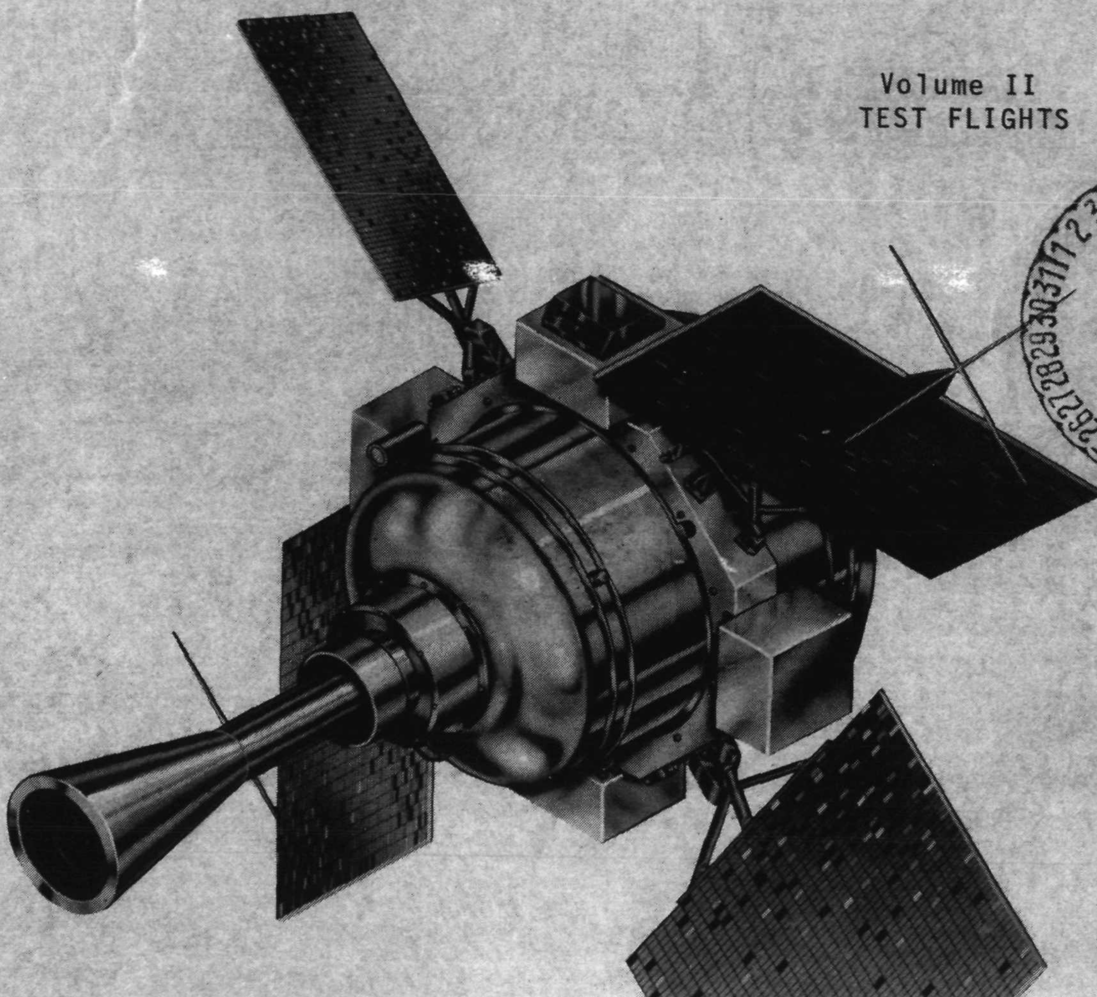
MISSION DEFINITION

for

STANFORD RELATIVITY SATELLITE

FINAL REPORT F71-07

Volume II
TEST FLIGHTS



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BALL BROTHERS RESEARCH CORPORATION
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VOLUME II

ENGINEERING FLIGHT TEST PROGRAM

Final Report

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FOR
STANFORD RELATIVITY SATELLITE

22 December 1971

Report F71-07

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Section 1
SUMMARY

Early in the planning of the Stanford Relativity Experiment, it became clear that the gyroscopes and dewar could not be ground tested in a manner that would establish their performance because of the presence of the earth's gravitational field. This report examines the need for orbital flight tests of these components, and others, in order to reduce the technical and financial risk in performing the final experiment. A program is described that will generate engineering data to allow the final performance to be predicted with sufficient accuracy to proceed with confidence.

Two flight tests are recommended. The first flight would test a dewar smaller than that required for the final flight but of size and form sufficient to allow extrapolation to the final design. The second flight would use the same dewar design to carry a set of three gyroscopes which would be evaluated for spin-up and drift characteristics for a period of a month or more. A proportional gas control system using boil-off helium gas from the dewar, and the ability to prevent slosh of the liquid helium would also be tested.

Cost of the program is kept low by flying piggyback on the improved Delta vehicle and by operating the program in a manner similar to the sounding rocket programs with documentation and controls commensurate with an engineering test.

Target dates would be mid-1973 for the first flight and mid-1974 for the second.

Section 2
SELECTION OF TESTS

The Stanford Relativity Experiment hardware can be divided into four major pieces of equipment: the dewar, the gyroscopes, the telescope and the control system. Each of these is vital to the experiment and each is advancing the state of the art. The tests required for determining their performance will be examined in the next section to determine which tests cannot be performed adequately on the ground.

2.1 DEWAR

The dewar for the Stanford Relativity Experiment must hold enough superfluid liquid helium to last for a year in orbit. It must have a central cavity large enough to contain the experimental apparatus and must have a window through which the telescope can observe the reference star. In addition, the helium within the dewar must not be allowed to move about enough to change the C.G. of the overall satellite by more than a centimeter or to interact with the control system. Helium vapor must be provided in sufficient quantities to the control system. Finally, the requirements of other experimenters needing helium dewars in space should be investigated to determine whether their test requirements can be satisfied by this program without serious conflict.

2.1.1 Venting

The problem of venting the dewar in zero gravity is a complicated engineering task. In the case of subcritical fluid storage, the problem is that liquid may be pushed overboard by a small heat input to the dewar, the consequence being a greatly shortened mission. In zero gravity the liquid will seek a minimum energy geometry and will tend to creep into holes (i.e. vent lines) so there is a tendency for the container to vent liquid.

Two solutions have been proposed: a superfluid plug invented at Stanford, and a boiler concept. The superfluid plug has been tested on the ground with an invertible helium dewar to verify its performance at +1 g and -1 g. It appears to work well for both normal and superfluid helium; its operation is independent of whether the heat sources are internal or external to the liquid helium bath. Some difficulties might occur in applying it to situations subject to sudden large heat inputs (as happens, for example, when a superconducting magnet accidentally goes normal). The boiler concept has been tested on the ground with oxygen and nitrogen but not with helium. It appears to be feasible for normal helium provided the internal heat sources are not too great. It has the advantage of eliminating the need for two cryogenic valves and probably allows rapid emergency venting of the dewar. There are reasons to doubt whether the boiler concept would work in space for superfluid helium. Both methods appear worth pursuing at the present time.

Before the time of a flight test, one or the other may be eliminated from contention. In any case, it is doubtful that the true performance of either system in zero g can be predicted with sufficient confidence to avoid an orbital test. Therefore, one of the first objectives of the flight test will be the evaluation of the two systems with superfluid helium. This can be accomplished in a single test flight.

The venting problem in a dewar with normal helium and very little internal heat generation is of interest to the HEAO experiment being studied by the University of California at Berkeley under Dr. Alverez. This test could and should be incorporated in the test flight with little additional effort.

2.1.2 Helium Management

The second major problem associated with the dewar in this experiment is the location and stability of the helium within the dewar. The helium initially represents nearly 25 percent of the mass of the spacecraft. Holding the C.G. of the spacecraft within a centimeter of the center of the experiment is a critical requirement. Without constraints, the helium could move to locations that would cause much larger C.G. changes. In addition, the free motion of the helium within the dewar could degrade the control system performance in an unpredictable manner. We have suggested a scheme, using surface tension, to control the location of the helium so that the C.G. does not change appreciably throughout the life of the mission and so that only a very small fraction of the helium can slosh at any time. This scheme, and others considered so far, provides very small controlling forces (10^{-4} g) which are nevertheless adequate in orbital flight. Although analysis and scaled ground tests using other fluids may give an indication of the performance, an orbital test is required.

2.1.3 Temperature Distribution and Creeping Film

The temperature distribution within the central cavity is important to this experiment as well as to possible infrared astronomy missions. The temperature variation is limited by the conductivity of the cavity wall and by the creeping superfluid film. This film is expected to be several orders of magnitude thicker in orbit than on the ground and should significantly change the temperature variation near local heat sources. Measurement of both the temperature distribution and the thickness of the film are important orbital test objectives. The temperature distribution in the presence of normal helium which does not exhibit the creeping film phenomenon is of interest to other experiments which do not contemplate the use of superfluid helium.

2.1.4 Other Tests

All other aspects of dewar performance can be tested on the ground with sufficient confidence. The boil-off rate is a measurement of prime interest since this is a direct measure of lifetime. This can be determined quite accurately on the ground assuming vapor venting. Orbital tests would verify the results.

The ability to survive launch can be determined by proper centrifuge and shake tests. The satisfactory operation of retractable launch locks can also be determined by ground tests. The orbital flight would confirm the design. The ability to deliver additional vapor for the control system by adding heat to the dewar can be determined on the ground.

2.1.5 Summary

The important orbital dewar tests are:

- Dewar venting tests of superfluid plug and boiler with Helium II and Helium I
- Helium management
- Temperature distribution within dewar
- Thickness of creeping film

Secondary orbital tests are:

- Launch survival

- Verification of boil-off rate calculation
- Verification of retractable support system
- Verification of the ability to quickly accelerate boil-off

2.2 GYROSCOPES

The Stanford gyroscopes operate within a liquid helium dewar. They need to be electrically suspended, spun up in orbit to a given speed and aligned to a given orientation. They must remain essentially drift free for a period of one year, and be read out to an accuracy of better than 0.001 arc-second. An elaborate test and evaluation program is required to assure that such performance can be achieved.

2.2.1 Suspension System

The suspension system used for ground testing the gyroscopes must support the ball reliably in a 1 g field and be capable of supporting the ball for considerably higher g loads caused by shocks, earthquakes, and mishaps occurring during testing. This requires the application of several thousand volts to the gyroscope suspension plates and results in gyro drift rates about 10^6 higher than in orbit. The orbital suspension system operates at a few volts, supports the gyroscope against a nominal 10^{-8} g's acceleration and has emergency suspension capabilities to protect the gyroscope against acceleration of 10^{-4} g's. For spin-up in orbit, the suspension system must provide about 0.2 g's.

It is clear that the ground and orbit suspension systems have very different requirements and that the orbit system cannot suspend the gyro in realistic ground tests. Simulations of the orbit suspension system could be made using a hollow ball or one floated in a dense fluid, but the mass and damping properties would be considerably different. Because of the very wide change in parameters between any ground test condition and the orbital conditions, it is clear that the orbital suspension system must be flight tested.

2.2.2 Spin-up and Orientation

The gyroscope is spun up to a speed of approximately 200 rps using cold helium vapor. Spin-up takes approximately 40 minutes. The spin axis must be aligned within 10 arc-seconds of the desired direction. After spin-up, the pressure within the gyro chamber must be reduced to 10^{-8} mm Hg as rapidly as possible to avoid excessive spin down and torques. Realistic ground testing of the gyroscope spin-up may be possible. Initial tests at room temperature can give a qualitative measure of the spin-up characteristics as a function of flow rate and pressure, and later tests at cryogenic temperatures will give much more accurate information. Even though the 1 g suspension system is used, the force required to center the ball against spin gas disturbances can be determined by observing the suspension system error signals during spin-up. This would allow proper scaling of the orbital system. One area of doubt with the data gathered in this way is that the difference in frequency response of the two systems may cause the ball to act differently. This could possibly be resolved by tests in which a DC bias in the vertical direction could be used to buck out the force of gravity, with the orbital spin-up suspension electronics providing the difference signals required to keep the ball centered. Great care would be required to avoid vibration or bumping of the apparatus during the test.

Two of the gyroscopes must have their spin rates matched to 1 percent. It is doubtful that this will be achieved on an open-loop basis. Probably the spin rates will have to be controlled either by ground command or by a closed-loop system in the satellite. This could be done by stopping the spin-up process at the appropriate time or introducing a small amount of gas into the spin-up cavity of one gyro to cause the spin to reduce to match the other gyro. In either case, this can be determined by ground testing.

The ability to pump the gyroscope cavity down to the proper pressure can be determined by ground tests, although it takes large pumps and a clean system.

Tests to determine the accuracy of spin axis orientation must be done at cryogenic temperatures using the London Moment readout. Presently it is proposed to rotate the entire satellite about the desired spin axis during spin-up in order to average the effects of spin gas cross-torques. This can be simulated in the laboratory by rotating the gyroscope housing at the same rate. This test must be performed with the spin axis parallel to the earth's axis to remove the effects of the earth's rotation.

In general, the spin-up characteristics seem determinable on the ground. However, because of the requirement for a gyroscope drift test in orbit, to be discussed later, an orbital test of gyroscope spin-up will occur as a matter of course. It should be instrumented to get confirming data.

2.2.3 Gyro Drift

Gyro drift on earth is predominantly caused by suspension-dependent torques due to gravity. These torques may cause laboratory drift rates many millions of times greater than the orbital drift rates. It is clearly not reasonable to extrapolate from such a drift rate to the orbital drift rates. Further, this large drift rate obscures the much smaller but very important drifts due to other causes.

An orbital drift test is the only way to establish limits on what the gyro drift can be in the final flight. For useful results, the test spacecraft orientation must be maintained to a few arc-seconds and the acceleration on the gyroscope must be kept low (10^{-8} g). In addition, the test must be long enough to clearly establish the drift rate parameters.

2.2.4 Gyroscope Readout

Since the magnetic shielding of the gyroscope is the same on the earth as it is in orbit, there is no reason to believe that very adequate tests of the readout system cannot be performed on the ground. However, since this readout must be used in the orbital drift test, an orbital test will confirm its performance.

2.2.5 Summary

The important orbital gyroscope tests are:

- Gyroscope Suspension
- Gyroscope Spin-up
- Gyroscope Drift

The secondary orbital tests are:

- Gyroscope Spin Axis Orientation
- Gyroscope Readout

2.3 TELESCOPE

The telescope provides the prime reference for the experiment and also provides the error signals to the control system. It must be made of diffraction-limited optics and must have a mechanical and electrical null stability of better than 0.001 arc-second over a period of a year. The linearity of the error signal must be better than 0.001 arc-second over a range of ±0.05 arc-second.

2.3.1 Stability

Mechanical instability of the telescope is caused by bending because of thermal gradients, viscous flow and elastic bending of the quartz under the forces of gravity, and creep due to the release of elastic strains inherent in the quartz. By operating on the ground in a cryogenic environment the problems of thermal gradients are removed because the coefficient of expansion is nearly zero and the temperature is very constant. The creep due to the release of elastic strains is also reduced. Problems associated with viscous flow from gravity forces are reduced by annealing properly and by keeping the telescope vertical so the flow affects mostly focus which is not as critical as alignment. Holding the telescope vertical also removes the elastic bending due to gravity.

Performing tests on the ground to determine null drift will be very difficult. Probably these will have to be indirect tests using differential auto-collimation techniques to check the alignment of two parallel but separated portions of the telescope while it is in a cryogenic environment. Accuracy approaching 0.01 arc-second may be possible with great care. Performing an orbital test to determine stability would be almost as expensive as a final flight. A gimbal and control system similar to that planned for the main flight, instrumentation electronics, to readout the telescope pointing error would have to be incorporated. This is not a reasonable approach. If laboratory tests approaching 0.01 arc-second accuracy can be performed, the extrapolation to orbital conditions is not unreasonable.

2.3.2 Diffraction-Limited Optics

The determination that diffraction-limited optics are being used can be made by the examination of the piece parts and the assembly, using careful optical manufacturing quality-control techniques.

2.3.3 Linearity

The linearity of the star image is a result of the quality of the optical parts and their assembly. If diffraction-limited optics are aligned properly and the focus is correct, the image will have the required linearity. Testing for the ultimate linearity on the ground will require a precision artificial star with collimation much better than 1 arc-second. With such an artificial star, linearity measurements to better than 0.01 arc-second can be achieved but will be difficult. Using real stars for this test is not possible due to atmospheric shimmer. An orbital test is ruled out for the same reason as in the case of mechanical stability.

2.3.4 Summary

No reasonable orbital tests of the telescope can be made which would greatly improve the confidence gained from careful ground tests.

2.4 CONTROL SYSTEM

The two-stage control system contemplated for the final flight keeps the entire satellite oriented with respect to the telescope optical axis to better than ±2 arc-seconds using a proportional helium gas thruster system while keeping the telescope pointed toward the star to 0.05 arc-second using an internal gimbal system operated by cryogenic actuators. Error signals for both systems are derived from the telescope error signal. In addition, the satellite is kept in a drag-free orbit to an accuracy of 10^{-9} g by the use of the same proportional thrusters. Error signals for the drag-free control will come from a proof mass mounted with the telescope-gyroscope assembly. Satellite roll attitude is controlled to about ±2 arc-seconds using the proportional thrusters. The roll error signal is derived from the secondary readout signals from a perpendicular gyroscope.

2.4.1 Outer Body (Satellite) Control

The boil-off helium gas from the dewar is used for control of the satellite by means of proportional thrusters. Ground testing of this control system using an air bearing in the conventional way is complicated by the extremely low torque levels available, a few thousand dyne-cm, and the fact that the exhaust gas pressure is about 1/50 atmosphere. An elaborate ground test setup is being considered in which the body motions which would be caused by the thrusters are caused by powerful gimbal actuators whose signals are derived from analog computations based on the actual thrust delivered. The thrusters and the force measuring apparatus would be in a vacuum chamber. Care must be taken to be sure that the model used for the computations is an adequate representation of the real spacecraft.

Orbital testing of this control concept on a small test satellite suffers from a similar scaling problem as the ground tests, but might contribute significantly to the confidence of the system.

Since orbital testing of the dewar and gyroscopes requires use of the helium boil-off gas and a proportional thruster system, an orbital control system test is attractive and would space-qualify these components for the final flight with little effort.

2.4.2 Inner Body (Telescope) Control

The inner body control system is a two-axis gimbal arrangement using a cryogenic actuator to apply the forces. Obtaining the ultimate accuracy in a 1 g field using the artificial star may not be possible, but the performance should not be seriously in doubt.

2.4.3 Drag-Free Control

The drag-free control system planned for this satellite is to be an adaptation of the DISCOS system. The major difference will be the use of proportional thrusters instead of a bang-bang system. Testing the system functionally on the ground is not possible because of gravity. However, two-dimensional simulations are possible using an "air puck" system similar to the one in existence at Stanford University. DISCOS is to fly in 1972 and an ESRO drag-free satellite is to fly in 1973. Presumably, drag-free systems will be well demonstrated within a few years and there will be no need to flight-test this system. Should there be problems with DISCOS, this decision should be re-evaluated.

2.4.4 Summary

Important orbital tests:	None
Secondary orbital tests:	Testing proportional valves and thrusters as a part of the dewar and gyro tests.

2.5 CONCLUSION

Orbital tests of a dewar and the gyroscopes in a dewar are required. The dewar test will be relatively straightforward and inexpensive. The gyroscope test will be more difficult and more expensive because of the additional cost of the gyroscopes and the difficulty of testing and instrumentation. For this reason, gyroscope tests should follow the dewar test and not be combined with it. The dewar of the first test should be designed to carry the gyroscope package of the second test to reduce the cost of the overall program. The basic control system used on the dewar flight should be adaptable to the gyroscope test flight to avoid having to test a control system on this critical flight and also to reduce the overall cost.

Section 3
LAUNCH VEHICLE POSSIBILITIES

We have surveyed the launch vehicle possibilities, for a satellite of roughly the size and weight required for the two orbital tests. We have examined the use of an entire vehicle for the test and also looked at the possibilities of a piggyback ride. In the interest of cost, a piggyback ride is certainly most attractive and with the capability of present boosters it is quite possible. For this reason, little effort was spent on further examination of a prime payload situation.

The two best possibilities for a piggyback ride are the Air Force Titan III-C and the NASA Thor-Delta.

3.1 TITAN III-C

The characteristics of the Titan III-C piggyback possibilities are shown in Table 3-1.

There is a great deal of room directly beneath the prime payload into which a piggyback satellite could be placed. Cost would be minimal, but because many of the prime payloads are classified there could be some very difficult launch pad interface problems. Table 3-2 shows the launches which might be available.

Because of the fact that some of these flights are classified and because they are controlled by a different agency, it has been difficult to get very specific about flight opportunities. However, it seems possible that should a specific requirement be generated for a flight, space could be made available.

Table 3-1

TITAN III-C PIGGYBACK CHARACTERISTICS

Location	Transtage Engine Compartment	Beneath Primary Payload	Viking Proof Flight
Volume & Shape	Several 0.46 m cubes	Cylinder: Diameter - 2.75 m Height - 0.93 m	Large, since no payload is present.
Payload Weight	90 kg	90 - 360 kg	Large, as ballast is being flown to simulate payload
Attachment	Explosive clamp and separation springs	→	→
Orbits	Geosynchronous	→	→
Prime Payload Constraints	None	Cannot interfere with prime payload	None
Launch Environment	Hot, due to engine. Higher acoustic levels	Dynamics altered	Not known
Cost	Nominal	Truss only	Nominal
General	Has not been done before, but has been studied	Classified payloads could pose problems. Limited access during integration of prime payload	To test centaur restart. Some risk involved

Table 3-2

TITAN III-C FLIGHT OPPORTUNITIES

Launch Date (Qtr./Year)	Mission	Orbit	Excess Payload
2/1973	ATS-F	Geosynchronous	90 kg
1/1974	Viking Proof Flight	Geosynchronous	Very large
3/1974	SESP (Solrad High)	Not well defined	Not defined
1973-1975	Several Classified Missions	Geosynchronous	225-400

3.2 THOR-DELTA

The Thor-Delta project office at GSFC is actively studying the piggyback possibilities of the improved Thor-Delta launch vehicle. With the addition of strap-on solid propellant rockets, the Thor-Delta can place 1100 kg in a moderate altitude earth orbit. With the new eight-foot diameter shroud, there is a great deal of payload space. Both of these factors make the Thor-Delta a very good candidate for piggyback rides.

Table 3-3 describes the Thor-Delta piggyback possibilities. A small amount of space is available in the engine compartment which could be used but which is not very desirable. There is adequate space available in the Payload Experiment Package (PEP) being considered by the Delta project office and there is also adequate space in place of the PEP package. These possibilities will be examined later.

Table 3-4 shows the Delta launches scheduled over the next three years. With only one exception, ERTS-B, there is additional payload capability if strap-ons are added. The orbit in which the second stage ends up is listed in the orbit column. The most

Table 3-3

DELTA PIGGYBACK CHARACTERISTICS

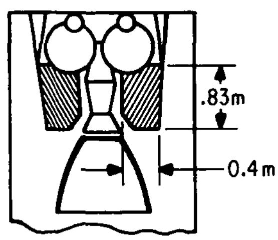
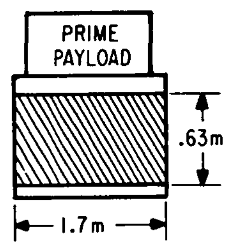
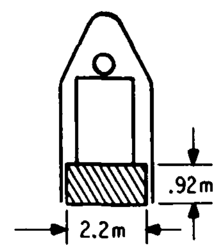
Location	Delta Engine Compartment	PEP or PAC Package	Beneath Primary Payload
Volume & Shape			
Payload Weight	22.5 - 45 kg	45-90 kg	45-225 kg
Attachment	Permanently bolted to second stage	Permanently bolted to PEP	Explosive bolts & spring eject
Orbits	185 - 1480 km I = 28.5 → 110°	185 - 1480 km I = 28.5 → 110°	185 - 1480 km I = 28.5 → 110°
Prime Payload Constraints	None	Moved up and supported by PEP	Moved up into fairing
Launch Environment	Hot due to engine higher acoustic environment	Altered dynamic environment	Altered dynamic environment
Launch Costs	Nominal	$700 K + development for PEP	Truss ≈ $200 K
General	OSCARS, TETR's set precedent	Concept proposed but not funded 2 year lead time for entire concept	Strong impact on primary interface

Table 3-4

DELTA FLIGHT OPPORTUNITIES

Launch Date (Qtr./Year)	Mission & Launch Vehicle	2nd Stage Final Orbit Inclination Altitude km	Excess** Payload kg
2 or 3/1972	IMP H 1603	28.5° 185	0
1/1973	Nimbus E 0600	110° 1100	11
1/1973	ITOS E 0300	101° 1460	22
2/1973	SIRIO (Italy) 1603	?	?
2/1973	ERTS-B 0900	99° 930	34
2/1973	*RAE-B 2314	28.5° 185	40
2/1973	*IMP-J 2613	28.5° 185	?
3/1973	*Skynet IIA 2313	28.5° 185	0
3/1973	*AE-C 1610	63° 185	45
4/1973	*OSO-I 2310	33° 550	90
4/1973	*Skynet IIB 2313	28.5° 185	0
1/1974	ITOS-F 1300	101° 1460	22
4/1974	Nimbus F 2600	110° 1100	22-34

* 8 Foot fairing

** Additional 45-225 kg obtained by adding three solid strap-ons to Delta desirable ones for this test would be the middle altitude (550-1300 km) circular orbits.

Although the space and weight capacity exist with the Thor-Delta, piggyback flights of additional payloads have not been very common in the past. With exception of the 112 kg PAC experiment flown along with OSO-6, the piggyback payloads have been small and simple. What we wish to fly here is neither small nor very simple. In addition, it requires some launch pad auxiliary equipment to pump on the dewar until a few hours before launch. An adequate structure must be designed and built to hold the experiment that does not alter the vehicle dynamics significantly. Also, great care must be taken to not jeopardize the success of the prime payload. Very close coordination with the prime payload project office will be required.

In spite of these problems, the Delta piggyback approach seems the most reasonable and cost-effective approach.

Section 4
ORBITING HARDWARE

The orbiting hardware described in the following sections show some of the possibilities for experiments to take advantage of the Delta piggyback capability. Ideally, the orbiting hardware design of the dewar flight should be directly applicable to the dewar-gyroscope flight. In the first case examined this is not at all true and is only presented as an extreme fallback position. In the second case examined, the dewar design is applicable and in the third case the entire hardware package is directly applicable.

4.1 DEWAR FLIGHT TEST HARDWARE

In Section 2.1 we determined that the important characteristics to be tested on a dewar flight were the following:

- Dewar venting with superfluid plug and boiler with Helium II and Helium I
- Helium management
- Temperature distribution within the dewar
- Launch survival
- Verification of boil-off calculation
- Verification of retractable support system.

The flight test dewar must be large enough to perform a gyroscope drift test of two months duration. With a boil-off rate of 0.225 kg per day, the estimated usage rate, the dewar must carry at least 14 kg of helium plus enough for the launch pad standby and must have a central cavity large enough for the gyroscope package.

4.1.1 Dewar in Second-Stage Engine Compartment

This concept is presented as an inexpensive test which measures some of the important dewar characteristics but not all of them and does not result in a usable design for the follow-on gyroscope tests.

A small dewar holding approximately 2.7 kg of superfluid helium is mounted in the engine compartment of the Delta second stage. A separate electronics package is mounted on the opposite side as shown in Fig. 4-1. The spent second stage is allowed to tumble freely after its control gas is used up. Because of the long cigar shape of this stage, it will generally try to become earth-oriented due to the gravity gradient torques, and tumble at a once-per-orbit rate. Because of its large magnetic moment, it will also try to align itself with the earth's magnetic field and try to tumble twice per orbit. However, tumble rates higher than this are not anticipated. The test duration would be a week to ten days.

Because of the short duration of the test, we would abandon an effort to test the boiler concept and the Helium I tests. Because of the lack of control over the second stage dynamics, we could perform only limited tests of the helium management and could not perform slosh tests. Test of the performance of the superfluid plug could be made, the temperature distribution within the dewar could be measured, the boil-off calculation could be confirmed, and the ability to survive the launch could

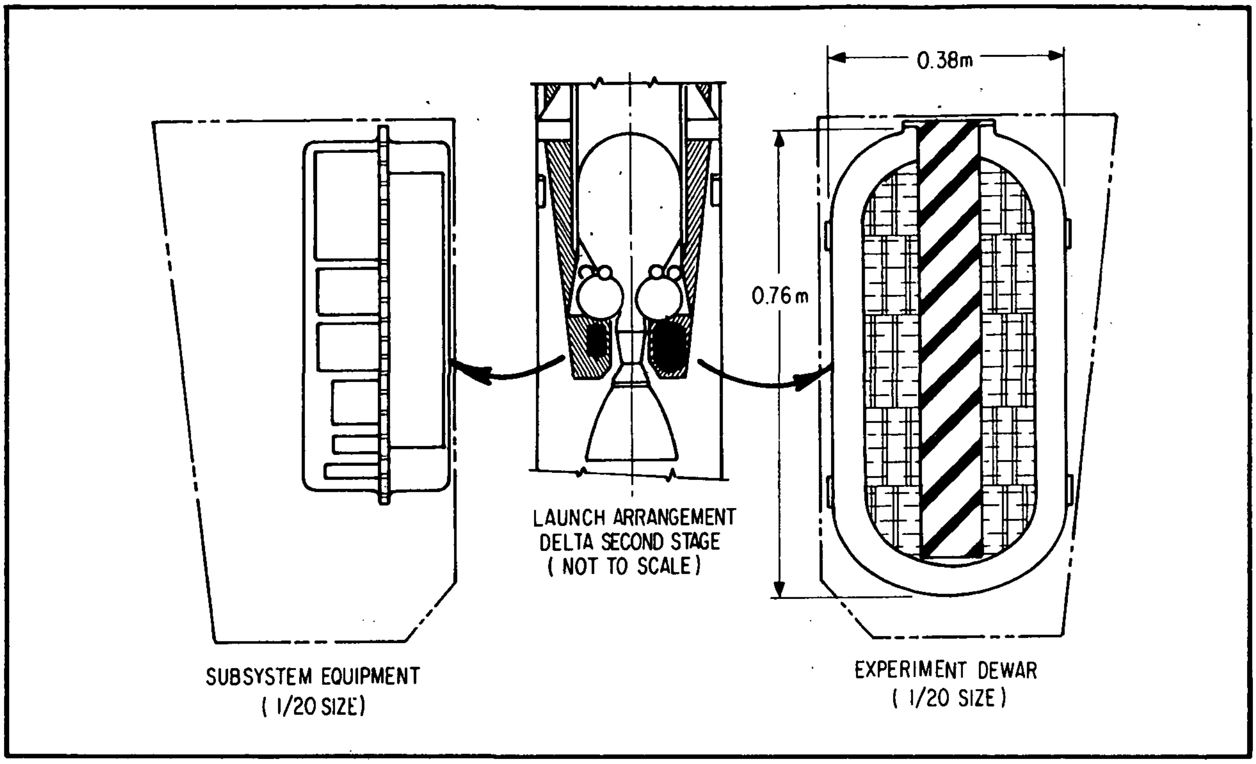


Fig. 4-1 Installation in Delta Second-Stage Engine Compartment

be determined. Because of the small size of the dewar, there could be problems in extrapolation to much larger dewars.

A battery pack of silver-zinc cells would be used for power, and a simple VHF, STADAN compatible, telemetry and command system would be used.

The characteristics of this concept are summarized in Table 4-1. A block diagram is shown in Fig. 4-2.

4.1.2 Concept B: Dewar Only in Delta PEP Module

This concept uses a Dewar, compatible with the gyroscope test flight, mounted in the PEP package on top of the second stage of the Delta (Fig. 4-3). PEP is a system studied by McDonnell-Douglas for the Delta Project Office at GSFC. The PEP package would

Table 4-1

DESCRIPTION OF TEST FLIGHT NO. 1 CONCEPT A

Launch Vehicle: Delta Piggyback

Location: Second Stage Engine Compartment

Total System Weight = 29 kg

Dewar

- Shape = Cylinder
- Size = 0.38 m dia x 0.76 m long
- Insulation = 0.051 m
- Volume = 0.023 m^3
- Helium Wt. = 2.7 kg
- Super Fluid Plug - no Boiler Bypass

Spacecraft Control

- First 2 hrs using Delta 2nd Stage
- None Thereafter
- Slow Tumble or Libration due to gg

Power

- No Solar Arrays
- ~1000 Watt hr AgZn Primary Cells

Communications

- VHF Down Link
- "COSMOS" Data Storage ≈ 100 K bits
- VHF Up Link

Dewar Instrumentation

- Helium Location
- Temperature Probes
- Super Fluid Film Thickness
- Helium Flow Rates

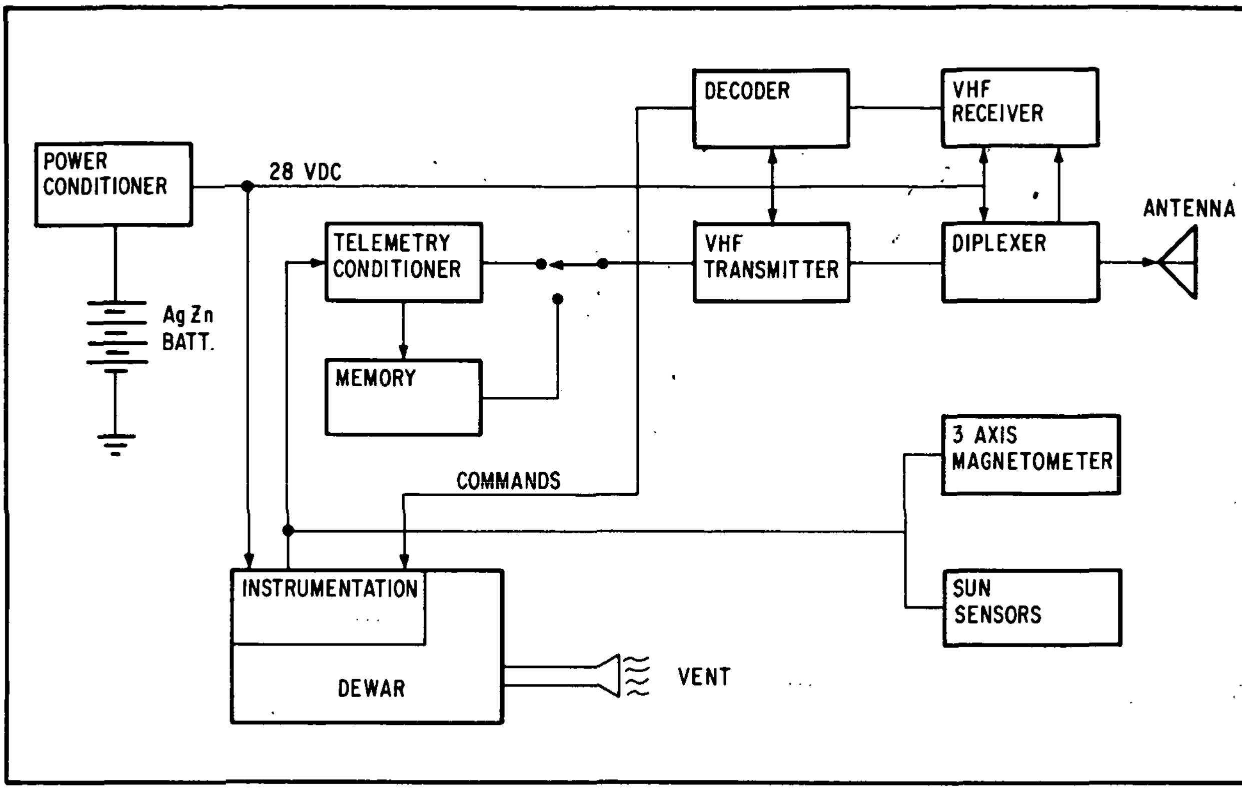


Fig. 4-2 Concept A: Functional Block Diagram

provide mounting for experiments, power, telemetry and command, thermal control and stabilization. It converts the second stage into a gravity gradient stabilized earth-oriented satellite. The active stablization system provides three-axis control. The PEP program is not yet funded and is expected to take 18 months until the first test flight.

The dewar would hold approximately 18 kg of Helium II so the test could last up to two months. All important parameters of the Dewar test objectives could be measured with the possible exception of some of the measurements of the helium management scheme. Evaluation of the proportional valves and thrusters needed for the next flight would not be accomplished.

The characteristics of this configuration are listed in Table 4-2.

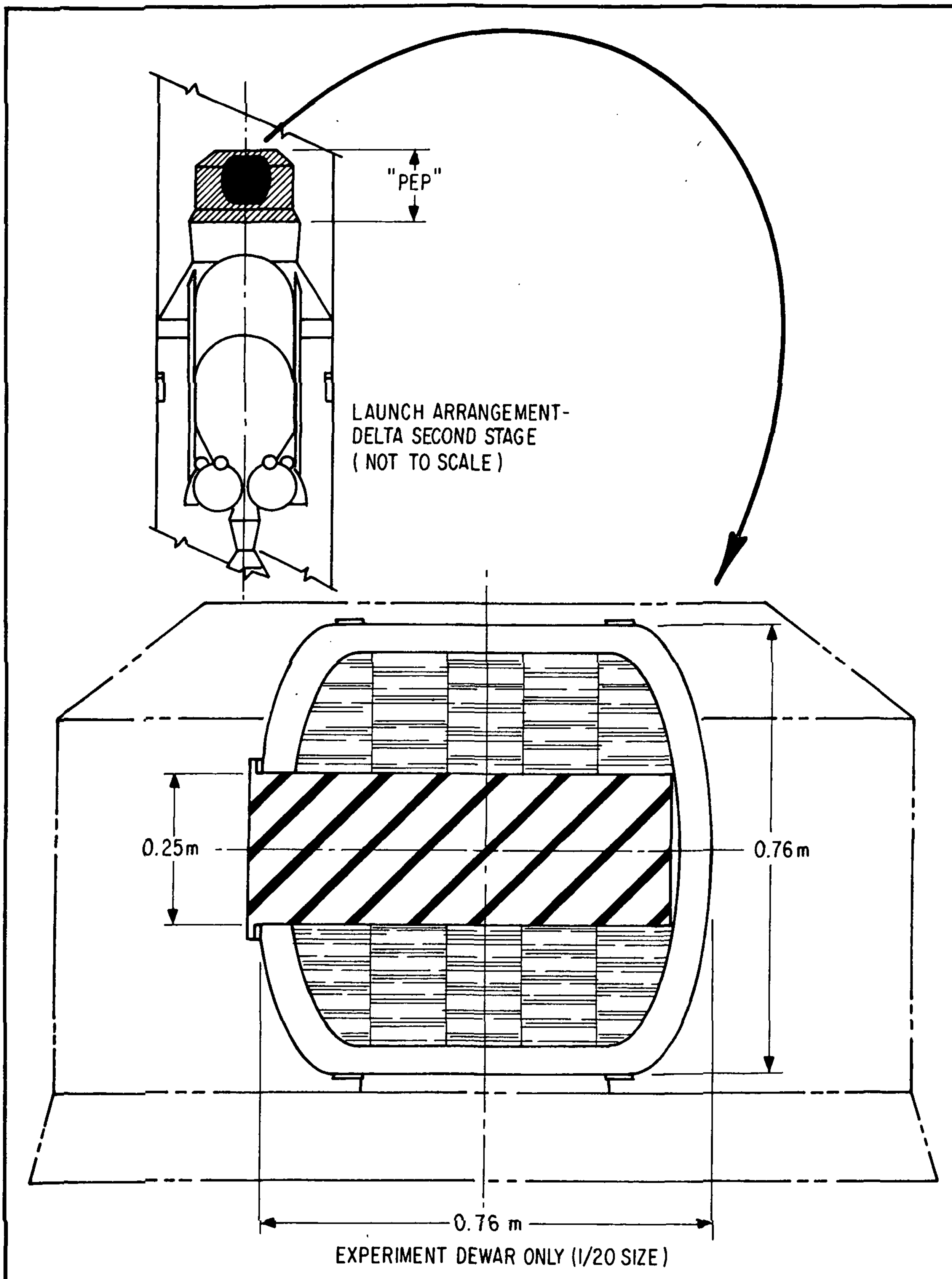


Fig. 4-3 Concept B: Dewar Only in Delta PEP Module
Independent Satellite in PEP Package

Table 4-2

DESCRIPTION OF TEST FLIGHT NO. 1 CONCEPT B

Launch Vehicle: Delta Piggyback

Location: Attached to PEP Module

Dewar

- Shape = Cylinder
- Size = 0.76 m dia x 0.76 m long
- Insulation = 0.05 m
- Volume = 0.15 m^3
- Helium wt = 18 kg
- Superfluid Plug & Boiler Bypass

S/C Control

- Earth Pointing Provided by PEP

Power

- Over 50 watts avg Provided by PEP

Communications

- PEP Provides Telemetry, Command & Tape Recorders

Dewar Instrumentation

- Helium Location
- Temperature Probes
- S.F. Film Thickness
- Helium Flow Rates

4.1.3 Concept C: Independent Satellite in PEP Package

A third scheme satisfies all the requirements for the dewar test and results in the tested spacecraft design required for the follow-on dewar-gyroscope test, and therefore is recommended as the proper approach. A separable payload is mounted on top of the Delta second stage beneath the prime payload in the space being planned for PEP. After release of the prime payload, the piggyback payload is ejected into a different orbit, to separate the two.

The separable paylod (Fig. 4-4) consists of the same dewar as discussed in Concept B, attached to a spacecraft module containing a nitrogen cold-gas attitude control system, a STADAN-compatible VHF telemetry and command system, a foldout solar array and batteries for power, and the instrumentation package. There would also be an alternate proportional cold-gas control system using the boil-off helium as part of the test program to evaluate the proportional valves and the thrusters.

The characteristics of this concept are listed in Table 4-3. The block diagram is shown in Fig. 4-5.

The system is quite conventional; however, the control system may need some explanation. The satellite is solar-oriented, using coarse and fine sun sensors similar to those used on the OSO series. A three-axis, non-floated, integrating rate gyro package is used to provide damping during acquisition, to provide a nighttime reference for the solar direction, and to provide the roll reference for a low roll rate. A pressure vessel containing 1.8 kg of nitrogen gas at 20 x 10^6 n/m^2 provides the gas for attitude control initially. A two-stage regulator reduces the pressure to a low level so the acceleration on the spacecraft can be kept low when the on-off valves are actuated. This allows a proper test of the helium mangement system.

After the initial tests of the dewar, control is transferred to the proportional helium gas control system to stabilize the spacecraft toward the sun. The accuracy required for the gyroscope test flight is of the order of 10 arc-seconds. The ability to achieve this accuracy would be tested on this flight using the proportional gas system.

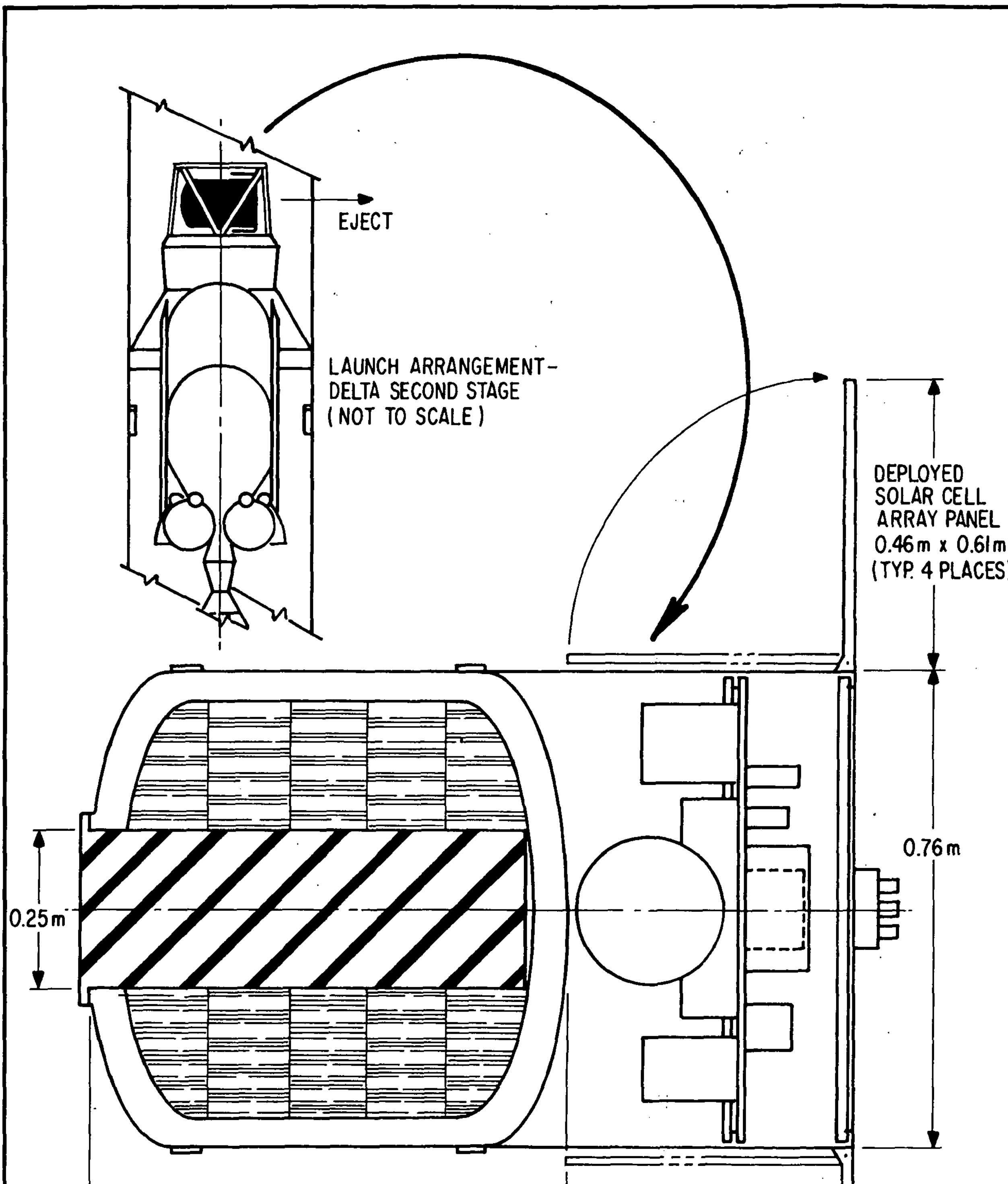


Fig. 4-4 Concept C: Independent Satellite as Secondary Payload

Table 4-3

TEST FLIGHT NO. 1 CONCEPT C

Launch Vehicle: Delta Piggyback

Location: Separable Payload - Attachment beneath primary payload in special Piggyback Truss

Spacecraft:

- Size = 0.61 dia x 1.22 long
- Shape = Cylinder with Fold-Out Array
- Weight ≃ 100 kg

Dewar

- Size = 0.76 dia x 0.76 length
- Insulation - 0.05 inch
- Volume = 0.15 m^3
- Helium Weight = 18.4 kg
- Superfluid Plug & Boiler

Helium Controller

- 3-Axis Proportional Valves & Thrusters
- Outer Body Controller
- Backup to N_2 System

S/C Control

- Nitrogen System & Crude Gyros for Acquisition and Night Time
- Always Points Roll Axis at Sun

Power

- 4 Solar Array Panels ~0.28 m^2 each
- ≈60 watts Average Power Total
- NiCad Batteries for Night Operation
- 28 VDC

Communications

- VHF Up & Down Links
- Tape Recorder for Data Storage

Dewar Instrumentation

- Helium Location
- Temperature Probes
- Superfluid Film Thickness
- Helium Flow Rates

Control Instrumentation

- Accurate Sun Sensors

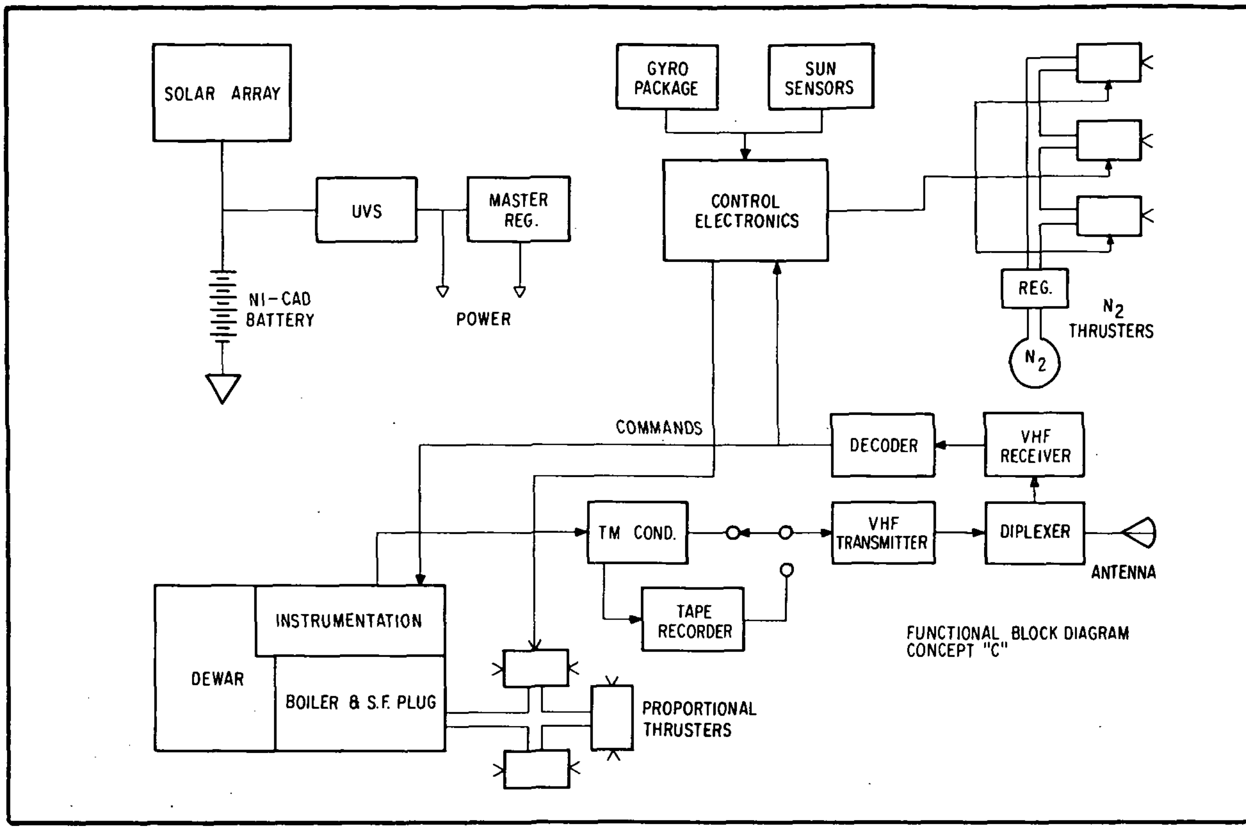


Fig. 4-5 Concept C: Functional Block Diagram

<u>Dewar Description</u>. The dewar recommended for this test would be essentially the same as the dewar of Concept B, and would have similar geometry to the SRS dewar. It would hold approximately 18 kg of helium in an annular chamber surrounding a 0.25 m diameter central cavity. The cavity is large enough to hold the gyroscope assembly needed for the next flight. The neck area of the dewar would be designed to accept the gyroscope package and its associated vents and spin-up system.

Six fiberglass bands would support the inner container while in the orbital condition. Six rigid titanium, retractable supports would be used in addition to take the launch loads.

Both the superfluid plug and the boiler venting systems would be incorporated, with valves to select the venting system being tested. The dewar would also contain heaters to allow conversion to normal helium late in the test program.

Honeycomb-like cells similar to those suggested for the SRS dewar would be incorporated in the liquid helium space, to provide a helium management scheme.

Dewar Instrumentation. For the most part, the instrumentation needed for the dewar flight test must be developed. In some cases, extensions of standard laboratory techniques can be used and in others new methods must be devised. In both cases, development is involved and the task is not trivial.

In order to instrument the venting characteristics of the superfluid plug and the boiler, a technique must be devised to measure mass flow rates in the range of a few milligrams of helium per second without causing temperature fluctuation within the dewar. Further study and development is required to devise a system which would work well in an orbital environment.

Differential temperature measurements to 0.001° K must be made to evaluate the superfluid plug and to determine the temperature distribution throughout the dewar. Relatively standard laboratory techniques can be adapted to this problem. Absolute temperature measurements must be made to 0.01° K. This may take some development.

Determining the location of the helium liquid, gas and creeping film and measuring the thickness of the film in a zero "g" environment will be difficult. Methods normally used for cryogenic fluids with the thin superfluid film on Earth may have difficulty distinguishing between liquid and vapor in orbit because of much thicker creeping superfluid film which covers the sensor. A new sensor or set of sensors must be developed to solve this problem.

4.2 GYROSCOPE FLIGHT TEST HARDWARE

In Section 2.2 we determined that the important orbital gyroscope tests are:

- Gyroscope suspension
- Gyroscope drift
- Gyroscope spin-up
- Gyroscope spin speed control
- Gyroscope spin axis orientation
- Gyroscope readout

The test flight described in this section provides a realistic environment for determining all the above parameters, except that gyroscope accelerations might be a factor of 10 larger than for the final flight. This is true because:

- The test flight is not drag free
- The gyroscopes are not constrained to lie close to the C.G.
- Control accelerations are not as tightly constrained as for the final mission

As a result of these additional accelerations, the drift rate of the gyroscope might be considerably greater than expected for the final flight. Because this flight takes place early in the gyroscope development, a higher drift rate might also be expected due to the state of development of the gyroscope. It would be considered a successful test if the gyroscope could be shown to have drift characteristics under these flight conditions of 1 arc-second per year (~0.1 arc-second per month) and with 0.1 arc-second per year as a goal. Such performance would be approximately five orders of magnitude better than an exceedingly fine inertial quality gyroscope on earth. The instrumentation described for this test would have the sensitivity to detect drfits of 0.01 arc-second. A two month flight is suggested.

A completely successful test flight would show:

- Gyroscopes can be spun up in zero "g".
- Gyroscope spin axis can be controlled to 10 arc-seconds in orbit.
- Gyroscope spin speeds can be adjusted.
- Drift could be shown to be less than:

  0.1 arc-second per year (0.01 arc-second per month) maximum sensitivity required for test.

- Instrumentation loop is feasible.
- AGC loop is feasible.
- Rolling the satellite is or is not effective in reducing gyro and/or instrumentation drift.

- Proportional gas control system for outer loop is feasible.

4.2.1 System Concept

The system suggested for performing the gyroscope flight test is diagrammed in Fig. 4-6. Three gyroscopes are mounted on a quartz block with their spin axes co-linear and along the roll axis of the satellite. The block and gyroscopes are rigidly mounted within a cavity of the same dewar design as was used in the dewar test flight. The same spacecraft control system is used as in the dewar flight with the addition of a roll-reference star tracker and an offset solar detector. In this case, the proportional valve control system using boil-off helium would be the prime system, the nitrogen system being the backup. One gyroscope would be used as the prime reference to control the spacecraft. Drift of the other two gyroscopes would be measured with respect to the prime gyroscope.

Once one of the gyroscopes becomes the reference for the spacecraft, the spacecraft is inertially stabilized. In order to take advantage of a sun-oriented solar array, we can choose the initial inertial orientation so that, during the two-month period of the test, the sun travels from 30° on one side of normal to the array to 30° on the other side, passing across the front of the array. This will result in only about 15 percent variation of the power input to the array over the life of the mission.

The sequence of operation would be as follows:

1. Separate from launch vehicle.

2. Reduce tip-off rates using rate gyroscope reference.

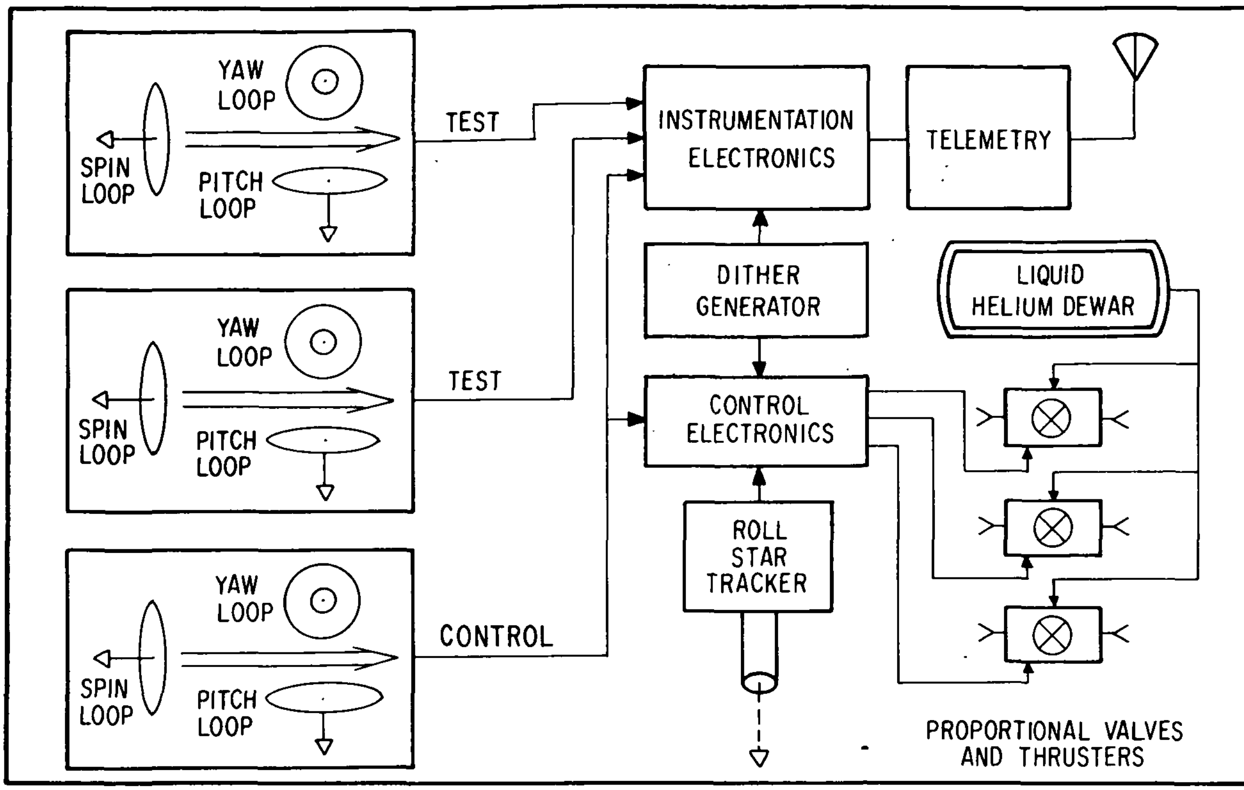


Fig. 4-6 Block Diagram, Gyroscope Flight Test System

3. Acquire the sun using coarse and fine sun sensors.

4. Roll to find Canopus near the ecliptic pole.

5. Offset from the sun approximately 30° in the ecliptic plane.

6. Spin up the control gyroscope while locked on the offset sun and Canopus.

7. Transfer pitch and yaw control to the control gyroscope.

The satellite is now inertially stabilized to the gyroscope reference in pitch and yaw. Roll is stabilized toward the star Canopus. When Canopus is occulted, the roll rate gyroscope is used as a reference.

Spin-up of the other two gyroscopes begins now. The spin speed is to be adjusted to the speed of the first gyroscope, to 1 percent and the spin axes are to be aligned with the first gyroscope to 10 arc-seconds. The method presently proposed to control the spin axis alignment is to roll the spacecraft about the control gyroscope spin axis during spin-up of the other gyroscopes. In order to have the spin rates identical, spin-up must be terminated at the appropriate time or the spin rate slowed down later by introducing some gas into the spin-up chamber. The choice between these two methods has not been made.

The next sequence of steps is as follows:

1. Roll spacecraft about control gyroscope spin axis at approximately 1/10 rpm.

2. Spin up the two test gyroscopes.

3. Stop roll, reacquire Canopus.

4. Measure gyroscope spin rates.

5. Measure orientation of spin axes.

6. Adjust gyroscope spin rate.

7. Null the readout loops.

The instrumentation loop is shown in Fig. 4-7. The method is essentially the same as in the final experiment where the telescope and gyroscope signals are continuously subtracted. In this case, the signals from the control gyroscope and the test gyroscope are subtracted to determine the drift of one with respect to the other. Since the spacecraft is controlled to approximately

10 arc-seconds and drifts to be measured are as small as 0.01 arc-second, the gains of the two channels must be matched to 0.1 percent in order that the subtraction of two large numbers does not give an error greater than 0.01 arc-second. This is done by introducing a deliberate sinusoidal dither into the control system. The signals resulting from the dither, as seen by the two gyroscopes, are subtracted and synchronously demodulated. The error signal is integrated to remove noise and is used to change the gain of the test gyroscope AGC amplifier until the two signals are matched.

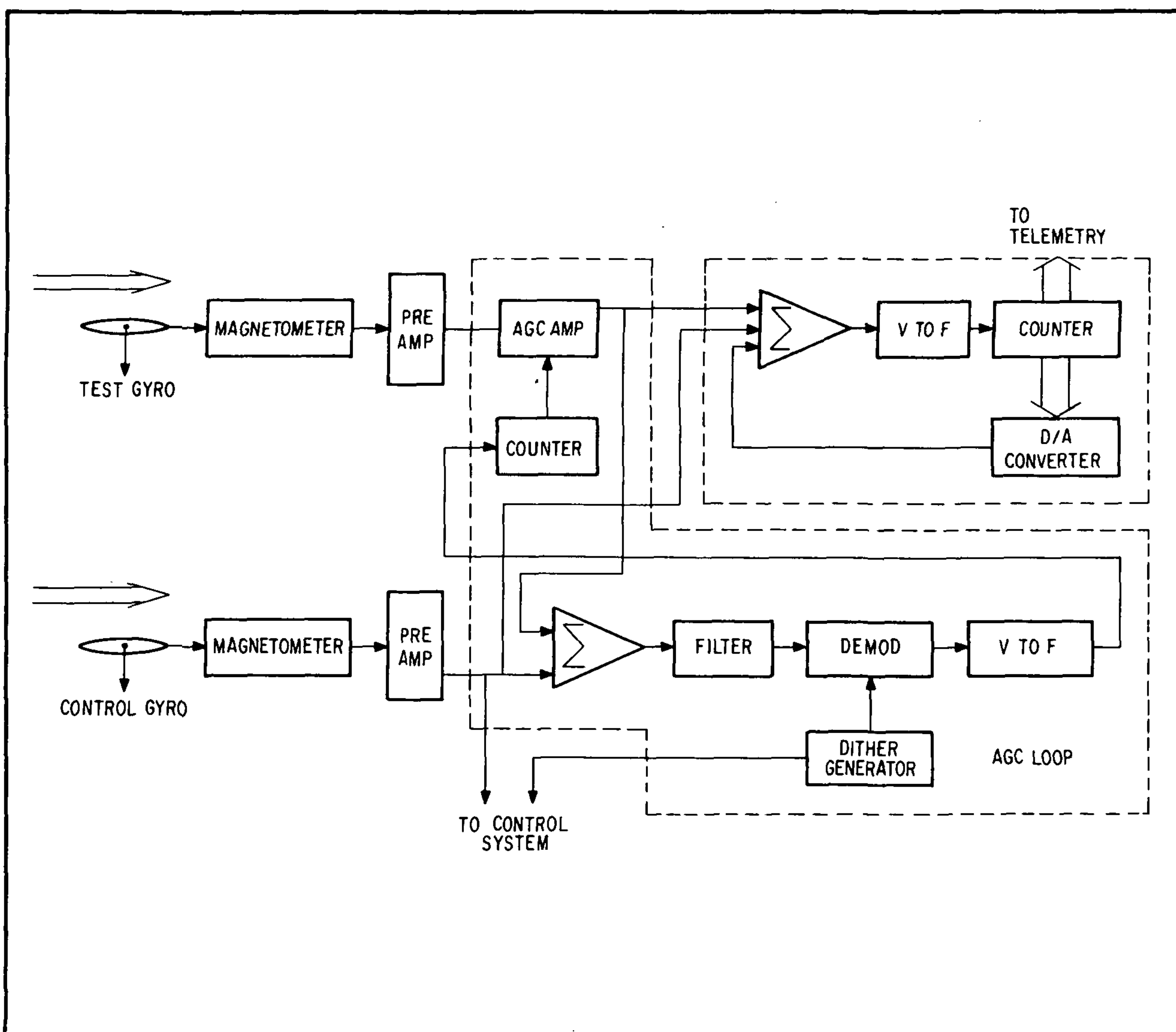


Fig. 4-7 Typical Single-Channel Instrumentation

With the gains matched, the outputs of the two gyroscopes are compared in the instrumentation loop. The resulting difference signal is integrated to reduce the noise, and the drift signal is telemetered in digital form.

After data has been taken for a month with the satellite stabilized in roll, the satellite will be rolled slowly about the control gyroscope spin axis to allow averaging of the suspension dependent torques as is planned for the final flight. This test will determine how rolling improves the gyroscope drift characteristics.

Section 5
PROGRAM PLAN

The program for the engineering flight test of the critical components of the SRS program supports the overall program in a way that allows a modest expenditure of funds to obtain the vital data required for an orderly conduction of the main program. The flight test program must start near the beginning of 1972 to assure a launch of the final experiment by mid-1976. The flight test being proposed must be a very closely coordinated operation between Stanford University and the contractor for the flight hardware. Stanford must supply both scientific support to the program and many of the test articles that are to be flown. The program described here assumes that Stanford supplies the superfluid plug, the gyroscopes, and the suspension system, their instrumentation, proportional valve and nozzle design and major scientific input to the design. It is assumed that a spacecraft contractor will provide the dewar, the spacecraft and all integration, test and launch support. A schedule for the flight test program is shown in Fig. 5-1.

5.1 DEWAR FLIGHT TEST

On the schedule, the commitment to the dewar flight is shown as January 1, 1972. At this time, it is important that the launch vehicle be chosen so work on the truss can be started and so proper coordination with the appropriate program office can be maintained.

Work would start immediately to generate the engineering data and to design a superfluid plug for the first dewar. This effort should take place at the Stanford University Hansen Laboratories under Drs. Fairbank and Everitt, where the plug was invented. The result should be a flyable plug which can be delivered to

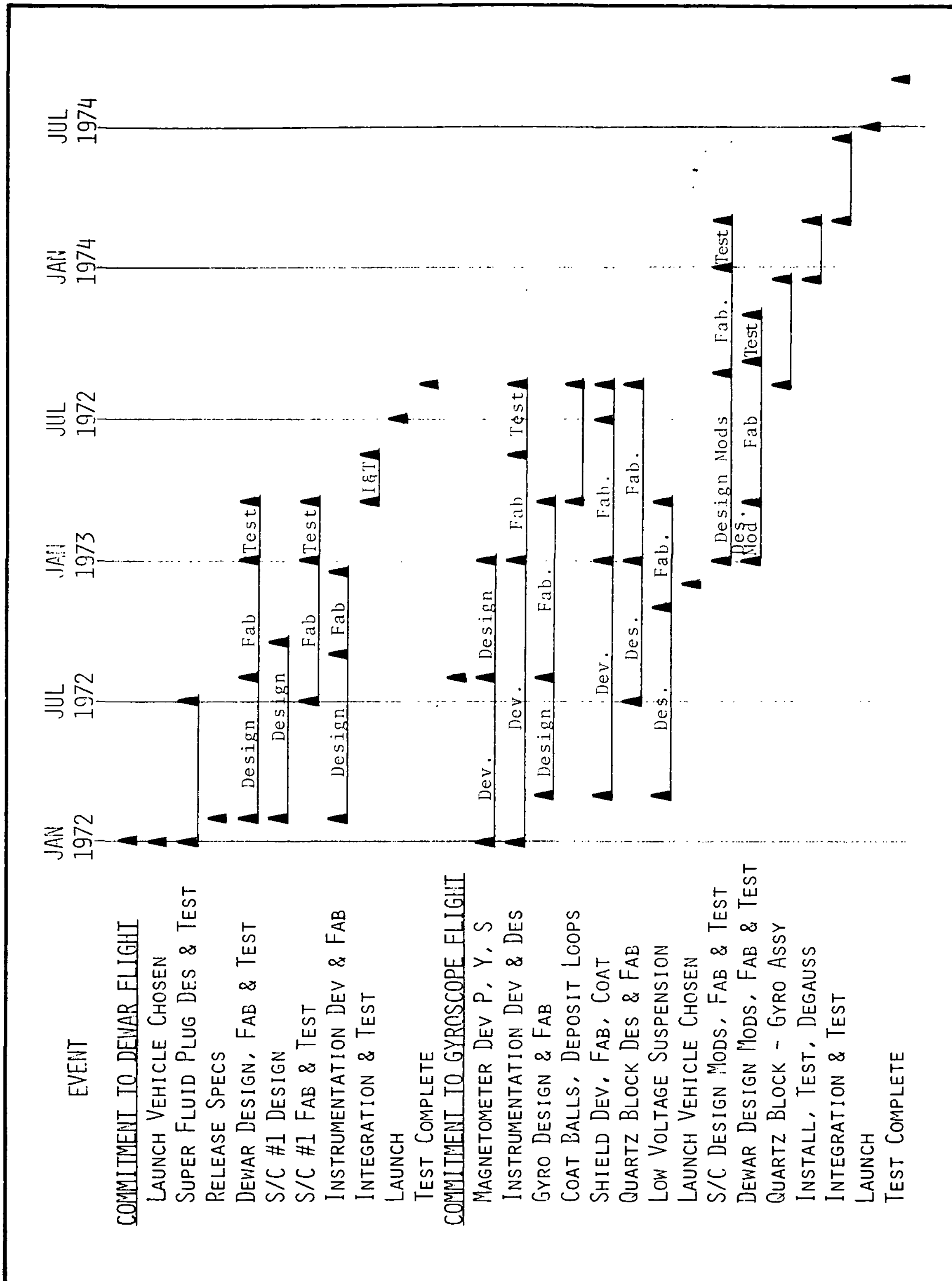


Fig. 5-1 Flight Test Program Schedule

the dewar designer for inclusion into the flight dewar. This effort should take approximately six months.

At the same time, work on specifying the systems and subsystems will be initiated. After a month, the specification will be issued to allow design of the spacecraft and dewar to proceed. Both the dewar and the spacecraft must be designed with the gyroscope test flight in mind. Development and fabrication of these two articles is expected to take approximately one year.

Along with the development of the dewar and spacecraft, the instrumentation for determining the boil-off rate, the location of the helium liquid and vapor, the creeping film thickness and the temperature distribution must be developed. This is a very important task, since the whole purpose of the flight is to gather the data to be generated by the instrumentation.

Testing of the dewar and the spacecraft separately would take approximately three months. This would be followed by a two-month integration and test phase. One month is allowed for launch operations. The flight test would be over in one to two months.

5.2 GYROSCOPE FLIGHT TEST

Work must start on getting ready for the gyroscope flight test at the same time as work begins on the dewar flight test. This work is preliminary to a major commitment of funds for the gyroscope flight test.

Development of a flyable magnetometer for this test is one of the long lead items that can start immediately. Design and development of the actual flight gyroscope and its magnetic shield should start as soon as there is enough data from the laboratory spin-up tests to proceed with confidence. Development of the low-voltage flight suspension system should start at about the same time.

The real commitment to the gyroscope flight comes about July, 1972, when funds must be made available for the fabrication of the gyroscopes. By this time, there will be a considerable amount of laboratory experience with the existing gyroscope. By August, 1973, the gyroscope, quartz block and the necessary instrumentation are ready for assembly into a single package.

Work on the second spacecraft and dewar design does not have to begin before January, 1973. By this time, the first unit will be assembled and ready for subsystem tests and most of the problems will have been discovered. The modifications to the design include the addition of the roll star tracker, the offset sun sensor, and the accommodation of the experiment package into the dewar. Modification to the control system to operate from the gyroscope signals is also required.

By the time the quartz block and gyroscopes are assembled and tested, the dewar is ready for installing the instrument package and testing. This starts late in 1973 and extends into early 1974. Integration and test takes place in the second quarter of 1974 with the launch taking place on July 4, 1974.

5.3 PROGRAM PHILOSOPHY

The cost estimate for this program is based on a philosophy that treats the program as an engineering test. The goal is to obtain engineering data on some very sophisticated pieces of hardware, not to provide exemplary spacecraft design or data-handling techniques. Emphasis must be put on carefully designed tests, good instrumentation, obtaining the simplest supporting systems, and having a system which will not jeopardize the prime payload.

We assume that the program will be conducted by a small, highly skilled project team which will be with the program all the way. Documentation will be limited to that necessary to reproduce the hardware using a similar skilled team. Supporting systems will tend to be over-designed to reduce extensive analysis. It is also assumed that the prototype hardware will be flown.

Following this philosophy will result in a high probability of success at as low a cost as is reasonable.

Ball