the GSFC mission requirements of small, scientific satellites dictated despinning the satellite to some lower spin rate. To accomplish this requirement, the satellites were despun by means of a yo-yo mechanism.

The rigid yo-yo despin mechanism is essentially two small, identical weights attached to two separate, but equal, lengths of wire (Figure 18). These weighted wires are wrapped symmetrically around the satellite in the apposite direction to spin, and the weights are held in place by an electromechanically actuated device. At some preselected time, normally after last stage burnout or after separation, both release mechanisms are activated simultaneously by small electric pyrotechnics. Both weights then unravel in the same direction as satellite rotation. Both wires release simultaneously at a time when the wires are perpendicular to the satellite spin axis. Despin 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 despin a satellite to a zero-rpm condition and, if necessary, to spin up the satellite in the opposite direction. Spinup in the opposite direction requires a somewhat complex wire-release device.

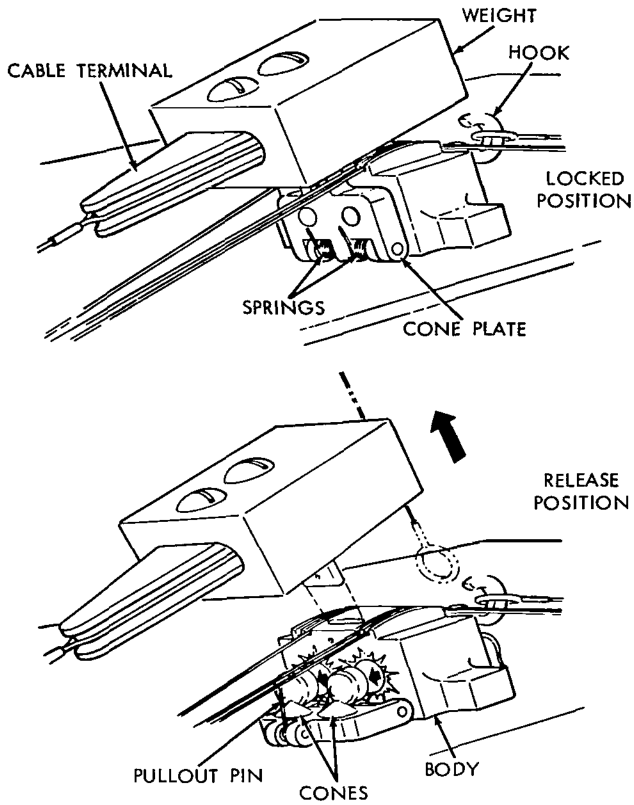


Figure 18—Despin cable-release device.

The final spin obtained is dependent on error (less than 1-1/2 percent in calculations and the tolerance of vehicle spinup. The vehicle spinup tolerance is ±10 percent. This same tolerance applies when calculating the final despin rpm. For example, if a satellite is to be despun to 10 rpm, the tolerance is ±1 rpm. Notice, that in designing for an rpm close to zero, the tolerance approaches zero.

If mission requirements are such that a ±10 percent tolerance is not acceptable, it is recommended that a despin system be designed that can at times provide a ±1 percent tolerance of the final spin rate. This system is called the stretch yo-yo and is similar in operation to the system discussed earlier, except that 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 despin system consult References 3, 4, and 5.

The total weight of the despin yo-yo system depends primarily on the wire length and satellite radius. If the system is designed with two complete turns of wire, the total weight should be less than 1 percent 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 weights not spaced diametrically opposite each other (180 degrees apart)
2. One weight slightly heavier
3. Weights and wires not released simultaneously
4. One wire slightly heavier or longer.

The further the yo-yo is located above or below the center of gravity, the greater the induced coning. Despin system design is accomplished by completing the equations on the Yo-Yo De-Spin Calculation Sheet (Radial Release) (see Figures 19 and 20). Some other important considerations that must be considered in despin design are:

1. Accounting for the inertias of all parts of the system if the satellite is to remain attached to the final rocket stage during despin
2. Proper accounting for despin caused by inertia changes which can be caused by appendage erection, gas depletion, and other such factors.

Linear Acceleration

Acceleration caused by rocket thrust has not been a problem in the past, but as new propellants are utilized, the thrust is increasing to such high levels that it is becoming the dictating factor in designing satellites under 125 pounds. 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 vehicles list a 3.0-g thrust axis level for a 25 to 250-Hz frequency. This means that the acceleration caused by thrust very likely would 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 vehicles, all of the examples will be based on this vehicle. As an example, consider 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 as follows.

$$\text{Acceleration} = \frac{1.5 \times 3000}{125 + 77} \text{ g's}$$

$$= 22.3 \text{ g's} ;$$

for the X258 solid-stage motor,

$$\text{Acceleration} = \frac{1.5 \times 6700}{125 + 77} \text{ g's}$$

$$= 50 \text{ g's} ,$$

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)

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

I = ____________ slug - ft^2 ω_0 = ____________ rad/sec

a = ____________ ft ω_f = ____________ rad/sec

ℓ = ____________ ft.

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

$$\omega_0^2 = \_\_\_\_\_\_\_\_\_\_ /sec^2;$$

$$F = 1.3\frac{m}{2}\omega_0^2\lambda = 1.3\frac{(\quad)}{2}(\quad)^2(\quad) = \_\_\_\_\_\_\_\_\_\_ \text{ lbs.}$$

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 $1\frac{1}{2}$ percent of the theoretically correct value.

Figure 19—Yo-yo despin calculation sheet.

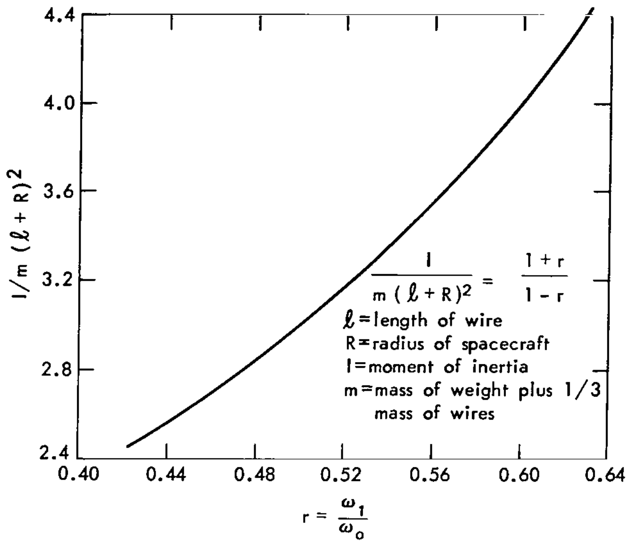


Figure 20—Yo-yo despin curve.

where

1.5 = safety factor (prototype level),

125 lb = satellite weight,

3000 lb = thrust of X248,

77 lb = expended weight of X248 and X258, and

6700 lb = thrust of X258.

Notice that the 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).

Angular Acceleration

Angular acceleration resulting from spin rockets has not been a problem, but it should not be ignored. Angular acceleration Figures are normally in a Vehicle Restraints Manual, which provides curves showing angular acceleration for spin rates versus moments of inertia.

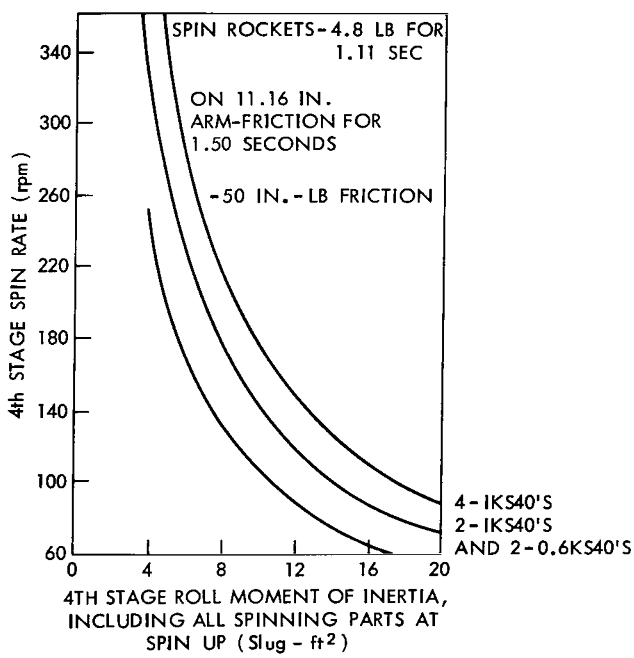


Figure 21—Spin rate versus fourth-stage moment of inertia (vacuum).

The spin environment for various fourth-stage total moments of inertia is shown in Figure 21 for vehicles using the cold (spring ejection) separation system. The spin rates shown in Figure 21 are those which will occur at fourth-stage ignition. As a result of the internal gas dynamics, the spin rate at fourth-stage burnout could be approximately 11 percent greater than at ignition.

A parallel effort should also be undertaken at the outset of stress calculations 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 (Appendix D). To calculate static unbalance, let

$$S = W\delta$$

where

S = static unbalance, in. lb.

W = payload weight, and lb.

δ = axis shift in.

For dynamic unbalance,

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

(For small angles, let $\tan\alpha = \alpha$) .

Therefore:

$$D = g\alpha(I_z - I_x) \qquad \text{(Plotted in Appendix D)}$$

where:

D = dynamic unbalance, (ft-lbs)

g = gravitational constant, (32.2 ft-sec^2)

α = principal axis tilt, (degrees)

I_x = moment of inertia (MOI), lateral, (slug-ft^2)

I_z = MOI spin axis (slug-ft^2).

Figure 22 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 spin axis with the vehicle 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.

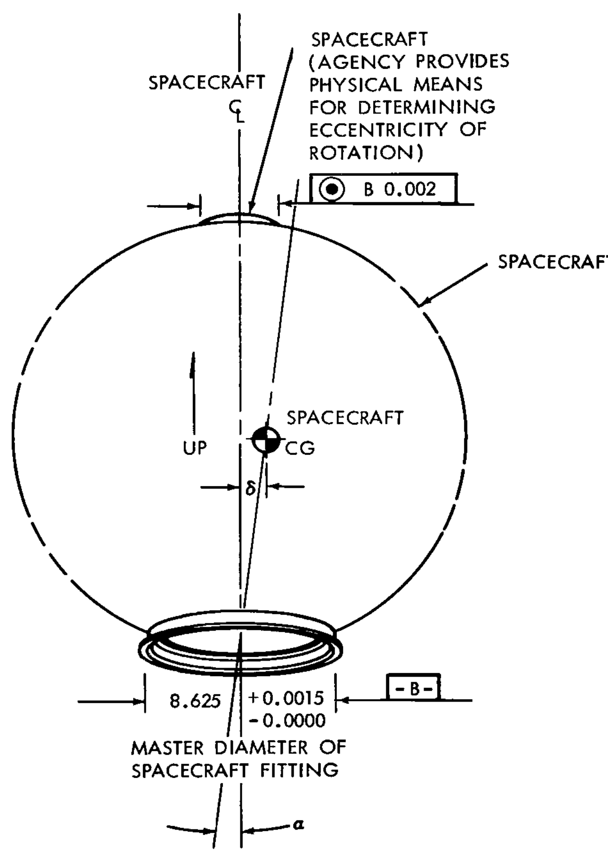


Figure 22—Balancing considerations.

The vehicle usually has a requirement for what is considered to be an acceptable maximum static and dynamic unbalance. The purpose of this requirement is to assure a good alignment of the thrust vector with the spin axis. On most of the GSFC-built small scientific satellites, the dynamic unbalance requirements dictated by the experimenters have been more stringent than the vehicle requirements. Therefore, based on the most critical requirement, the structural engineer must try to balance the spacecraft mathematically by shifting or interchanging electronic components and subsystems. However, the weights of all experiments and subsystems are primarily approximations. Although every effort is made to balance a satellite mathematically, actual balancing cannot be accomplished except by luck or by continued balance computations that continue until the design freeze. This may seem an unnecessary exercise for testing the engineers' mathematical capabilities; however, it is not, for this is one area in which the engineer can help reduce weight by avoiding the use of unnecessary counterweights. 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 designed to measure radiation or it could produce secondary particles.

Consideration should be given to 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. They should be located in areas that won't necessitate removing parts of the satellite during the balance operation.

It takes approximately 1 year to launch a satellite from the time that it is first assembled. Hence, 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 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 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; the satellite should also be covered with a protective cover during periods of idleness. The protective cover should be fabricated from material without 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 circuits within the satellite or ignite an electric squib, cutter, or dimple motor. These electric actuators perform a special task, e.g., yo-yo release. The same precautions should be undertaken with working clothes and assembly areas.

Antenna Pattern Mockup

An additional critical item that should be fabricated is an antenna pattern mockup (Figure 23) used to determine the effect of the satellite configuration on the antenna pattern. The exterior shape or configuration and the location of these antennas on the mockup determine the pattern. Therefore, the mockup can be welded or riveted sheet aluminum. It need not be a precise fabrication, but it should resemble the final expected shape; i.e., solar paddles and all appendages should be simulated properly.

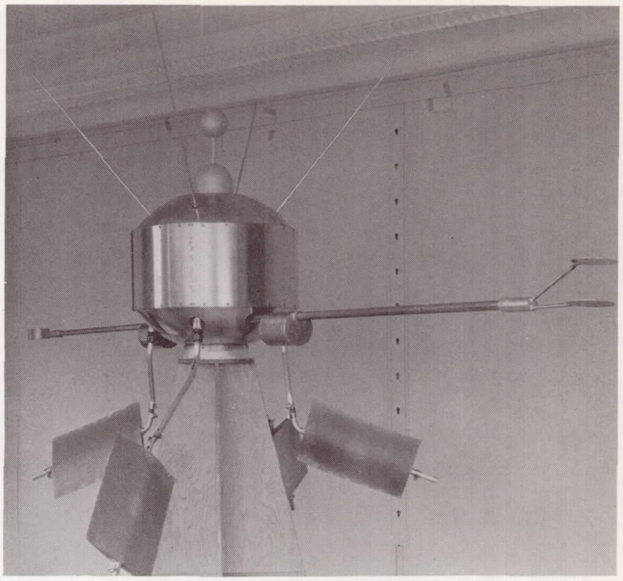


Figure 23—Antenna mockup (Ariel I).

Engineering Test Unit

After all computations and design drawings are completed, an engineering test unit (ETU) is fabricated and assembled (Figure 24). The primary purpose of this unit is to test thoroughly the satellite's structural integrity before ordering hardware for the prototype and flight units.

Fabrication

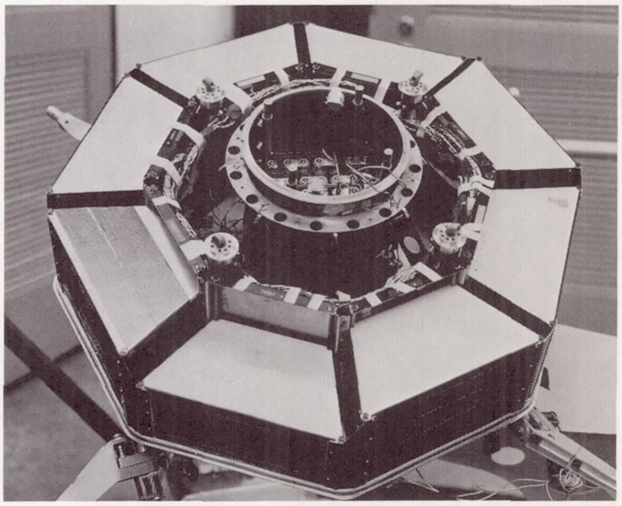


Figure 24—Engineering test unit (AIMP).

To evaluate the ETU properly, 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 percent more. This can be accomplished easily by bolting weighted wooden blocks to simulate all the subsystems. 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. These data are also helpful for testing (hard table vibration) of prototype and flight experiments at realistic Q's. It can also be used as a thermal model by installing resistors and thermistors in each card to determine the thermal temperature gradients.

Vibration Test

After the ETU is assembled and ready for testing, the prototype thrust vibration test is performed in accordance with Tables 1 through 5 as applicable. If any failures are detected during or upon completion of this test, the satellite should be removed from the vibration shaker, inspected, analyzed, and redesigned before proceeding further with the test. If no failures occur, the lateral vibration test then should be performed. If failures occur during this test the fault should be corrected and testing resumed. Random vibration generally follows the sinusoidal vibration combined with 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.

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; however, if they were, this could be a source of trouble.

Spin

The spin test, which follows the acceleration test, has never given the structural designers any trouble at the Goddard Space Flight Center. This results mainly from 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 has been no more than 12 inches from the spin axis. In checking the g level that these packages underwent, it was found that the level was 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, the following relationship should be used:

$$\text{force in g's} = \frac{r\omega^2}{g}$$

where,

r = 1 foot

ω = 18.9 rad/sec.

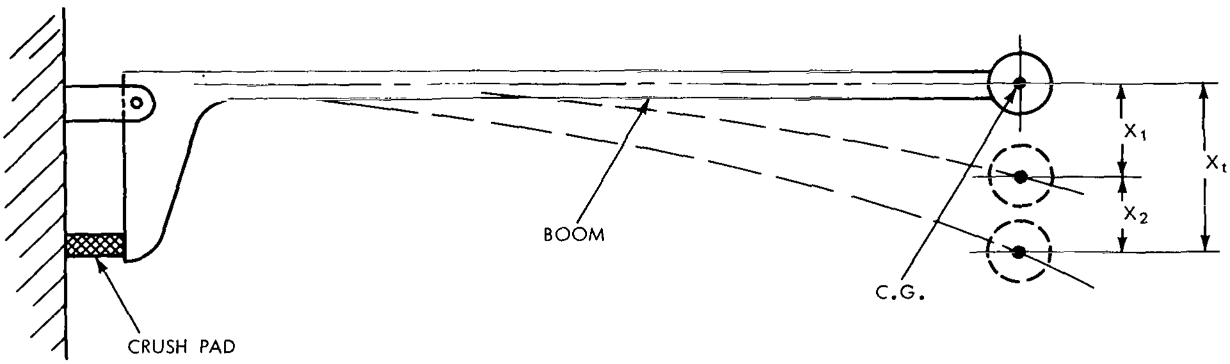
Therefore,

$$\frac{1}{32.2}(18.9)^2 = 11.16 \text{ g's} .$$

In the future, this force may become a problem, when satellite diameters become much larger. Also if the satellite were 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.

Appendage Erection

The last test, if applicable, is to check appendage erection. The theory used at the Goddard Space Flight Center is to assume a despin 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 despin sequence. If the satellite does not despin, 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 energy dissipator is based on the crushing or yielding of materials. For example, consider a simple appendage (Figure 25) with the mass concentrated at the extreme end; 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 the 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

Figure 25—Schematic diagram of a simple appendage showing its deflection excursions and crush pad.

If a despin failure occurred the appendage would erect at a higher spin rate. Since the appendant boom was designed to pass the lower spin rate, the additional kinetic energy in the appendage would cause structural failure if the crush pad were not included in this system.

To consider the preceding example quantitatively, assume

1. The appendage kinetic energy is equal to (E_K) 1800 in.-lb;
2. The spring constant K is equal to 75 lb/in. (from the previous example on the design and test of the 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 gives

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

$$1000 = 1/2\left(75X_1^2\right)$$

$$X_1^2 = 26.7$$

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

Up to this point, we have stored 1000 in.-lb of potential energy. We must dissipate the remaining 800 in.-lb of energy by allowing the boom to travel a distance of X_2. To solve for X_2, we must calculate the peak force F of the 1000 in.-lb stored energy, or

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

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

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

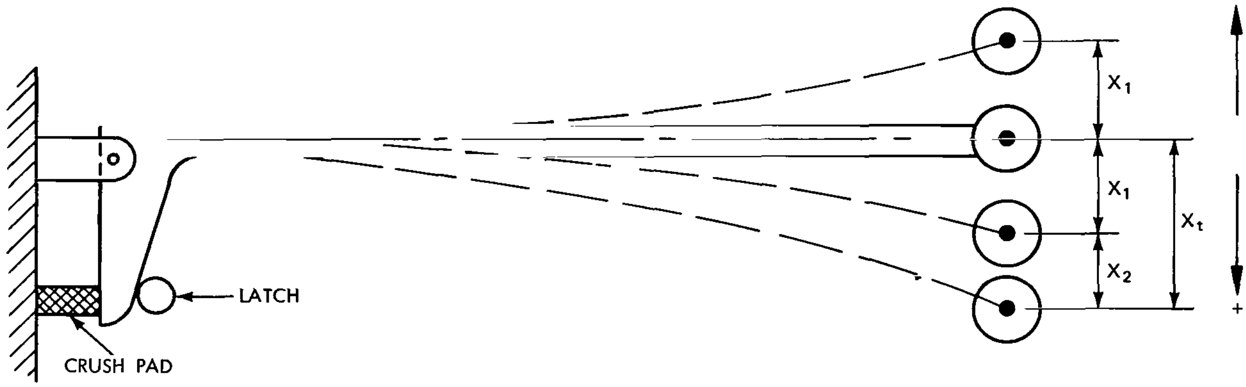
or $F = 388$ lb. At this point, the pad begins to yield. The distance X_2 it must yield is equal to the energy remaining divided by the peak force, or

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

$$= 2.06 \text{ in.}$$

Figure 26 shows the preceding calculations graphically. The area within the upper enclosure is equal to the energy dissipated, and the area within the lower enclosure is the rebound potential energy. Energy dissipation caused by 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 for some period of time depending on the damping factor or friction or possibly an additional energy dissipator. 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 that the boom had on the up-swing plus the energy generated by the earth's gravitational pull. High-speed photography will normally establish whether failure occurs on the up-swing or down-swing. Since this condition



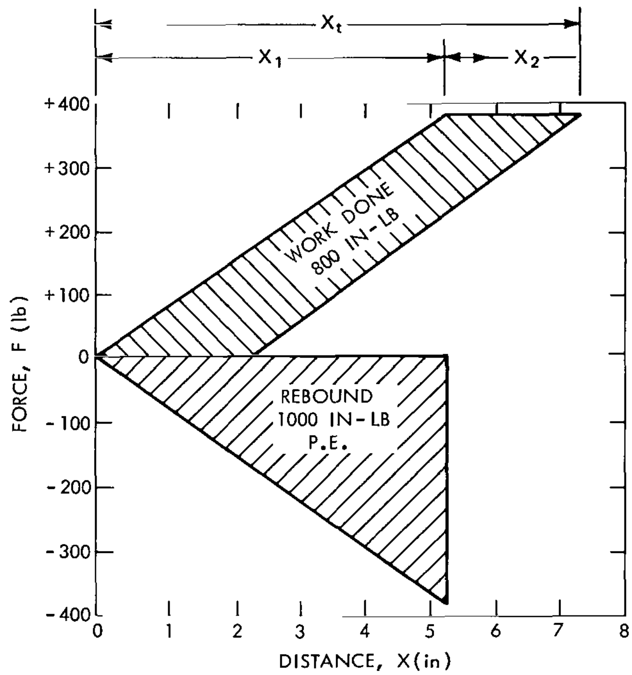


Figure 26—Schematic diagram and graphic presentation of a simple appendage after despin failure.

is peculiar to tests conducted on earth using a rigid fixture, the solution is to either test by free-falling the simulated satellite or, if the pictures definitely establish a failure on the down-swing, to determine the increase of energy caused by gravity and, if it is appreciable, to test the appendage under an equivalent static load as determined by the use of the previously mentioned equations. Realistic tests can be simulated by using a rigid fixture, provided that gravity is accounted for by overspin (see Reference 4 and Appendix E).

The foregoing 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; however, the calculations are valid, and the error is relatively small.

The crush pad is only one of many ways in which kinetic energy can be dissipated during the erection of an appendage. Some other methods that are just as effective are

1. Escapements
2. Friction brakes
3. Hydraulic dampers

Despin

Following the appendage erection tests, the ETU can be submitted to yo-yo or despin tests or 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 despin and appendage erection test, one can design a flat circular disk with adjustable weights for varying the moments of inertia. Attached to this disk is a shell or cover similar to the flight satellite cover with a duplicate, flight-expected despin system. The despin and appendage erection tests are performed in a large vacuum chamber under a free-fall condition. Remember that this is the ideal method. If the free-fall cannot be utilized, the error will be very small, usually less than 1 percent for the despin tests; however the error could be much larger for appendage erection tests made under atmospheric conditions. The aerodynamic drag could also be appreciable if the despin weights are physically large, and the cable long.

As a precaution, the despin wires, mechanisms, and the structure to which the mechanism is attached should be exposed to a pull test that is 1.5 times the force calculated on the form shown in Figure 19.

Prototype Unit

After the appendage and despin tests, the prototype structural hardware should be ordered. This unit will be an excellent indicator of the final outcome and configuration of the flight unit, since this will be the first time that all the experiments and subsystems will be mechanically integrated with the prototype structure. Since prototypes are sometimes launched for economy reasons or because of last-minute flight failures, the prototype hardware must be identical to the flight hardware and of flight quality. 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 percent. A log book is assigned to the prototype unit to record all the components and their respective serial numbers. The book records all events on a daily basis so that an accurate record is kept as to who worked on this unit, what was done to it, what problems were encountered, how the problems were resolved, and by whom were the problems resolved.

The first step is to begin the assembly of the main structure—usually a joint effort by the structural personnel and the electronic integration team so that the wiring harness can be installed early to prevent difficulty in trying to force-fit the harness. The subsystems and electronic components are then installed, carefully inspected, and examined for (1) hole alignment, (2) proper connector mating, (3) freedom from 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 checkout. 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 8 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 purpose of this operation is to prevent the vibration shaker and the spacecraft from becoming damaged during vibration testing by a force (couple) created by an excessive center of gravity shift from the geometric axis. In the lateral vibration mode, this same couple would induce torsional vibration. If the torsional frequency and bending frequency were the same, the resulting motion would expose the spacecraft to much higher stresses and probable failure.

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 do problems develop in the structure. Acceleration which follows does not usually pose any difficulties in any part of the spacecraft.

Temperature and humidity testing is a 1-week test that uncovers defects and weaknesses in electronic circuitry. The structure should pass this test without any difficulty.

Thermal vacuum testing is normally a 3-week test that may extend to more than 3 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 1 to 3-day operation, depending on the complexity of the satellite. The personnel performing the balance operation should be informed concerning the eventual location of the balance weights; when 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 it is shipped to the launch facility with the flight unit.

Flight Unit

The ideal time to begin assembling the flight unit is after the completion of the environmental testing of the prototype. But this is very seldom the case since many problems develop during prototype integration. 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 decreasing, so that more sensors and circuitry can be packaged in smaller volumes. This microminiaturization adds more complexity to existing problems, and the result is a longer time needed for correcting these problems. This situation causes schedule slippage, and one way to make up this loss is to begin assembling the flight unit before the completion of prototype tests. A flight unit log book is also assigned to this unit, and the same type of 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 normally disclosed by environmental testing of the flight unit are generally in the area of workmanship, e.g., poor solder joints, faulty connectors, defective fasteners, etc.

Thermal patterns are not finalized until the completion of flight unit thermal vacuum testing. Based on these results, the patterns can be corrected or changed as late as 2 weeks before launch. The reason for this is the short time left between the end of thermal vacuum testing 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

Before the satellite design freeze date, the task of designing and fabricating encapsulation molds for the experiments and subsystems should be undertaken. These molds prevent the experiment and subsystem frames or containers from buckling during encapsulation.

Encapsulation and conformal-coating of electronic and other components are usually done during and after completion of electronic integration since the cards and experiments must be removed many times during electronic integration 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 reassembled into the prototype to determine 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 semirigid epoxy approximately 2 mils thick. The coating 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 with a low-density material. This material is a closed-cell, polyurethane foam or similar substance, with densities varying between 2 pounds per cubic foot to over 20 pounds per cubic foot. The polyurethane foams used by the Goddard Space Flight Center are the Eccofoam FP and Eccofoam FPH (Reference 4). Eccofoam 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 capability of long exposure to high vacuum. As a note of caution, it must be remembered that rigid or non-resilient epoxies and insulation compounds are not recommended because the coefficient of expansion of these coatings is different than the coefficient of expansion of the electronic components coated. During thermal cycling, the rigid coating would stress and fracture some of the electronic components.

INTEGRATION AND TEST

The mechanical and electronic integration of the subsystems into the structure will begin to reveal discrepancies in the spacecraft such as interference of one experiment or its electronic circuit with another experiment, RF leakage into some other circuitry, or an additional requirement or change as a result of previously mentioned troubles. These problems may require structural changes. Most of these changes are minor, but occasionally some subsystem must be redesigned, thereby causing a structural design change. Therefore, it is desirable to delay production of the flight unit until the prototype has undergone complete environmental testing. This procedure represents 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; however 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 occur are fatigue failures resulting from repetitive testing. This is natural and should be expected; however, 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 the prototype structural members.

The problems associated with the flight unit are oriented strictly to subsystem defects. From here, the structural engineer concentrates mainly on excellence of workmanship and ensuring that the flight unit is assembled so that, upon completion of environmental testing, the unit will be shipped to the launch facility and launched. To ensure flight readiness, the structural engineer should have prepared a field checkoff list for the flight unit. The purpose of this list is to keep a running log of everything that has been removed and reinstalled 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 it 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 checkoff list for IMP-B is given in Appendix F.

SHIPPING

The shipping containers should be designed and fabricated before the ETU dynamic tests. It is recommended that these shipping containers be fabricated from either plywood or sheet aluminum. Aluminum is preferred because 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 Hz. 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 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, desiccant should be included inside the shipping container for absorbing moisture. A tight (not pressure-tight) container is beneficial; on an extremely humid day, there is very little chance that the interior will be drastically affected. However, satellites are normally packed indoors where the temperature is 70°F or more, and the humidity could be as high as 50 percent. However, when this container is exposed to a cold, outdoor temperature, condensation can form within the container. Thus, desiccant should be used at all times in sufficient quantities to prevent any moisture accumulation. The amount of desiccant used is determined by container volume and the manufacturer's instructions, plus some additional safety factors.

Exterior container size and mode of transportation are other worthwhile considerations. If, for example, the container is too large, it may not fit aboard an airplane. It is best to obtain dimensions of cargo space and the size of the opening to the cargo space. If a passenger airline cannot get this package through their doorway one may have to hire a special cargo plane for the sole purpose of shipping the 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 that 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 to witness any damage that may have been done to the spacecraft so that it can be analyzed immediately and corrected if it is decided that it could cause a failure.

FIELD OPERATIONS

Small scientific satellites normally are shipped to the field 3 to 5 weeks before launch. The last week of this time is devoted to vehicle/satellite operations. Prior to this, the satellite undergoes operational checks and calibration. If the satellite is very complex, then approximately 3 weeks would be required for operational checks and calibration; but if it is simple, 1 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 perform all the scheduled tasks on both units simultaneously.

The first few days are generally spent in operating both the prototype and flight unit. All voltages and currents are carefully checked and compared with 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 then are transported to an antenna range to test RF transmission—generally about a 1-day test.

The following day, the flight unit may undergo further checkout and calibration, and the prototype would be taken to the spin facility for determining last-stage and satellite compatibility; also 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). A final test is to determine if any RF interference exists on the gantry by turning the spacecraft on, transmitting a signal, and receiving and analyzing this signal approximately a mile distant. Completion of this test occurs approximately 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 ensure proper alignment of both the satellite and last stage axis so that the complete assembly is dynamically balanced. Misalignment would require more weight to balance the assembly. If the satellite has solar paddles, then assembly should be balanced using dummy-weighted paddles; the weight and center of gravity should be identical with the flight paddles. The reason for balancing the assembly with dummy paddles is that vehicle personnel must attach and remove lead weights in the general area of the solar paddles. These paddles therefore are 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 before fairing installation. After the fairing is installed, the only tasks remaining are installation of the turn-on plug and removal of a cord to release the antennas, so that they may rest on the inner surface of the fairing. When these items have been completed, the satellite is ready for launch. Appendix F, a field operations checkoff list, provides an insight into the field operations.

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 in accordance with 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 caused by premature paddle and boom deployment. The Explorer XV experienced yo-yo despin 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 deployment. 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 despin failure on Explorer XV was never really established. The complete despin system was tested to try to duplicate this failure, but without success. The end result was redundant wiring in the electrical system and an improved despin weight-release mechanism. The low apogee of Explorer XVIII and Explorer XXI was attributed to subperformance of the Delta vehicle third stage.

These are examples of problems that required investigation, 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; he must carefully analyze any deviations in operational sequence and determine if these 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 experience; experience is the best teacher. The information presented in this paper is based on the experience gained from several of Goddard Space Flight Center's small scientific satellites. The information presented is not complete, but it will provide valuable assistance to satellite structural engineers.

The paper provides a step-by-step approach which has been used successfully on small scientific satellite structures from inception to launch and orbit. Included have been such topics as shape determination, design loads for structural members and 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. The paper includes sample calculations to aid the engineer in designing and testing appendages, moment-of-inertia fixtures, yo-yo despin systems, dynamic loads, and section modulus. It also provides recommended procedures for handling and shipping the satellite to the launch facility.

ACKNOWLEDGMENTS

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Goddard Space Flight Center
National Aeronautics and Space Administration
Greenbelt, Maryland, March 22, 1966
861-51-75-01-51

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Appendix A

Fastener Torque Value Reference

Table A1 lists 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 are used on all AIMP hardware to equalize properly the loads throughout the spacecraft, unless exception is taken by the cognizant engineer.

Table A1

Fastener Torque Values in In.-lb for AIMP Spacecraft.*

Bolt Size	18-8 and 300 Series SST	Material				Tolerance
		Brass	Phosphor Bronze	Aluminum 2024-T4	Magnesium ZK60-T5	
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

*Choose the smaller torque value for any combination of bolt and insert/fastener; for threaded inserts (helicoils, etc.), compare screw and insert materials. For example,

#4-40 Al screw in phosphor 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 phosphor bronze helicoil in magnesium = 4.3 in.-lb.

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Appendix B

Mechanical Interface Requirements (IMP-F, -G)

Dimensions of Body-Mounted Electronics and Experiments

Dimensions are given in length, depth, and height using the sample body illustration shown in Figure B1. The basic electronics module and experiment package mounted on the octagonal platform in the main body of the spacecraft shall be a trapezoidal-shaped card as shown in Figure B2. The height shall be not less than 0.9375 in. and not more than 9.000 in. Any height greater than 0.9375 must be approved by GSFC. If an experiment is of such dimensions that it cannot fit within the dimensions shown in Figure B2, the experimenter should plan to extend the package through the back of the card as shown in Figure B3. For those requiring a smaller volume than that shown in Figure B2, a half card is available as shown in Figure B4.

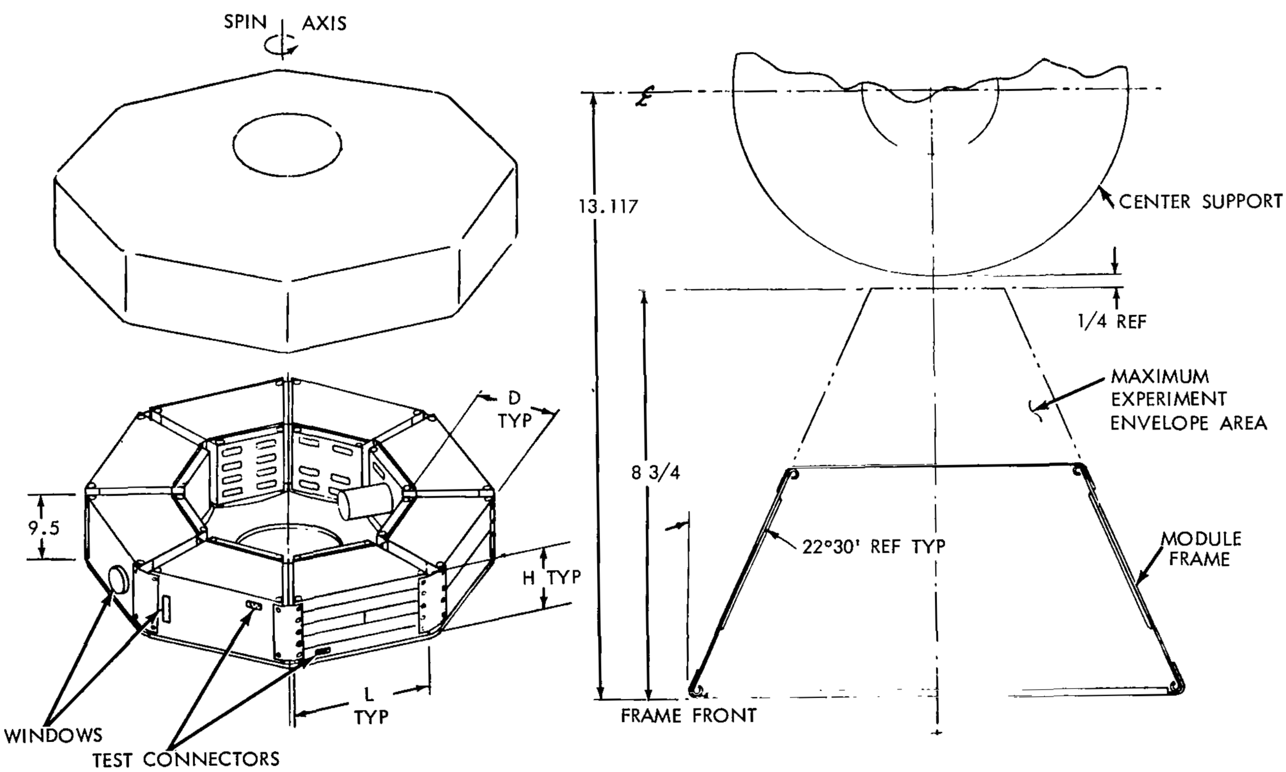


Figure B1—Dimensional outline of IMP F and G.

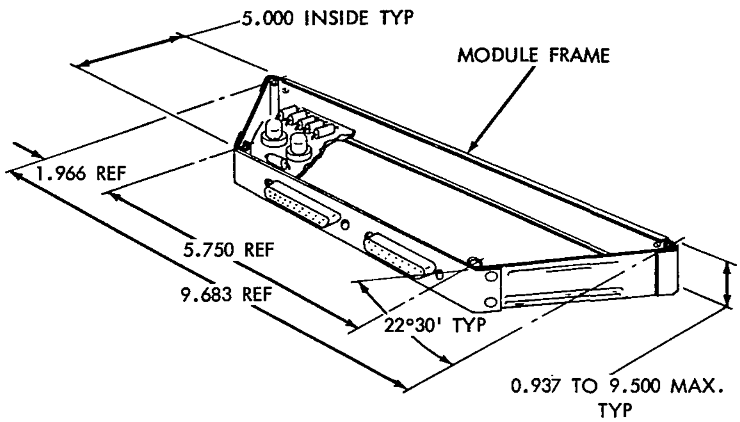


Figure B2—Basic electronics module.

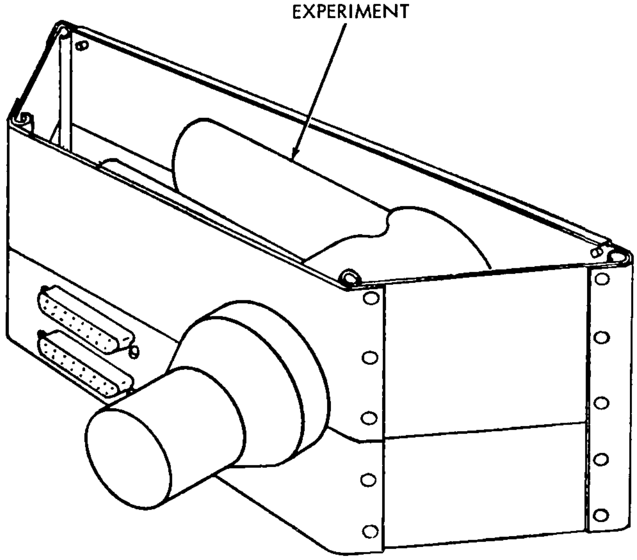


Figure B3—Electronics module with protruding, oversize component.

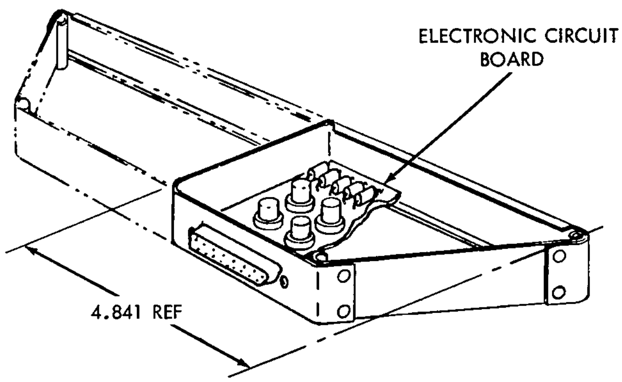


Figure B4—Electronics module containing half-size card.

Connector Location

All main harness connectors shall be oriented horizontally as illustrated in Figures B1, B2, B3, and B4. All test connectors must be accessible from the front of each package and be provided with plastic dust covers. Connectors must also be flush with the front face of the module card.

Weight

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

Windows and Experiment Look Angles for Body-Mounted Experiments

Windows (openings) can be located on the top and bottom of the octagon. All windows located on the facets of the octagon shall be located perpendicular to the spin axis as shown in Figure B1. Look angles for the body-mounted experiment should be submitted to GSFC for approval. The experiment window shall not extend more than 0.031 inch beyond the outer frame of the trapezoidal module as shown in Figure B5.

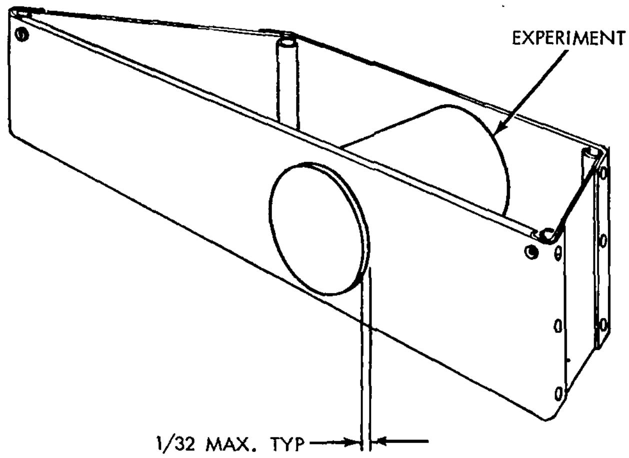


Figure B5—Experiment module with experiment window.

Materials

The use of magnetic materials will be avoided. In order to minimize or prevent a buildup of magnetic fluxes, the materials used in the construction of the experiments should be either aluminum or fiberglass. Certain brass, magnesium, and other nonferrous metals exhibit some magnetic properties. Prior approval from GSFC should be obtained for use of brass or magnesium. The magnetic restrictions of each subsystem or experiment will meet the following requirements:

a. Residual magnetism of 32 gamma at 18 inches, after a 25-gauss exposure

b. Residual magnetism of 2 gamma at 18 inches, after a 50-gauss deperm

c. Stray magnetism of 4 gamma at 18 inches

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.

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 should be notified of the location of all hotspots.

Approval of Mechanical Interface

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 foregoing requirements shall be submitted to GSFC for approval through the IMP F and G project office.

Appendix C

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 by means of the torsion rod principle.

1.0 Standard Disc

1.1 Standard Disc Weight: 50.156 lb

1.2 Standard Disc Inertia

The standard disc moment of inertia about the Z-Z axis (see Figure C1) is:

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

$$= \frac{(50.156 \text{ lb})}{64.4 \text{ ft/sec}^2}\left(0.5885 \text{ ft}^2\right) =$$

$$I_s = 0.265 \text{ slug ft}^2 .$$

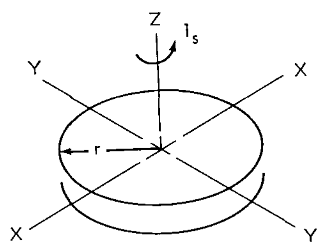


Figure C1—Standard disc moment of inertia about Z-Z axis.

1.3 Standard Disc Period $\left(T_{s_r}\right)$ for I_{roll}

The disc is mounted to the torsion rod and a picture is taken; then it is torqued approximately ±10 degrees, released, and the free oscillation timed. The torsion rod is used 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 suspending or hanging the spacecraft from the torsion rod in various orientations, obtaining the period T_p of each configuration, and calculating the moment of inertia from

$$I_T = \frac{I_s}{T_{s_r}^2} T_T^2 = k_{s_r} (T_T)^2 ,$$

where

$$k_{s_r} = \frac{I_s}{T_{s_r}^2} = \frac{0.265}{(\quad)^2} = \underline{\qquad} .$$

1.5 Inertia of Roll Attachments (I_{ra})

A. The attachment plate, adapter, two stainless steel marmon clamps, and two eyebolts are attached to the standard disc; a picture is 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 stainless steel marmon clamps, two eyebolts, and the folded hardware are attached to the standard disc; a picture is 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. $I_c = \frac{I_s}{T_{s_r}^2} T_c^2 = k_{s_r} (\quad)^2 = (\quad)(\quad)^2 = \underline{\qquad}$ slug ft² ;

b. $I_{c_f} = \frac{I_s}{T_{s_r}^2} T_{c_f}^2 = k_{s_r} (\quad) = (\quad)(\quad)^2 = \underline{\qquad}$ slug ft² .

The inertias of the roll attachments are

a. $I_{ra} = I_c - I_s = \underline{\qquad} - \underline{0.265} = \underline{\qquad}$ slug ft² ;

b. $I_{raf} = I_{cf} - I_s = \underline{\qquad} - \underline{0.265} = \underline{\qquad}$ 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 T_T (sec)
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:

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

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

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

Configuration 2:

$$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 = \underline{\hspace{3cm}} \text{ slug-ft}^2 ,$$

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

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

Configuration 3:

$$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 = I_T - I_{ra} + I_{motor} = I_{roll\ total} ,$$

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

Configuration 4 (determined analytically with data from Configuration 3):

$$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 = \underline{\hspace{3cm}} \text{ slug-ft}^2 ,$$

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

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

Configuration 5:

$$I_T = k_{s_r} T_T^2 = (\quad)(\quad)^2 = \underline{\hspace{3cm}} \text{ slug-ft}^2 ,$$

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

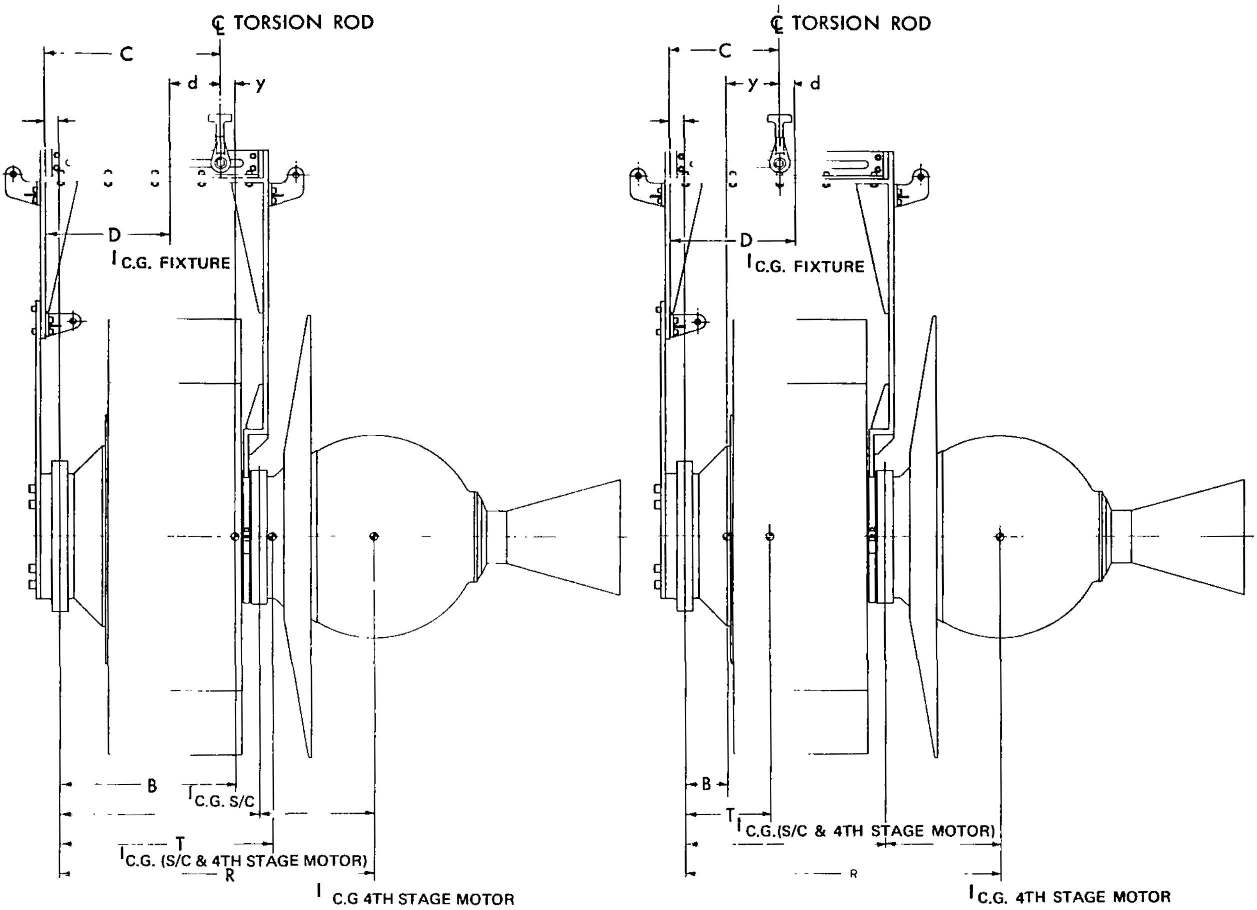
$$(\quad) - (\quad) = \underline{\hspace{3cm}} \text{ slug-ft}^2 .$$

2.0 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 C2.

2.1 Standard Disc Period $\left(T_{s_p}\right)$ for I_{pitch}

The disc is mounted to the torsion rod, and a picture taken; it then is torqued approximately ±10 degrees, released, and the free oscillations timed. The torsion rod is employed without an extension for I_{pitch}.



d = + TO LEFT OF OF TORSION ROD
d = + TO RIGHT OF OF TORSION ROD

Figure C2—AIMP moment-of-inertia test setup.

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

Therefore,

$$I_p = \frac{I_s}{T_{s_p}^2} T_p^2 = k_{s_p} \left(T_p\right)^2 = \frac{(0.265)}{(\quad)^2} T_p^2 = (\quad) T_p^2 .$$

2.2 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

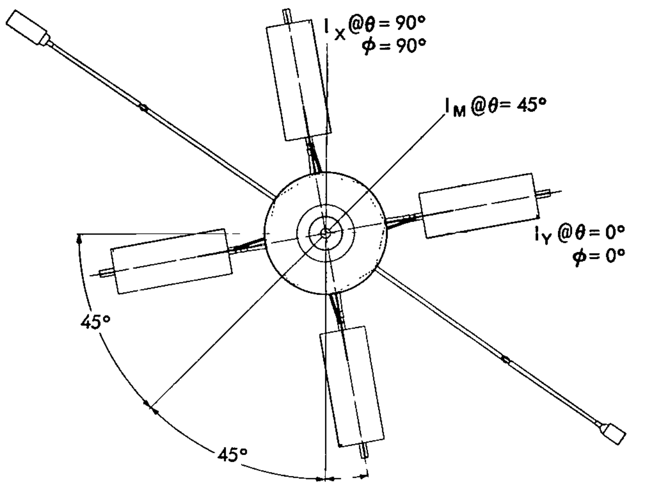


Figure C3—Measurements I_x, I_m, and I_y in the transverse plan.

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 degrees apart as marked on Figure C3. Each inertia of the spacecraft alone $I_{s/c}$ is obtained by measuring the 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 fourth stage retromotor, since it is not feasible to use the live fourth stage.

Knowing the center of gravity and $I_{c.g.\ 4th\ stage}$ of the fourth stage and determining the center of gravity and $I_{c.g.\ s/c}$, the total transverse inertia of the various configurations can be obtained by assuming a configuration (Figure C2), and determining the distance d and c, and D as a cross-check. Then,

$$\Sigma M @ \text{Rod} = W_{fixture}(d) - W_{s/c}(y) = 0 \text{ (when fixture is leveled)} ,$$

$$d = + \text{ to left of } ℄ \text{ of torsion rod} ,$$

$$d = - \text{ to right of } ℄ \text{ of torsion rod} ;$$

therefore,

$$y = \frac{\left(W_{fixture}\right)(d)}{W_{s/c}} ;$$

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

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

The data on the fixture (with stainless steel marmon clamp attached) used to hold the spacecraft during the transverse measurement are shown in Figure C2.

a. Fixture without folded attachment hardware

Weight =

Oscillations =

Time in sec =

Period in sec (T_F)

D =

b. Fixture with folded attachment hardware

Weight =

Oscillations =

Time in sec =

Period in sec (T_{FF}) =

D =

Determining the Maximum and Minimum Transverse Moments of Inertia*

For this calculation, refer to Figure C3. Thus

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

In this case θ = 45 degrees (angle between I_x, I_y, I_M). Solving for I_{xy} gives

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

$$\tan 2\phi = \frac{2 I_{xy}}{I_x - I_y} .$$

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. Thus,

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

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

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

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

*J. L. Synge, and B. A. Griffith, "Principles of Mechanics," McGraw Hill Book Co., Inc., 1949.

DATA SHEET FOR $I_{transverse}$

Configuration	Axis	Run	Oscillations	Time (sec)	T_{total} period (sec)
Paddle folded, F/G booms folded, with 4th stage hdw	x-x	1			
		2			
		3			
	M-M	1			
		2			
		3			
	y-y	1			
		2			
		3			
Paddle extended, F/G booms folded, with 4th stage hdw	x-x	1			
		2			
		3			
	M-M	1			
		2			
		3			
	y-y	1			
		2			
		3			
Paddles extended, F/G booms extended, with 4th stage hdw	x-x	1			
		2			
		3			
	M-M	1			
		2			
		3			
	y-y	1			
		2			
		3			
Paddles extended, F/G booms extended, without 4th stage hdw	x-x	1			
		2			
		3			
	M-M	1			
		2			
		3			
	y-y	1			
		2			
		3			

Calculation Sheet for $I_{transverse}$

Configuration ( )

$d =$

$c =$

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

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

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

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

$I_{c.g.\ fixture} = k_{s_p}(T_p)^2 = (\quad)(\quad)^2 = \_\_\_\_\_\_$ slug-ft^2

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

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

Axis (x-x)

$I_{total} = k_{s_p}(T_{total})^2 = (\quad)(\quad)^2 = \_\_\_\_\_\_$ slug-ft^2

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

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

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

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

$= \_\_\_\_\_\_$ slug-ft^2

Axis (y-y)

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

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

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

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

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

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

Axis (M-M)

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

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

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

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

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

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

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

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

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

$$\phi =$$

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

$$= \frac{(\quad)[0.5 + 0.5(\cos \quad)] - (\quad)[0.5 - 0.5(\cos \quad)]}{(\quad)} =$$

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

$$= \frac{(\quad)[0.5 + 0.5(\cos \quad)] - (\quad)[0.5 - 0.5(\cos \quad)]}{(\quad)} =$$

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.

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Appendix D

Dynamic Balance Versus Mass Moments of Inertia

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

$$\tan 1/2\phi = \frac{\text{dynamic unbalance}}{\left(I_{roll} - I_{transverse}\right) g},$$

where ϕ is the total tilt angle.* Figure D1 illustrates the relationship between dynamic unbalance versus the moments of inertia.

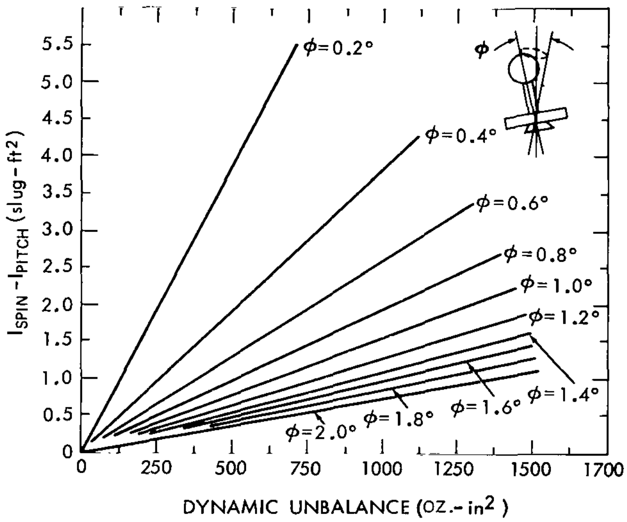

Figure D1—Graph of relationship between dynamic unbalance and moments of inertia.

*N. C. Schaller and J. M. Lewallen, "Methods of Expressing Mass Unbalance," NASA Technical Note D-1446, May 1963.

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Appendix E

Derivation of the Overspin Equations to Compensate for Gravity and Inertial Variations During Appendage Erection

Derivations

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

$$KE_1 = \frac{1}{2} I_1 \omega_1^2 \tag{E1}$$

for the folded configuration; for the erected configuration,

$$KE_2 = \frac{1}{2} I_2 \omega_2^2 , \tag{E2}$$

$$\Delta E_S = \frac{1}{2} \left(I_1 \omega_1^2 - I_2 \omega_2^2\right) , \tag{E3}$$

$$\Delta E_S = \frac{1}{2} I_1 \omega_1^2 (1 - R) \qquad \text{where} \qquad R = \frac{I_1}{I_2} . \tag{E4}$$

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

$$KE_1 = \frac{1}{2} I_1 \omega_{1T}^2 , \qquad P.E. = 0 \tag{E5}$$

for the folded configuration; for the erected configuration,

$$KE_2 = \frac{1}{2} I_2 \omega_{2T}^2 , \qquad P.E. = \Sigma\, wh ; \tag{E6}$$

$$\Delta E_G = \Delta KE + \Delta PE , \tag{E7}$$

$$\Delta E_G = \frac{1}{2} \left(I_1 \omega_{1T}^2 - I_2 \omega_{2T}^2\right) - \Sigma\, wh , \tag{E8}$$

$$\Delta E_G = \frac{1}{2} I_1 \omega_{1T}^2 (1 - R) - \Sigma w, \tag{E9}$$

where

$$R = \frac{I_1}{I_2}.$$

The purpose of the ground test is to subject the appendage to the energy experienced in a space erection; therefore let

$$\Delta E_G = \Delta E_S, \qquad \text{and assume} \qquad I_{1T} = I_{1S}. \tag{E10}$$

Then,

$$\frac{1}{2} I_1 \omega_{1T}^2 \left(1 - R_S\right) - \Sigma\,\text{wh} = \frac{1}{2} I_1 \omega_{1S}^2 \left(1 - R_S\right); \tag{E11}$$

$$\omega_{1T} = \sqrt{\omega_{1S}^2 + \frac{2\,\Sigma\,\text{wh}}{I_1 \left(1 - R_S\right)}}. \tag{E12}$$

This derivation assumes that test inertias and flight inertias are identical. The term 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 occurred on the AIMP. In flight, the initial inertia is the total inertia 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 mockup 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 $I_{2T} - I_{1T} = I_{2S} - I_{1S}$. (E13)

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

c. Using Equation E9

$$\frac{1}{2} I_{1T} \omega_{1T}^2 \left(1 - R_T\right) - \Sigma\,\text{wh} = E_S, \tag{E14}$$

$$\omega_{1T} = \sqrt{\frac{2\left(E_S + \Sigma\,\text{wh}\right)}{I_{1T}\left(1 - R_T\right)}}, \tag{E15}$$

where

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

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

Both of the preceding derivations neglect the erection spring energy. This neglect was intentional since this energy is present in identical amounts for both flight and test conditions.

In the case of the AIMP ETU paddle erection tests, it was necessary to alter the initial spin rates to compensate for gravity and the inertial differences mentioned earlier.

Comparison of Initial Spin Axis Inertias (I)

Item	Flight	Test
Spacecraft*	= 3.422 slug-ft^2	3.422 slug-ft^2
X258	= 0.730 slug-ft^2	1.046**
I_{1S}	= 4.152 slug-ft^2	I_{1T} = 4.468 slug-ft^2

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

**This value includes X258 dynamic mockup, DAC attach fitting, marmon clamp and 6 appendage cradles.

Adjustment

The preceding comparison 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. Thus,

4.468 slug-ft^2, test inertia
-0.315 slug-ft^2, loaded kickmotor
4.153 slug-ft^2
+0.070 slug-ft^2, empty kickmotor
4.223 slug-ft^2, ETU for test.

Flight Despin Sequence

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

ω_3 = 27.5 rpm
ω_0 = 150.0 rpm

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

$$\omega_2 = 41 \text{ rpm}$$

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

$$\omega_1 = 107.5 \text{ rpm} .$$

Parameter Spin Rates

a. Nominal = nominal DAC spinup and successful yo-yo

$$\omega_{1A} = 107.5 \text{ rpm}$$

b. Overspin = 10 percent DAC spinup and successful yo-yo

$$\omega_{2A} = 118 \text{ rpm}$$

c. Yo-yo failure = 10 percent DAC spinup and yo-yo failure

$$\omega_{3A} = 165 \text{ rpm} .$$

Flight Energies (Using Equation E4)

$$E_S = \frac{1}{2} I_{1S}\,\omega_{1SA}^2 \left(1 - R_S\right) , \qquad R_S = \frac{4.15}{10.85} = 0.383$$

$$E_S = \frac{1}{2}(4.15)\left(\omega_{1SA}\right)^2(1 - 0.383)$$

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

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

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

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

Test Spin Rates (Using Equation E15)

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

$$W = 6.15 \text{ lb/paddle}, \quad h = 2 \text{ ft}, \quad 4 \text{ paddles} ,$$

$$R_T = \frac{4.22}{4.22 + (10.85 - 4.15)} = 0.386 ,$$

$$\omega_{1TA} = \sqrt{\frac{2\left[162 + 4(6.15)\ (2)\right]}{4.22(1 - 0.386)}} \approx 12.8 \text{ per second} ;$$

therefore,

$$\omega_{1TA} \approx 122 \text{ rpm} .$$

Similarly,

$$\omega_{1TB} = 13.7/\text{sec} \approx 131 \text{ rpm}$$

$$\omega_{1TC} = 18.2/\text{sec} \approx 174 \text{ rpm} .$$

Summary

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 were 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 to 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 (Using flight inertias only):

I_{3s} = ____ slug-ft^2, paddles and 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 spinup rate.

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

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

ω_2 = ____

ω_2 = ____ rpm (boom erection)

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

ω_1 = ____

ω_1 = ____ rpm (paddle erection)

3. SPIN RATES TO BE USED FOR ENERGY INPUT

A. Nominal spinup + nominal yo-yo, $\omega_{1A} = \omega_1$ — ω_{1A} = ____ rpm

B. 10% overspin + nominal yo-yo, $\omega_{1B} = 1.1\,\omega_1$ — ω_{1B}^2 = ____ rpm

C. 10% overspin + yo-yo failure, $\omega_{1C} = 1.1\,\omega_0$ — ω_{1C} = ____ rpm

4. ENERGIES IMPARTED TO PADDLES FOR ZERO 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. TEST SPIN RATES TO COMPENSATE FOR GRAVITY AND INERTIA VARIANCES

A. Weight per appendage, W = ________ lb

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

$\boxed{\Sigma Wh = \text{\_\_\_\_\_\_\_\_}}$

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

$\boxed{\omega_{1tA} = \text{\_\_\_\_\_\_\_\_ rpm}}$

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

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

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

$\boxed{\omega_{1tC} = \text{\_\_\_\_\_\_\_\_ rpm}}$.

Terms

ΔE = energy absorbed by appendages (ft-lb)

h = height to 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 per seconds

Subscripts:

1 = initial position, appendages folded

2 = final position, appendages erected

S = in space, zero gravity

T = on ground test, 1g

A = nominal DAC spinup with successful yo-yo

B = 10% DAC overspin with successful yo-yo

C = 10% DAC overspin with yo-yo failure.

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Appendix F

Field Checkoff Document
(IMP-B Field Operations Checkoff List)

Introduction

The function of this checkoff list is to ensure that all mechanical systems (including experiments, fasteners, screws, despin systems, etc.) are properly and permanently inserted to ensure a successful mission of the IMP B Interplanetary Monitoring Platform. The spacecraft will not be considered ready for flight until it has been checked thoroughly and doublechecked by the cognizant project engineer or his designated alternate. Any defect noted, no matter how insignificant, should be brought to the attention of the project engineer immediately.

Spacecraft Mechanical Personnel on IMP B Field Operations

(a) Mechanical Systems Branch Personnel

		Motel	Phone
J. M. Madey	Head, Space Probe and Satellite Section		
E. W. Travis	Project Engineer		
X. W. Moyer	Asst. Project Engineer		
D. K. McCarthy	Research Engineer		
F. N. LeDoux	Head, Structural and Mechanical Applications Section		
A. J. Pierro	Lead Technician		
P. E. Henley	Technician		

(b) EMR Personnel

R. W. Forsythe	Project Engineer		
L. S. Mamakos	Technician		
W. B. Leidig	Technician		

Fastening Procedure Instructions

Long-Lok screws shall be used wherever possible, with Nylok screws as second choice and blue Lock-tite on standard screws as third choice. To indicate that the screws are properly installed and that they are to remain in the spacecraft permanently, the head of every screw will be painted with a white dot partly on the head of the screw and partly on the adjoining surface, after which the spacecraft technician shall initial the appropriate item in the first column with the project engineer's (or alternate) initials in the second column. This operation is necessary in that it affords an immediate visual indication that the screws are locked and ready for flight.

If a screw must be removed, it must be discarded, paint removed from the adjoining surface, and a new screw used and repainted as indicated previously. Fill in the comment column for removal of marked screws and state the reason.

Change and removal sheets (blank) are provided herein and any defects or changes in procedure are to be recorded.

One master checkoff list shall be recorded for the spacecraft that is launched and one master maintained on the spare spacecraft. Extra copies shall be maintained for reference use only by the MSB and EMR personnel.

IMP B Schedule at AMR

F–20	F–19
Stud and trailer personnel arrive Deliver and check out stud and trailer (GSE)	Check out GSE
F–18	**F–17**
Delivery of IMP B, IMP C and radioactive sources IMP B and C personnel arrive; check out GSE	IMP B spacecraft checkout IMP C spacecraft checkout
F–16	**F–15**
IMP B to antenna range for RF, R & RR, and magnetic checks IMP C spacecraft checkout; to antenna range for R & RR and magnetic checks	IMP B spacecraft checkout and calibration IMP C to spin and balance facility; fit for brackets on live stage; mount on dummy stage

F–14 IMP B spacecraft checkout and calibration IMP C mate with vehicle; GSE to gantry; check blockhouse interface; spacecraft checkout	**F–13** IMP B spacecraft checkout and calibration; μ-meson run at night IMP C RFI checks with vehicle
F–12 IMP B spacecraft checkout and calibration; μ-meson run at night; Chicago calibration IMP C spacecraft checkout	**F–11** IMP B spacecraft checkout and calibration; install mag. boom D.M. IMP C spacecraft checkout and calibration
F–10 IMP B to spin and balance facility; Mount on live stage IMP C μ-meson run at night; Chicago calibration	**F–9** IMP B alignment and rough balance IMP C μ-meson run at night; Chicago calibration
F–8 IMP B rough balance IMP C standby	**F–7** IMP B prepare for final balance IMP C standby
F–6 IMP B final balance IMP C standby	**F–5** IMP B final balance IMP C standby
F–4 IMP B to gantry; install on vehicle; complete checkout IMP C standby	**F–3** IMP B complete checkout IMP C standby
F–2 IMP B complete checkout IMP C standby	Contingency
F–1 IMP B F–1 day checks; strip and touch touch up thermal coating (2 hours) IMP C standby	**F–0** IMP B F–0 day checks; Launch IMP C standby
F+1 Close field operations and pack for return trip	**F+2** Return to GSFC

IMP B

Item	Action	Initials: Performed by	Initials: Checked by	Comments
	F-11 R_b Magnetometer Erection System Dimple Motor Installation			
1	Measure resistance of two prepotted and insulated Hercules DM 29Ao Dimple Motors. Use special squib checker. (See data sheet on pg 9 .) Green leads - D.M. #1 ________ ohms Red leads - D.M. #2 ________ ohms			
2	Install Dimple Motors into mechanism. Check for proper fit & shim if necessary. Refer to pictorial schematic which follows.			
3	Solder D.M. #1 to terminals 4 and 8. Solder D.M. #2 to terminals 1 and 7. Use heat sink.			
4	Again measure resistance of each D.M. as before. D.M. #1 ________ ohms D.M. #2 ________ ohms			
5	Measure resistance of each 5 ohm nominal resistor on each terminal board. R_1 = ________ ohms R_2 = ________ ohms			
6	Measure total resistance of each D.M. and resistor system. R_1 = ________ ohms R_2 = ________ ohms			
7	Have 1G5 leads soldered to both terminal boards and inspect.			

Item	Action F-11 R_b Magnetometer Erection System Dimple Motor Installation	Initials: Performed by	Initials: Checked by	Comments
8	Install mechanism into center tube 12 screws loctited.			
9	Insert release pin into bushing.			
10	Remove IMP Card IG5.			
11	Using test connector which mates with "D" frame connector IG5-J2, measure and record the following with special squib checker: Pins 4 & 8 R_1 = ________ohms Pins 1 & 7 R_2 = ________ohms Above resistances each should read between 6.4 and 7.6 ohms. Otherwise reject and install new Dimple Motors and/or resistors.			
12	Brush-coat terminals with Epon 828 or equivalent.			
13	Add mechanism safety tape using glass tape 1/16 inch wide by 1/2 inch long.			
14	Visually inspect assembly. Insure that all wires to the assembly are supported.			
15	Reinstall card IG5.			

INTERPLANETARY MONITORING PROBE
IMP B
MECHANICAL DATA SHEET

1. ASSEMBLY (TITLE)
Dimple Motor DM29AO

2. SUBSYSTEM
Magnetometer Release System

3. DRAWING NO.
Hercules Powder Co. Dwg. HD-920

4. RESPONSIBILITY
E. W. Travis
Mechanical Systems
X 5096

5. OUTLINE DRAWING:

I. Physical Data

Body size — .293" dia., .51" long
Wire leads — #24 AWG, copper solid
Seal — Phenolic
Bridge resistance — 1.4-2.6 ohms, wire type
Ignition — Lead styphnate
Main charge — LMNR/Black powder type

II. Performance Data

Test current (maximum) — 10 MA.
Max. nonfire (MNFC) — 0.25 amp, one 30 sec. pulse
Min. fire (MFC) (Borderline, not recommended) — 0.45 AMP.
Recommended (all fire) (RFC) — 1-3 AMPS
Ignition time:
Amps — 1.0, 2.0, 3.0, 5.0
Milliseconds — 4.0, 3.5, 3.0, 2.7
High temperature — Functions after 2 hours at 250°F
Low temperature — Functions at −65°F
Reliability — 99.9%

6. POTTING INFORMATION:

7. WEIGHT (GRAMS)
UNPOTTED
POTTED

8. LOCATION IN SPACECRAFT:
Under battery can

PREPARED BY
E. W. Travis

DATE
6/6/63

REVISION

APPROVALS AND DATE
E. Travis — DESIGNER
P. Butler — PROJ MGR

TITLE

IMP ID NO.

DATE OF ISSUE
X 125

Δ REVISION MARKER
GSFC 6-2 (8/62)

IMP B

Item	Action Below Platform Before Cone Installation Prior to F-10		Initials: Performed by	Initials: Checked by	Comments
1	Paddle arm resistors	8 scr			
2	Paddle arm attach. scr	16 scr			
3	Paddle arm	16 nuts			
4	Energy absorber housing	8 scr			
5	Paddle arm microswitches	8 scr			
6	Filter-con (2 scr per unit)	6 scr			
7	Sep. switch (2 scr per unit)	6 scr			
8	Mag. brake and release	15 scr			
	(a) install new cord				
	(b) tape release door				
	(c) tape on cord guide				
	(d) shim squibs as required				
	(e) Check wiring as per Figure F1				
	(f) Check mag-squibs if required				
9	Mag. connector (a)	4 scr			
	(b)	2 std offs			
10	Spring seat	8 S.S. scr			
11	Spring seat cover	8 scr			
12	"D" frame bolt nut	4			
13	Strut nuts and screws (bottom)	16			
14	Ray pan struts (bottom)	16			
15	XMTR Cu screws with conductive epoxy (max. torque 2 in. lb)	2 scr			
16	Copper foil				
17	Cable clamps—scr. & lacing				
18	F/G boom conn.	4 scr			
19	F/G boom	4 nuts			
20	F/G leaf spring	4 scr			
21	F/G torsion spring	2			
22	F/G brkt "A" umbilical	1 scr			

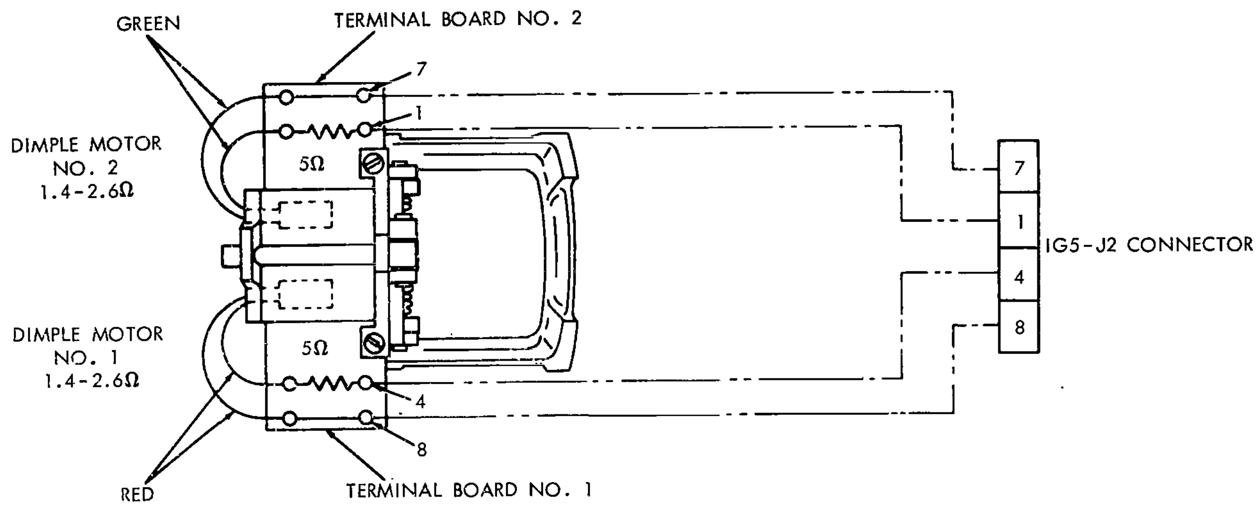


Figure F1—I #9—wiring release mechanism viewed from below spacecraft

IMP B				
Item	Action Lower Cone Installation Prior to F-10	Initials: Performed by	Initials: Checked by	Comments
1	Base of cone — 13 screws & washers ____			
2	Middle of cone — 12 screws & washers ____			
3	Top of cone — 24 screws & washers ____			
4	Cone split — 6 screws & washers ____			

IMP B

Item	Action Top of Platform Prior to F-10		Initials: Performed by	Initials: Checked by	Comments
1	Base of "D" frames	16 screws Stainless steel			
2	Strut to center tube	72 screws			
3	Platform to center tube	8 screws			
4	Mag. spring cup	4 screws			
5	Battery bolts	4			
6	Battery lower connector	2 screws			
7	"D" frame bolts	16			
8	F. G. plugs	4 nuts			
9	Turn-on plug bracket	10 screws- 4 stand off			
10	Connectors each facet				
11	Battery connector	2 nuts			
12	Antenna system				
	(a) Board bracket	2 screws			
	(b) Hybrid board	4 screws			
	(c) Antenna redline (4)	thick Glyptol			
13	Range and range rate redline connectors (10)	Glyptol			
14	All card connectors—8 facets				
15	Killer timer shorting plug	2 screws 2 nuts			
16	Test connectors facets A-1, D-2, E-4	black glass tape			
17	Balance weight	4 screws			
18	Facet F top D frame	3 screws			
19	Record all card numbers on payload drawing in Section XX				

IMP B				
Item	Action Center Tube and Mag. Outer Tube Prior to F-10	Initials: Performed by	Initials: Checked by	Comments
1	Base of mag. outer tube — 6 screws			
2	Top of ant. cone — 6 screws			
3	Ant. seat cups (4) — 24 screws			
4	Ant. mag. conn. with grd. wire — 6 screws			
5	Wire holder-outer tube — 10 screws			
6	Top outer tube-guide screw — 2			
7	Mag. erect switch — 2 nylon screws			
8	Ball ring bottom — 4 screws			
9	Ball connector bracket — 2 screws			
10	Ball top — 4 screws			
11	Ball connector — 2 screws			

IMP B				
Item	Action Top Cover Prior to F-10	Initials: Performed by	Initials: Checked by	Comments
1	Topside and center — 8 screws			
2	Topside-outside (not A&C facets) — 4 screws			
3	Sides — 18 screws			
4	Install optic aspect sensor (a) 2 screws 1" long (b) 4 screws (short) backed off 1/2 turn & locktite			
5	Observe thermal coat-touch up and add strip coating			

IMP B

Item	Action F-10 to F-1 Balance Operations	Initials: Performed by	Initials: Checked by	Comments
1	Mount spacecraft on X-258 motor.			
2	Observe DAC installing clamp band and check clearance of separation switch plunger.			
3	Install full size dummy paddles.			
4	Observe DAC alignment and bonding of the paddle and boom cradles and standoffs.			
5	Observe DAC compatibility check of umbilical cord and nose fairing.			
6	Observe payload runout measurements and record.			
7	Observe DAC transfer of payload and bottle to balance fixture.			
8	Install weighted paddle spars.			
9	Observe DAC assembly of tie-down system.			
10	Install safety cord around bottom of appendages.			
11	Remove all sensor covers and install antennas.			
12	Observe DAC rough balance and installation of weights.			
13	After rough balance, remove weighted spars and install four flight paddles in accordance with the preselected orientation.			
14	Observe DAC final balance operations.			
15	Remove four flight paddles.			
16	Observe DAC install final balance weights.			
17	Install weighted paddle spars and observe installation of tie-down system.			
18	Observe DAC final balance check.			
19	Record all pertinent weights in Section XVII.			
20	Replace all sensor covers and tie down antennas.			
21	Observe assembly of transport container and shipment to launch pad.			

Time (min.)	Item	Action F-1 Day Fairing Off	Initials: Performed by	Initials: Checked by	Comments
T-400	1	Mount range vs dE/dx and plasma probe relay boxes on top cover.			
	2	Attach the OA exciter, jig, and strobe light to the cover.			
	3	Release the payload antennas.			
	4	Remove dummy turn-on plug and insert meter panel plugs.			
	5	Remove meter panel plugs and insert live turn-on plug.			
	6	Remove live turn-on plug and insert dummy turn-on plug.			
	7	Remove OA jig, strobe light, and the range vs dE/dx and plasma probe relay boxes.			
	8	Secure the antennas against the magnetometer boom and tie with a cord.			
	9	Replace all experiment protective covers.			

IMP B

Time (min.)	Item	Action F–0 Day Fairing Off	Initials: Performed by	Initials: Checked by	Comments
T-755	1	Mount range vs dE/dx and plasma probe relay boxes on top cover.			
	2	Attach the OA exciter, jig, and strobe light to the cover.			
	3	Release the payload antennas.			
	4	Remove dummy turn-on plug and insert meter panel plugs.			
	5	Remove meter panel plugs and insert live turn-on plug.			
	6	Remove live turn-on plug and insert dummy turn-on plug.			
	7	Remove OA jig, strobe light, and the range vs dE/dx and plasma probe relay boxes.			
	8	Secure the antennas against the magnetometer boom and tie with a cord.			
	9	Replace all experiment protective covers.			
	10	Install covers over: (a) antenna access hole in cover top (b) test connector holes on sides of top cover.			
	11	Remove 2 large OA covers and install center cover with Long-Lok screws or Loc-tite.			
	12	Loc-tite or install Long-Lok screws in top cover where not loctited—4 screws.			

IMP B

Time (min.)	Item	Action F–0 Day Fairing Off	Initials: Performed by	Initials: Checked by	Comments
T-755	13	Remove (a) MIT experiment protective cover (b) Thermal ion electron experiment protective cover (c) University of Chicago experiment protective cover (d) University of California experiment protective cover			
	14	Remove mag. safety screws (2)			
	15	Remove separation switch safety lock system (2 screws)			
	16	Open each paddle arm to ensure that latch locks properly: Arm No. 1 ________ Arm No. 2 ________ Arm No. 3 ________ Arm No. 4 ________			
	17	Check proper closure of microswitches on: Arm No. 3 ________ Arm No. 4 ________			
	18	Remove protective strip coating.			
T-355	19	Install solar paddles and tie-down Long-Lok screws on connectors (8)			
	20	Inspect DAC tie-down system.			
	21	Inspect paddle arm wiring for nicks, cuts, etc.			
	22	Install live turn-on plug.			
	23	Release antennas.			
	24	Remove live turn-on plug and install dummy turn-on plug.			
	25	Remove spring retainer screws on paddle arms (4 screws).			

IMP B

Time (min.)	Item	Action F–0 Day Fairing Installed	Initials: Performed by	Initials: Checked by	Comments
T-235	1	Release antenna.			
	2	Install umbilical connector to spacecraft.			
	3	Install flight turn-on plug (2 screws). Use red Loc-tite.			
	4	Visual inspection and final approval of all systems.			
	5	Confirm removal of all tools taken up to the gantry.			

Sequence of Events

Table F1 lists the significant engineering events which occur from liftoff to third-stage burnout of the Delta vehicle. All event times are listed in seconds-after-liftoff. In addition, all second-stage and subsequent events are referenced to the start of the second-stage program timer at the time of main engine cutoff (MECO).

Table F1

Sequence of Events.

Event	Initiated by	Time (seconds) Expected	Actual
Uncage stage I gyros	Liftoff switch	T+0	
Start stage I programmer	Liftoff switch	T+0	
Enable MECO circuitry	Stage I programmer	T+139	
MECO	FIP switch	T+148.56 (M+0)	
Start stage II programmer	MECO relay	T+148.56 (M+0)	
Sequence 1 a. Blow blast-band bolts b. Blow transition skirt bolts	Stage II programmer	T+152.56 (M+4)	

Table F1 (Continued)

Event	Initiated by	Time (seconds)	
		Expected	Actual
c. Uncage pitch and yaw gyros d. Enable stage II roll control e. Start stage II engine f. Transfer guidance reference power			
Jettison fairing	Stage II programmer (sequence 2)	T+182.56 (M+34)	
Stage II engine cutoff command	BTL ground station	T+317.40 (M+168.8)	
Switch to coast control	SECO relay	T+317.40 (M+168.8)	
Turn off hydraulics	SECO relay	T+317.40 (M+168.8)	
Sequence 3 a. Arm oxidizer probes b. Arm TPS	Stage II programmer	T+317.56 (M+169)	
SECO		T+317.74 (M+169.2)	
Sequence 4 a. Fire spin rockets b. Start ignition wire cutter TDR c. Start pyrotechnic TDR for sequence 5 backup d. Start stage III ignition time delay e. Start stage III sequence timer f. Fire stage III ignition wire cutters	Stage II programmer	T+338.56 (M+190)	
Sequence 5 a. Blow stage III/II separation bolts b. Fire retros	Stage II programmer	T+340.56 (M+192)	
Stage engine ignition	Pyrotechnic time delay	T+344.56 (M+196)	
Stage III burnout	Depletion	T+367.16 (M+218.6)	
Erect payload solar paddles	Stage III sequence timer	T+401.56 (M+253)	
Start pyrotechnic TDR for payload boom erection	Stage III sequence timer	T+401.56 (M+253)	
Erect spacecraft booms	Boom erection pyrotechnic TDR	T+403.56 (M+255)	
Start spacecraft separation TDR	Boom erection pyrotechnic TDR	T+403.56 (M+255)	
Spacecraft/stage III separation	Spacecraft separation pyrotechnic TDR	T+444.56 (M+296)	
Fire second-stage tumble rockets	Stage III sequence timer	T+446.56 (M+298)	

Satellite Checkout
Change Sheet

Step	Action and Comment	Performed by	Checked by

Inertia and Weight Data

A. Spacecraft Folded (Launch) Configuration

C. G. = ____________ above separation plane

I_{roll} = ____________ slug-ft^2

$I_{pitch\ max}$ = ____________ slug-ft^2

B. Spacecraft Paddles Extended

C. G. = ______________ above separation plane

I_{roll} = ______________ slug-ft^2

$I_{pitch\ max}$ = ______________ slug-ft^2

C. Spacecraft Fluxgate Booms and Paddles Extended

C. G. = ______________ above separation plane

I_{roll} = ______________ slug-ft^2

$I_{pitch\ max}$ = ______________ slug-ft^2

D. Spacecraft Fluxgate Booms, paddles, and Magnetometer Extended (occurs after separation)

C. G. = ______________ above separation plane

I_{roll} = ______________ slug-ft^2

$I_{pitch\ max}$ = ______________ slug-ft^2

$I_{pitch\ max}$ = ______________ degrees measured counter clockwise looking down from paddle No. 1 (Figure F2)

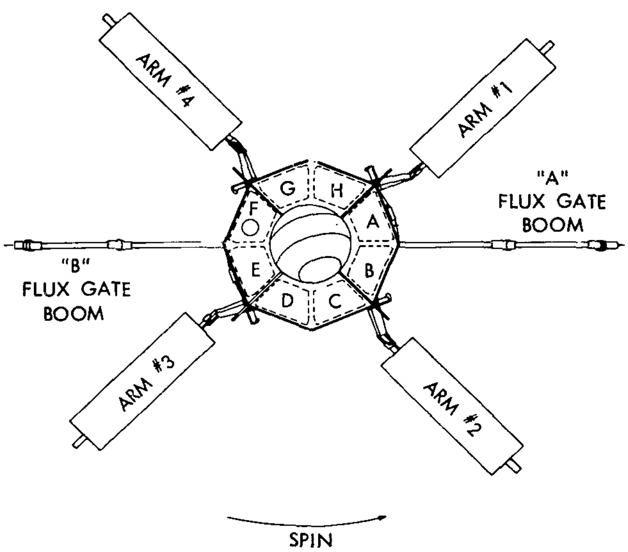


Figure F2—IMP B transverse plane.

E. X-258 Expended Motor and Separation Mechanism

Item	Estimate at GSFC (incl. one 0.005 alum. foil)	Actual at Cape
I_{roll}	0.71 slug-ft^2	
I_{pitch}	6.49 slug-ft^2	
C. G. station	230.0	
Weight	78.4 lb	

F. Folded Spacecraft on Expended X-258

Weight = ______________ lb

C. G. = ______________ the separation plane

I_{roll} = ______________ slug-ft^2

$I_{pitch\ max}$ = ______________ slug-ft^2

G. IMP A Spacecraft

After GSFC dynamic balance

Weight = __________ lb

Static unbalance = ________ oz-in.

Dynamic unbalance = _________ oz-in.^{2}

H. Weights (pounds)

(1) loaded bottle __________

(2) separation mechanism __________

(3) rocket tumble system __________

(4) aluminum foil __________

(5) attach fittings __________

(6) spin table __________

(7) nose fairing __________

(8) final stripcoat __________

I. Solar Paddle Weights and C. G.'s

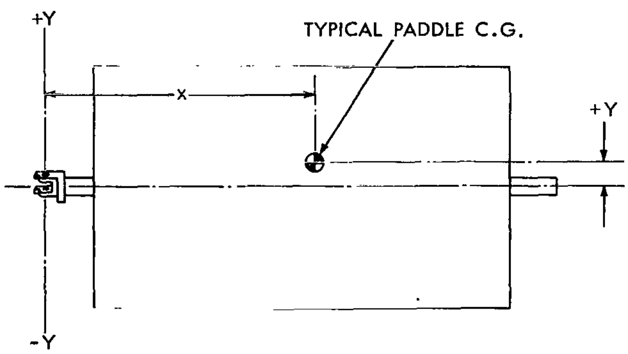


Figure F3—Solar paddle, top view.

Paddle no.	X (inches)	Y (inches)	Weight (grams)
102	14.818	0.014	2898
103	14.850	0.107	3045
106	14.834	-0.028	2958
107	14.849	-0.002	2955
108	14.881	0.035	3039

Spin Rate Calculations

Symbols

ω_0 = stage III spin up rate (rpm)

F_p = each pet rocket thrust (lb)

n = number of pet rockets

Δt = duration of each pet rocket firing (sec)

r = perpendicular distance from stage III spin axis to each pet rocket thrust vector (ft)

I_{ff} = spin inertia of full stage III, attachments, and spacecraft folded (slug ft^2)

I_{ef} = spin inertia of expended stage III, attachments, and spacecraft folded (slug ft^2)

I_p = spin inertia of expended stage III, attachments, and spacecraft with paddles extended (slug-ft^2)

I_{pb} = spin inertia of expended stage III, attachments, and spacecraft with paddles and booms extended (slug-ft^2)

Spinup:

$$\omega_0 = \frac{9.55}{I_{ff}} \left[\left(F_p\, r\Delta t\right)_1 + \left(F_p\, r\Delta t\right)_2 + \cdots \left(F_p\, r\Delta t\right)_n \right] .$$

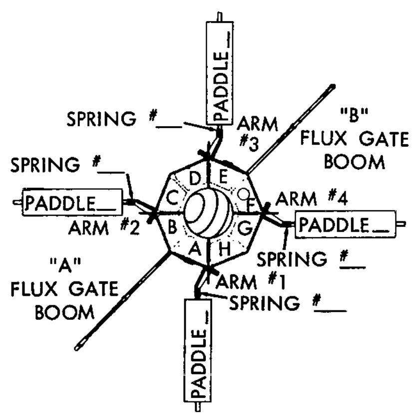
Spin Rate, Paddle Extended $\left(\omega_p\right)$:

$$\omega_p = \frac{I_{ef}}{I_p} \left(\omega_0\right) .$$

Final Spin Rate, Paddles and Booms Extended $\left(\omega_{pb}\right)$:

$$\omega_{pb} = \frac{I_p}{I_{pb}} \left(\omega_p\right) \qquad \omega_{pb} = \frac{I_{ef}}{I_{pb}} \left(\omega_0\right) .$$

IMP B Paddle Arm Assembly Orientations (See Figure F4)



ARM POSITION	SPRING NO.	PADDLE NO.	CRUSH PAD L.
1			
2			
3			
4			

Figure F4—IMP B paddle arm assembly orientations.

Spacecraft Drawings (See Figures F5 through F7)

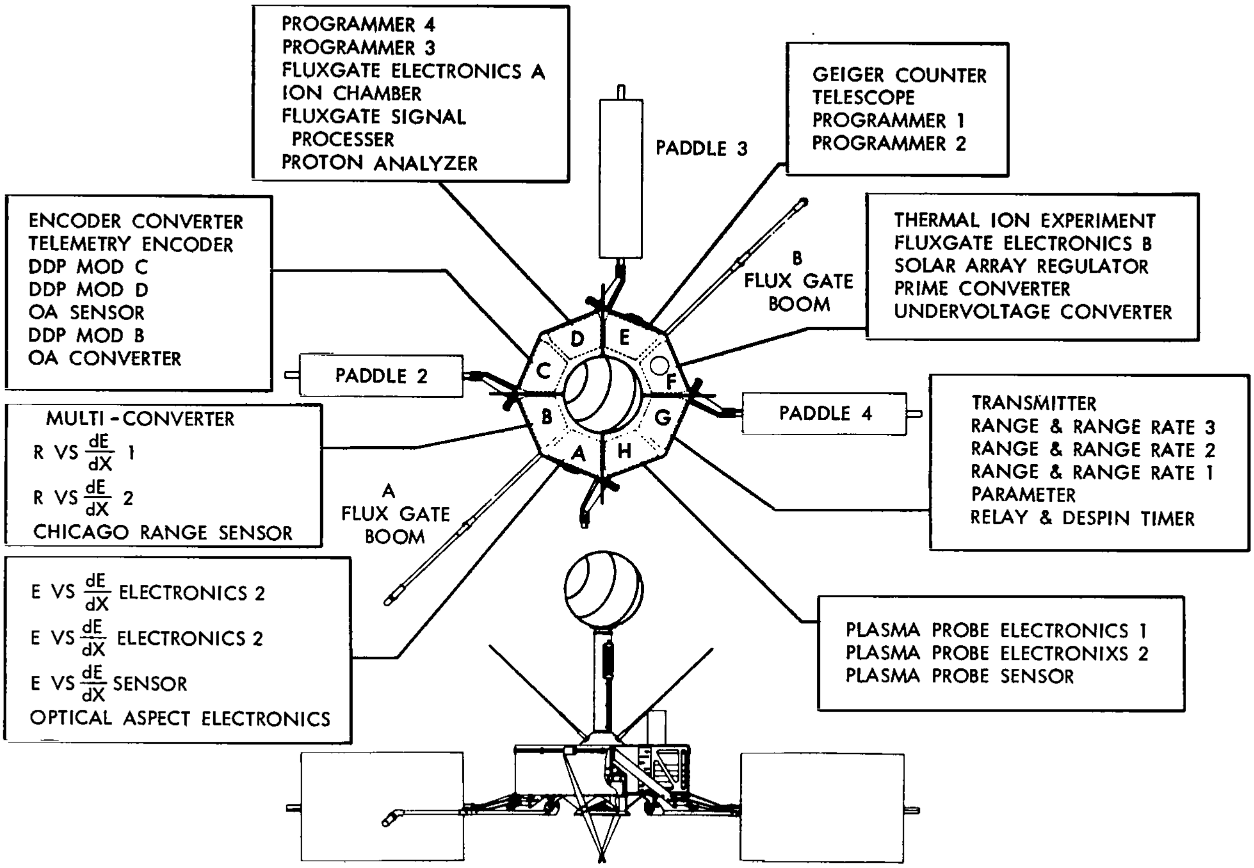


Figure F5—IMP B experiment arrangement.

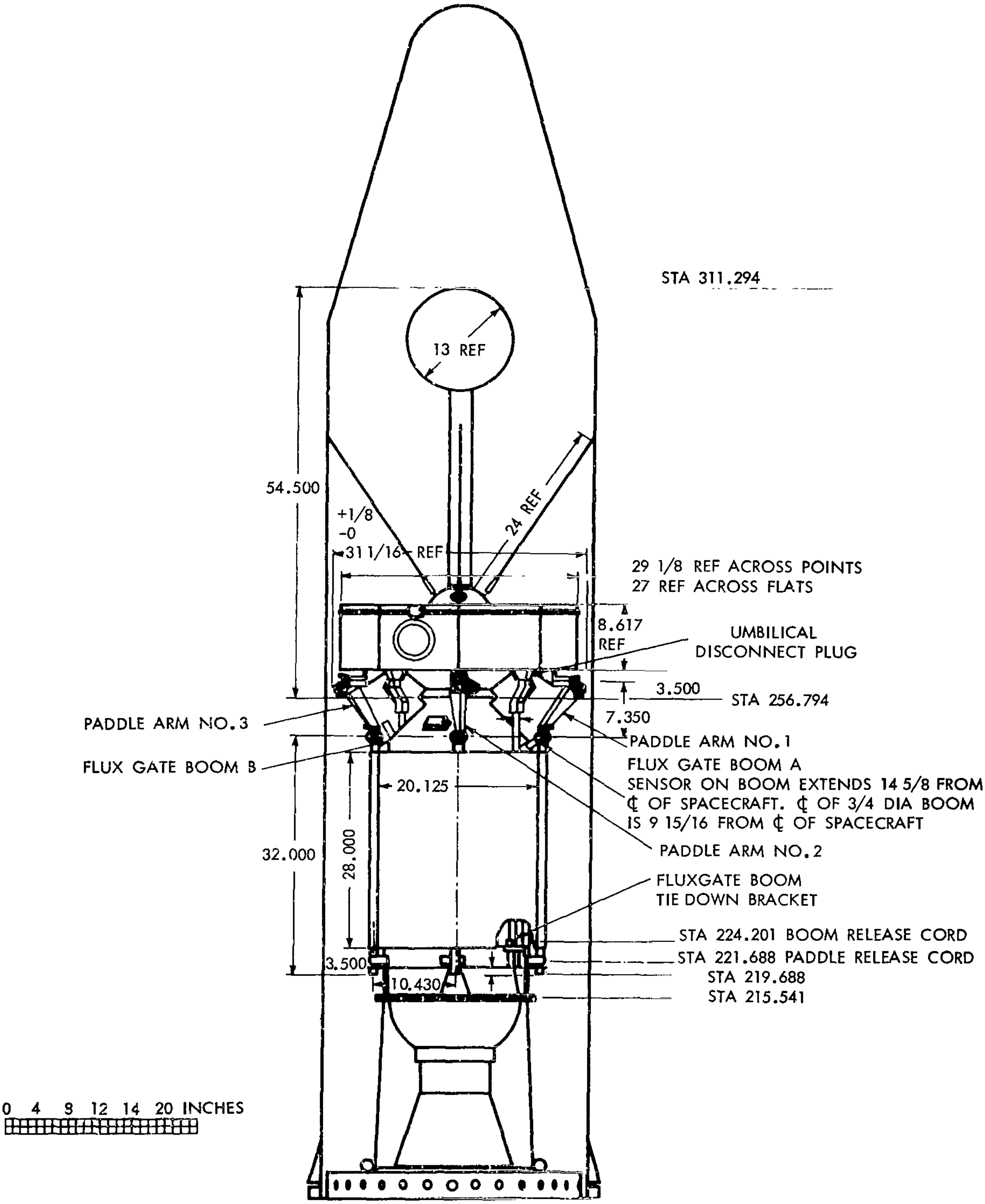


Figure F6—IMP B payload installation.

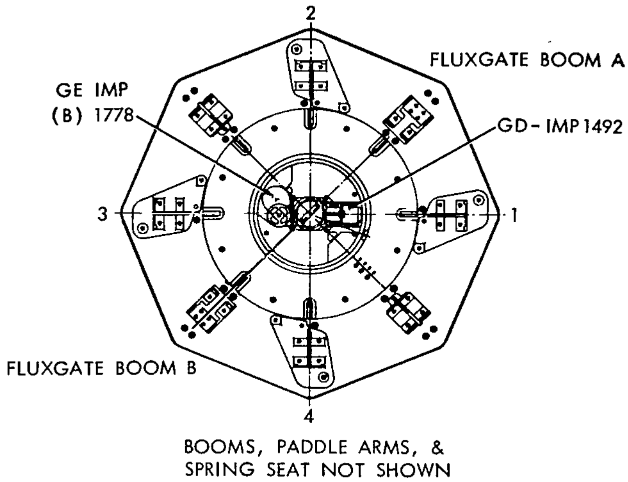


Figure F7—IMP B structure assembly.

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