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14 APRIL 1967

PRELIMINARY DESIGN STUDY OF SLAMAST SCOUT-LAUNCHED ADVANCED MATERIALS AND STRUCTURES TEST-BED

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RE-ENTRY SYSTEMS DEPARTMENT

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Prepared for

National Aeronautics and Space Administration
Langley Research Center
Langley Station
Hampton, Virginia

GENERAL ELECTRIC
RE-ENTRY SYSTEMS DEPARTMENT
A Department Of The Missile and Space Division
3198 Chestnut Street, Philadelphia 4, Penna.

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ABSTRACT

This report presents the results of a feasibility study of a small, lifting re-entry vehicle test-bed capable of being launched by a Scout launch vehicle and capable of being recovered. The purpose of the study was to determine if it was possible to conduct meaningful, sub-scale, thermostructural experiments involving panels of interest of representative full scale, manned, lifting re-entry vehicles. The study was based upon the HL-10 vehicle concept.

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INTRODUCTION

This volume of the SLAMAST Design Feasibility Report contains an overall summary of the work done and the conclusions reached during the study. Since the study was composed of three major substudies, this summary is organized accordingly. Hence, summaries are presented for (1) Experiment Feasibility, (2) Design Feasibility, and (3) Program Considerations.

EXPERIMENT FEASIBILITY

Introduction - This section of the report describes an approach to thermostructural similitude for lifting re-entry vehicles in general, with specific emphasis on the HL-10 vehicle. Analysis to date shows that an application of margin of safety philosophy combined with a utilization of traditional similarity parameters, test panel designs, and trajectory shaping holds promise of resulting in useful and valid simulation of full scale MLRV thermostructural systems with an economical subscale vehicle. In addition, it is shown that in the process of flying a thermostructural similitude mission, valuable data can be simultaneously obtained concerning aerodynamics, thermodynamics, heat shield performance and flow transition.

An approach to the simulation process is presented which permits more flexibility than the usual method involving the duplication of a large number of dimensionless parameters related to an idealized thermostructural situation. This approach is discussed in detail along with its application to the HL-10. Also presented is the numerical substantiation that could be generated during the study period, as well as consideration of the limitations of the simulation technique and a comparison of ground test capabilities versus flight test.

It is believed that the approach developed is both sound and workable. The work done during the study period has not yielded any information to the contrary. The great flexibility in environment generation potentially available with the SLAMAST vehicle tends to reinforce the judgment. However, the numerical analyses which would be generated during the span of this study are not extensive enough nor complete enough to permit an unequivocal statement of feasibility. Regardless of the foregoing, it is important to note that a test bed of the type analyzed is not and should not be limited to thermostructural testing alone and, as stated earlier, significant testing can be accomplished in the areas of aerodynamic,thermodynamics, heat shield performance and flow transition.

Approach to Thermostructural Simiiltude. - The design of a structure, whether it be a flight structure, a civil structure, or any other type of structure, involves three basic elements: (1) the environment to which the structure will be exposed during its useful life, (2) the characteristics of the materials being employed in the design (i.e., the material properties), and (3) the analysis techniques used to predict the structure's performance, behavior, or response. The main concern here is, of course, flight structures. The primary items which contribute to each of the three elements for flight structures are shown in Figure 1.

The next point to consider in the development of the simulation approach is the consideration of testing. Tests are performed primarily for the purpose of proving that the structure will not fail in its anticipated environment, or that it does not have excessive strength in its environment. In other words, it should be neither over-designed nor underdesigned.

In the development of manned lifting entry vehicles, there are questions regarding environment, material characteristics, and analysis techniques, hence, the need for testing. While the primary purpose of SLAMAST is thermostructural response simulation, it is impossible to completely isolate this from the environmental and material characteristic considerations. The similitude philosophy developed here takes advantage of this fact, and it will be shown that with the use of the SLAMAST flying laboratory, some verification of environment and material characteristics will be obtained as part of the thermostructural similitude.

The specification of structural sizes, thicknesses, stiffnesses, etc. depends on a knowledge of all three of the basic elements. A deficiency in any one of these areas could result in a structure which is incapable of withstanding the environment or one that is overdesigned and, hence, overweight. Testing with proper instrumentation will reveal these deficiencies, if they indeed exist. Testing, therefore, becomes a job of demonstrating the accuracy with which environments (pressures, temperature distributions, loads) can be predicted, material behavior in the presence of that environment is known, and structural response in the presence of both these can be predicted.

The final design of a structure is specified in terms of detail drawings of the structure and a structural analysis of the design. This structural analysis includes a consideration of many or several types of potential failure modes such as:

(1) Buckling
(2) Thermal stress
(3) Mechanical stress
(4) Excessive deflection or strain

These various structural behaviors need investigation in all the possible flight regimes including:

(1) Launch
(2) Abort
(3) Space flight
(4) Re-entry

a. Nominal
b. Overshoot
c. Undershoot

Approach to touchdown. - Once the structure has been designed, its capability can, and usually is, presented in terms of margins of safety. A margin of safety is given for each potential mode of failure. The margin of safety is defined as

$$\text{M.S.} = \frac{\text{Allowable stress (or strain, etc)}}{\text{(Actual expected stress)} \times \text{F.S.}} - 1$$

The factor of safety is a number which the expected stress (or load or strain, etc.) is multiplied by to account for variations in material quality and manufacturing variances. There is a factor of safety for yield (usually 1.0) and an ultimate factor of safety (usually 1.4 or 1.5 for manned vehicles). Design criteria, by definition, essentially says that no yielding shall exist at yield load and no failures (breaking or buckling) shall exist at ultimate load.

If the margin of safety for each potential mode of failure is plotted as a function of time, a curve similar to the one shown in Figure 2 would result. If the curves correspond to the modes discussed above, it is seen that the minimum margin of safety (the critical mode of failure) or the "weak-link" in the structure is buckling (point 1) at time t_1. This is a prediction based on analytical or empirical techniques. It tells us that based on our knowledge of the environment, based on our knowledge of material properties, and based on our knowledge of structural response, the structure is closer to buckling than to any other mode of failure. However, since the margin is positive, no failure should occur.

Simulation Philosophy. - With these thoughts in mind, a simulation philosophy or approach was formulated. This approach consists essentially of reproducing, in an experimental panel on a subscale maneuvering vehicle, the minimum margins of safety of the full scale prototype at a particular time which will be called the experiment time. This is to be done in a flight environment sufficiently similar to the prototype environment that the same phenomena are encountered. It is most significant to note that the primary difference between this approach and the traditional one is that this is similitude from a practical engineering viewpoint rather than a completely theoretical viewpoint. The latter approach to the problem usually leads to the conclusion that anything short of one-to-one matching is unacceptable or at least highly questionable.

Exact similitude is impossible because aerodynamic, thermal and structural modeling do not have the same scaling laws. Some physical factors such as thermal contact resistance, boundary layer transition between laminar and turbulent flow, and structural details are impossible to scale, and the structural concepts being considered for HL-10 are very involved.

Even if perfect similitude (by a one to one correspondence of all parameters) was obtained, the distortions arising in an actual structure caused by attachments, resulting in thermal shorts or leaks and structural edge effects and differences in edge fixity, would negate, in any physical structure of this type, the identical similitude calculated analytically.

However, this in no way negates the utilization of a test bed, such as the SLAMAST vehicle, to obtain valuable and useful information for the HL-10 and other lifting re-entry vehicles of this class. SLAMAST can be used to confirm or correlate the three basic elements of design; i.e., the environment, the material characteristics, and the analytical and design tools. In particular, it can provide:

(1) additional knowledge about the aerodynamic environment associated with lifting re-entry vehicles.

(2) a means to check the adequacy of present analytical and empirical methods of analysis for pressures, transition points, flow separation, and effectiveness of control surfaces.

(3) additional knowledge about the thermal environment, and the heat transfer to and through the vehicles, including ablation rates.

(4) a means to check the adequacy of present analytical and empirical methods of thermal analysis for convective heat transfer, ablation, conductive through complex structures including honeycomb sandwich, and multilayer insulation.

(5) a means of checking structural methods of analysis and design procedures for complex structures of this type subjected to thermal loads, in-plane loads, and lateral pressure loads.

(6) a means of checking the actual performance of material systems under actual flight conditions, and comparing them with data obtained by laboratory tests, including the ablation and bonding materials.

To insure that valuable information is produced which is applicable to prototype lifting re-entry vehicles:

(1) It is necessary that SLAMAST have a sufficiently similar environment to include all phenomena that will exist in the flight of a prototype lifting re-entry vehicle.

(2) It is most desirable for the structural test elements of the SLAMAST to use the same materials as the prototype elements on the full scale vehicle.

(3) It is necessary for the SLAMAST test components to employ the same structural concepts as the corresponding prototype subsystems.

(4) It is necessary that the test elements represent the same thermostructural response characteristics.

(5) The identical factors of safety used in the design of the prototype should be used in the design of the model test elements.

(6) The SLAMAST test section must be designed to have the same critical mode of failure as the prototype section.

(7) The margin of safety for the critical loading condition of the prototype section should be identical to the margin of safety of the test element under the same loading condition. The higher margins of safety should be reproduced in the same proportions as in the prototype to a degree dependent on their closeness to the minimum margins to accommodate interaction effects.

This final requirement needs some additional discussion since it is an important key to the whole approach. If the minimum margin of safety is 0.01, for example, and the next lowest one is 0.5, again for example only, this clearly indicates that the mode of failure represented by the 0.01 is certainly the weak link with all other potential failure modes far removed from possibility. This would be the case unless there are major inaccuracies in the environment, material property, or structural response predictions. If on the other hand, the minimum margin is 0.01 and the next lowest is 0.015, it could be most important to reproduce both these margins in the test panel. The reason for this is that even though the mode of failure represented by the 0.01 margin is predicted to be critical, in reality the mode represented by 0.015 could be the critical one due to tolerances on all factors going into the prediction process. This is shown in Figure 3. It is seen that the tolerances could act in a way to make mode "a" the actual critical mode, whereas mode "b" was predicted to be the weak link. Therefore, the goal in trajectory and test panel design will always be to reproduce, at SLAMAST experiment time, all the HL-10 margins of safety. But since this is an extreme condition to insist upon, it will be sufficient to reproduce only the minimum margin and others sufficiently close to the minimum as to represent potential critical modes of failure.

Another important factor which must be considered in the test design is that the margin of safety for the critical mode in the test panel reaches its minimum value at the experiment time, and that no other margin reaches a value at other flight times which could possibly fail the test panel prematurely.

This philosophy will derive the greatest value from SLAMAST and insure that the prototype vehicle will accomplish its mission. By this procedure, the SLAMAST will uncover, by suitable instrumentation, any unsatisfactory knowledge of the environment, any deficiencies in analysis, design, or material behavior, and any unforeseen phenomena. In other words, by the SLAMAST flight profile simulating as nearly as possible the critical aspects of the prototype flight profile, by using the same materials and the same structural concepts, and by forcing the model structural element to have the same critical mode of failure, with the same factors of safety and the same margin of safety, the most important elements of thermostructural simulation are achieved, without the complications of less important considerations.

The test panel designed with this philosophy will be equally close to failure as the full scale prototype. This is perhaps a unique feature of this approach since many models are designed such that failure thresholds are far removed from expected loading levels.

Even though a large number of the critical conditions for the HL-10 structure occur at approach to touchdown, or abort, through the use of the philosophy stated above, the SLAMAST can study these critical conditions although it has no abort or touchdown considerations itself.

Test Panel Design and Trajectory Shaping. - As discussed above, the goal in the design of the test panel and the shaping of the trajectory is to arrive at a situation, at a specific SLAMAST flight time (the "experiment time"), such that the margins of safety against the various modes of failure are all the same as the margins in the HL-10 or in whatever prototype is being simulated. It is not expected that this can be completely accomplished since this would be perfect similitude, most difficult, if not impossible to achieve. It is necessary only that the minimum one, or ones be matched. To assist in shaping the trajectory to reach the desired experiment conditions, the classified similarity parameters germain to rectangular panels undergoing small deflection are used. These parameters deal with buckling, yielding, fracture, thermal and mechanical stress, extensional and flexural stiffness, etc. which are representative of the type structure under investigation.

The procedure for designating the SLAMAST trajectory and panel design to simulate on HL-10 panel design is demonstrated in Figure 4.

It is conceivable that one flight of the SLAMAST vehicle would contain more than one experiment. By carefully designing the trajectory, it is possible that one critical load condition for the prototype can be simulated at one time in the SLAMAST flight while other conditions are being simulated at other times. The feasibility of achieving this is further enhanced due to the fact that there are two thermostructural experiment locations on SLAMAST, namely, a windward panel and a leeward panel.

Simulation of the prototype entry environment with the SLAMAST vehicle must include operating the SLAMAST vehicle in an entry corridor that results in heat transfer rates comparable to those anticipated for the prototype. Figure 5 summarizes the peak heat transfer rate distribution predicted for the HL-10 vehicle for the various design trajectories. Superimposed upon Figure 5 is the predicted SLAMAST environment for entry velocities of 20,000 fps (6.58 km/s) and 25,000 fps (8.22 km/s) and entry path angles of 1 and 10 degrees dfh. Note that the environment of the SLAMAST vehicle is comparable to the HL-10 environment ranges for the complete range of path angles at V_E = 20,000 fps (6.58 km/s). However, at an entry velocity of 25,000 fps (8.22 km/s) the SLAMAST environment is well above that of the HL-10 and well outside the efficient operating range of typical low density ablators. Hence, it appears that an entry velocity of 25,000 fps (8.22 km/s) provides much too severe an environment at the steeper path angles. In addition, note that for an entry velocity of 25,000 fps (8.22 km/s), the SLAMAST heat transfer environment falls well above the HL-10 environment, in a regime where most low density ablators exhibit relatively poor performance.

In addition, the effect of guidance errors must be considered. A nominal tolerance of $\pm 3/4^{o}$ on entry path angle exists. Figure 6 illustrates the effect of entry path angle and velocity on maximum stagnation heating rate. Since the local heating distribution is proportional to the stagnation heating, similar trends with path angle will exist for both the stagnation and local body points. Note that the variation of maximum heat transfer rate with entry path angle is much smaller for the 20,000 fps (6.58 km/s) entry velocity than for the 25,000 fps (8.22 km/s) case. Hence, guidance errors would introduce less variation in local heat transfer for the 20,000 fps (6.58 km/s) case than for the 25,000 fps (8.22 km/s) case and thus make the 20,000 fps (6.58 km/s) entry case a more desirable one for experimental purposes.

The environmental simulation technique for trajectory definition requires two major inputs:

(1) Histories of those environmental parameters which must be matched to satisfy a particular experiment.

(2) Closed form expressions for the instantaneous local values of the environmental parameters.

The histories of requirement (1) may be a function of velocity or time. Two or more parameters may be treated simultaneously; however, solution time increases exponentially with the number of simultaneous parameters. When multiple parameters are considered, weights reflecting the relative importance of each must be assigned to the parameters. The closed form solutions of requirement (2) may, in general, contain terms which are tabular functions of other variables. With these inputs plus an initial trajectory to start the iteration, the ODC (Optimum Discrete Control) digital program can be used to generate the SLAMAST trajectory which optimizes the weighted match with the experiment parameters. The computation time required to iterate to a solution is a function of the initial trajectory and can be reduced by using parametric studies to provide a judicious initial trajectory profile. One important environmental parameter which is useful in simulation studies is heat rate. Figure 7 shows integrated stagnation heat rate for the initial SLAMAST trajectory and the eleventh iteration made by the ODC program in attempting to match the desired stagnation heat history indicated. Further iterations could be made to improve the match, if necessary, since the program was still converging on the desired heating history at the eleventh iteration.

Other SLAMAST benefits

Boundary layer transition studies: Magnitude of the local heat flux is strongly influenced by the state of the local boundary layer, i.e., laminar or turbulent. Hence, an optimum lifting entry vehicle design is influenced by the boundary layer transition criteria employed.

A study of available ground test transition data on SLAMAST-type elliptical cones at an angle of attack was undertaken to determine whether existing sphere-cone transition prediction techniques, can be applied to an elliptical configuration. It was concluded that because of the paucity of transition data on elliptical bodies, a valid comparison with sphere-cone transition data could not be made. Therefore, no method is currently available by which a logical extension of sphere-cone correlation techniques will allow qualitative estimates of the expected conditions under which transition will occur on an elliptical vehicle at an angle of attack.

An additional application of the SLAMAST vehicle would be to measure the onset and propagation of boundary layer turbulence as it is influenced by the various combinations of flight parameters like angle of attack, surface temperature, mass injection rate, local Reynolds number, and local Mach number. This understanding of boundary layer transition can then be logically extended to application on the larger prototype vehicles with the subsequent optimization of the low density ablator heat shield.

Materials performance: Over the past several years, ablation performance anomalies have been noted in both low and high density materials. Several of the anomalies are described below:

(1) Surface sealing of silicone elastomers observed in Langley 2500 KW ARC - leads to uncontrolled swelling.

(2) Boundary layer combustion observed on phenolic nylon at Hi oxygen partial pressures.

(3) Boundary layer combustion observed on low density filled epoxy in honeycomb (Avcoat 5026-39-HC G) in low heat flux regime.

(4) Significant surface recession occurs on low density filled epoxy in honeycomb (Avcoat 5026-39-HC G) for low pressure, low heat flux, low aerodynamic shear conditions due to combustion of pyrolysis gases at char surface.

As more emphasis is placed on optimization of the low density class of ablators, a good understanding of the in-depth chemical reactions as well as the aerothermochemical interactions of the injected decomposition products with the hypersonic boundary layer is required. The complex chemical reactions are dependent on several parameters of which temperature and pressure are of key importance. Internal pressure buildup is directly a function of the molecular weight and rate of the gas generation, and the char layer porosity. The char porosity is very dependent on time; hence, only a true simulation of a time history of the prototype environment will result in an accurate duplication of the in-depth and surface chemical reactions. SLAMAST will be an excellent tool to help illuminate these materials behavior phenomena.

Aerodynamic Flight Experiments: SLAMAST class test beds provide excellent facilities for investigation of aerodynamic characteristics in a true re-entry environment not otherwise obtainable in present ground test facilities. The SLAMAST flight investigations will provide aerodynamic data of considerable interest in the hypersonic-low-density flight environment over a large portion of the trajectory. Although these data may be of limited scope (depending on the primary experiment), they will include control effectiveness (derived from trim angle of attack as a function of control deflection), associated control loads, control duty cycles, stability characteristics, and lift and drag performance. These data are of importance to lift and drag performance. These data are of importance to lifting vehicle technology and will be correlated with available ground test and analytical results. SLAMAST flights specifically oriented to provide aerodynamic information offer even greater opportunities for significantly advancing aerodynamic technology.

Other testing alternatives. - At present, there are only two approaches available to verify or validate a design of a manned lifting re-entry vehicle, or for that matter, any flight system. These are flight testing of a prototype vehicle or individual ground testing of critical items.

Although the actual flight testing of a prototype vehicle under its associated real flight environment does provide the most definitive data, the costs associated with these design verifications are usually extremely high. The second approach, namely ground testing, while reduced in overall cost, does however only provide a limited amount of data on singular phenomena (i.e., the ground test results usually do not include interaction effects, which are quite important in total structural response). There are basically four types of facilities in which structural re-entry environments simulations can be performed. These are: (1) large conventional wind tunnels; (2) arc heated tunnels, (3) rocket exhaust, or (4) pressurized chamber with radiant heat sources.

Few wind tunnels are available which are large enough to accommodate test panels of the sizes proposed on SLAMAST {about 14 inches (.35 meters) by 16 inches (.42 meters)}. None of these facilities can provide experiment times available on a flight vehicle such as SLAMAST.

At the present time, there are no arc heated facilities which are capable of operating for long times with a large flow field, nor, are there any planned for the near future.

Rocket exhausts such as those at the Malta facility are generally used for thermostructural tests. However, for the conditions being considered, they are not acceptable because the maximum flow field is limited to 15 inches (.38 meters) and the heat fluxes and pressures will be much too high for a lifting vehicle environment.

The remaining choice would be to place the test panel in a chamber designed that a differential pressure could be applied to the panel, with radiant heat sources providing the necessary thermal environment. If this were possible, which it is not today, it would be a rather expensive process.

The facilities described above would not be capable of varying the heat flux or pressure during testing, which would more closely simulate actual flight conditions. This effect could be minimized if the facilities had enough latitude on environmental control, i.e., such that those environments producing the minimum margins of safety in the critical modes of failure of the test panel could be simultaneously achieved. It is essential that the thermostructural response of the test panel is not seriously impaired by producing artificial environments either too quickly or at unacceptable rates. The entire discussion in the philosophy section highlights the goal of matching as closely as possible the similarity parameters during the earlier flight times so as to approach the "critical conditions" in the same way as in the prototype vehicle.

The real limitations of ground testing lies in its inability to combine simultaneously the effects of aerodynamics, variable heating rates, ablation phenomena, inertia loading and dynamic loading which can only be achieved in flight. It is particularly important when the design margins of safety for several of the most critical failure modes are close. The interaction of these failure modes under a combined environment may seriously affect the thermostructural response of the test specimen.

It is during these combined environments that unforeseen events occur; events that were not predicted on the basis of ground tests, which at best could only combine two parts of the flight environment, and not at the desired level, at that. For example, only after flight testing in which aerothermal effects were included was it found that the fuselage and tail surfaces of the X-15 research airplane had to be reinforced. It was determined that the vibration problem exhibited by these surfaces was compounded by thermal stresses that were induced simultaneously by other factors.

Flight tests are also extremely useful in uncovering unrelated but significant phenomena. On the GTV program, which was primarily aimed at evaluation of ablative materials characteristics in re-entry environments, the phenomenon of roll resonance was observed. This completely unpredicted phenomenon, it was found, can seriously affect the total performance of the re-entry system unless special precautions are taken. SLAMAST flight tests, aimed at simulating as closely as possible the actual flight conditions and environments of the prototype manned vehicle could well reveal phenomena, heretofore unknown, since the flight regime associated with lifting re-entry vehicles and their associated behaviors are still in the early stages of investigation.

DESIGN FEASIBILITY

This section of the report summarizes the system design effort performed during the SLAMAST study.

It includes a general description, Aerodynamic, Flight Dynamics, Thermodynamics, Instrumentation and Recovery Studies as well as subsystem trade-off studies, subsystem specifications and preliminary hardware definition of the following subsystems:

(1) Attitude Measurement and Control

(2) Instrumentation and Communication

(3) Recovery

(4) Electrical Power and Distribution

(5) Separation

(6) Structural Design and Packaging

In addition, an Integrated Ground Test Plan and general descriptions of the mechanical and electrical Aerospace Ground Equipment (AGE) are presented along with a program description and detailed schedules

The proposed system is the result of several interations. As originally conceived, it was thought the re-entry velocities of 25K fps (8.22K m/s) or greater would be required to obtain environmental similitude with HL-10. This belief necessitated the use of a 4 stage Scout system and required a spacecraft length of approximately 48 inches (1.46 meters), the maximum which could be accommodated within the Scout shroud.

Subsequent progress in the trajectory analysis resulted in an established requirement for re-entry velocities of 20K fps (6.58K m/s) or less. This allowed the use of a 3 stage Scout booster and relaxed the geometric constraints so that it was not necessary to use a 48-inch (1.46-meter) spacecraft. This, in turn, permitted a design consideration of spacecraft length as a function of approximate historical packaging efficiency using non "tailor-made" components.

Although this was not a design optimization effort, the "test-bed" philosophy dictated a preliminary consideration of the costs involved and a reasonable effort to keep these costs compatible with the technical program objectives. Final selection of the "feasible" subsystems for design purposes was, therefore, predicated upon economic viability as well as technical feasibility.

The "test-bed" philosophy also resulted in consideration of the inherent versatility of the as-designed configuration and of alternate component or subsystem selections which could increase specific spacecraft capabilities.

The SLAMAST spacecraft design chosen for feasibility demonstration is a modified elliptic cone vehicle with a length of 63 inches (2.07 meters) and a weight of 330 pounds (150 kilograms). The ellipse major axis is 28.04 inches (0.805 meter) and the minor axis is 16.04 inches (0.788 meter).

It incorporates two flaps; the top (or pitch) flap is servomotor powered for variable angle control, whereas the bottom (or drag) flap is driven by a pneumatic one-shot device to a constant angle. These flaps are faired into flat areas on the spacecraft. This reduces the flap pivotal point design and thermal insulation problems without significantly affecting the aerodynamic performance of the craft. The vehicle has a nominal $W/C_L S$ of 250 lbs/ft^2 (1232 Kg/m^2) with a maximum lift-to-drag ratio of 2.4 at 8 degrees angle-of-attack.

The selected structure is aluminum and is comprised of two integrally machined keel members which are stiffened by a series of webs and formers. This configuration provides integral hard points for separation and recovery subsystem attachments and is compatible with a removable access and test panel concept.

A mechanically attached graphitic nose tip and an ESM (GE-Silicone) heat shield are provided for thermal protection.

Figures 9 and 10 show the general configuration discussed above. Figure 11 is an inboard profile showing the internal packaging arrangement and Figure 12 shows the test panels. Tables I and II show the weight statement and preliminary component list for the spacecraft. The instrumentation and communication subsystem consists of a C-band tracking and ground command link, an S-band PCM continuous data transmittal link (utilizing a "micro-miniaturized" multicoder) and record and playback capability. It includes those diagnostic and performance sensors necessary to gather the data defined in the measurements list.

The recovery subsystem is a subsonic subsystem and consists of a FIST ribbon decel parachute, a ballute, and an rf beacon location aid. The ballute is ram-air inflated and provides inherent spacecraft flotation capability. (The drag flap mentioned above may be considered a part of the retardation sequence prior to parachute deployment.) This system has been sized to provide a spacecraft water impact velocity of $\leq$ 100 fps (32.9 mps).

The separation subsystem consists of four collet assemblies, an in-flight disconnect (IFD) and auxiliary hardware. The collets are "finger and piston" devices which mechanically attach the spacecraft to the spacer. They are pneumatically operated and, upon command, gas pressure pushes the pistons forward. This releases the mechanical attachment and, by continued piston travel, imparts a separation velocity to the spacecraft.

The electrical power and distribution subsystem consists primarily of harnesses, a power switching module, a battery, an in-flight disconnect (SC to spacer) and a spacer-to-shroud-access-door umbilical for ground power accommodation.

The selected attitude measurement and control subsystem derives its reference orientation from a Whittaker PRYS platform. It utilizes a programmed sequence for mission accomplishment. Exospheric, 3 axis reaction control and attitude orientation after separation is provided by a Freon-14 gas expulsion system. Upon re-entry, the pitch control (only) is relinquished to the top flap. Roll and yaw control are maintained through the Freon-14 system. The design trajectories were characterized by a constant altitude trajectory from pull-out to drag brake initiation and this is obtained by modulation of the pitch flap angle, roll nulling, and yaw rate limiting during flight in the atmosphere. At the end of the experiment, the drag and pitch flaps are used in conjunction with each other to begin the ballistic mission termination.

The experiment specimens consist of two panels; one on the top (leeward) surface, and another larger one on the bottom (windward) surface. The reference design for these panels was based on thermostructural modeling requirements.

A cylindrical spacer is proposed. This spacer provides the spacecraft-to-booster mating interface. It is designed so that the spacecraft-to-spacer mechanical attachment takes advantage of the inherent hard-points in the spacecraft. The spacer assembly also includes the mounting arm* of the umbilical for ground power access and the separation subsystem hardware previously described.

Capability. - As shown, the SLAMAST spacecraft will endure boost loads and will separate from the third stage of Scout at a timed interval from booster engine cut-off. It will orient itself at a predetermined pitch angle-of-attack (up to a maximum of 16 degrees) and will stabilize itself in roll and yaw within a ±5 degrees dead band.

It will re-enter, pull out of its initial ballistic path (by using aerodynamic force), and will re-establish the time reference based on the pull-out maneuver. (This compensates for booster burn out altitude variations with resulting error effects on separation time from launch and its potential perturbation of range.)

The angle-of-attack is modulated (through the programmed pitch flap control) to fly the predetermined trajectory.

*Modification of the umbilical mounting arm would permit the selected SLAMAST to be flown on 3 stage SCOUT without the shroud.

After the timed glide portion of flight, the spacecraft will begin its mission termination by deploying both flaps. After a set time from this event, the recovery sequence proper begins with the deployment of the FIST type ribbon parachute.

The spacecraft will survive water impact, at 100 fps (32.9 mps), and will subsequently float for 72 hours minimum. The rf beacon location aid will operate for a minimum of 10 hours.

During boost, re-entry, and glide, all information is gathered from the included sensors and is transmitted continuously over an S-band PCM system. During re-entry and glide, this information is also recorded. During mission termination (both flaps deployed) the continuous data transmittal and recorded data playback occurs simultaneously. The Electrical and Mechanical AGE required to handle, test, ship, and check-out the spacecraft has also been included in this capability.

Flexibility. - Boost and Re-entry: The boost phase design loads arose from the 3 stage Scout environments as reflected in the Systems Specification, and the proposed design will meet these requirements.

The design bounds on re-entry path angle were from γ_E = -1 degree to γ_E = -10 degrees. The craft is capable of re-entering with any γ_E between these bounds. It is also flexible enough to re-enter at either steeper or shallower path angles. The factors which would have to be assessed against a specific γ_E outside the -1 degree to -10 degree bounds are: heat shield, attitude sensing and reaction control impulse requirements.

Pull-Out: The pull-out altitudes specified from the feasibility demonstration point of view are predicated on a pitch flap initial deployment angle of 40 degrees and are dependent primarily upon γ_E (i.e. γ_E = -1 degree causes a pull-out at approximately 145K feet (44.2 Km) and γ_E = -10 degrees causes a pull-out at approximately 115K feet (35.06 Km). This attitude could also be varied by changing the initial pitch flap deployment angle ($\delta_F >$ 40 degrees for higher pull-out, $\delta_F <$ 40 degrees for lower pull-out). The factors which would have to be assessed against specific δ_F not equal to 40 degrees are: heat shield, structural strength, and spacecraft stability.

Glide: The pitch flap program chosen for feasibility demonstration provides for constant altitude flight through modulation of the angle-of-attack. This program could be modified to provide for angle-of-attack/altitude composite variations, etc.). The factors which would need to be assessed against a specific glide path control scheme are: pitch flap actuator capability, available electrical power, programmer modifications, heat shield requirements, spacecraft stability.

The roll control implementation scheme chosen for feasibility demonstration provides for roll nulling within a deadband of ±5 degrees (from separation through to parachute deployment). This could be varied by establishing a desired roll history program. For example the craft could be banked approximately 90 degrees, kept in that attitude for a specific time then reverse banked through 180 degrees and finally returned to null for recovery initiation. The primary factors influenced by implementing something other than roll nulling are: programmer changes and total impulse and thrust level changes. (A significant increase in available impulse and thrust level could be provided by using a hot-gas system. This system also requires significantly less volume for a given impulse than the Freon-14 system).

The level flight duration chosen for design feasibility is approximately 500 seconds. This can be varied by a simple timing change. The factors which would need to be assessed against a specific glide duration are: heat shield (thermal insulation) requirements, retardation effectiveness for recovery, and reaction control impulse requirements.

Recovery: All design trajectories employed the drag flap as a retardation device and showed subsonic, low dynamic pressure conditions at parachute deployment. If less high altitude retardation was desired for higher ballistic termination velocities, the subsonic recovery system could be elevated to a Mach 1.5 capability by changing to a hemisflo parachute with small weight and volume penalties.

Control System: The attitude reference system chosen for design feasibility is a modified MARS platform (termed PRYS by Whittaker Corp.). The mission accuracy predicated on the errors derived from this platform (and ancillary hardware) is presented in Section 4.1 of Volume II. Other reference systems are available which could, at some increase in expense, increase the total mission accuracy.

Redundancy. -

Data Transmittal: The design provides for both continuous real time S- band data transmittal and data recording and playback without interruption of the real time capability. This assures data gathering even if factors such as plasma attenuation or ground receiver availability prevent continuous reception of telemetered data.

The playback mode will be initiated by either of two events:

(1) Primary, the passage of a specified time from pull-out.

(2) Back-up, the receipt of a ground command through on-board C-band via the FPS-16 radar.

Mission Termination: The deployment of the drag/pitch flaps for mission termination initiation will be effected through either of two events:

(1) Primary, the passage of a specified time from pull-out.

(2) Back-up, the receipt of a ground command through on-board C-band via the FPS-16 radar.

Structure: The experiment panels are mounted so that they form the "lid" of a structural pan. The pan itself is an integral part of the spacecraft structure and has its own heat shield and structural integrity. A failure of the panel should not result in spacecraft failure.

Destruct: Although no design implementation of a destruct system was pursued, the wealth of reference sensors and performance data measurements assure the availability of existing on-board equipment which can be used to signal the need for destruction and to provide initiating action to the chosen devices. The sensing of this need would be derived from totally redundant parameters.

Pyrotechnic Initiation Devices: All pyrotechnic devices will be provided with redundant and isolated circuits.

PROGRAM CONSIDERATIONS

This section of the report describes the program devised to translate the SLAMAST spacecraft design developed during the design feasibility study into flight hardware. The cornerstone for this description is the overall program schedule, the detailed analyses schedule, subsystem development schedules and test schedules (See Figures 13 through 16).

The program philosophy adopted for development of the SLAMAST spacecraft was largely dictated by the premise that SLAMAST would be a test bed for conducting a variety of experiments. As a result, emphasis was placed on maximizing confidence in the test bed developed in order to maximize the opportunity for conducting the experiments. This philosophy had its major impact in two areas. The first was in the design definition and the second was in the development schedules and tasks.

In the design definition, the emphasis was placed on simplicity and specification of conventional subsystems and components in order to maximize confidence. For example, a relatively simple preprogrammed flight path control scheme was adopted; a conventional cold gas reaction control subsystem was specified in lieu of a less conventional hot gas subsystem which had lower weight and volume requirements; the basic attitude measurement sensor selected was one which is relatively inexpensive and has a substantial flight history; the recovery subsystem utilizes a conventional subsonic decel chute and location aids along with a somewhat less conventional but flight proven ballute; the structural design is a conventional semimonocoque design utilizing conventional materials and fabrication techniques; structural design margins of safety were kept high; the separation subsystem was based on designs which have been successfully fabricated and flown on other GE-RSD programs. Thus, with the exception of the S-band communication system, which was a contractural requirement, the spacecraft could be characterized as being composed of yesterday's subsystem designs. Even in the S-band communications area, hardware specification was based on the current state of development and well defined plans for development so that major breakthroughs in the state-of-the-art are not required for successful implementation.

The program schedules developed were based on the assumption that the design evolved during this study would be implemented, and that a conventional four stage information release system would be implemented. The data generated during the study satisfies the requirements for a stage 1 release. As such only three more stage releases will be required during the development cycle. Figure 13 shows the overall program schedule that was evolved. The schedule shows only the elapsed time from go-ahead to first flight since this is a key input to a project development plan and sufficient data exists to develop this schedule. Subsequent flight schedules would be dependent upon definition of the experiments to be conducted, and the philosophy of their conduct, e.g. are the tests to be independent of one another or are subsequent test definitions going to be based on the results of previous tests?

The analysis schedules are shown in Figure 14, the subsystems development schedules are shown in Figure 15 and an integrated test schedule is shown in Figure 16. In general, stage 2 information releases (PDR) are coordinated to occur 3 months from go-ahead and stage 3 releases (FDR) are scheduled to be made approximately 10 months from go-ahead. Note that completion of order of qual hardware is always scheduled to occur after fourth stage component releases are made. This is in keeping with the conservative approach adopted. Frequently, on other programs, qual hardware is ordered on the basis of third stage release data.

It should also be noted that the program defined includes an engineering development vehicle (EDM) and a qual vehicle (prototype spacecraft), as well as a flight vehicle. It is possible, and has been done on other programs, that the development vehicle or the qual vehicle can be eliminated if the program philosophy is changed. Another element of the schedule that can be subject to revision is the adopted philosophy that component qual will be completed before system qual tests are initiated. Programs have been conducted where the requirement was to achieve component qual only some time before flight. Finally it should be noted that before the qual spacecraft is subjected to the qual environments, it will be subjected to acceptance test environments. On other programs, this step is frequently omitted.

In summary, the spacecraft development schedule evolved results in a flight two years after go-ahead. Assuming a program start in mid 1968, this means the first flight would occur in mid 1970. This time span is believed to be in keeping with a relatively conservative program philosophy. A shorter schedule with potential economic benefits can be generated by implementing any or all of the alternatives identified in previous paragraphs.

TABLE I. - SLAMAST WEIGHT & BALANCE

ITEM	WT. (LB.)	C.G. STA. (IN. FROM NOSE)	MOMENT OF INERTIA (SLUG-FT^2)
[1] Structure	45.1	46.2	
[1] Heat Protection	88.7	40.1	
Electrical Equipment	19.0	37.2	
T/M & Tracking	27.0	41.0	I_{Roll} = 2.24
Attitude Contr. & Meas.	22.7	26.7	I_{Yaw} = 28.7
Recovery	11.8	59.0	I_{Pitch} = 27.4
Flap Control	7.0	59.0	
Brake Control	3.0	59.0	
Reaction Control	36.9	53.6	
Nose Tip/Ballast	68.8	7.1	
Total Spacecraft	330.0	35.8	
Adapter Structure	22.4	81.2	
Separation System	4.9	78.0	
Total Adapter	27.3	80.6	
Total S/C + Adapter	357.3	39.2	

NOTES -

(1) Includes Flaps & Test Panels

(2) The weights etc. stated above are the result of the last iteration and have not been totally factored into the report. However, the differences are not crucial to the feasibility demonstration.

TABLE II. - SLAMAST: PRELIMINARY PARTS LIST

Subsystem	Component	Qty	Comments
1. Recovery	1. Drogue chute Assy. . (Canopy, risens, and deployment bag)	1	(Goodyear, Aerospace)
	2. Ballute Assy. . (Ballute and deployment bag).	1	(Goodyear, Aerospace)
	3. R-F beacon & antenna	1	
	4. Drogue Chute Release Mech.	1	
	5. Ejection Mortar & Squib	1	
2. TT&C (I&C)	1. Telemetry Antenna	1	
	2. Telemetry Transmitter (S-band)	1	Similar to Microcom Model T-40
	3. PCM Multicoder (Micromin)	1	Similar to MK-12 (recent developmental award)
	4. Baseband Assy Unit	1	
	5. Tape Recorder	1	Similar to Borg-Warner R310
	6. Analog Signal Conditioner	1	Similar to GE/RSD 47C138575
	7. Sensor Power Supply	1	Similar to GE/RSD 111C5250
	8. C-band Transponder	1	Similar to VEGA Model No. 302 C-2
	9. Decoder Module	1	Similar to VEGA Unit
	10. Single Axis Accelerometer	3	Similar to Honeywell GG 322
	11. Sensors: Pressure Temp etc.	22 100	
3. Separation	1. Collet Assy.	4	
	2. Freon Tank; 3000 psi	1	
	3. Pyro. Actuated Valve	1	Holex Inc. 5090A
	4. Fill & Vent Valve	1	MS 28889-1
	5. Pressure Transducer	1	
	6. IFD - 26 pin	1	Cannon PN 1020570001 (26 pin)
	7. Separation Switch	1	GE PN R 2336
4. AM&C	1. Autopilot	1	
	2. PRYS Attitude Ref. Unit	1	Similar to Whittaker MARS
	3. Tri-axial Rate Gyro	1	GE/RSD 47C142658
	4. Tri-axial Accelerometer	1	GE/RSD 47C142659
	5. Controller Programmer	1	
	6. Pitch Flap Actuator	1	Bendix PN 122 dwg 3170164
	7. Reaction Control		
	a) Tanks; 3500 psi	6	
	b) Pressure Regulator	1	Similar to Sterer Mfg. PN 99643
	c) Filter	1	GE/RSD 47C139969
	d) Solenoid Valves	6	Carleton PN 1809001
	e) Pyro Actuated Valve	1	Holex Inc. 5090A
	f) Nozzles	6	
5. EP&D	1. Battery	1	Yarnley 7AH
	2. Power Controller		

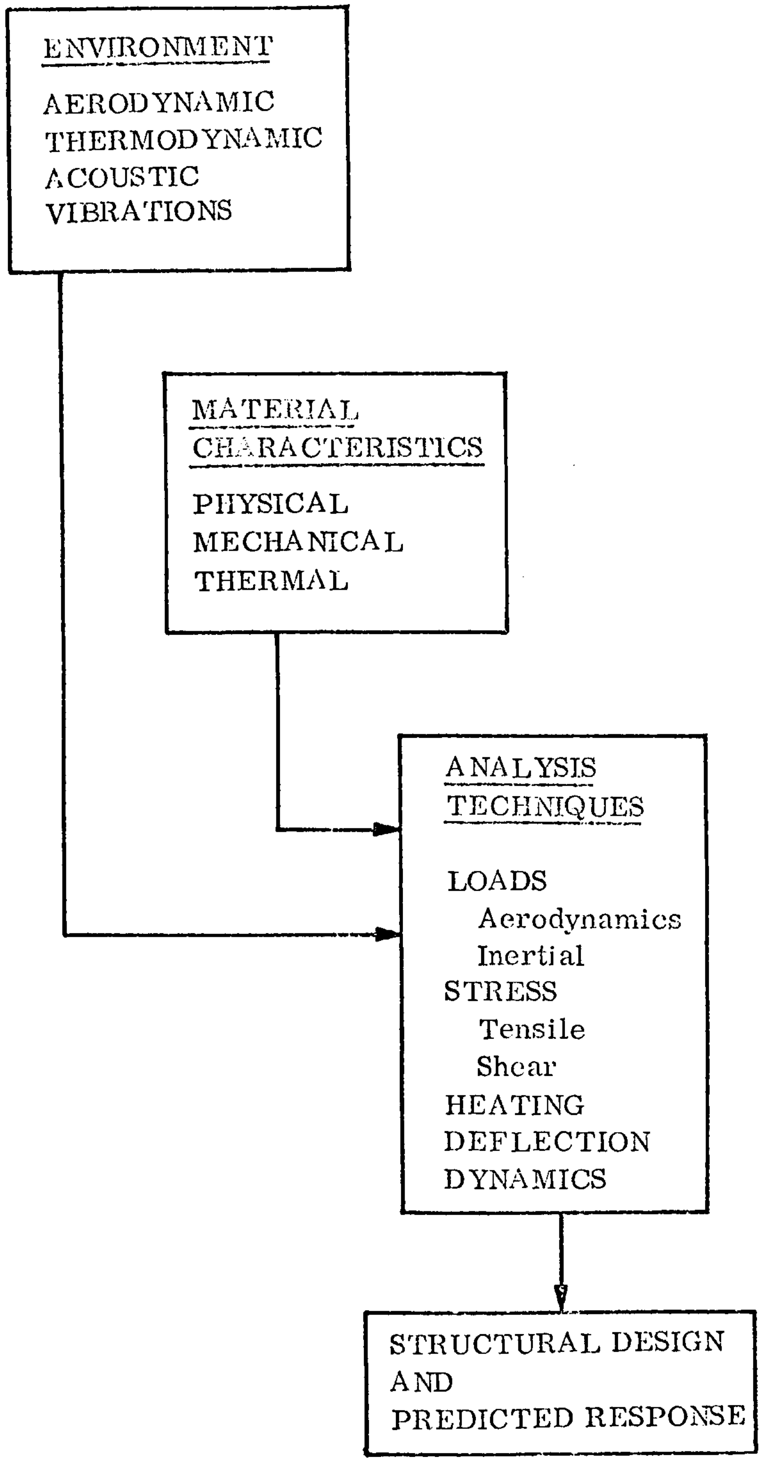

Figure 1. - Elements of Structural Design

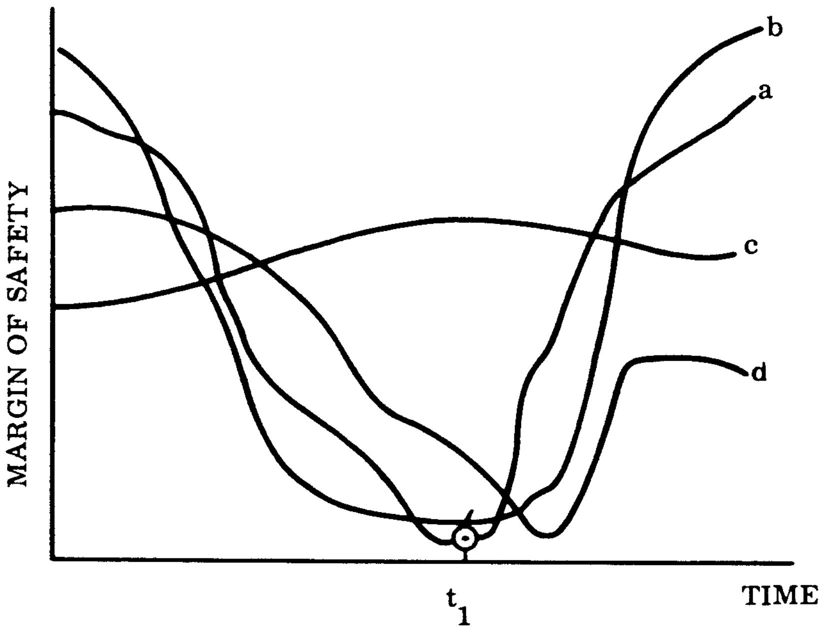

Figure 2. - Margin of Safety as a Function of Time

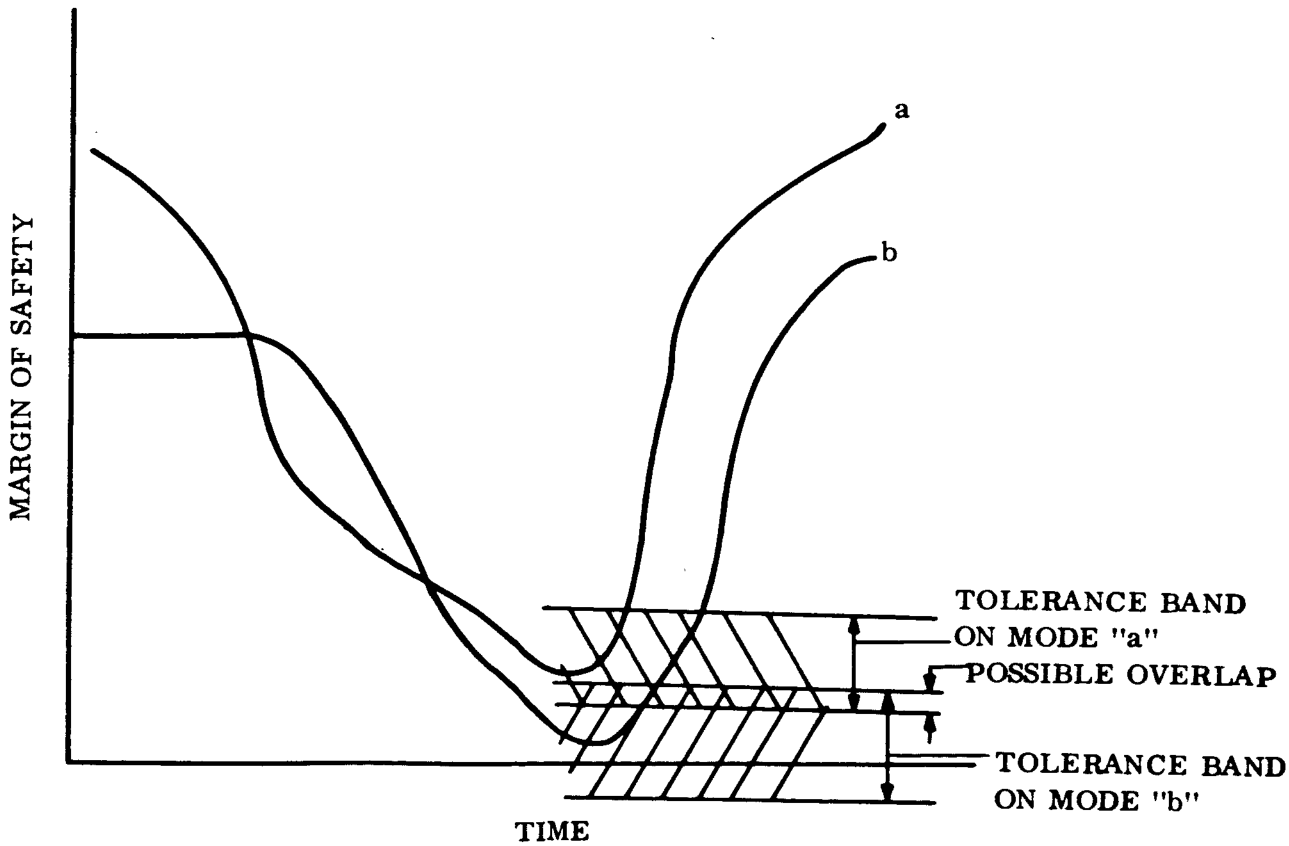

Figure 3. - Margin of Safety - Critical Failure Modes

Z1-381

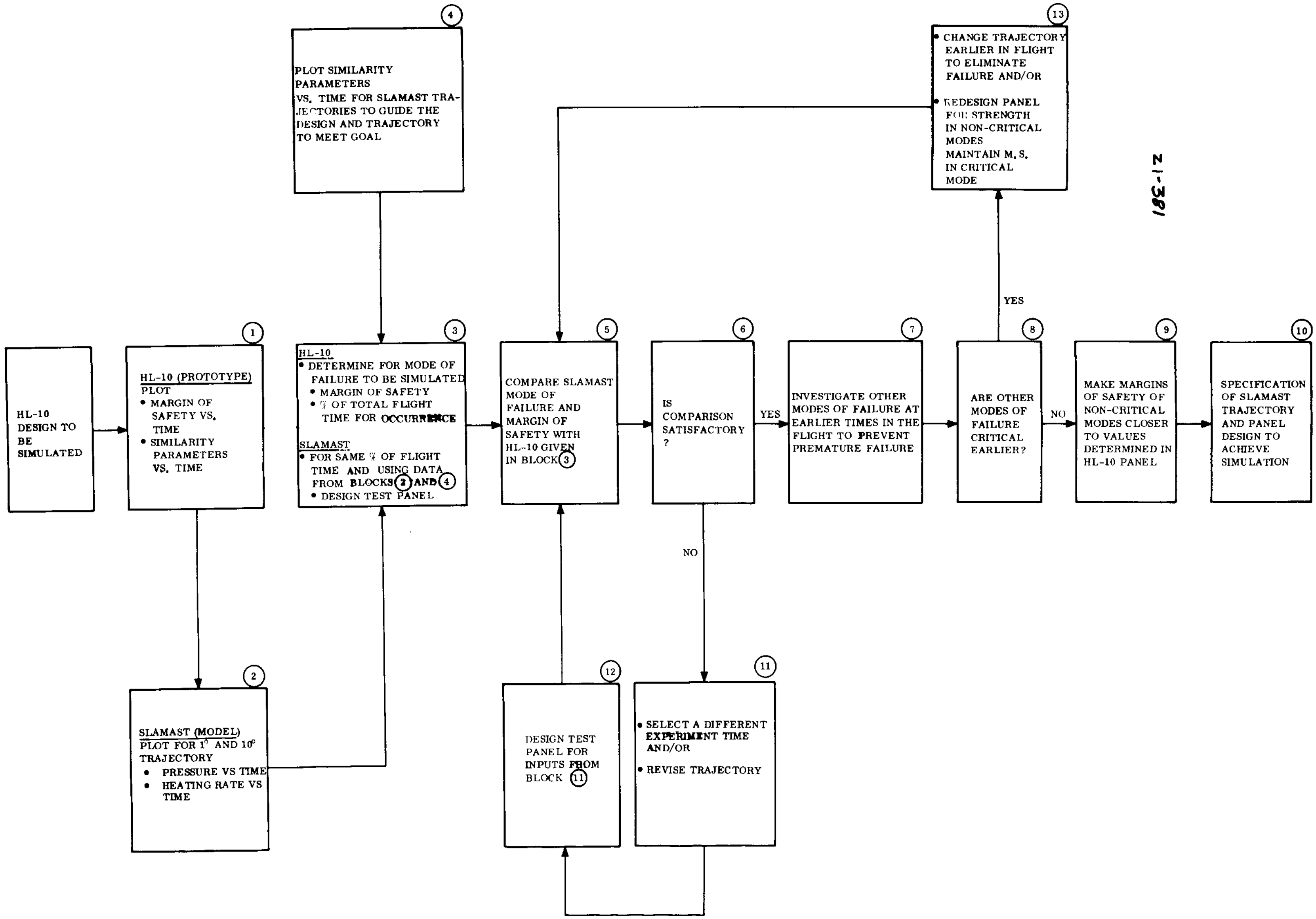

Figure 4. - Flow Chart Demonstrating Procedure for Selecting SLAMAST Trajectory and Panel Design for HL-10 Simulation

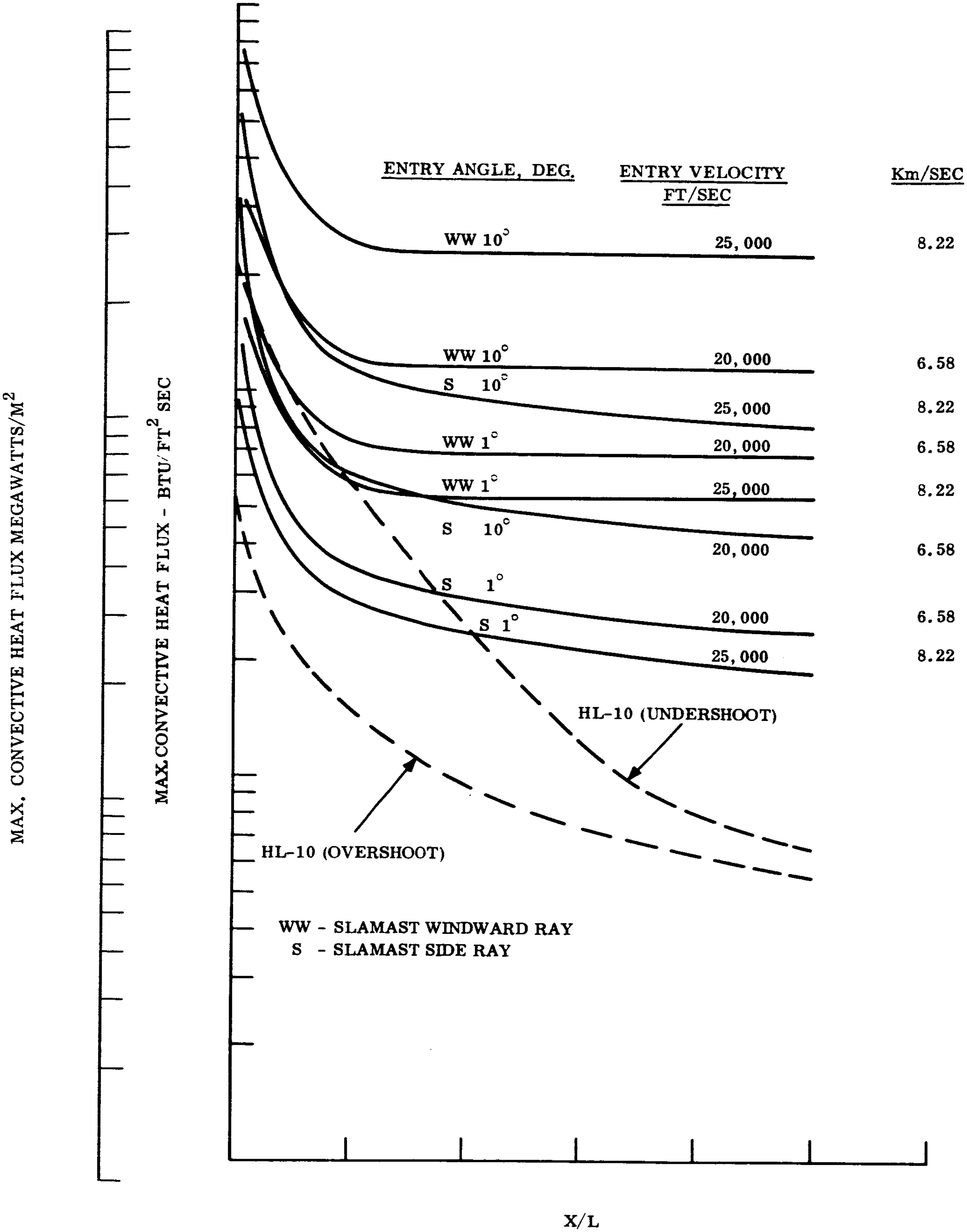

Figure 5. - Comparison of Maximum Convective HeatFlux Rates for HL - 10 and SLAMAST Vehicles

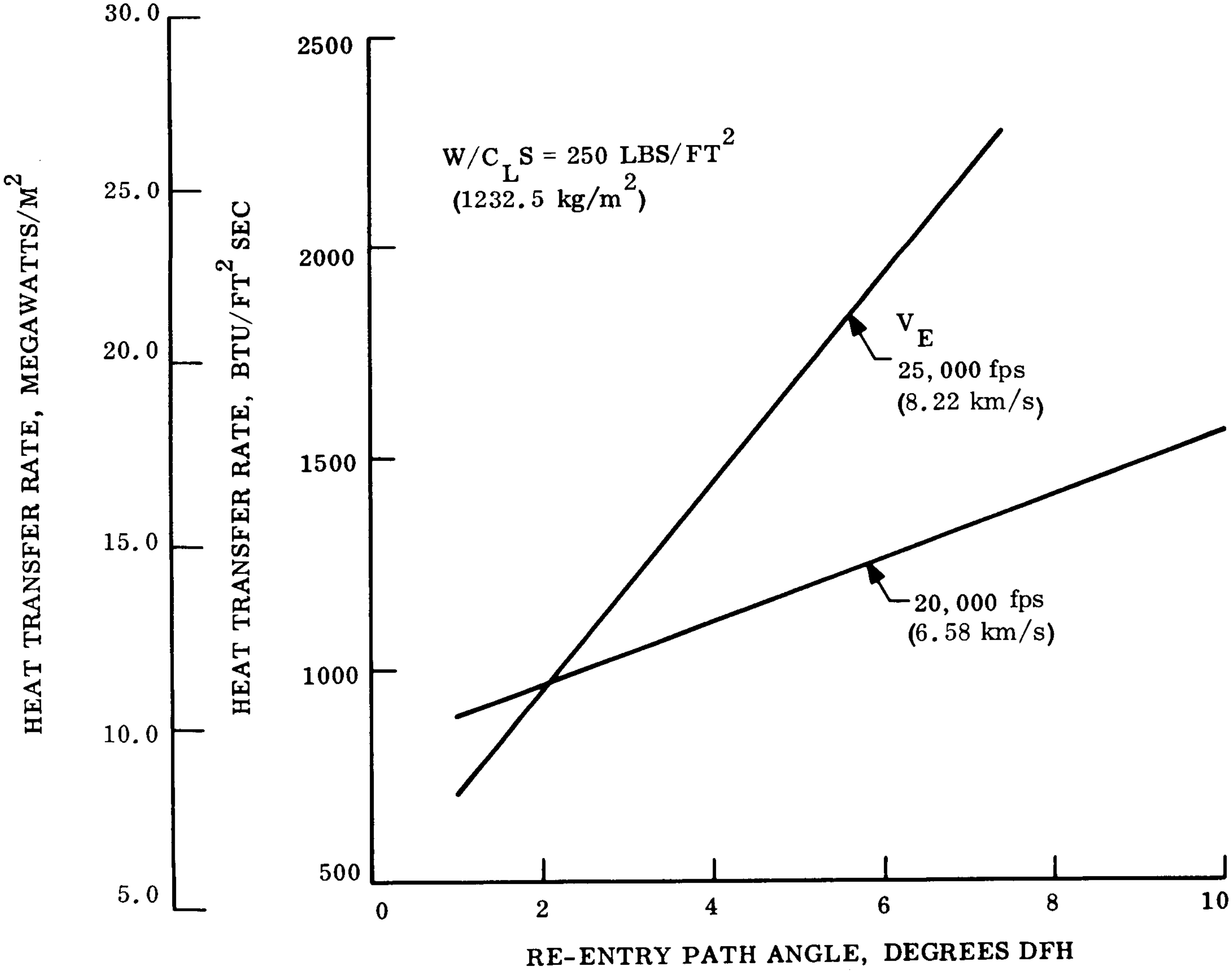

Figure 6. - Maximum SLAMAST Stagnation Heat Transfer Rates

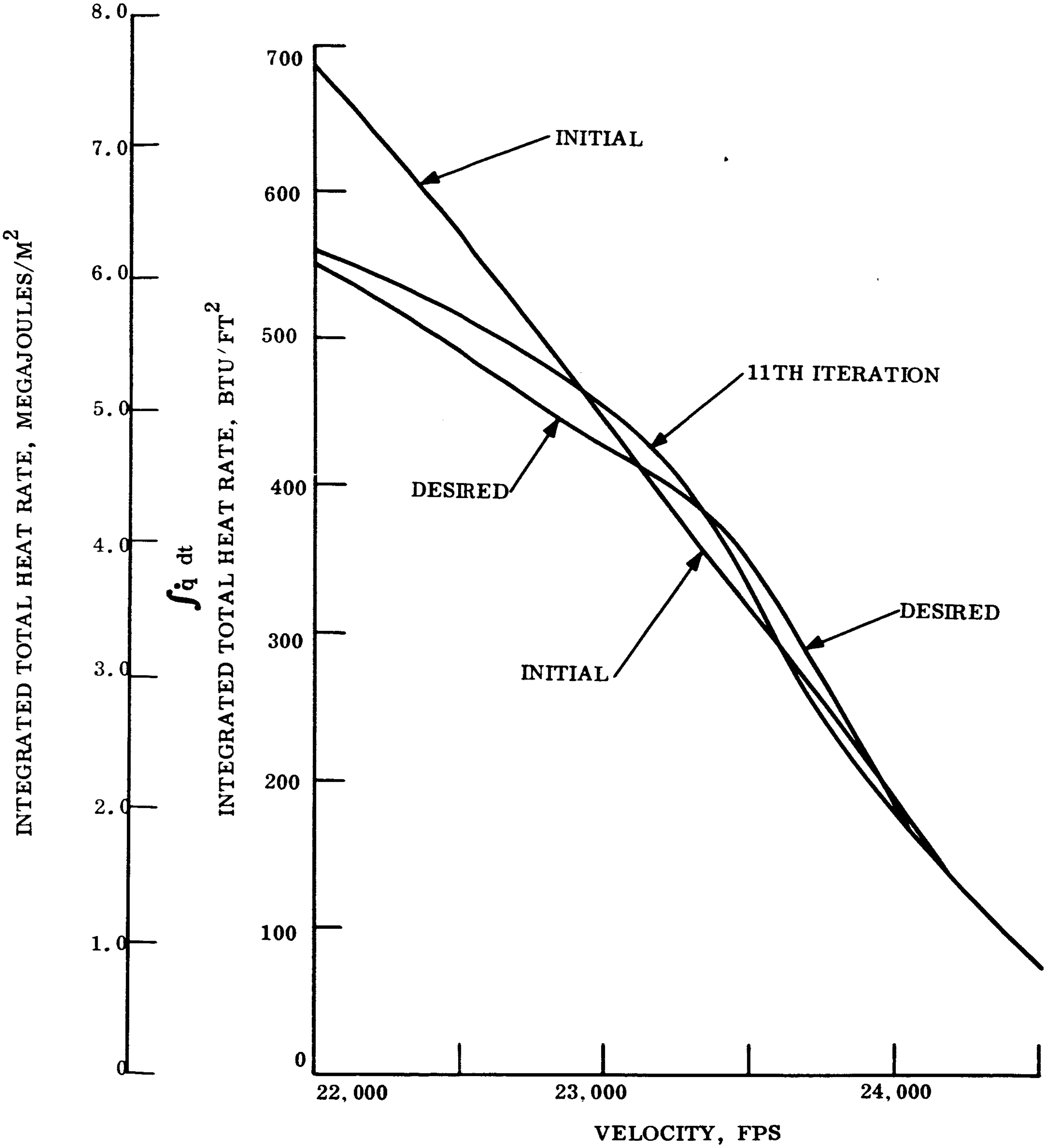

Figure 7. - SLAMAST Integrated Stagnation History versus Velocity

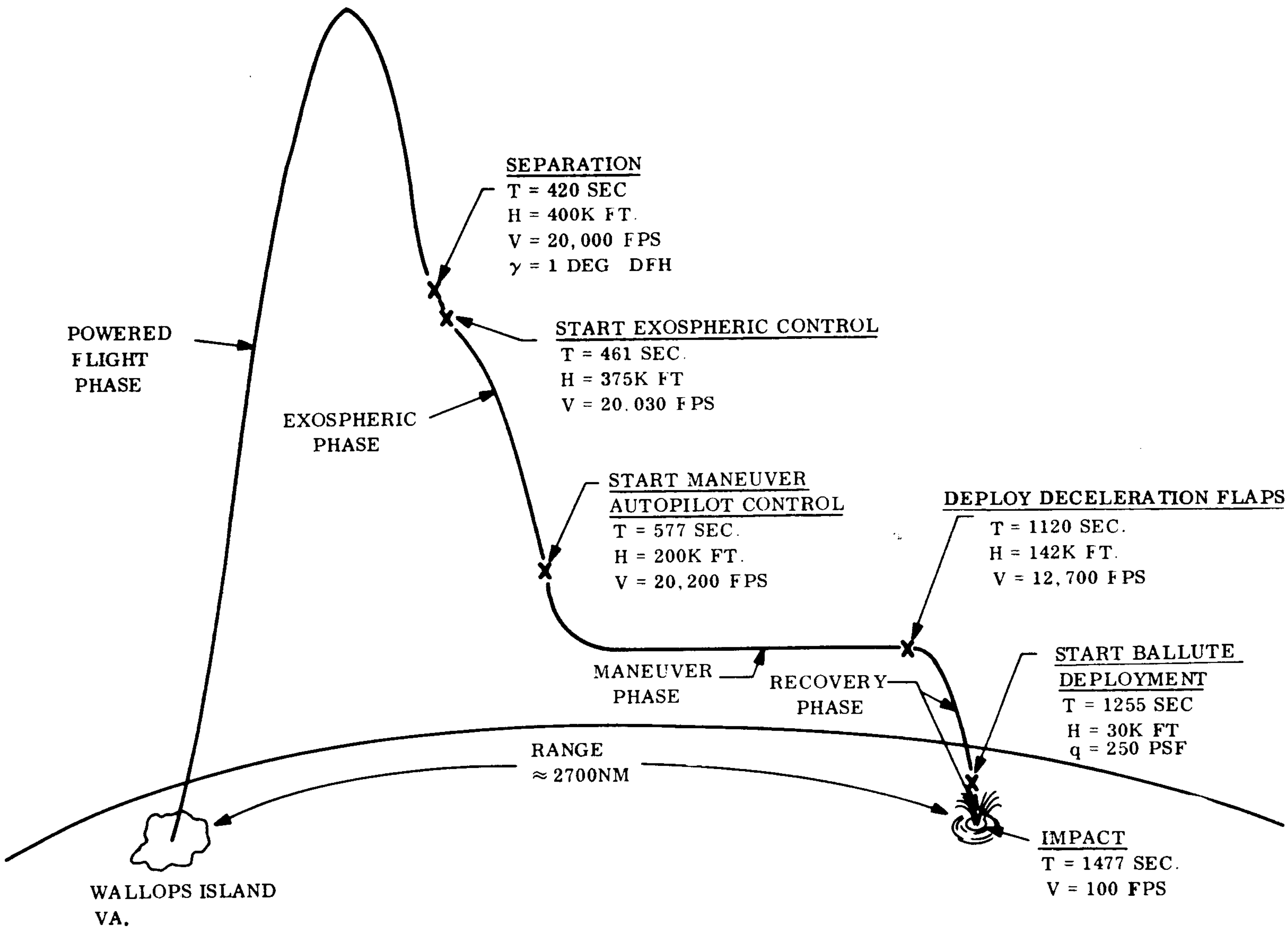

Figure 8 SLAMAST Flight Mission Profile, $\gamma_e = 1$ Degree DFH

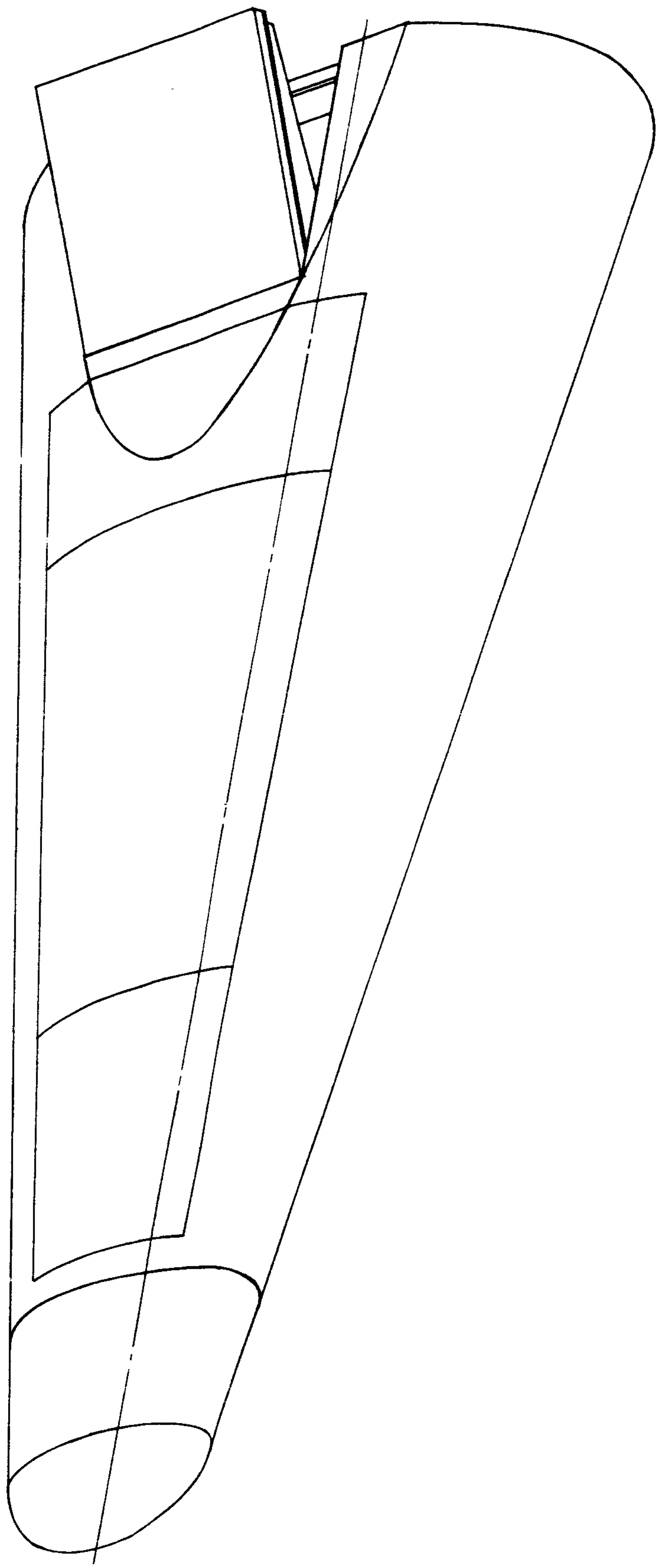

Figure 9. - SLAMAST S/C

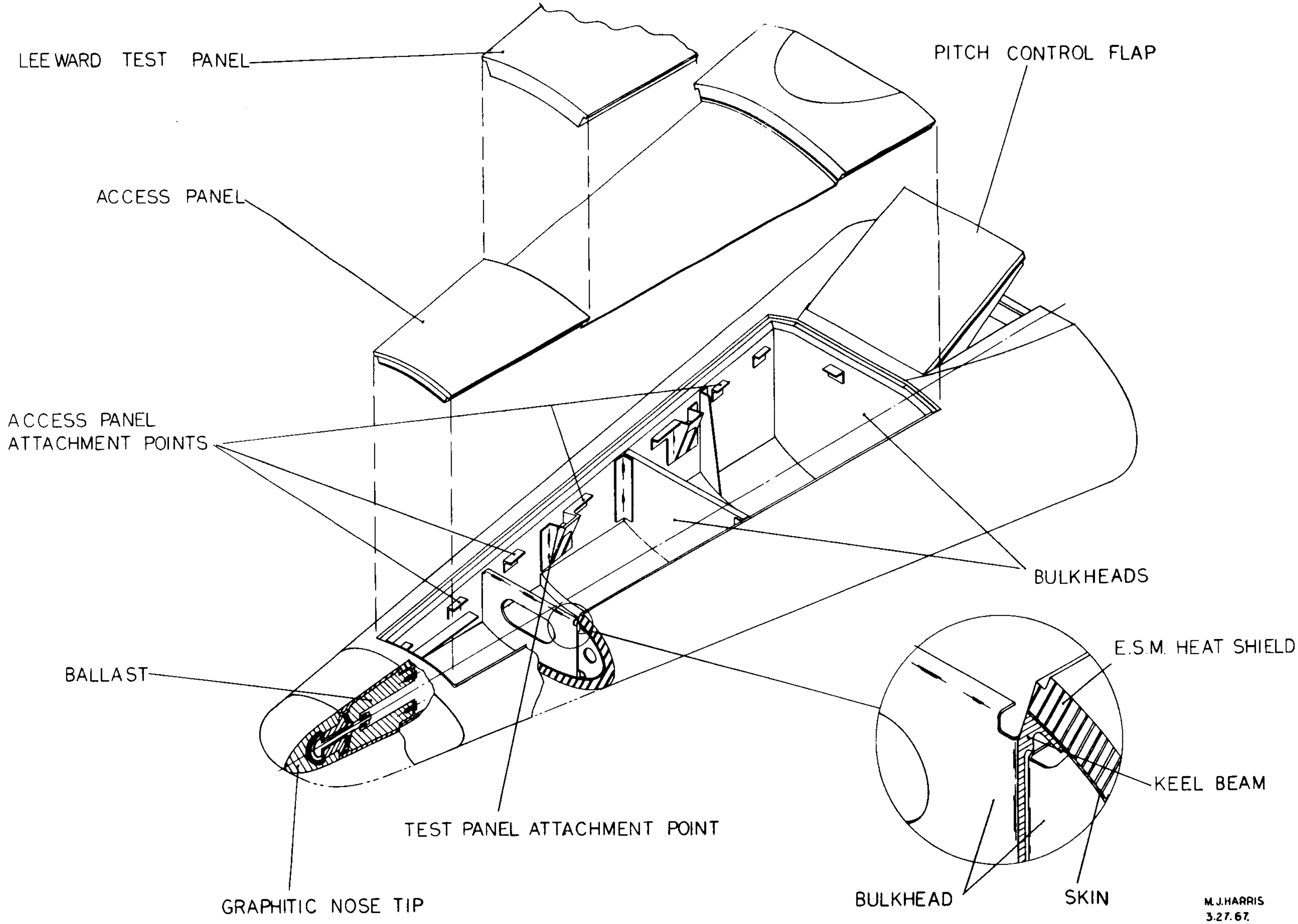

Figure 10. - SLAMAST Structural Arrangement

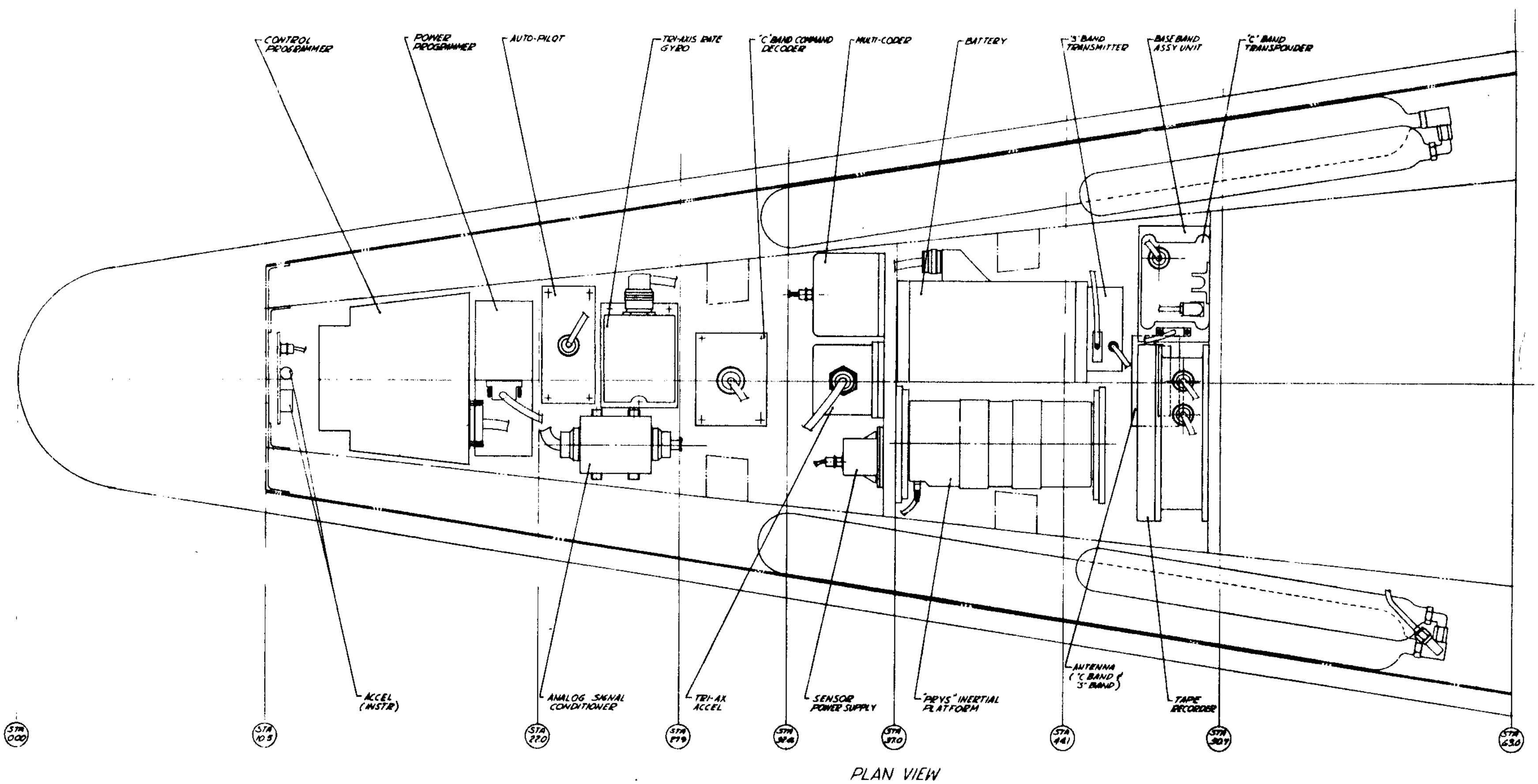

Figure 11. - Component Arrangement - Plan View

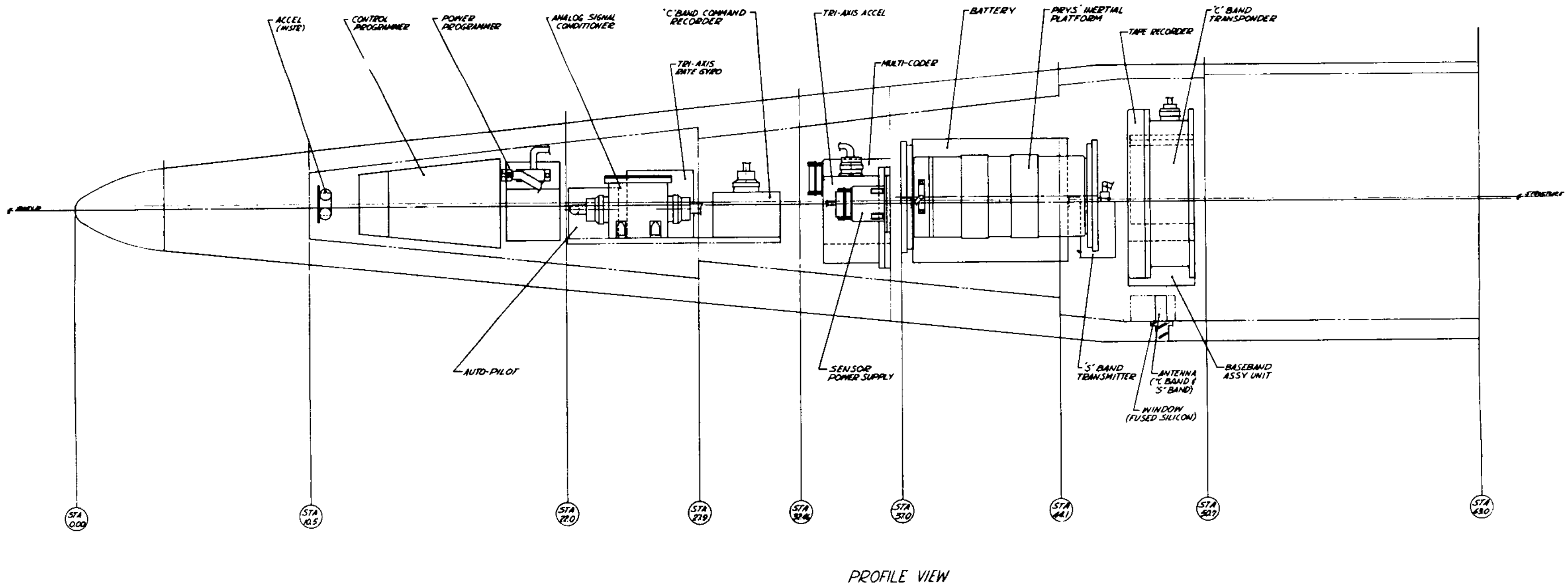

Figure 11. - Component Arrangement - Profile View (Concluded)

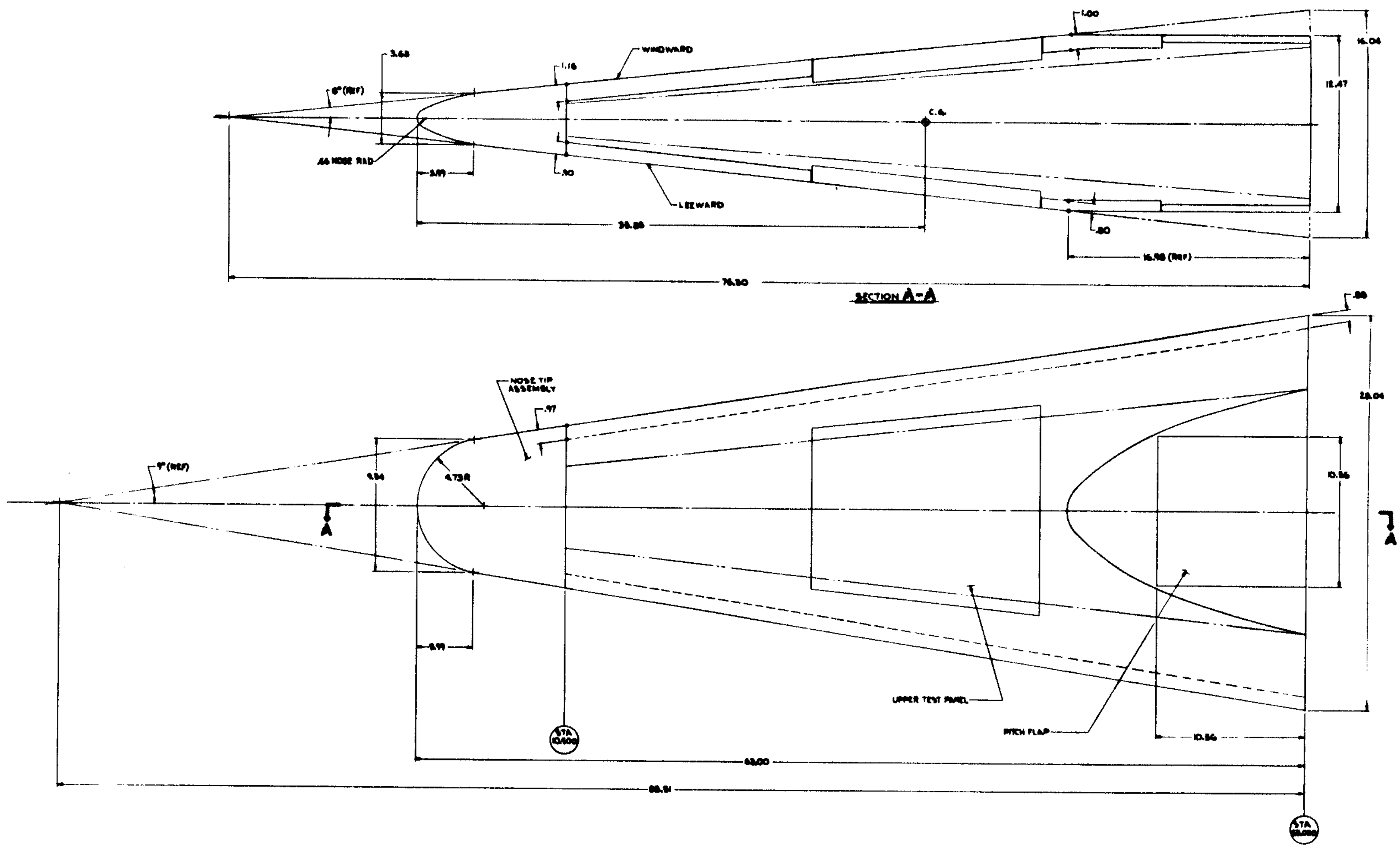

Figure 12. - Spacecraft Geometry

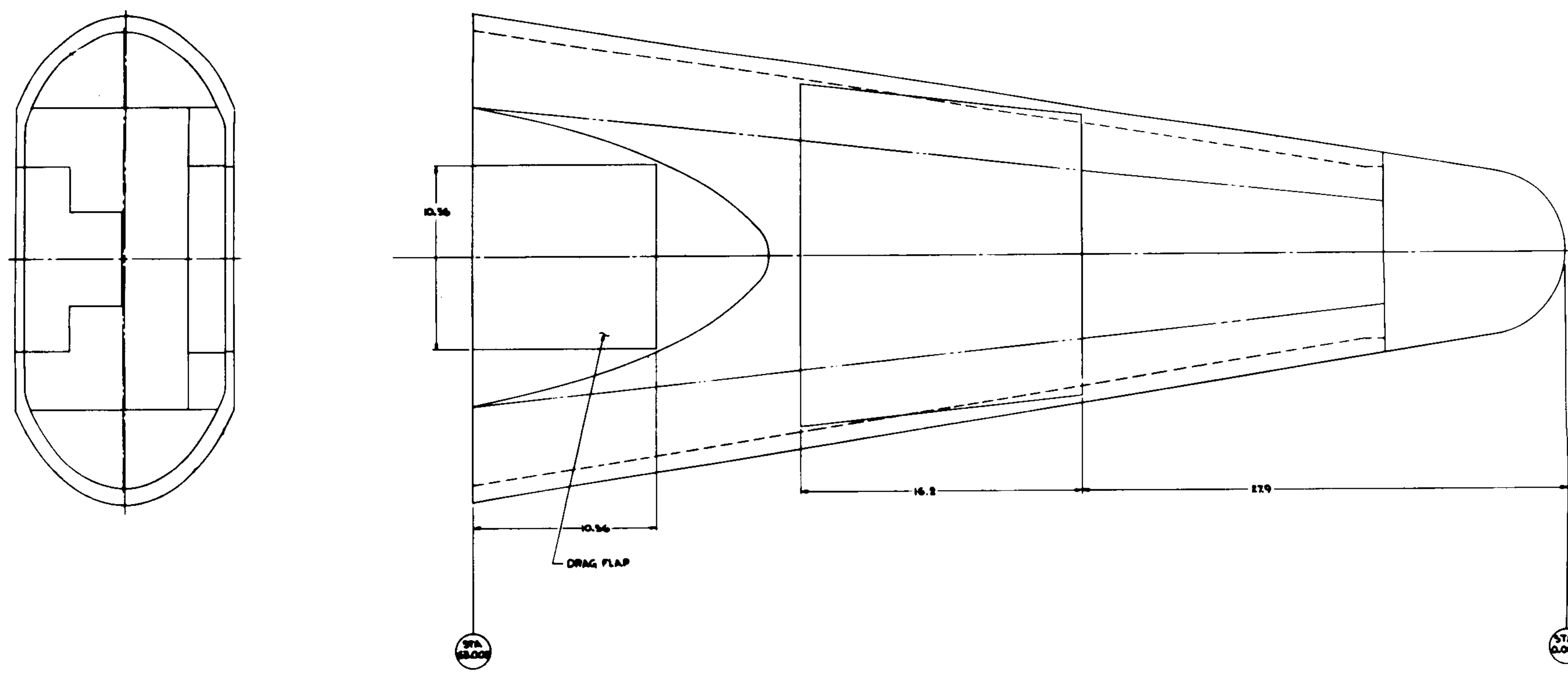

Figure 12. - Spacecraft Geometry (Concluded)

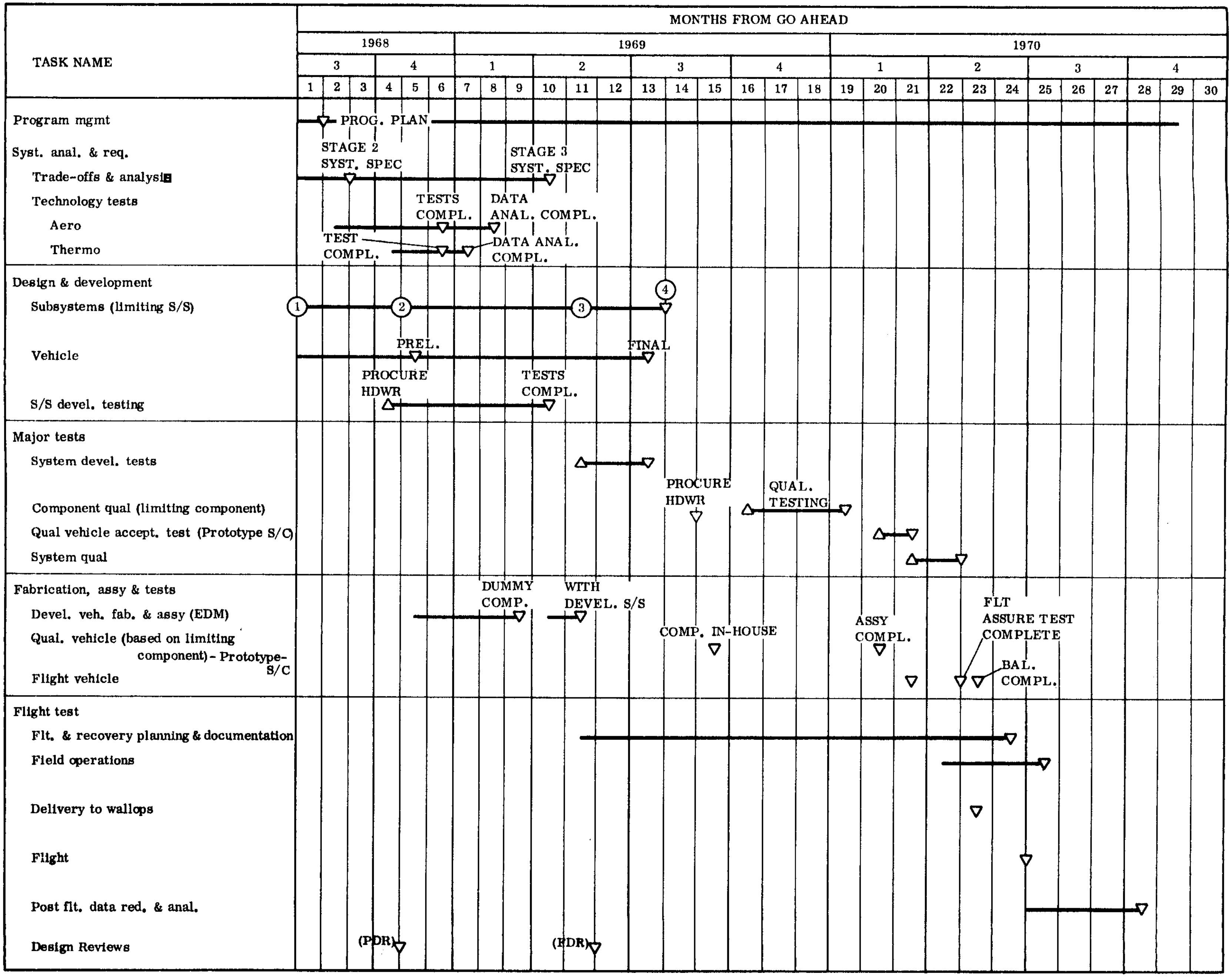

Figure 13. - Overall Program Schedule

△ START

▽ COMPLETE

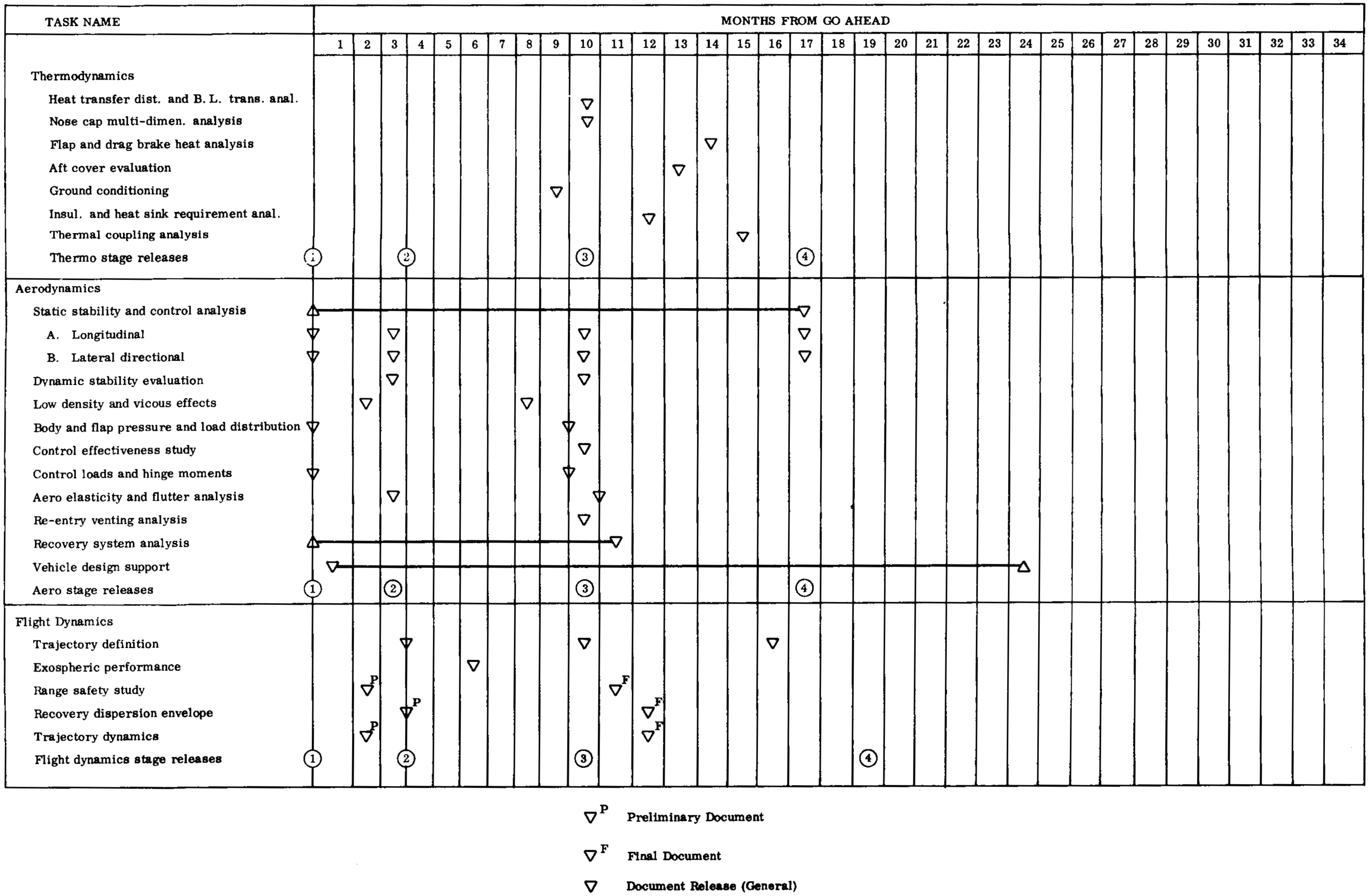

Figure 14. - Analysis Schedules

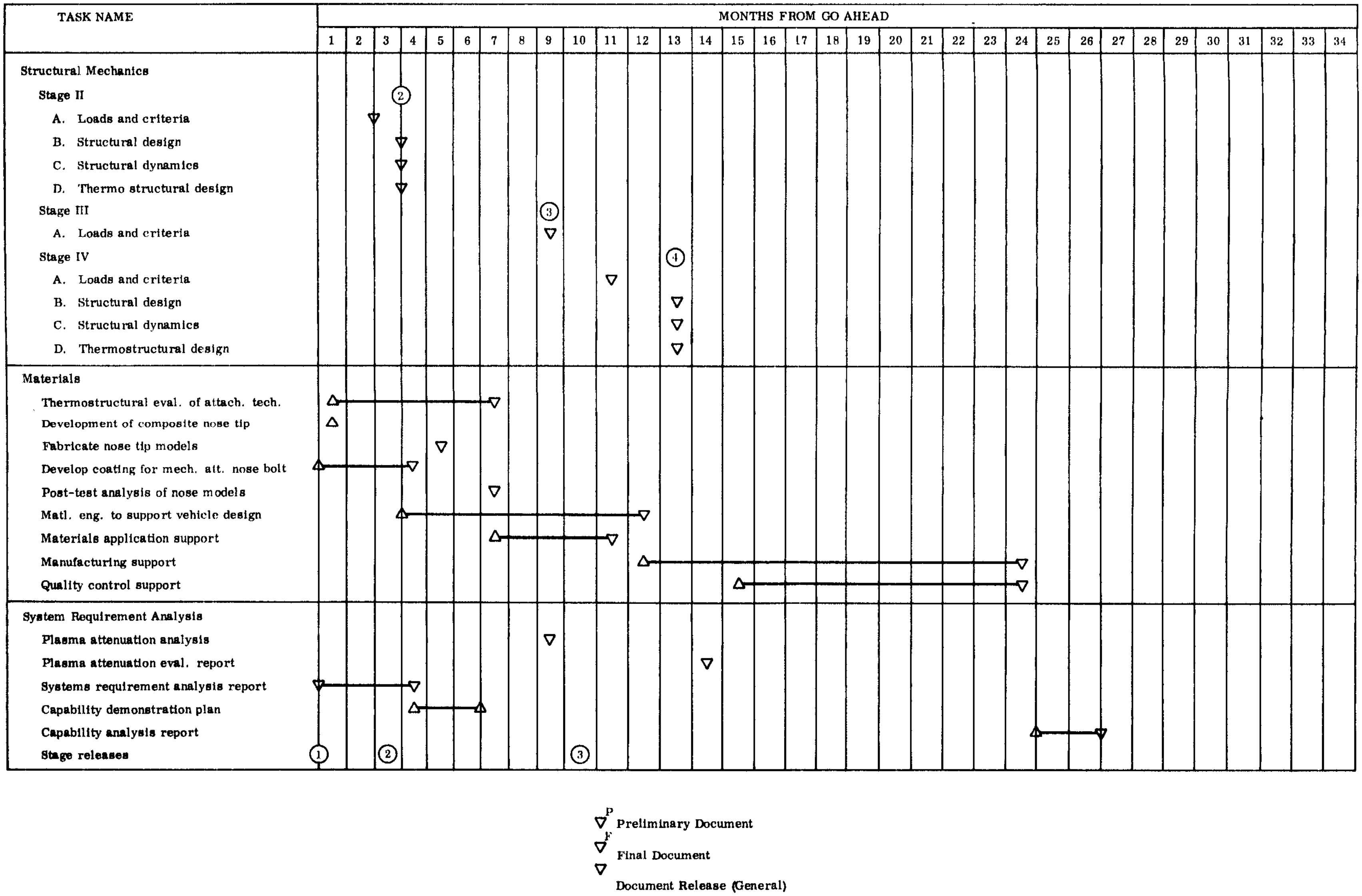

Figure 14. - Analysis Schedules (Continued)

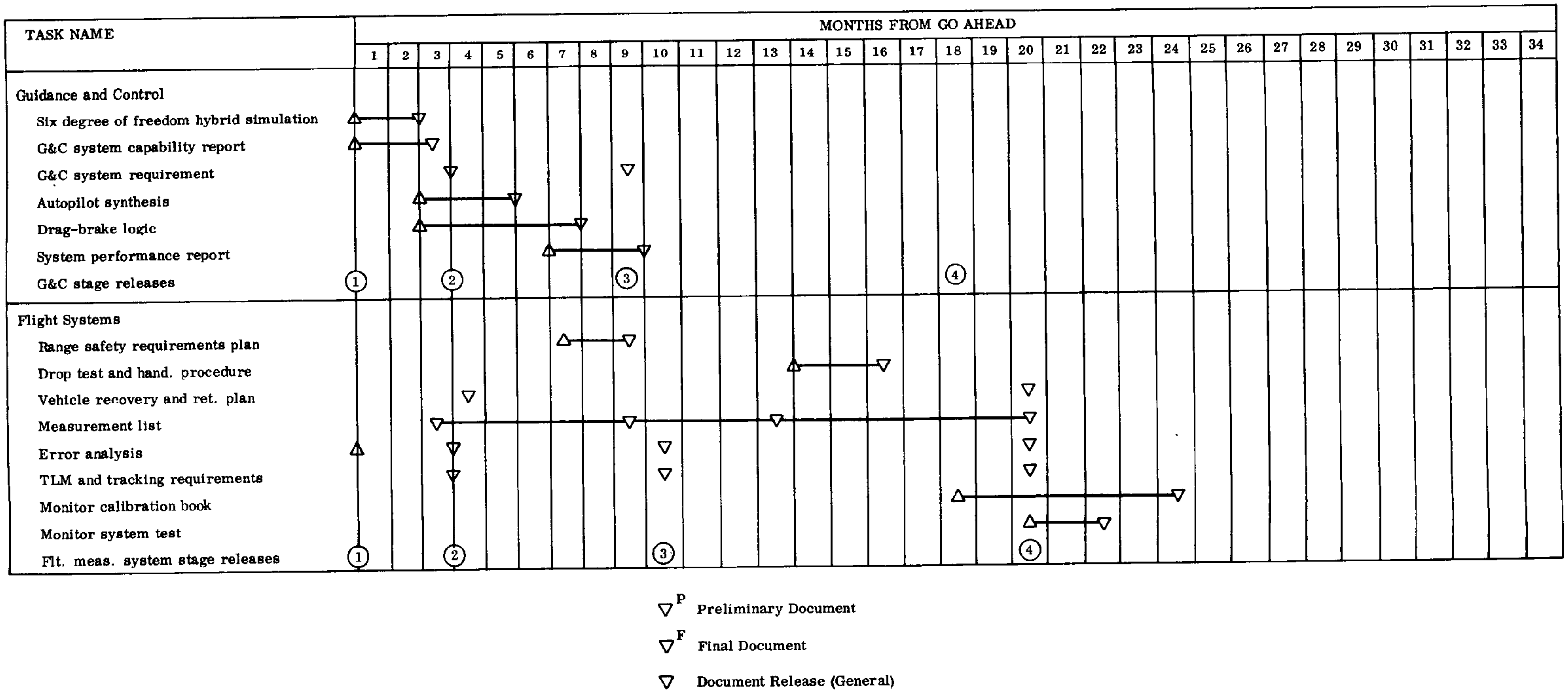

Figure 14. - Analysis Schedules (Continued)

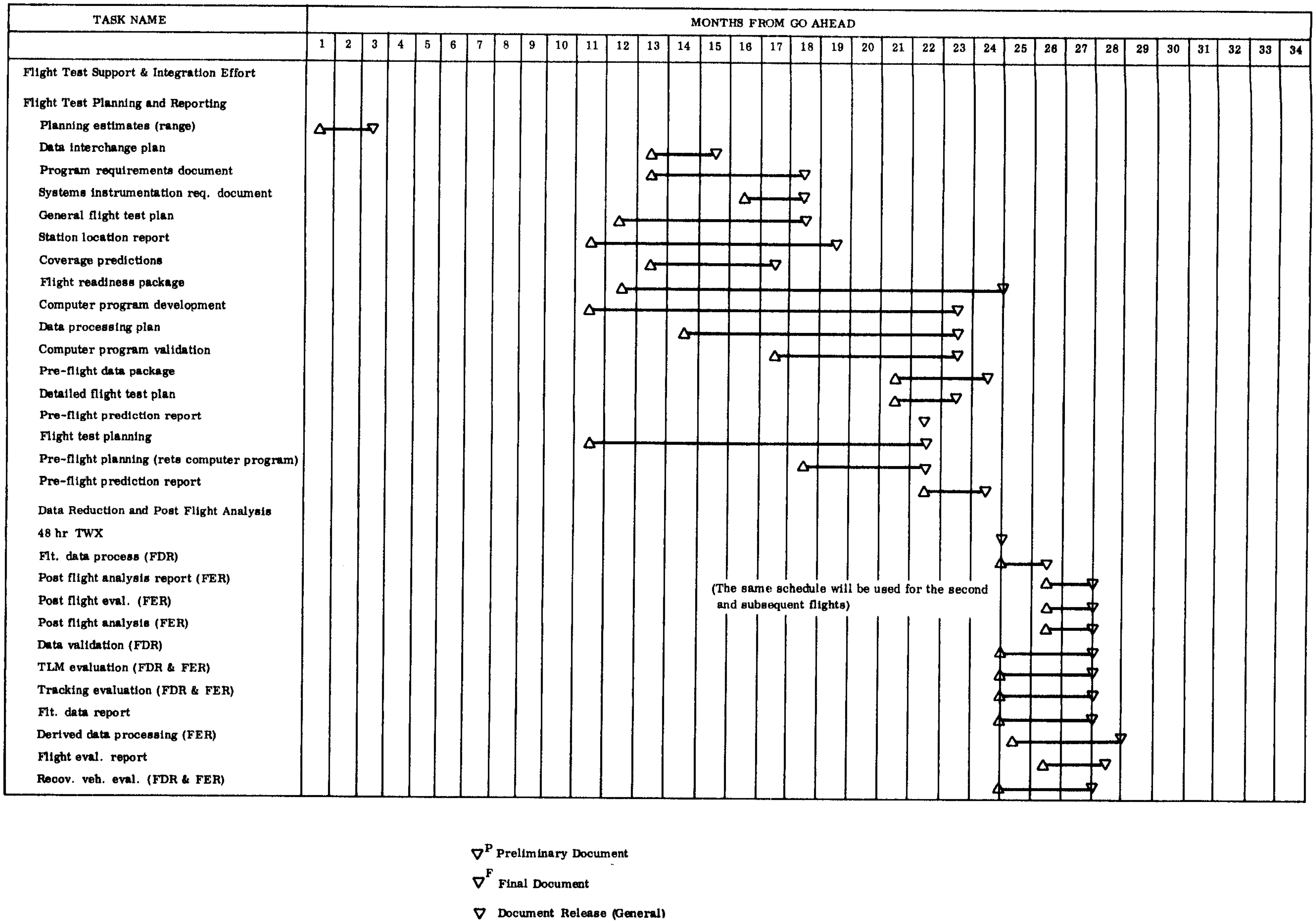

Figure 14. - Analysis Schedules (Concluded)

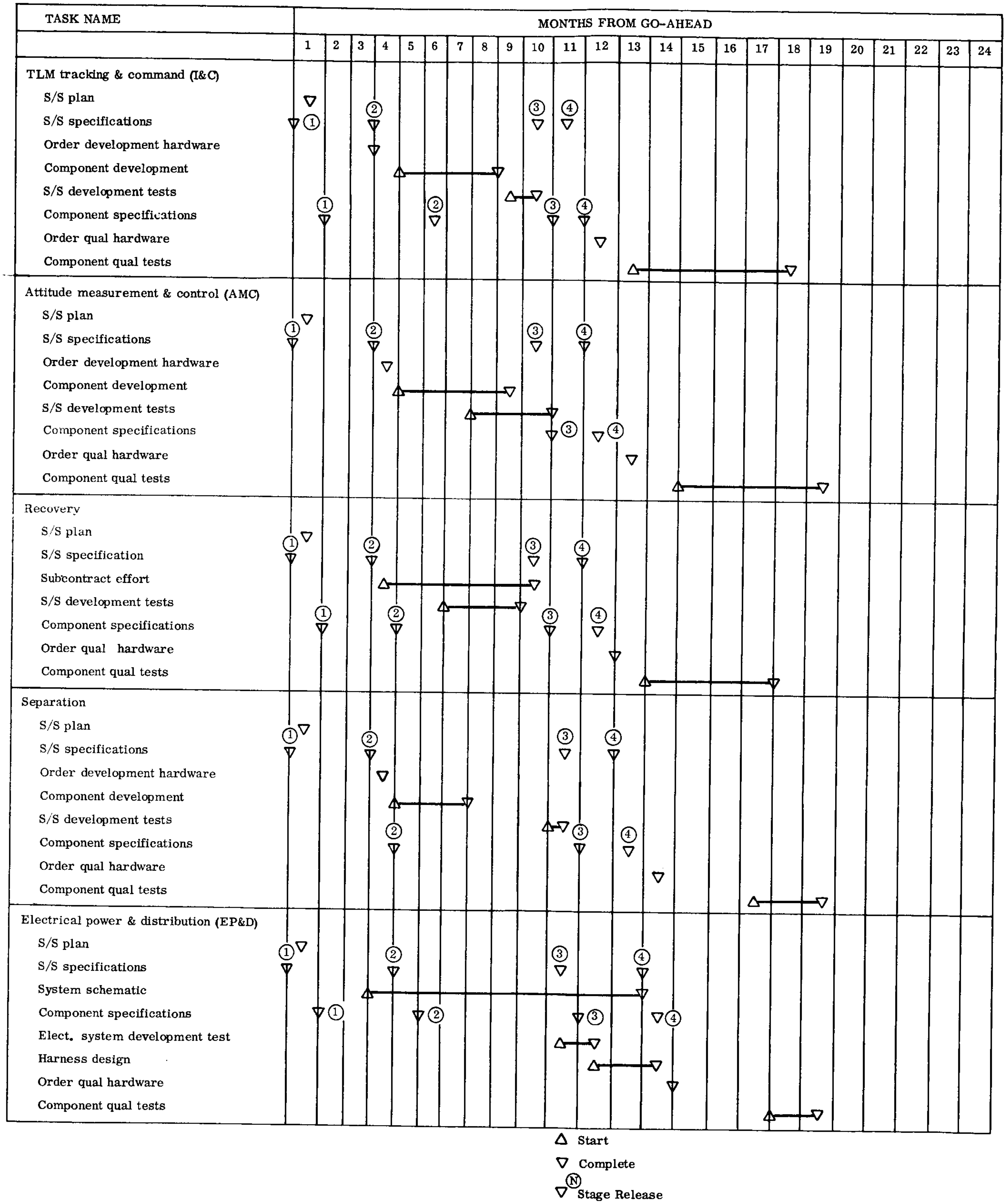

Figure 15. - Subsystem Development Schedule

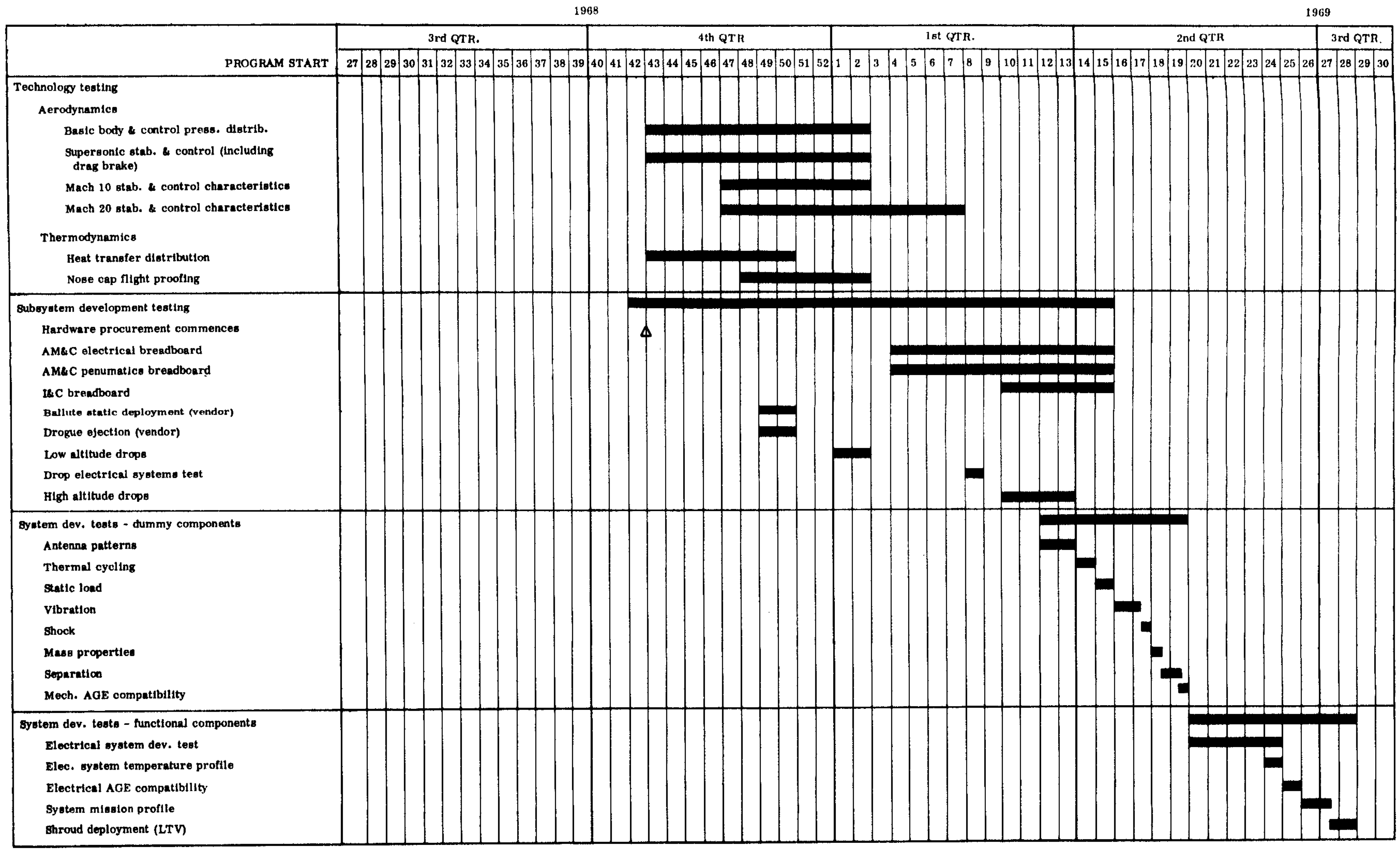

Figure 16. - Integrated Test Schedule

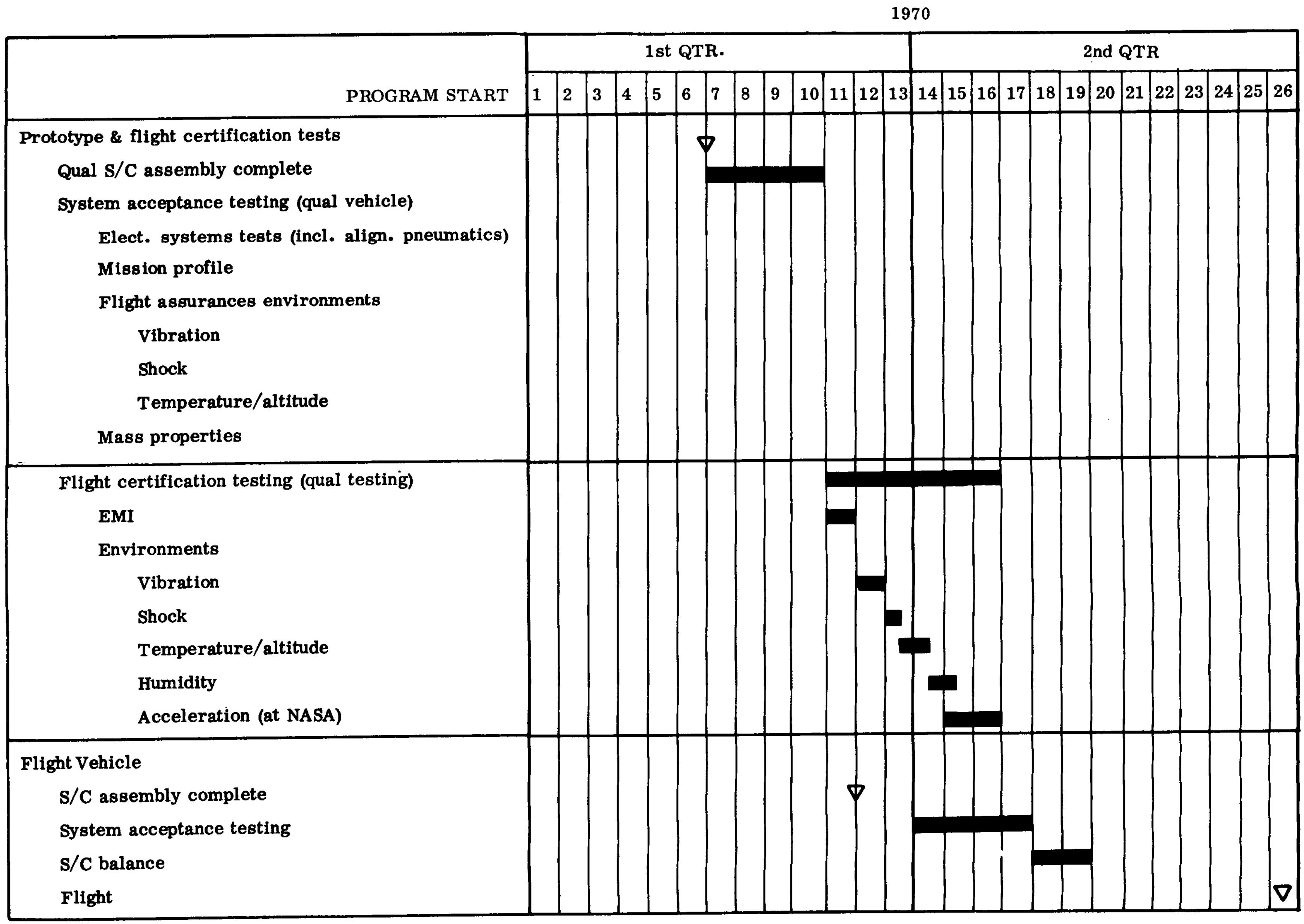

Figure 16. - Integrated Test Schedule (Concluded)