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REPORT

NO. LPL-610-1 DATE: 6 Sept. 1963

FEASIBILITY STUDY AND DETAILED TEST PLANS FOR LEM-1 AND LEM-2

[U]

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GRUMMAN AIRCRAFT ENGINEERING CORPORATION

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1.0 Summary.-

A failure analysis for LEM showed the propulsion subsystem to be a flight safety item for all manned LEM flights. Review of the propulsion development program indicated that there was a requirement for in-flight development tests. Ground development tests could not adequately simulate the in-flight environment. The purpose of the study contained in this report was:

1. to establish the requirements for flight test.

2. to select the boost vehicle.

3. to determine the mission objectives for each flight.

4. to prepare detailed test plans for each flight, and

5. to investigate the feasibility of conducting the tests.

The result of the study is a recommendation to conduct two flight tests with the LEM vehicle launched by a Little Joe II booster. In addition, it is recommended that a booster qualification flight be conducted prior to the first LEM test.

2.0 Discussion.-

2.1 Requirements for Flight Tests.-

A failure in the propulsion (or reaction control) subsystem which permits the contact of fuel and oxidizer outside of the thrust chamber will be catastrophic to the astronaut. The development program must prove the subsystem's integrity prior to a manned LEM flight. A review of the propulsion & RCS development programs has indicated several areas where flight conditions, and/or flight environments, can not be adequately simulated on the ground. Unmanned flight tests are, therefore, required to demonstrate and prove the system's integrity in the design environments. The specific results which are to be sought from flight tests and which are unattainable in ground testing are:

a. Evaluation and certification of the safe operation of the propulsion subsystem.

   1. Evaluation of the performance in the space environment of zero "g", hard vacuum, and unrestrained vibration.

      a. Zero "g" - LEM-1 will be the first time the system can be subjected to this condition. Valve timing and sequencing are two of the many functions which might be affected. LEM-1 will also provide information on the behavior of hypergolic propellants in this environment.

      b. Hard vacuum- The finite lag between contact of the propellants and actual ignition varies with pressure. LEM-1 test altitudes will exceed those obtained in test chambers, thereby providing a lower pressure envrionment.

      c. Vibration - LEM-1 will provide the first test of unrestrained vibration.

      d. Combinations of the above environments are not ground tested and failures due to combined environments could occur.

   2. Evaluation of full duration firings in space. LEM-1 and -2 provide the only full duration firings until LEM-8.

b. Evaluation and certification of the safe operation of the Reaction Control Subsystem.

c. Demonstration of the fire-in-the-hole condition for the lunar launch and abort phases.
   Ascent engine exhaust impingement on the descent stage could cause combustion instability and/or structural damage. Exact ground simulation is not possible as

2.1 (cont)

exhausting into the altitude chamber will cause an immediate loss of altitude, and the earth's gravitational force prevents dynamic simulation of the separation. The ground tests will use a simulated descent stage (i.e. Heavy-weight rig) to preclude the overall program delays which would be caused by damage to a test article.

d. Evaluation of the system dynamics across a wide range of propellant loads.

LEM development problems must be defined early in program. LEM-1 and -2 provide the only opportunity prior to LEM-8 to test the system at full fuel loads, in a space environment.

1. The discussions presented for the propulsion subsystem in paragraph 2.1C (1) are applicable to the RCS.

2. According to current NASA restrictions, the RCS will not be tested in an altitude chamber as a system, although GAEC does not concur with this approach to the ground test program.

e. Evaluation of the system performance after subjection to the combined vibratory, acoustic, and acceleration environments of the booster.

Combined environments are not ground tested due to the complexity involved.

f. Evaluation of the performance in the LEM engine and RCS induced environments (vibratory, thermal, acoustic, and acceleration.)

Ground test of the combined environments are not conducted due to facility constraints.

g. Evaluation and certification of the system structural integrity.

1. Evaluation of integrity subsequent to boost and combined boost environment.
Combined environments are not ground tested.

2. Evaluation of the LEM engine induced thermal distribution.
The LEM engines are buried inside structure and as such have primary input to the thermal distribution of the system. In an altitude chamber the need for a diffuser, which restricts plume expansion, prevents determining the input of the plume to the structure and landing gear.

h. Demonstration of LEM staging.

LEM-1 will demonstrate clear spearation of the descent

2.1 (con't)

stage (i.e. no impact with ascent stage) for the first time in the dynamic condition.

2.2 Additional Gains from the Test.-

The performance of the LEM-1 test will provide:

a. Evaluation of the LEM engine-induced environments (thermal, vibratory, acoustic).
   The test results will be used to determine the adequacy of ground simulations.
b. Evaluation of the compatibility of the applicable GSE.
c. Evaluation of microwave attenuation, absorption, or interference caused by engine exhausts.
d. Improved reliability.
   The reliability goals (.999, etc.) cannot be demonstrated to the desired degree of confidence within reasonable cost and schedules by conventional testing. Statistics show a prohibitive number of tests are required. The test of LEM-1, by increasing the number of tests, will increase the confidence level due to the tests being conducted in the vehicle's operational environment.

2.3 Selection of the Boost Vehicle.-

The particular boosters available for the LEM-1 tests were the Little Joe II and the Saturn C-1B. The Little Joe II was determined to be the better choice for the following reasons:

a. Little Joe II will have completed development and the LEM test will not be subjected to the requirements of a booster development program. The C-1B boost for the LEM would be a booster development flight.

b. Little Joe II is more economical than the C-1B.

c. Little Joe II offers a test which is independent of all other Apollo components and thereby permits the LEM program to progress freely in the early development days and the booster scheduling to be dictated by LEM requirements.

d. Little Joe II availability permits an earlier flight date and one not subject to development delays, as in the C-1B. The early flight permits feedback for modification and/or redesign without severely delaying propulsion subsystem qualification.

e. The time required for Little Joe II prelaunch checkout is less than the Saturn, again permitting an earlier launch date with the same delivery date and a more economical launch.

f. Little Joe II availability permits the scheduling of a back-up booster for the contingency of a mishap. The Saturn booster schedule does not have this flexibility. In the event of mishap,

2.3 (con't)

f. (con't)
the entire Apollo booster phasing would be required to slip to provide a back up capability.

g. Little Joe II satisfies the development test requirements of LEM without the complexity of a C-1B test (i.e. it's simpler).

2.4 Feasibility.-

The detailed feasibility analysis is presented in the text of the LEM-1 test plan (section B). This paragraph summarizes the areas of particular interest.

2.4.1 Booster Aerodynamics.-

2.4.1.1 Shroud Configuration.-

The design criteria for the shroud which are of principal interest are the nose cone half angle, the overall shroud length, and the boattail angle. The nose cone half angle should be kept small to avoid severe buffet behind the shoulder, the overall length should be as short as possible to minimize the destabilizing pitching moment contribution, and the boattail angle should be very small (less than 6°) to preclude separation of flow on the boattail and introduction of unsteady air loads.

Joint Grumman and MSC studies have resulted in a primary and an alternate shroud configuration. (Figure 6-1). The final selection of shroud configuration will be made subsequent to wind tunnel tests.

2.4.1.2 Stability and Control.-

Grumman and MSC digital studies indicate that the booster is capable of performing a satisfactory launch, in a 99% wind, with the present aerodynamic and reaction control systems. These studies included consideration of the booster's mechanical limitations (i.e. maximum elevon deflection, elevon actuator stall torque, etc.), the reduction fin effectiveness in the shroud wake, and the variations in aerodynamic parameters (C_L, etc.) due to aeroelastic effects. The data for these studies was estimated based on the wind tunnel results in the Little Joe II/CM/SM configuration.

The digital analysis indicated that the Little Joe II/LEM flight requires only a single change in the boost vehicle as configured for the CM/SM flight. This change can be either reactivation of the RCS earlier in the flight (RCS is shut off at t+8 until reactivated) or an increase in autopilot gains. Grumman has satisfactory results for each of these changes; however, the data presented in this report reflects earlier reactivation of the RCS.

The RCS was reactivated at the earliest practical time (t+51 seconds) consistent with the burn time available at maximum thrust (40 seconds).

2.4.1.2 (con't)
This is not to be interpreted, however as a marginal flight which does not consider contingencies. It is noted that the RCS burn time is actually longer for lesser thrust levels, that the autopilot gains can be increased in combination with the RCS change, and finally that the 99% wind is a stringent design condition and is not a practical nor a realistic launch condition.

Grumman concludes on the basis of digital analysis that control of the Little Joe II/LEM is feasible and recommends that wind tunnel testing be scheduled and that the booster manufacturer begin the analog studies which are required for final firming of the configuration.

2.4.1.3 Aerodynamic Drag at Separation.-

The dynamic pressure at booster engine cutoff (Figure 4-7) creates a sizeable drag force on the LEM and on the launch vehicle. This force has two fold input to the test. It affects the LEM separation from the adapter, and is also "reverse ullage", that is it causes the propellant to collect at the top of the LEM tanks.

2.4.1.3.1 Separation.-

Grumman desires the simplicity of separating the LEM from the adapter using the force from preloaded springs. The drag at separation increases the size of springs required, and study of this area is not complete.

Should spring actuated separation prove undesirable, separation will then be accomplished by retrograding the booster with small rocket motors. The latter, however, requires some booster development as the present vehicle does not have this capability.

2.4.1.3.2 Ullage.-

The "reverse ullage" created by the drag force can be overcome by either coasting to a higher altitude and lower dynamic pressure, or by installation of posi-grade ullage rockets in the LEM. The loss of test altitude and test time due to coast appears to be undesireable at the present time. This report, therefore, includes a description of a rocket motor installation to demonstrate the feasibility of that approach.

2.4.2 Structural.-

A structural design analysis of the launch vehicle indicated that the following are the results of that analysis:

a. The LEM shroud can be designed to minimize panel flutter.

b. The booster is flutter free up to dynamic pressures of 1580 psf and Mach 5.0.

2.4.2 (con't)

c. Preliminary calculations indicate no extensive design problems due to buffeting.

d. The Little Joe II vibration environment may be higher than the C-5, however, the difference is not believed large enough to effect equipment or structural design.

e. No difficulty is expected in designing for the acoustic noise levels though they exceed C-5 levels.

f. The basic booster structure is well within the strength required for the LEM payload. Doubling the number of attachment bolts in the booster/adapter interface attachment ring will provide adequate strength to sustain the LEM load.

g. The ultimate fin loads of the launch vehicle for the LEM test exceed the booster design ultimate by 2000 lbs., however, the GD/C stress report indicates a strength margin of 15,000 lbs. is still available.

2.4.3 Launch Considerations.-

Payload, booster motor burn sequencing, maximum dynamic pressure, etc. are all intimately related in planning a launch. The principal considerations used to guide preparation of the Little Joe II/LEM test plan were:

a. Due to the doubtful aerodynamic situation which exists transonically, the trajectory should result in a single transonic pass and should not cause the vehicle to dwell in the transonic region.

b. The boost vehicle is flutter free to dynamic pressures of 1580 psf and a Mach Number of 5.0.

c. The design longitudinal load factor for the LEM is 5.6 g.

d. The Little Joe II boost vehicle should not require major modification and redevelopment.

2.4.4 Range Safety.-

A nominal no perturbation trajectory indicate the LEM ascent stage will impact on the range cenerline 80 miles down-range from the launch site. An analysis of the impact dispersion for possible perturbations indicated that worst case perturbations would result in wide variations, some of which exceeded the range boundaries. A root sum square analysis indicated the test could be restrained within the lateral range boundaries while overshooting the longitudinal boundary slightly.

2.4.4 (con't)

Due to the inability to predict the perturbations to be encountered and the proximity of the range boundaries to the vehicle flight path, a capability to apply flight path corrections has been included in the LEM. In addition, Grumman recommends that early discussions of range safety be held with WSMR personnel to correctly assess the range safety aspects of the LEM test.

2.5 Mission Objectives.-

2.5.1 General.-

Early LEM test planning indicated a LEM-1 flight test with LEM-2 providing a spare capability. During the preparation of the LEM-1 feasibility study it became evident that a single flight could not satisfy all of the test requirements, and that the complexity of the single flight LEM-1 test was undesireable for an initial test. A study was therfore conducted of alternate flight plans to satisfy the mission objectives.

2.5.2 Individual LEM Stage Tests - Two Boosters.-

Consideration was given to testing the LEM descent stage on the first Little Joe booster and the ascent stage on the second. This approach would simplify the tests through elimination of a propulsion subsystem, reduction in the instrumentation required, and by a reduction in the requirements for the command system (less commands). This approach would also provide a full duration firing of each stage.

The disadvantages of individual tests are that only one fire-in-the-hole configuration (abort or lunar launch)could be evaluated; there would be no spare available in the event of mishap; and an undesireable delay is created before results are available on both propulsion subsystems.

2.5.3 Ascent Stage Alone - Combined Ascent/Descent - Two Boosters.-

A test of ascent stage with a dummy descent stage for fire-in-the-hole evaluation, followed by a flight with combined ascent/descent testing would satisgy all test requirements. The simplification of the individual stage testing would be attained for the first test, and a reduction in the complexity of the second test could be anticipated based on having previous results.

The disadvantages of this approach to LEM flight testing is the undesireable delay before results become available on both propulsion systems, and also the delay prior to evaluating the exhaust plume effect on the landing leg. A failure or malfunction of the ascent propulsion susbystem on the first flight could result in a fruitless flight.

2.5.4 Combined Ascent/Descent - Two Boosters.-

Two flights which tested both the ascent and descent stage propulsion subsystems would provide fire-in-the-hole tests for both the abort and lunar launch conditions, full duration firing of both the ascent and descent propulsion subsystems (one system per flight), evaluation of the LEM system dynamics across the widest range of propellant loading, and satisfy all other test requirements. The flights would also reduce the complexity required of the single LEM-1 test by dividing the test emphasis between the flights. The LEM-1 flight would evaluate ascent and demonstrate descent propulsion at maximum thrust only. LEM-2 would reverse the emphasis to evaluate descent and demonstrate ascent. This approach reduces the instrumentation and the number of commands required on both flights, and provides results for both propulsion subsystems and the plume effects on the landing leg at the earliest possible date. Another advantage of this test phasing is that for single failures, results are obtained on one of the stages as a minimum. If a descent propulsion malfunciton were to occur, ground command would stage the LEM and advance the programmer to ascent propulsion tests. If a malfunction were experienced in the ascent stage, the descent stage and fire-in-the-hole results would have been obtained prior to the failure.

2.6 Booster Qualification Test Vehicle.-

A study has indicated that a booster QTV is desirable for the following reasons:

a. The Little Joe II/LEM launch vehicle has a bulbous forebody of the type generally referred to as a "hammer head" configuration. Previous vehicles in the category have experienced structural and control problems due to separation of the flow along the shroud boattail. The separation can lead to correlated unsteady air loads which induce primary structural modes and contribute strongly to vehicle dynamic instability. A qualification test vehicle would demonstrate the freedom of the launch vehicle from this adverse condition.

b. A booster QTV, with LEM mass and inertia simulation, would provide the opportunity to demonstrate the shroud and LEM separations without jeopardizing a LEM vehicle.

c. The QTV would be a demonstrator vehicle for retrograde rockets if they become a requirement.

2.7 White Sands/Atlantic Missile Range.-

Section 2.4.4 of this report indicated that discussions concerning the range safety would be held with White Sands Missile Range personnel. The purpose of this section is to provide cursory comments on the effect of conducting the Little Joe II/LEM test at AMR should range safety be a problem at WSMR.

2.7.1 Launch Complex.-

There are no Little Joe launch facilities at AMR. Modification of an existing Redstone complex would be required.

2.7.2 Checkout Facilities.-

The Merritt Island Apollo checkout facilities are planned for completion and could be used at AMR. The problem in this area is the availability of the static firing stands. Present plans call for funding of the stands in the fiscal 1965 budget. Delays in construction might result in delaying the LEM-1 launch date.

2.7.3 G.S.E. Requirements.-

A detailed study is required in this area; however, it is considered that a Little Joe II launch at AMR would reduce the LEM GSE requirements. Fluid and transportation equipment provided for LEM-1 which was applicable to later LEM's would be available for test of the later vehicles. The move to AMR, however, would not eliminate the need for a LEM-1 pre-launch checkout unit. The Little Joe II launch would occur from a Redstone complex and the Apollo PACE would not be available at the site.

2.7.4 Test Operations.-

Conduct of the Little Joe II launch at AMR would be advantageous to general test operations. The lesser scope of the LEM-1, -2 subsystems and test objectives would permit a more gradual introduction to the AMR operation and build efficiency earlier in the program.

2.7.5 Personnel.-

Again further study is required; however, initial thoughts indicate that conducting the Little Joe II test at AMR would simplify rotation of personnel from the LEM-1 to later LEM tests, reduce the resident personnel at WSMR, and thereby decrease the overall personnel requirement.

3.0 Conclusions.-

The following are concluded as a result of this study:

a. Unmanned flight tests are required to certify the safe operation of the propulsion and reaction control subsystems.

b. The Little Joe II should be used to boost the LEM test.

c. The Little Joe II/LEM flight test is feasible. Booster retrograde rocket motors may be required.

d. Two flight tests are required to satisfy all of the development requirements.

e. The optimum mission planning requires testing of both the ascent and the descent stage on each of the flight tests. The test emphasis, however, should be varied (i.e. first flight - ascent; second flight - descent).

f. A booster qualification flight (QTV) is desirable.

4.0 Recommendations.-

It is recommended that:

a. NASA approve the detailed test plans presented for LEM-1, and LEM-2, and the booster QTV (Sections B, C, and D).

b. A booster qualification flight (QTV) be approved and included in Grumman's LEM development plan.

c. Wind tunnel testing to select the final shroud configuration commence as soon as possible.

d. Convair perform a detailed analog analysis of the Little Joe/ LEM configuration.

e. Discussions on range safety be held with White Sands Missile Range personnel as soon as possible.

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1.0 Introduction.-

1.1 Purpose.-

The purpose of this section of the report is:

a. To present the detailed test plan and feasibility data for LEM-1.
To designate the objectives of the test, the data required for LEM-1 evaluation, configuration, instrumentation, trajectory, and flight sequences.

b. To familiarize all cognizant and interested activities with the test.

c. To provoke comment on the test plan and seek NASA MSC approval.

1.2 Revisions.-

This, the initial presentation of the LEM-1 detailed test plan, will be revised to provide additional information which is unavailable at this time. The revisions will be presented in the LEM general test plan, Volume III. The information contained in this report supercedes the May 15 issue of the LEM Test Plan (LPL600-1)

1.3 Definitions.-

1.3.1 Objectives.-

The objectives for this test are described as being either first-order or second-order.

1.3.1.1 First-Order Test Objectives.-

These test objectives define the main reason for making the flight and must be achieved for the flight to be a success. Malfunctions of launch-vehicle systems, ground equipment, or instrumentation which jeopardize the attainment of first-order objectives will be cause to hold or scrub the flight until such time as a fix can be made.

1.3.1.2 Second-Order Test Objectives.-

Second-order test objectives are those desired to support future test-vehicle flights or to supply supplementary data for overall test-vehicle evaluation. Malfunctions of launch-vehicle systems, ground equipment, or instrumentation which jeopardize the attainment of second-order objectives will not require a hold after start of countdown. Corrections thereafter may be made

1.3.1.2 (cont)

at the discretion of the NASA test director.

1.3.2 Definition of System Priorities.-

The launch-vehicle systems are assigned either primary or secondary priorities.

1.3.2.1 Primary Systems (P).-

Primary systems are those functionally required for the launch vehicle to perform its planned mission. A hold or scrub will be mandadatory if any of these systems indicate malfunction prior to launch. Positive indication of satisfactory performance must be available.

1.3.2.2 Secondary Systems (S).-

Secondary systems are those not functionally required for the spacecraft to perform its planned mission. Malfunction of any of these systems may or may not cause a flight countdown hold or scrub, as dictated by the order of the objectives which they support.

1.3.3 Objective Terminology.-

The definition of terms used in connection with objectives for this directive are as follows:

1.3.3.1 Demonstrate.-

Denotes the occurrence of an action or an event. The accomplishment of an objective of this type requires a qualitative answer. The answer will be derived through the relation of this action or event to some other known information or occurrence. This category of objective implies a minimum of airborne instrumentation and/or that the information be obtained external to the launch vehicle.

1.3.3.2 Determine.-

Denotes the measurement of performance of any system or subsystem. This category implies a quantitiative investigation of overall operation which includes, generally, instrumentation for measuring basic inputs and outputs of the system or subsystem. The information obtained should indicate to what extent the system is operating as designed. The instrumentation should allow performance deficiencies to be isolated to either the system or to the system inputs.

1.3.3.3 Evaluate.-

Denotes the measurement of performance of an operation involving

1.3.3.3 (cont.)

people, or the measurement of performance of any system or subsystem as well as the performance and/or interaction of its components that are now under investigation. The accomplishment of objectives of this type requires quantitative data on the performance of both the system or subsystems and its components. The performance levels will then be analyzed for their contribution toward performance of the system. This category will provide the most detailed information of any of these categories.

1.3.3.4 Obtain Data.-

Denotes gathering engineering information which is to be measured to augment the general knowledge required in the development of the overall launch vehicle. This category may also be used for supplemental investigations, such as environmental studies, ground equipment studies, et cetra. The degree of instrumentation is not implied by this definition.

1.3.3.5 Establish.-

Denotes gathering engineering information for the development of ground procedures and operating techniques. Objectives in this category are not necessarily dependent on analytic studies.

2.0 Test Objectives & System Priorities.-

2.1 General.-

The LEM-1 is the first of the unmanned flight tests required to man-rate the LEM vehicle. Test emphasis will be concentrated on the ascent propulsion subsystem and the effects associated with it.

2.2 Test Objectives.-

2.2.1 First Order Test Objectives.-

a. Demonstrate that the rocket motors which have been attached to the descent stage for LEM-2 tests, function satisfactorily and provide sufficient ullage for descent stage ignition.

b. Demonstrate the safe operation of the descent propulsion subsystem at maximum thrust.

c. Evaluate the thermal effect of the descent engine exhaust plume on the landing gear.

d. Evaluate the fire-in-the-hole condition during a simulated lunar launch.

e. Evaluate and certify the safe operation of the ascent propulsion subsystem.

f. Evaluate the thermal distribution of the LEM in the space environment with a fully expanded plume.

g. Re-evaluate the in-flight dynamics of the LEM system throughout a wide range of propellant loadings.

h. Demonstrate clear separation of the descent stage in the zero "g" environment.

i. Evaluate and certify the safe operation of the reaction control subsystem.

2.2.2 Second Order Test Objectives.-

a. Demonstrate the system structural integrity.

b. Evaluate cabin venting during launch.

c. Evaluate the LEM engine induced environments.

d. Demonstrate the compatability of the applicable GSE.

2.3 System Priorities.-

2.3.1 White Sands Missile Range.-

	Item	Priority
a.	Range Communication Network	P
b.	Range Wind Measurement System	P
c.	Range Computer System for determining launcher elevation and azimuth adjustment	P
d.	Motion-picture camera system for engineering coverage of lift-off	P
e.	Range radar tracking network and plotting board system	P
f.	Range optical tracking network	P
g.	Station command destruct transmitter	P
h.	Telemetry station	P
i.	WSMR real-time computer system	P
j.	NASA telemetry trailer	P
k.	Closed-circuit television camera system coverage of lift-off	P

2.3.2 Launcher.-

	Item	Priority
a.	Launcher elevation and azimuth correction system	P
b.	Launcher launch-vehicle umbilical system	P
c.	Launcher retract arms	P

2.3.3 Launch Vehicle.-

	Item	Priority
a.	Algol propulsion system	P
b.	Aerodynamic & RCS Control System	P
c.	Airframe	P
d.	Electrical power system	P
e.	Command destruct system	P

2.3.3 (con't)

	Item	Priority
f.	Instrumentation system	P

2.3.4 LEM Subsystems.-

	Item	Priority
a.	Structure	P
b.	Stab. & Control	P
c.	Environmental Control	P
d.	Electrical Power Supply	P
e.	Propulsion	P
f.	Reaction Control	P
g.	Instrumentation	P
h.	Command	P
i.	Command Destruct	P

2.3.5 Shroud.- P

2.3.6 Adapter.- P

3.0 Mission Description.-

3.1 General Flight Plan.-

The launch is planned to take place from the Little Joe facility of the White Sands Missile Range. The vehicle will be launched at an elevation of 85° and on an azimuth which will depend on the prevailing wind. Four Algol motors will be ignited simultaneously at launch. The launch attitude will be maintained by the booster aerodynamic and reaction controls. At launch plus 39 seconds, the remaining three Algols will be ignited. At BECO, the LEM shroud will be separated followed by LEM separation and booster retrograde. The LEM will be rotated and stabilized with the X-X axis parallel to the weight vector and Z-Z axis in the plane of the planned trajectory. The test ullage rockets will be ignited if they are necessary and the descent engine will be started. The descent engine will be maintained at maximum thrust until all descent fuel is consumed. After DECO, the descent stage will be separated in a sequence which will simulate the lunar launch (fire-in-the-hole). Ascent propulsion tests will commence immediately upon separation and will include steady state performance, multiple starts, including zero "g", operation in the redundant modes, interface tests with the RCS, and possibly malfunction tests by simulated failure (i.e. loss of electric power, etc.). At the completion of the ascent propulsion subsystem evaluation, the LEM reaction control subsystem will be evaluated as the vehicle free falls toward the earth.

3.2 Detailed Test Sequence.-

The time history of test events is tabulated in the following section:

Time, seconds	Event
t-	Electric Power on
t-	Activate Environmental Control
t-	Erect gyros (or Align Platform)
t+0	1st stage booster ignition 4 motors
t+39	2nd stage booster ignition 3 motors
t+44	1st stage booster cutoff
t+62	Fire T.C.A.'s 1 through 16 in 0.15 second intervals (air purge)
t+74	Shutdown T.C.A.'s 1 through 16
t+75	Pressurize RCS (fire squibs)
t+77	Select descent engine on Engine Mode Select
t+81	Actuate descent helium tank squibs
t+82	Actuate descent propellant tank squibs
t+83	2nd stage booster cutoff (BECO determined by acceleration sense) Initiate shroud separation
t+85	Separate LEM from adapter Initiate booster retrograde

3.2 Con't.

Time, seconds	Event
t+86	Rotate LEM X-X axis to earth radial and stabilize
t+87	Select maximum descent engine thrust Initiate RCS ullage firing (if required)
t+88	Ignite LEM ullage rockets
t+89	Fire descent engine
t+90.25	Ullage rocket burnout
t+456	Actuate ascent helium tank squibs
t+457	Actuate ascent propellant tank squibs
t+458	Descent engine cutoff (DECO)
t+459	Select ascent engine on engine mode select Initiate ullage firing of RCS
t+460	Start ascent engine
t+461	Separate descent stage
t+561	Select redundant pressurization and propellant valve
t+621	Shut RCS "A" & "B" fuel and oxidizer shut-off valves
t+622	Select "A" manifold for ascent feed
t+623	Actuate RCS-ascent interconnect valves
t+624	Actuate maniforld crossfeed
t+635	Shut off RCS-ascent interconnect valves
t+636	Open RCS fuel & oxidizer S/O valves
t+838	Shut down ascent engine
t+840	Start
t+843	Shutdown
t+845	Start
t+848	Shutdown
	Stabilize at zero (0) gravity
t+865	Start
t+866	Shut off all RCS regulator S/O valves
t+867	Fire steady-state T.C.A.'s 13 and 14 for four (4) seconds (to decay propellant tank pressures)
t+871	Actuate redundant regulator S/O valves
t+872	Shut off isolation valves affecting T.C.A.'s 1 and 3
t+873	Roll using T.C.A.'s 8 and 16 for two (2) seconds
t+876	Stop roll using T.C.A.'s 4 and 12 for two (2) seconds
t+879	Open isolation valves to T.C.A.'s 1 and 3 Close isolation valves to T.C.A.'s 2 and 4
t+880	Roll using T.C.A.'s 3 and 4 for two (2) seconds
t+883	Stop roll using T.C.A.'s 7 and 15
t+886	Open isolation valves to T.C.A.'s 2 and 14 Close isolation valves to T.C.A.'5 and 8
t+887	Translate in + Z direction using T.C.A.'s 7 and 11 for two (2) seconds.

3.2 (cont.)

Time, seconds	Event
t+893	Stop translation using T.C.A.'s 3 and 15 for two seconds
t+894	Open isolation valves to T.C.A.'s 5 and 8 Close isolation valves to T.C.A.'s 6 and 17
t+896	Translate in - Y direction using T.C.A.'s 4 and 8 for two (2) seconds.
t+900	Stop translation using T.C.A.'s 12 and 16
t+903	Open isolation valves to T.C.A.'s 6 and 7 Close isolation valves to T.C.A.'s 9 and 12
t+904	Thrust in + X direction using T.C.A.'s 6 and 14 for two (2) seconds
t+907	Thrust in -X direction using T.C.A.'s 5 and 13 for two (2) seconds
t+910	Open isolation valves to T.C.A.'s 9 and 12 Close isolation valves to T.C.A.'s 14 and 16
t+911	Thrust in - X direction using T.C.A.'s 1 and 9 for two (2) seconds
t+914	Thrust in + X direction using T.C.A.'s 2 and 10 for two (2) seconds
t+917	Open isolation valves to T.C.A.'s 14 and 16 Close isolation valves to T.C.A.'s 13 and 15
t+918	Positive pitch using T.C.A.'s 9 and 14 for one (1) second
t+920	Stop pitch using T.C.A.'s 1 and 16 for one (1) second
t+922	Open isolation valves to T.C.A.'s 13 and 15 Close isolation valves to T.C.A.'s 10 and 11
t+923	Positive yaw using T.C.A.'s 1 and 14 for one (1) second
t+925	Stop yaw using T.C.A.'s 2 and 13 for one (1) second
t+927	Shut off "A" system fuel and oxidizer main feed S/O valves
t+929	Thrust in + X direction using T.C.A.'s 2, 6, 10, and 14 for one (1) second
t+931	Open system "A"; shut off system "B"
t+932	Thrust in + X direction using T.C.A.'s 2, 6, 10, and 14 for one (1) second
t+934	Open system "B"
t+	Using T.C.A.'s 1 and 13 introduce oxidizer before fuel then vice versa

3.3 Flight Parameters.-

The following parameters are given as descriptive information and are not requirements of the test. The test requirement is simply that the test altitude during the flight be sufficient to provide a "space environment."

3.3.1 Launch (t+).-

Flight path angle - 85°
Azimuth - dependent on the wind

3.3.2 Booster Engine Cutoff (t+83).-

Mach number	4.29
Altitude (above MSL)	162,000 ft.
Dynamic pressure	24 psf
Range	3.2 naut. mi

3.3.3 LEM Descent Engine Cutoff (t+458)

Mach Number	.42
Altitude (above MSL)	1014000 ft.
Dynamic Pressure	0
Range	34.6 naut. mi.

3.3.4 Ascent Engine Cutoff (t+866)

Mach Number	1.90
Altitude (above MSL)	443,000
Dynamic Pressure	0
Range	67.5 naut. mi.

3.4 Trajectory.-

The normal, no perturbation trajectory is shown in Figure 3-1. Since the Little Joe is stabilized only in an attitude hold mode, there will be disturbances from this nominal. The range dispersion due to disturbances and/or the unknowns in prediction is discussed in Section 13.0.

3.5 LEM Orientation.-

Prior to launch the LEM attitude reference will be aligned to the local vertical with the Z axis in the plane of the nominal trajectory. The LEM orientation after initial stabilization in flight will then hold with the X axis vertical and the Z axis parallel to the planned trajectory path until the ascent engine cutoff (AECO).

During the free fall subsequent to AECO, LEM attitude will be programmed for RCS system tests. At any time the attitude command for the RCS test is removed the system will revert to the pre-AECO attitude hold mode.

3.6 Flight Path Corrections.-

If flight path corrections are required prior to AECO, they will be made by applying a component of the ascent or descent engine thrust. A step signal from the ground command will tilt the vehicle about the Z axis and thereby result in a lateral thrust component from the operating engine. This method of control provides a backup to the primary mode, increasing the reliability of the R & D Command System.

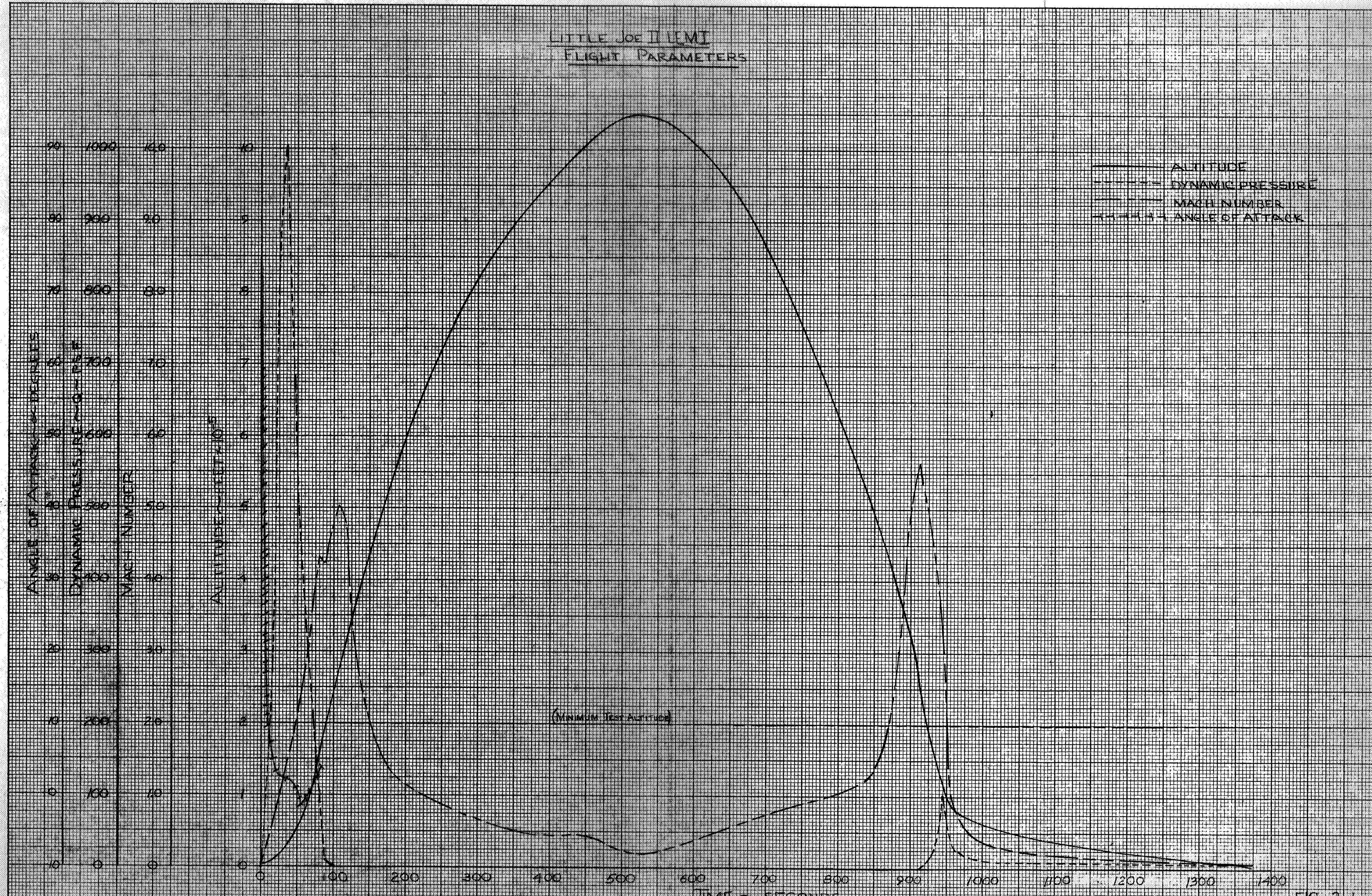

LITTLE JOE II LEM I
FLIGHT PARAMETERS
ALTITUDE
DYNAMIC PRESSURE
MACH NUMBER
ANGLE OF ATTACK
ANGLE OF ATTACK ~ DEGREES
DYNAMIC PRESSURE ~ Q ~ PSF
MACH NUMBER
ALTITUDE ~ FEET × 10^5
(MINIMUM TEST ALTITUDE)
TIME ~ SECONDS
0
100
200
300
400
500
600
700
800
900
1000
1100
1200
1300
1400

FIG 3-1

4.0 Description of the Test Vehicle.-

4.1 General.-

The test vehicle comprises the LEM, shroud, and an adapter, supported on the Little Joe II booster. The LEM is supported by the adapter which is attached to the booster. The shroud is supported by the adapter through a continuous band of skin. The configuration is illustrated in Figure 4-1.

4.2 Vehicle Weight and Balance Summary.-

The mass characteristics of the test configuration prior to launch are presented in Table 4-1. The mass characteristics of the launch vehicle (LEM, shroud, adapter, and Little Joe II booster) during boost are presented in Figure 4-2.

4.3 Shroud.-

4.3.1 Structure.-

The shroud (Figure 4-3) is the aerodynamic fairing for the payload. The maximum diameter (212 inches) is determined by the clearance required at the reaction control system thrusters. An aluminum alloy skin supported by ring frames and longerons comprise the primary shroud structure. Structural continuity between the shroud and the adapter is provided by a continuous circumferential splice.

The shroud contians provisions for access to the LEM and is vented to equalize the internal and external pressures during ascent.

4.3.2 Separation.-

The shroud is composed of two halves spliced together to form a structural seam as shown in Figure 4-4. Separation is effected by severing the spliced joints, both internal and external, by means of a continuous shaped charge. Probability of failure is minimized by using two explosive paths, each capable of breaking the joint. Ignition is started at a minimum of three places insuring deflagration of the complete charge.

Upon separation of the shroud, a pair of thrusters will impart sufficient lateral velocity to each half to allow LEM release without interference.

4.3.2.1 Analytical Data.-

Shroud cutting time	-	1.5 milliseconds
Shroud thruster	-	approx. 400 lbs.
Shroud velocity (avg)	-	5 ft/sec
Shroud clear time	-	2 sec

TABLE 4-1

Item	Weight Lbs.		C.G. Arm - inches			Moments of Inertia, slug - ft^2		
	Ascent	Descent	X	Y	Z	Ix	Iy	Iz
			* Station 0 taken at LEM Adapter/Little Joe II Separation Line					
LEM Subsystems:								
Structure	1219	806						
Stab. & Control	88							
Nav. & Guid.								
Crew Systems								
Environmental Control	380							
Landing Gear								
Instrumentation								
Electrical Power Supply	321							
Propulsion - inert	671	2171						
- usable	Below	Below						
Reaction Control	633							
Communications								
Spares								
Instrumentation	300							
R & D Command	100							
Command Destruct	30							
LEM-1 Totals								
Ascent - inert	3742		-150.1	-1.1	-17.9	1013	623	844
- usable prop	4329		-155.7	0.0	0.0	2258	733	2856
Descent - inert	2977		- 84.7	0.0	0.8	719	380	504
- usable prop	12578		- 71.5	0.0	0.0	8802	6357	4529
Total LEM	23626		-101.1	-0.2	- 2.7	13014	15591	16011
Shroud - N_5	4744		-161.4	0.0	0.0	9026	13962	13962
Adapter	300		- 15.0	0.0	0.0	364	182	182

TABLE 4-1 (con't)

Item	Weight lbs.	C.G. Arm - inches			Moments of Inertia, slug - ft^2		
		X	Y	Z	Ix	Iy	Iz
		* Station 0 taken at LEM Adapter/Little Joe II Separation Line					
Little Joe II:							
Forebody	1919	98.5					
Afterbody	4238	321.5					
Fins - 50 sq ft	3029	331.5					
Electrical	134	95.3					
Measurement & destruct	258	112.7					
Surface Contr. & Guid.	529	278.8					
Reaction Controls	711	318.3					
Algol instl. prov.	91	348.0					
R.C. Propellant	640	318.3					
Algol cases	20846	243.0					
Algol Propellant	133154	186.6	0.0	0.0	62851	233135	233135
LEM-1/Little Joe II Booster B.O.	61065	83.8	-0.07	-1.06	54680	540319	540715
Lift-off	194219	154.3	-0.02	-0.3	117541	869023	869408

4.4 Adapter.-

4.4.1 Structure.-

Aerodynamic and inertia loads from the payload are transmitted to the booster structure through the adapter which is attached to the booster by thirty-two (32) bolted connections. The basic adapter structure (Figure 4-5) consists of a stiffened skin supported on ring frames. The LEM is supported at eight (8) load bearing points on the adapter, four (4) of which carry tension loads. The length of the adapter is determined by the clearance between the descent engine and the booster structure.

4.4.2 LEM-Adapter Separation.-

Four structural pick up points on the LEM descent stage (Figure 4-6) provide a means for adapter tension attachment. Explosive bolts having dual bridge-wire type igniters with dual ignition circuits, are used to make connections. Preloading is accomplished by the initial installation and tightening of the explosive bolts. Compression and shear loads are distributed by eight (8) structural fittings on the underside of the LEM.

Retrograde rockets will be installed on the booster to provide sufficient clearance for safe descent stage engine ignition. The specific rockets to be used shall be determined subsequent to further studies.

4.4.3 Ullage Rocket Motors.-

An estimated drag force of 1800 points on the LEM at the proposed time of descent engine ignition (Figure 4-7) creates a "reverse ullage" condition causing the propellants to collect at the top of the tanks. This condition may be overcome by allowing the vehicle to coast to a higher altitude and lower dynamic pressure where the 400 lbs. of available RCS thrust can provide the necessary ullage or by using posigrade rockets on the LEM. Deterioration of test time and altitude due to coast appears to be undesireable; therefore, the following ullage rocket motor installation has been presented to demonstrate the feasibility of such an approach.

Space allocated to cryogenic and water storage in the descent stage is not utilized and, therfore, is available for mounting ullage rockets (Figure 4-8). The areas involved are diametrically opposed offering symmetrical motor locations and satisfy the following conditions when a motor with a canted nazzle(such as the MK 25, Mod O, JATO unit).

a. The thrust vector of each ullage rocket motor must pass through the center of gravity of the vehicle..

b. The ullage rocket exhaust plume must not impinge upon any portion of the descent structure.

4.4.3 (con't)

c. The ullage rocket motor nozzles must be clear of the descent engine exhaust plume (-110° from descent thrust vector).

d. The ullage rocket motor nozzles must be clear of the locus of the 6° gimbal excursion of the descent engine nozzle skirt.

e. There must be adequate local "hard" structure to permit attachment of the ullage motors.

Although such an installation requires truss type engine supports and local reinforcement of descent structure, it entrails no major structural redesign.

4.5 LEM.-

4.5.1 Structure.-

The LEM is essentially the prototype vehicle less landing gear and landing gear support outriggers. Support of the LEM in the adapter therefore, is not typical of the prototype. The eight (8) adapter support points mate with local structure provided at the ends of the beams comprising the primary structure of the descent stage.

4.5.2 Propulsion Subsystem Description.-

The LEM vehicle incorporates two independent propulsion subsystems, the descent engine and the ascent engine. Both engines use liquid propellants with a pressurized feed system. The two thrust chambers are cooled by ablation of the chamber lining. The descent engine is throttleable from 10,500 lbs. thrust to 1,050 lbs. thrust and the chamber can be gimballed. The ascent engine thrust is fixed at 3,500 lbs. and the chamber can not be gimballed.

4.5.2.1 Descent Engine (Figure 4-9).-

4.5.2.1.1 Descent Engine Helium Pressurization Section.-

The descent engine propellant tanks are pressurized by the helium pressurization section. In this section, gaseous helium is stored at 4500 psia in two spherical vessels. The helium spheres discharge into a manifold. A normally-closed, squib-actuated valve at the manifold discharge isolates the helium supply prior to start. The squib valve is followed by a filter which is included to prevent contamination of the helium pressure regulation assembly. Downstream of the filter, the system divides into two flow paths. There are, in each flow path, a normally-open, solenoid-actuated, latching valve and two pressure regulators in series. Each of the four regualtors has a different setting. If the lowest regulator discharge pressure were designated 1 and the other settings were designated 2, 3, and 4 in the order of increasing pressure, the regulator arragy could be described as follows: in the primary leg the first reg is set at 2 and the downstream regulator is set at 3;

4.5.2.1.1 (con't)

in the secondary leg the first reg is set at 1 and the downstream reg is set at 4. The output of the pressure regulation assembly is normally determined by the upstream regulator of the primary leg. Should the upstream primary regulator fail closed, the output of the pressurization assembly would revert to the upstream regulator in the secondary leg. In the event either leg is disabled by regulator failures, the solenoid valves can be used to shut off that leg. Check valves are provided downstream of the regulator assembly in each leg to prevent backflow. Downstream of the check valves the flow paths combine to manifold the primary and secondary helium supply. There are two paths out of that manifold, a fuel tank helium supply, and an oxidizer tank helium supply. These supply lines incorporate quad check valves to prevent backflow of the propellant vapors. In addition, a normally-closed, squib-actuated valve isolates the tank ullages from the helium supply section prior to start. There are two fuel tanks and two oxidizer tanks. The two fuel tank ullages are manifolded as are the oxidizer tank ullages. Overpressure in any tank is vented through a relief valve on the fuel ullage manifold or a relief valve on the oxidizer ullage manifold.

4.5.2.1.2 Descent Engine Propellant Feed System.-

Descent engine fuel and oxidizer are carried in four cylindrical tanks with hemispherical caps. The fuel tank outlets are manifolded and the manifold discharges to a common fuel line. The common fuel line incorporates a burst disc which isolates the propellant from the chamber prior to pressurization of the propellant tanks. The disc is ruptured at start by the use of tank pressure. Downstream of the burst disc is a 200 micron filter to prevent contamination of the propellant valves. In the same manner the oxidizer tank outlets are manifolded, discharging through a burst disc and filter.

4.5.2.1.3 Rocketdyne Descent Engine Section.-

The Rocketdyne descent engine is an integrated package which includes a thrust chamber and injector assembly, upstream main propellant valves, isolation valves, a helium injection assembly, and a gimbal mount.

Immediately downstream of the calibration orifices at the engine/feed system interface, the common fuel supply line and common oxidizer supply line divide to provide two flow paths to the chamber. Each flow path contains two propellant valves, a main propellant throttle valve, and, downstream, an isolation valve. In each flow path fuel and oxidizer valves are paired; that is, each throttle valve consists of a fuel valve and an oxidizer valve fastened together. The throttle valves are used to meter propellant flow to the chamber to regualte chamber pressure. Mixture ratio is held constant.

4.5.2.1.3 (con't)

Control is achieved in the following manner. Thrust chamber pressure is sensed by a transducer. The voltage analog produced by the transducer is summed algebraically with a command voltage to generate an error voltage. The error voltage is converted to a proportional current which drives a torque motor on the throttling valve. The throttling valve is positioned by a hydraulic actuator using fuel as the control fluid. The actuator is positioned by a three-way pilot spool which meters fluid flow to the actuator. The pilot spool is positioned by the torque motor. The actuator positions the oxidizer poppel which, in turn, moves the fuel poppel. The unit is spring-loaded closed and pressure unbalanced closed.

The isolation valves are also assembled in fuel/oxidizer valve pairs. The isolation valves are on-off valves, hydraulically actuated by fuel pressure through a normally-closed, three-way solenoid.

Helium injection is provided downstream of the propellant valves in each flow path. The helium supply is from the propellant tanks pressurization system. The supply for the fuel lines is separate from the oxidizer supply line. In each line control is provided by a series/parallel array of injection valves. This array consists of two normally-closed, DC operated, axial flow solenoid valves. Helium injection is to maintain the injector pressure drop when the engine is throttled. This contributes to combustion stability.

The dual flow paths are re-joined upstream of the chamber manifold. The propellant manifolds are segmented to provide uniform propellant distribution to the injector rings. The injector is a flat face with ring and groove construction and is fabricated from aluminum.

The thrust chamber utilizes a nozzle extension which attaches at the 6/1 expansion area ratio point. The nozzle extension is radiation cooled while the remainder of the chamber is ablation cooled. The ablative liner is a laminated structure utilizing high-silica-glass cloth impregnated with a phenolic binder. An asbestos-phenolic material is bonded to the exterior surface of the ablative liner to reduce the outer surface temperature. Structural loads in the chamber will be carried by a filament-wound fiberglas support shell employing both longitudinal and circumferential banding. The injector will be bonded to the thrust chamber and structurally supported by the filament-wound shell.

The engine start sequence is initiated with the pressurization of the propellant tanks. During the propellant tank pressure rise, the burst discs in the fuel and oxidizer lines rupture and the propellants are primed to the main valves. At the fire command signal both sets of throttle valves and isolation valves go full open. The valves are mechanically linked to provide an oxidizer lead. Ignition is hypergolic. In the event of a valve malfunction, the propellant flow path can be shut off and engine operation will continue using the remaining flow path. The cutoff command closes both sets of throttle valves and isolation valves.

4.5.2.1.4 Parallel Descent Engine (STL).-

STL is also developing, in a parallel effort, a throttleable descent engine for LEM. Either the Rocketdyne or STL engine will be selected before the LEM-1 flight. The STL Descent Engine Section consists of a single element variable area injector, a pair of cavitating venturi propellant flow control valves, an electromechanical thrust control actuator, mechanically linked fuel actuated bipropellant shut-off valves, an ablative chamber with a radiation cooled nozzle extension. The engine may be throttled from a maximum of 10,500 lbs. thrust to a minimum of 1,050 lbs. thrust. The thrust chamber may be gimballed.

Oxidizer and fuel are supplied from the propellant feed section which is described in Section 4.5.2.1.2. Downstream of the propellant feed section/eengine interface, the propellants flow through the flow control valve. The flow control valve is a pair (one oxidizer, one fuel) of mechanically linked variable area venturi. The valve incorporates a mixture ratio compensator which adjests the mechanical linkage and varies the relative rate of travel of the valve pintles. The mixture ration compensator is manually set. The valve pintles are contoured to vary the fuel and oxidizer flow rates linearly with pintle displacement. As the engine is throttled, the flow control area of the valve is decreased so that the pressure drop across the valve increases by an amount equivalent to the decrease in chamber presssure and injection occurs in the valve throats. From this level down to minimum thrust, the valve functions as a cavitating venturi, and the flow is controlled only by the throat area of the valve and propellant conditions at the valve inlet.

After flowing through the control valve assembly, the propellants pass through the parallel series shutoff valves. The shutoff valves are mechanically linked, pilot operated, fuel actuated ball valves. Fuel and oxidizer are completely isolated in separate valve bodies, with actuation of the oxidizer valve by a mechanical linkage driven from the fuel valve. The shutoff valve pilot valves are solenoid operated. All of the pilot valve solenoids are energized simultaneously at engine start.

Downstream of the shutoff valves the propellants pass through the injector assembly. This assembly includes the injector faceplate, the fuel manifold assembly, the propellant metering sleeve, and the oxidizer feed tube assembly, Fuel enters the combustion chamber through the annular gap formed by the faceplate and the metering sleeve. The meteting sleeve is the only moving part in the injector.

The thrust control is an electromechanical linear servo actuator. It contains redundant drive motors, clutch assemblies, rate generators, position transducer assemblies, and amplifiers. The motors are coupled to a common output shaft by "siamese" magnetic particle clutches and reduction gearing. When one side of the "siamese" clutch is energized, the output shaft moves in one direction; energizing the other portion of the clutch reverses the

4.5.2.1.4 (con't)

output. Because the servo has limited damping, it is supplemented with rate feedback supplied by a rate generator. The position feedback transducer provides a signal into the servo loop so that the output position is proportional to the input command. The rotary motion output of the clutches is converted to linear motion by a ball screw-and-nut arrangement. In the event of failure of either motor, the other motor has sufficient power to move the throttle linkage under a full load.

The thrust chamber consists of a composite ablative cooled chamber, nozzle throat, and nozzle divergent section to an area ratio of 16:1, and a radiation cooled nozzle extension to an area ratio of 49:1. The ablative sections are encased in a continuous titanium shell and jacketed by a pair of thin aluminum thermal radiation shields. The nozzle extension is flanged to the titanium shell. The assembly is supported by a gimbal ring.

4.5.2.2 Ascent Stage Propulsion Subsystem (Figure 4-10).-

4.5.2.2.1 Ascent Helium Pressurization Section.-

The ascent helium pressurization section is fully redundant. The gaseous helium is stored at 4500 psia in two spherical vessels. Either bottle can supply the helium required for mission completion. A check valve is installed at the discharge of each bottle to insure that the loss of one bottle would not deplete the entire supply. Downstream of each check valve is a squib-actuated valve which fires at engine start. Downstream of the squib valves is a crossover line connecting the parallel helium legs. There is a third squib-actuated valve in this line, the emergency crossover valve. The crossover valve is opened in the event of a malfunction.

Continuing downstream in the parallel flow paths the helium passes through a filter (to prevent regulator contamination) and then through a normally-open, latching, solenoid valve. Downstream are two pilot actuated, dome-loaded pressure regulators in series. The regulator discharge pressures are set at various levels. The scheme parallels that used in the descent engine (see 1:1). Downstream of the regulators in each leg is a quad check valve. Each quad check valve discharges in two paths, fuel and oxidizer. The primary leg fuel helium and secondary leg fuel helium lines meet at a second quad check valve. The same arrangement permits either helium supply to feed both fuel and oxidizer tanks and, at the same time, back flow between the fuel and oxidizer systems and between the parallel helium supplies is prevented.

Downstream of the quad check valves are squib-actuated valves which isolate the propellant tanks during helium section checkout. The flow paths divide again downstream of the squib valves to feed each of the two fuel tanks and two oxidizer tanks. The use of a common ullage pressure supply line minimizes the pressure differential between the oxidizer tanks. For the same reason the vent valves are located on the common lines. A burst disc is located upstream

4.5.2.2.1 (con't)
of each relief valve to protect the relief valves from the corrosive fuel and oxidizer vapors. A vent check valve is provided between the burst disc and the relief valve to prevent leakage from negating the purpose of the burst disc. The burst disc ruptures in the event of overpressure and the vent check valve closes due to the high pressure and the relief valve controls the high pressure.

4.5.2.2.2 Ascent Propellant Feed Section.-

The ascent engine fuel and oxidizer are carried in four spherical tanks, two each for fuel and oxidizer. The tanks for each propellant discharge into a common full line and common oxidizer line. Each of these lines contains a burst disc to prevent leakage prior to start, and a filter to prevent contamination of the main propellant valves.

4.5.2.2.3 Ascent Engine.-

Immediately below the engine/feed system interface are calibration orifices (one for each propellant). Downstream the propellant flow paths divide to provide two flow paths to the thrust chamber. As in the descent engine each flow path incorporates two main propellant valves in series. In the ascent engine both valves are on-off valves. The upstream valve is the isolation valve and the downstream valve is the bipropellant valve. Each valve assembly consists of a pair, one fuel valve and one oxidizer valve. The valves are normally-closed, mechanically linked, dual ball valves, controlled by hydraulic actuators. The actuating control fluid is fuel. The application of pressure to the actuators is controlled by normally-closed, solenoid valves. Each pair of fuel and oxidizer valves is controlled by a single actuator. Valves in the parallel legs are opened and closed simultaneously at start and cut-off. In the event of a malfunciton, the valves may be controlled independentlv.

The ascent engine thrust chamber uses a flat face injector. The chamber is ablation cooled.

4.5.3 Stabilization and Control Subsystem.-

4.5.3.1 Requirements.-

The Stabilization and Control Subsystem requirements for LEM #1 are as follows:

1. Stabilize vehicle about all three axes.
2. Hold attitude with X & Z axis in the plane of the trajectory.
3. Accept and respond to automatic programmer sequence outputs.
4. Accept and respond to emergency ground commands. The ground commands to have priority over onboard commands.
5. Accept and respond to ground initated flight path correction commands if required.

4.5.3.2 General Description.-

4.5.3.2.1 Hardware.-

The Stabilization and Control Subsystem for LEM-1 will consist of the:

1. Control coupler assembly (CCA)
2. Attitude and translation control assembly (ATCA)
3. Descent engine control assembly (DECA)
4. Rate gyro assembly (RGA)
5. Attitude reference assembly (ARA)
6. In flight monitor assembly (IFMA)
7. Gimbal drive actuators

The equipment in the above assemblies will be unmodified LEM equipment except for the control coupler assembly and the attitude reference assembly. This assembly replaces the guidance coupler assembly of the standard LEM subsystem. The control coupler assembly not only accepts and routes the onboard and ground command but also contains stored command signals.

A signal flow diagram for the SCS is given in Figure 4-11. The IFMA, not shown in the figure, is provided for purposes of instrumentation.

4.5.3.2.2 Functional Operation.-

4.5.3.2.2.1 Attitude Stabilization.-

For the attitude stabilization mode the control system will funcition as a rate command system with attitude hold (Figure 4-13). After the vheicle is initally stabilized, the reference loop (switch as shown) holds the vehicle at the desired orientation satisfying requirements 1 and 2. The vehicle attitude reference will be initially aligned to the local vertical with the Z axis in the plane of the nominal trajectory.

4.5.3.2.2.2 Programmed Testing.-

For the RCS portion of testing where attitude and translation are to be programmed, rate commands will be utilized for rotation and acceleration command for translations. For rotation about a particular axis, a rate command will be applied at the summing point 1, (refer to Figure 4-13), by closing the switch. While the vehicle is rotating, the attitude reference will be synchronized and the system will revert to attitude hold when the command is removed. Commands will be applied to one axis at a time to minimize crosscoupling effects. Translation commands will be routed to the desired jet driven amplifiers through the logic switching.

4.5.3.2.2.3 Flight Path Corrections.-

If flight path corrections are required they will be made by applying a component of the propulsion thrust vector to laterally displace the vehicle. To accomplish this, an attitude rate signal will tilt the vehicle resulting in the desired lateral thrust component of the propulsion engine thrust. This signal will be a ψ c command (rotation about the Z axis).

4.5.3.2.3 Mechanization.-

Both programmed and emergency commands enter the control coupler assembly (CCA). These commands are accepted with the emergency commands having priority over onboard commands. In the automatic mode the programmer drives relays or soled-state switches to activate stored command circuits in the C.C.A. These stored commands are sent to the appropriate assembly, resulting in a system response.

In order to perform the RCS portion (phase 5) of the test the thruster isolation signals will activate the thruster failure logic (located in ATCA) (see Figure 4-12) to simulate a thruster failure. This signal will select the desired alternate pair of thrusters to be fired.

Throughout the descent and ascent engine testing, vehicle attitude will be held to a local vertical unless trajectory corrections or other emergencies dictate otherwise. Stored commands for the propulsion subsystem will also be located in the C.C.A. and activated by the programmer.

4.5.3.3 Component Description.-

4.5.3.3.1 Rate Gyro Assembly (RGA).-

The rate gyro assembly is not to be modified for LEM #1. Two identical rate gyro assemblies are used with automatic failure detection and switchover circuitry (external to the RGA's).

Each Rate Gyro Assembly consists of three sub-miniature single-degree-of-freedom rate gyros so mounted that they sense vehicle roll, pitch, and yaw rates respectively. Each gyro is capable of operating with input rates up to ±20 degrees per second. The assembly operates on 26 volt 800 cps three phase power and supplies 800 cps rate signals to the appropriate channel of the Attitude and Translation Control Assembly. Each gyro contains self-test features consisting of a spin motor, rotation detector, and a self-test torquer.

4.5.3.3.2 Attitude and Translation Control Assembly (ATCA).-

The ATCA will not be modified for LEM #1; however, not all of its capabilities are required for the LEM #1 mission. Inputs not needed for this unmanned mission, such as direct mode commands, are grounded external to the ATCA (in the command coupler).

4.5.3.3.2 (con't)

The ATCA supplies the signals which control the firing of the reaction jets and the signals for automatic trim of the gimballed descent engine. The DC voltages required in all of the SCS sub-assemblies and the synchronized AC voltage for the RGA and command coupler are also generated within this assembly. The input signals from the CCA and RGA are processed in the ATCA and are directed to the appropriate jets and the descent engine control assembly. Figure 4-12 is a block diagram of the ATCA.

1. Input Signals - The input signals to the ATCA are the output signals from the Command Coupler Assembly and the Rate Gyro Assembly. In addition, power input of +28 VDC at 80 watts and the AC reference signals (from the ARA) for synchronization are required. Switching signals for the various selectable gains, rate limits and deadbands are sent from the Command Coupler Assembly. Jet failure compensation is accomplished by the Logic of the ATCA as a result of inputs from the Command Coupler. This feature will be utilized to implement the RCS test sequence.
2. Output Signals -

   a. Excitation signals to the solenoids of the Reaction Jets.

   b. Automatic Trim Signals to DECA consisting of two bi-polar proportional DC voltages (one each for yaw and pitch).

   c. In Flight Monitor and Test Signals which are being determined as part of the instrumentation effort.

   d. DC voltages to DECA, RGA, CCA.

   e. Synchronized 3 Ø, 800 cycle/second voltage to the RGA and single phase to the command Coupler and RGA.

3. Block Diagram Description - The following paragraphs detail the major subassemblies of the ATCA as shown in Figure 4-12. The autopilot electronics is concerned with the signal conditioning, feedback signals and required shaping. The logic follows the autopilot electronics and directs the error signals to the appropriate pulse modulators and solenoid drivers. The solenoid drivers control the reaction jet solenoids.

   a. Auto Pilot Electronics - The basic function of the Auto Pilot Electronics portion of the Attitude and Translation Control Assembly is to supply the Reaction Jet Logic and Descent Control Assembly with corrective attitude thrust command signals and c.g. offset compensation respectively. Attitude error or rate command signals and rate feedback signals are processed in the Auto Pilot Electronics to obtain optimum control loop performance.

4.5.3.3.2 3. a. (con't)
In the block diagram of the Attitude and Translation Control Assembly it is shown that the Auto Pilot Electronics consists of three similar channels of electronics for attitude control. Each channel receives its corresponding roll, pitch, or yaw error command signal which is then processed through a limiter and summed with the appropriate rate signal. This summed signal is then fed into the reaction jet logic. In the pitch and yaw loops this summed signal after demodulation is also sent to the Descent Engine Control Assembly for auromatic trim of the descent engine. The translation signals from the Command Coupler are directed to the logic assembly where they are combined with attitude signals and directed to the appropriate thrusters.

1. Selectable Rate Gradient - This element changes the amount of rate feedback in the attitude control loop to obtain optimum loop performance throughout the mission. Presently, only one gain change (at staging) is foreseen. The gain in all 3 axes will be switched upon receipt of the staging signal from the programmer coupler.

2. Position - Rate Signal Summation - Since the positional error and rate signals supplied to the Auto-Pilot Electronics Subassembly are synchronized 800 cps signals, it is possible to AC sum and amplify these signals prior to demodulation. One AC amplifier per channel will be needed to sum and amplify the error signals from the rate limiting and selectable rate gradient elements.

3. Dead Band - This element will be a selectable two level device, a large dead band ($\pm 5^{\circ}$) and a narrow dead band ($\pm 0.1^{\circ}$). The selection will be performed by a discrete signal from the Command Coupler (all 3 axes simultaneously).

b. Logic - The logic directs rotational and translational signals to the proper thruster channels for optimum system performance. Redundancy for thruster failures and means to prevent simultaneous ignition of opposing thrusters are provided in the logic configuration. For combinational commands the logic insures maximum moment control and optimum thruster use whenever possible.

Figures 4-14a and 4-14b contain the thruster logic configurations and thruster configurations for the vertical and horizontal thrusters respectively. Tables 4-2a and 4-2b list the thruster channels to be energized for independent rotational and translational inputs under various thruster pair failure conditions. False failure signals will be sent from the CCA to ensure specific thruster firings during RCS testing.

In order to describe the operation of the logic in the following paragraphs the term "contact" is used. In the actual system the switching may be accomplished using solid state devices.

HORIZONTAL THRUSTER IGNITION CONDITIONS

FOR

INDEPENDENT INPUT COMMANDS

AND

THRUSTER FAILURES

FUNCTION	NORMAL	FAILURE 1-3	FAILURE 2-4	FAILURE 6-7	FAILURE 5-8	FAILURE 10-11	FAILURE 9-12	FAILURE 13-15	FAILURE 14-16
P_1 +X Rotation	12, 4	12, 4	7, 15*	12, 4	12, 4	12, 4	7, 15*	12, 4	12, 4
P_2 -X Rotation	16, 8	16, 8	16, 8	16, 8	11, 3*	16, 8	16, 8	16, 8	11, 3*
Y_1 +Y Translation	16, 12	16, 12	16, 12	16, 12	16, 12	16, 12	16 (I)	16, 12	12 (I)
Y_2 -Y Translation	8, 4	8, 4	8 (I)	8, 4	4 (I)	8, 4	8, 4	8, 4	8, 4
Z_1 +Z Translation	11, 7	11, 7	11, 7	11 (I)	11, 7	7 (I)	11, 7	11, 7	11, 7
Z_2 -Z Translation	15, 3	15 (I)	15, 3	15, 3	15, 3	15, 3	15, 3	3 (I)	15, 3

* REDUNDANT SWITCH INITIATED BY FAILURE LOGIC
(I) INDICATES CONDITION OF INCOMPLETE CONTROL

TABLE 4-2

VERTICLE THRUSTER IGNITION CONDITIONS

FOR

INDEPENDENT INPUT COMMANDS

AND

THRUSTER FAILURES

FUNCTION	NORMAL	FAILURE 1-3	FAILURE 2-4	FAILURE 6-7	FAILURE 5-8	FAILURE 10-11	FAILURE 9-12	FAILURE 13-15	FAILURE 14-16
Q_1 +Y Rotation	5, 2	5, 2	9, 14*	5, 2	9, 14*	5, 2	5, 2	5, 2	5, 2
Q_2 -Y Rotation	1, 6	13, 10*	1, 6	13, 10*	1, 6	1, 6	1, 6	1,6	1, 6
R_1 +Z Rotation	1, 14	10, 5*	1, 14	1, 14	1, 14	1, 14	1, 14	1, 14	10, 5*
R_2 -Z Rotation	13, 2	13, 2	9, 6*	13, 2	13, 2	13, 2	13, 2	9, 6*	13, 2
X_1 +X Translation	10, 2	10, 2	14, 6*	10, 2	10, 2	14, 6*	10, 2	10, 2	10, 2
X_2 -X Translation	9, 1	13, 5*	9, 1	9, 1	9, 1	9, 1	13, 5*	9, 1	9, 1

* REDUNDANT SWITCH INITIATED BY FAILURE LOGIC

TABLE 4-2

4.5.3.3.2 3. b. (con't)

The failure logic is so designed that for specific failure indication and error signals present switching to redundant channels takes place. By combining the failure indications and error signals present through logic, full use of normal thrusters is realized. For example, if thruster (2) channel fails, redundant switching is required for positive Y rotational input signals, but not for negative Y rotation input signals.

The method of transfer for rotation signals about the Z and X axis is similar to that for the Y axis signals.

Translation signals are transferred in much the same manner as rotational signals. The Y and Z axis translation signals do not have a redundant transfer path since only one set of thruster channels are available to perform each required funciton.

Combination logic switching has been incorporated to eliminate selections of the same thruster for ignition by two independent signal inputs. This feature prevents possible summing amplifier saturation which would result if two large input command signals were added. The combination logic switching will not apply for conditions of three or more simultaneous input signals since selective thruster operation is limited by the basic thruster configuration. As a result, for for more than two simultaneous input signals, the possibility will exist of multiple signals commanding the same thruster ignition. The combinational logic is switch-off whenever a failure indication is received from the control panel and switching of dual contact pairs is then completely controlled by the failure logic. This feature allows selection of particular thrusters for test purposes.

4.5.3.3.3 Descent Engine Control Assembly (DECA).-

The DECA does not require modification for LEM #1. DECA contains circuitry for two functions of the descent engine, throttle control and c.g. trim gimbal electronics. In addition, the DECA transfers signals for engine ignition and shutdown.

1. Throttling Commands - Descent engine variable thrust is controlled linearily with an analog signal of zero to five (5) volts d-c. For LEM #1 this signal is programmed automatically in the Command Coupler.

2. Engine Start Signal - Engine ignition will be actuated by a 28VDC signal originated in the Command Coupler. The signal is routed from the CCA through the DECA and then to the descent engine. This same signal is used to arm the gimbal position actuators.

4.5.3.3.3 (con't)

3. Engine Shutdown - This is a separate 28 VDC signal initiated in the CCA. The signal is routed through the DECA and then to the descent engine.

4. Gimbal Positioning - When the engine is thrusting, gimbal commands are derived in the ATCA from steady state attitude errors and fed to the gimbal actuation system via the DECA Electronics. The descent engine gimbal is driven by an electromechanical actuator so that the thrust vector always passes through the moving c.g. The DECA contains the motor control current signal switching and malfunction select logic and drive amplifiers. Relay amplifiers (one for each of 2 axes) close the actuator motor power relays as a function of steady-state vehicle attitude command error. The gimbal drive electronics are completely redundant.

4.5.3.3.4 Attitude Reference Assy. (ARA).-

The Attitude Reference Assy. consists of a four-gimbal platform and the associated electronics. The accelerometer package which is part of the ARA is not required for LEM #1. Since the platform will be aligned remotely on the ground prior to launch, a platform control and align panel must be provided with umbilical connections to the LEM.

The output of the ARA is an attitude error signal that will be routed to the ATCA through the C.C.A. Power supply for the ARA is internal and is synchronized to the control electronics power supply.

4.5.4 Command System.-

4.5.4.1 General.-

In order to comply with the test sequence and to insure the greatest probability of success for the LEM #1 mission, the command system will have both onboard and ground command capabilities.

The three (3) modes of operation are:

1. Automatic program mode
2. Emergency ground command mode
3. Range safety mode

Figure 4-15 is an overall block diagram of the command loop.

4.5.4.1.1 Ground Rules.-

This system was selected based upon the following ground rules.

1. An automatic program should be used to command the normal test sequence for reasons stated below.

4.5.4.1.1 (con't)

2. The commands do not depend upon a communication link for transmission but are hard-wired circuitry.

3. The short time between commands might present a transmission problem for ground command system.

4. Onboard programmers have a large growth capability.

5. This type of programmer is qualified for space environment and is an "off the shelf item".

Range dispersion patterns indicate that some trajectory corrections on LEM flight path may be necessary. If this is a requirement, the capability can be implemented without any major change to the system. These commands will fall in the range safety catagory along with command destruct.

4.5.4.1.3 Command Priority.-

In the automatic mode, the onboard programmer will control the normal test sequence. If a ground safety problem or a LEM malfunction should arise, ground command will be established immediately until the malfunction is overrriden. The ground command system will have the following capability:

1. Command LEM vehicle destruct

2. Command compensation for some LEM system malfunctions
The range mode has top priority.

4.5.4.2 Description.-

4.5.4.2.1 General.-

The command system (Figure 4-15) will have three modes of operation. In the automatic mode, the onboard programmer will control the normal test sequence. A ground command mode will permit restricted corrective action should malfunctions occur, and a range safety mode will permit limited control of the flight path and command destruct capabilities.

4.5.4.2.2 Automatic Mode.-

In the automatic mode (primary command mode) of operation the onboard programmer will emit signals to the stored-command trigger relays. When these relays are activated, they will close circuits to send commands to the appropriate subsystem as required by the test sequence. (See Table 4-3) The programmer outputs (relay closures) will be tape recorded and also displayed in real time for monitoring purposes. In this way a malfunction in the programmer can be detected and corrective action taken on a cause and effect basis.

TABLE 4-3 COMMAND SCHEDULE FOR LEM #1 FLIGHT CONTROL

Time	Channel	Command Number	Description of Command	Remarks
62	1	1	Purge Thrusters 1 - 16 in .15 sec intervals	
75	2	2	Pressurize RCS fuel and oxidizer	
81			Pressurize Descent Engine (D.E.) fuel and oxidizer	
82	3	3	Activate Descent propellant squibs	
83	4	4	Separate Shroud	Interlock with booster burnout
85	5	5	Separate booster; Initiate booster retrograde	
86	6	6	Activate S & C (reference signal)	Emergency command
87	7	7	Select max thrust throttle signal	
			Initiate ECS ullage firing if necessary	
88	8	8	Ignite LEM ullage rockets (if required)	
89	9	9	Fire descent engine	Emergency command
456	10	10	Arm ascent engine (fire squibs)	Emergency Programmer Advance
458	11	11	Shutdown descent engine	
459	12	12	Select ascent engine on engine mode select	
			Initiate ullage firing of RCS	
460	13	13	Start ascent engine	Emergency command
461	14	14	Separate descent stage	
561	15	15	Select redundance (A.E.)	
621	16	16	Close RCS "A" & "B" fuel and oxidizer s/o valves	
622	17	17	Select "A" "A" manifold for ascent feed	
623	18	18	Actuate RCS-ascent interconnect valves	
624	19	19	Actuate manifold crossfeed	
635	$\overline{18}$	20	Close RCS-ascent interconnect valves	
636	$\overline{16}$	21	Open RCS "A" & "B" fuel and oxidizer s/o valves	
838	20	22	Shutdown ascent engine	
840	13	23	Start ascent engine	
843	20	24	Shutdown ascent engine	
845	13	25	Start ascent engine	
848	20	26	Shutdown ascent engine	
865	13	27	Start ascent engine	
866	21	28	Close all regulator s/o valves	Emergency Program Advance
867	22	29	Fire TCA's 13 & 14 (step input 4 sec)	
871	23	30	Actuate redundant regulator s/o valves	
872	24	31	Close isolation valves affecting TCA's 1 & 3	
873	25	32	Fire TCA's 8 & 16 to roll	
875	$\overline{25}$	33	Shutdown TCA's 8 & 16	
876	26	34	Fire TCA's 4 & 12 to stop roll	

TABLE 4-3 COMMAND SCHEDULE FOR LEM #1 FLIGHT CONTROL (con't)

Time	Channel	Command Number	Description of Command	Remarks
878	$\overline{26}$	35	Shutdown TCA's 4 & 12	
879	27	36	Open isolation valves for TCA's 1 & 3 Close isolation valves for TCA's 4 & 12	
880	28	37	Fire TCA's 3 & 4 to roll	
882	$\overline{28}$	38	Shutdown TCA's 3 & 4	
883	29	39	Fire TCA's 7 & 15 to stop roll	
885	$\overline{29}$	40	Shutdown TCA's 7 & 15	
886	30	41	Open isolation valves for TCA's 2 & 4 Close isolation valves for TCA's 5 & 8	
887	31	42	Fire TCA's 7 & 11 to translate in +Z direction	
889	$\overline{31}$	43	Shutdown TCA's 7 & 11	
893	32	44	Fire TCA's 3 & 15 to stop translation	
894	33	45	Open isolation valves for TCA's 5 & 8 Close isolation valves for TCA's 6 & 17	
895	$\overline{32}$	46	Shutdown TCA's 3 & 15	
896	34	47	Fire TCA's 4 & 8 to translate in -Y direction	
898	$\overline{34}$	48	Shutdown TCA's 4 & 8	
900	35	49	Fire TCA's 12 & 16 to stop translation	
902	$\overline{35}$	50	Shutdown TCA's 12 & 16	
903	36	51	Open isolation valves to TCA's 6 & 7 Close isolation valves to TCA's 9 & 12	
904	37	52	Fire TCA's 6 & 14 to thrust in +X direction	
906	$\overline{37}$	53	Shutdown TCA's 6 & 14	
907	38	54	Fire TCA's 5 & 13 to thrust -X direction	
909	$\overline{38}$	55	Shutdown TCA's 5 & 13	
910	39	56	Open isolation valves to TCA's 9 & 12 Close isolation valves to TCA's 14 & 16	
911	40	57	Fire TCA's 1 & 9 to thrust in -X direction	
913	$\overline{40}$	58	Shutdown TCA's 1 & 9	
914	41	59	Fire TCA's 2 & 10 to thrust in + X direction	
916	$\overline{41}$	60	Shutdown TCA's 2 & 10	
917	42	61	Open isolaiton valves to TCA's 14 & 16 Close isolation valves to TCA's 14 & 16	
918	43	62	Fire TCA's 9 & 14 for positive pitch	
919	$\overline{43}$	63	Shutdown TCA's 9 & 14	
920	44	64	Fire TCA's 1 & 16 to stop pitch	

TABLE 4-3 COMMAND SCHEDULE FOR LEM #1 FLIGHT CONTROL (con't)

Time	Channel	Command Number	Description of Command	Remarks
921	44	65	Shutdown TCA's 1 & 16	
922	45	66	Open isolation valves to TCA's 13 & 15	
923	46		Close isolation valves to TCA's 10 & 11	
923	46	67	Fire TCA's 1 & 14 for positive yaw	
924	46	68	Shutdown TCA's 1 & 14	
925	47	69	Fire TCA's 2 & 13 to stop yaw	
926	47	70	Shutdown TCA's 2 & 13	
927	48	71	Shut off "A" fuel and oxidizer main feed s/o valves	
928	49	72	Fire manifold cross feed squibs	Interlock
929	50	73	Fire TCA's 2, 6, 10, & 14 to thrust in +X direction	
930	50	74	Shutdown TCA's 2, 6, 10, & 14	
931	51	75	Open "A" fuel and oxidizer main feed s/o valves Close "B" fuel and oxidizer main feed s/o valves	
932	50	76	Fire TCA's 2, 6, 10, & 14 to thrust in +X direction	
933	50	77	Shutdown TCA's 2, 6, 10, & 14	
934	52	78	Open system "B" fuel and oxidizer main feed s/o valves	
935	53	79	Use TCA's program oxidizer lead	
940	54	80	Use TCA's program fuel lead	

4.5.4.2.3 Ground Command Mode (Stored Ground Commands).-

In event of subsystem malfunction, ground commands stored in the ground station will be sent to the vehicle via the communication uplink. The ground commands, which are not in the range safety category, will be activated manually by a specialist who will be monitoring the subsystems display.

If the specialist detects a malfuncition which is correctable, he will call for a particular ground-stored command. When activated, the command will pass through a coupler that will establish its priority, and then transmitted to the vehicle. The command coupler aboard the vehicle will interlock commands such that those from the ground will override any onboard commands.

The ground commands fall into three groups:

1. Those that will advance the automatic programmer to the next phase of testing.

2. Those that will switch to a redundancy within the subsystem.

3. Those that will alter the flight program.

The first group, as the name implies, will be those that advance the programmer to the next stage of testing. The second group will be those sent up as a result of subsystem malfunctions that are correctable by switching to redundant components or sources. The third group are commands which permit a restricted test should malfunctions prevent comprehensive testing.

4.5.4.2.4 Range Safety Mode.-

The range safety commands under the direction of the range safety officer, not only provide for a safe destruction of the vehicle, but can also provide for limited ground control of LEM trajectory. The destruct command is assigned top command priority and will utilize three or four of the command uplink tones. The method of implementing the trajectory commands would consist of a three button panel marked prot-off-starboard. Pushing a button would laterally accelerate the vehicle in the desired direction. The priority of the course correction command is a subject for further study.

4.5.4.3 Command System Component Description.-

4.5.4.3.1 Onboard Programmer.-

The onboard programmer will consist of clock driven switching circuitry that will close relays in the command coupler. The clock will be interlocked with an onboard accelerometer circuit or booster instrumentation so that the programmed testing can be started immediately command capability. The programmer will contain approximately 100 commands driving 65 relay circuits (see Table 4-3).

4.5.4.3.1 (con't)

The exact number will be established when the test plan and systems preliminary design are firmed up. These commands will also be transmitted to the ground for real-time display and records to be used for post-flight analysis. Since this device is pre-programmed, no deviations from the sequence can be made in the event of an emergency except to advance the program.

The programmer consists of two basic parts, a time accumulator and a decoder. The time accumulator uses a crystal as its clock source. Its primary frequency is counted down to one pulse per second by magnetic counting circuits. The one pulse per second is then fed to a simpler binary flip-flop counter. The decoder samples the states of the flip-flop counter and provides an output signal at the appropriate time. The sampling method is accomplished by means of logic "AND" gates. The time outputs are then routed to the proper relay or solid state switch.

4.5.4.3.2 Communication Link.-

The communication uplink that will be provided at the test site will be a single frequency FM/FM system. In order to be highly reliable, the commands should be combinations of three tones each. The expression for the number of commands available is:

$$\frac{n!}{(n-K)!\ K!}$$

Where n is the number of channels (tones) and K is three in this case. If ten channels are available for emergency commands,

$$\frac{n!}{(n-K)!\ K!} = \frac{10!}{(10-3)!\ 3!} = 120$$

assuming only half of these are reliable, sixty (60) will be available to be used for the ground commands. When received, the tones will be decoded and the decoded signal sent to the command coupler.

4.5.4.3.3 Ground Command Unit.-

The ground command unit will contain all the stored emergency commands. The circuits will be interlocked with top priority being assigned to the destruct command. Flight path correction commands will be superimposed on the normal and ground commands. When conflicts occur the flight path correction will prevail over onboard commands.

4.5.4.3.4 Real-time Visual Displays.-

The real-time visual displays, located in the control room, must display at least those parameters required to make decisions for manual selection of emergency commands. The required display parameters are indicated on the measurements list, Table 7-1.

Each subsystem will have panels to be monitored by specialists. Some displays need only be condition lights whiel others need to be continuous displays. The inputs to the displays will be decommutated and conditioned outputs from the ground receiver.

4.5.4.4 Study Areas.-

During this study some questions came up which will be the subject of future investigation. Some of the more important areas are:

1. Effects of exhaust plume on communications
2. Priority and interlock in control sequence
3. Optimization of flight path control
4. Methods of failure detection

If future studies indicate that less ground command will not impare mission success, a re-evaluation of these will be made.

4.5.5 Environmental Control Subsystem.-

4.5.5.1 Design Criteria.-

The standard LEM environmental control system will be modified for the following LEM-1 criteria:

1. LEM-1 is unmanned.
2. The total flight duration does not exceed 20 minutes.
3. The vehicle will be operated in a space environment.
4. The stabilization and control, instrumentation, and command subsystem electronics require cooling during prelaunch operations and flight (the latter with the exception of the first 150,000 feet of the ascent phase).

4.5.5.2 Description.-

The ECS for LEM-1 will consist of modified heat transport and water management sections to supply thermal control for the stabilization and control, instrumentation, and command subsystems.

The modified heat transport section will consist of a coolant recirculation assembly, GSE quick disconnects, a coolant water evaporator, a coolant accumulator and cold plates or other provisions as required to maintain thermal control of the electronics.

The modified water management section will consist of off-loaded or modified ascent stage water tanks, the coolant water evaporator water feed valves and provisions for pressurization of the water tanks.

4.5.5.2 (con't)

Prelaunch electronic cooling will be supplied by means of a GSE water-glycol service unit.

Should electronic cooling be necessary during the first 150,000 feet of the ascent, provisions for storage of freon and metering of its flow to the coolant water evaporator will be required.

In the event that in-flight cooling of the electronics is not required, the water management section will be completely eliminated and the heat transport section further modified by removal of the coolant recirculation assembly, coolant water evaporator and coolant accumulator.

4.5.6 Electrical Power Subsystem.-

4.5.6.1 Electrical Power Load Requirements.-

The LEM-1 load profile, based on latest available information, on equipment and trajectory, is presented in Figure 4-16.

4.5.6.2 Electric Distribution Bus Arrangement.-

The LEM-1 electric loads will be divided into five (5) categories with each category being supplied electrically from a bus similarly identified as follows (Figure 4-17 and Figure 4-18):

1. Command Destruct Loads - Command Destruct Bus
2. Pyrotechnic Loads - Pyrotechnic Bus
3. Reaction Control Loads - RCS Bus
4. All AC Loads - AC Bus
5. All other DC loads - Main DC bus

Each bus will have a redundant power supply.

4.5.6.3 Electric Power Supplies.-

The internal electric power supplies to each bus will be as follows:

	Bus	Primary Supply	Alternate Supply
1.	Main DC Bus	Main battery #1	Main battery #2
2.	RCS Bus	RCS battery	Main battery #1 or #2
3.	Pyro Bus	Pyro battery	"
4.	C.D. Bus	C.D. battery	"

The AC bus will be powered from the main DC bus through one of the two DC to AC inverters. The external electric power supply will be DC only, through an external power connector and umbilical, and will power the main DC bus. All other busses will be supplied from the main DC bus.

4.5.6.3 (con't)

The DC bus will be maintained at a nominal operating voltage of 28 ± 2 volts, steady state.

The AC bus will be maintained at a nominal operating voltage of 115/200 wye ± 2% volts at a frequency of 400 ± 1% cycles per second.

4.5.6.4 Control Scheme.-

Under normal operating conditions, each bus is supplied through an energized contactor from its primary power supply. On failure of any primary supply, its contactor will become de-energized and will lock out. The bus will be automatically supplied from its alternate supply through the de-energized contactor.

On command, each primary supply contactor may be reset.

4.5.7 Reaction Control Subsystem.-

4.5.7.1 General.-

The Reaction Control Subsystem (Figures 4-18 and 4-20) consists of three sections, (1) the Helium Pressurization System, (2) the Propellant System, and (3) the Thrust Chamber Assemblies.

Within the RCS there are two indipendent systems each containing its own Helium, Propellant, and Thrust Chamber assembly sections. Each system contains equal quantities of proprllant and helium.

4.5.7.2 Helium Pressurization System.-

Gaseous helium, at pressures between 3000 and 3500 psi and a nominal temperature of 70°F, is stored in a spherical titanium tank. The helium tank is filled through a port accessible from the exterior of the LEM vehicle. A pressure transducer, located on the tank inlet inlet-outlet port, coupled with tank temperature will suffice for helium quantity measurements. Redundant, normally closed squib valves isolate the helium tank from the rest of the system until just prior to LEM separation from the Little Joe II booster. A ground test point is located downstream of the squib valves for helium system checkout. A single filter is located downstream of this point containing possible squib debris and helium contamination. Downstream of the filter, the helium section splits into redundant legs. One leg begins with a normally open, solenoid operated, latch-type, shut-off valve while the other leg begins with a normally-closed shut-off valve. Only one leg of the helium system will be used at any time, but should this primary leg malfunction, it can be shut off and the secondary switched on. A two stage, line pressure sensing regulator follows the shutoff valves. The first stage reduces the helium pressure to 250 ± 4 psi and the second stage reduces the pressure to 190 ± 2 psi. A ground test point is located between the stages of the regulator for first stage checkout. Downstream of the regulator, the redundant legs are connected so that

4.5.7.2 (con't)
should one leg fail, the other will pressurize both propellant tanks. A pressure transducer is located on the connecting line to read regulator outlet pressure. In addition, a ground test connection is located on the same line to allow drawing off checkout gas. Quad check valves are placed on the propellant tank pressurization inlets to insure isolation of fuel and oxidizer.

A relief valve, consisting of a burst disc, filter, and relief mechanism, is situated close to the helium port on each propellant tank. The burst disc insures a sealed helium section during normal operation and the filter prevents relief mechanism fouling due to burst disc debris. Vent lines are brought from the helium ports of the propellant tanks to the propellant fill areas.

4.5.7.3 Propellant System.-

Each propellant system consists of two cylindrical titanium tanks with hemispherical ends. The oxidizer tank contains nitrogen tetroxide, N_2O_4. The fuel tank contains a 50-50 mixture of hydrzine (UDMH). (O/F ratio = 2.0) The propellants are contained within a 3-ply teflon bladder supported by a stand pipe running lengthwise in the tank. The helium pressurant flows between the bladder and tank wall, acting upon the propellant filled bladder for positive expulsion. The propellant is introduced into the tanks through the feed port from a fill point accessible from the exterior of the LEM vehicle. Tentatively, the propellant will be loaded under a pressure of 20 to 30 psig with a back pressure on the vent port of 5 psig lower than the fill pressure. A pressure transducer is located at the feed port. Nominal tank pressure will be 181 psia and nominal temperature will be $70^{\circ} \pm 20^{\circ}F$.

Propellant quantity gaging systems of the radiation type are presently under study for inclusion in the tanks. A burst disc, situated downstream of the tank feed port, holds the propellant in the tank until supply pressure is applied to the tank and reaches approximately 150 psig, the burst disc setting. At this point, a ground test point is located to provide for line, valve, and thruster checkout. A normally-open, test point, permits isolation of the tank. From this valve, the propellant flows into a manifold feeding eight thrust-chamber assemblies (TCA's).

The manifolds from each independent system can be connected by firing normally-closed, squib valves. Ascent engine propellant may be introduced into either one or both of the RCS propellant manifolds by actuating normally closed, solenoid operated, latch-type, shutoff valves.

Each independent RCS subsystem feeds eight thrust-chamber assemblies, two TCA's in each cluster, insuring control in all axes. The lines feeding these two TCA's may be closed by normally open solenoid

4.5.7.3 (con't)
valves, allowing isolation of malfunctioning TCA's or catastrophically broken clusters.

4.5.7.4 Thrust Chamber Assemblies.-

The thrust chamber assembly section consists of 16 TCA's grouped inclusters of four (quads). Each TCA consists of trim provisions (orifice), filters, two (2) propellant solenoid valves, injector, combustion chamber, and radiation nozzle.

The TCA's may be operated either steady-state, periods above one second, or pulse modulated, periods less than one second. Full thrust vacuum rating is 100 lbs. at a combustion chamber pressure of 90 psig.

4.5.8 Command Destruct System.-

4.5.8.1 General.-

To satisfy the range safety requirements of WSMR, it will be necessary to have the capability of destroying both stages of LEM-1 at any time from launch to termination. The purpose of the destructor is to disperse the propellant fuels.

4.5.8.2 System Description (Figure 4-21).-

The safe arm mechanism of the command destruct system utilizes two detonators and the ends of a loop of linear-shaped charge cable. In the "safe" position, the detonators are grounded and out-of-line with the terminals of the saped charge cable.

At the "arm" command, the detonators become ungrounded and move into axial alignment with the ends of the linear charge. The "fire" command initiates both detonators simultaneously into the ends of the explosive cable which encompasses the fuel and oxidizer tanks. Firing of on y one detonator in this destruct system provides a sufficient shock wave to the shaped charge to rupture the tankage.

As shown in Figure 4-21, both the ascent and descent stages will have complete system duplication.

4.5.8.3 Power Supply.-

Each command destruct system can be powered automatically from either of two battery power supplies.

4.5.8.4 Command Program.-

The commands are received from the ground station via antennae by the receivers to first arm the safe-arm device and then to close the fire switch.

4.5.9 Communications.-

4.5.9.1 Command System.-

Presently, the Motorola command receiver/voltage regulator combination, Mod. No. MCR 102A/MAD-101, is being considered for use in LEM-1. A separate command destruct receiver may be required to satisfy range safety requirements. The Command Subsystem willrequire an array of antennae mounted 90° apart on the exterior of the LEM. One command receiver will utilize the remaining pair. This arrangement provides the optimum configuration for satisfying the command subsystem and range safety requirements.

4.5.9.2 Tracking Transponder.-

Tracking of LEM-1 will be performed by the FPS-16 radar system. The transponder recommended for use on the LEM-1 is the DPN-66. Redundant components of the transponder will be required and will be GFE. The equipment required for prelaunch checkout of both the communications subsystem will be supplied by the ranges. The transponder antennae (CFE) will be mounted and connected in the same manner as the command system.

4.6 Little Joe II.-

4.6.1 Airframe.-

The airframe for Little Joe II is built by General Dynamics/Convair, San Diego, California. The vehicle airframe consists of cylindrical upper (forebody) and lower (afterbody) shells and four fins. Overall length of the Little Joe is 399 inches. The booster body (combined fore and after bodies) measures 350 inches in length and 154 inches in diameter. The four fins, each 50 square feet in area, are attached in a cruciform pattern to the afterbody. Each fin is equipped with a movable control surface at the trailing edge and a hydrogen-peroxide reaction control module.

Both body shells are of semi-monocoque construction and are fabricated from truncated-form corrugated sheets stabilized by ring frames. The main structural member of the booster is a large built-up bulkhead contains the rocket motor attachments and distributes the thrust load uniformly to the booster skin.

4.6.2 Electrical Power Subsystem.-

The Little Joe II electrical power system for the attitude controlled vehicle will be described at a later date.

4.6.3 Propulsion System.-

The Little Joe II propulsion system to be used for the LEM-1 test will consist of seven (7) Algol ID, Mod II solid propellant rocket motors. These motors are bolted to retaining rings in the thrust bulkhead of the booster afterbody.

4.6.3.1 Algol Rocket Motors.-

The algol ID Mod II rocket motor with a fixed, adjustable nozzle is manufactured by Aerojet-General Corporation, Sacramento, Calf. The motor is a solid propellant rocket rated at 103,200 pounds average thrust at sea level, firing at 70°F nominal propellant grain temperature.

4.6.3.2 Motor Arrangement and Firing Sequence.-

The algol motors will be ignited in two stages. Four motors will be ignited simultaneously at t+0 and the remaining three motors will be ignited simultaneously at t+39 providing a five second overlap. The nozzles for each set of motors will be adjusted so that the resultant thrust vector for each motor will pass through the vehicle center of gravity at the instant of ignition.

4.6.3.3 Command Destruct System.-

Booster thrust may be terminated by two command destruct systems upon radio command signal from an AN/FRW-2 UHF ground transmitter. Each of the two CD systems is powered by its own battery.

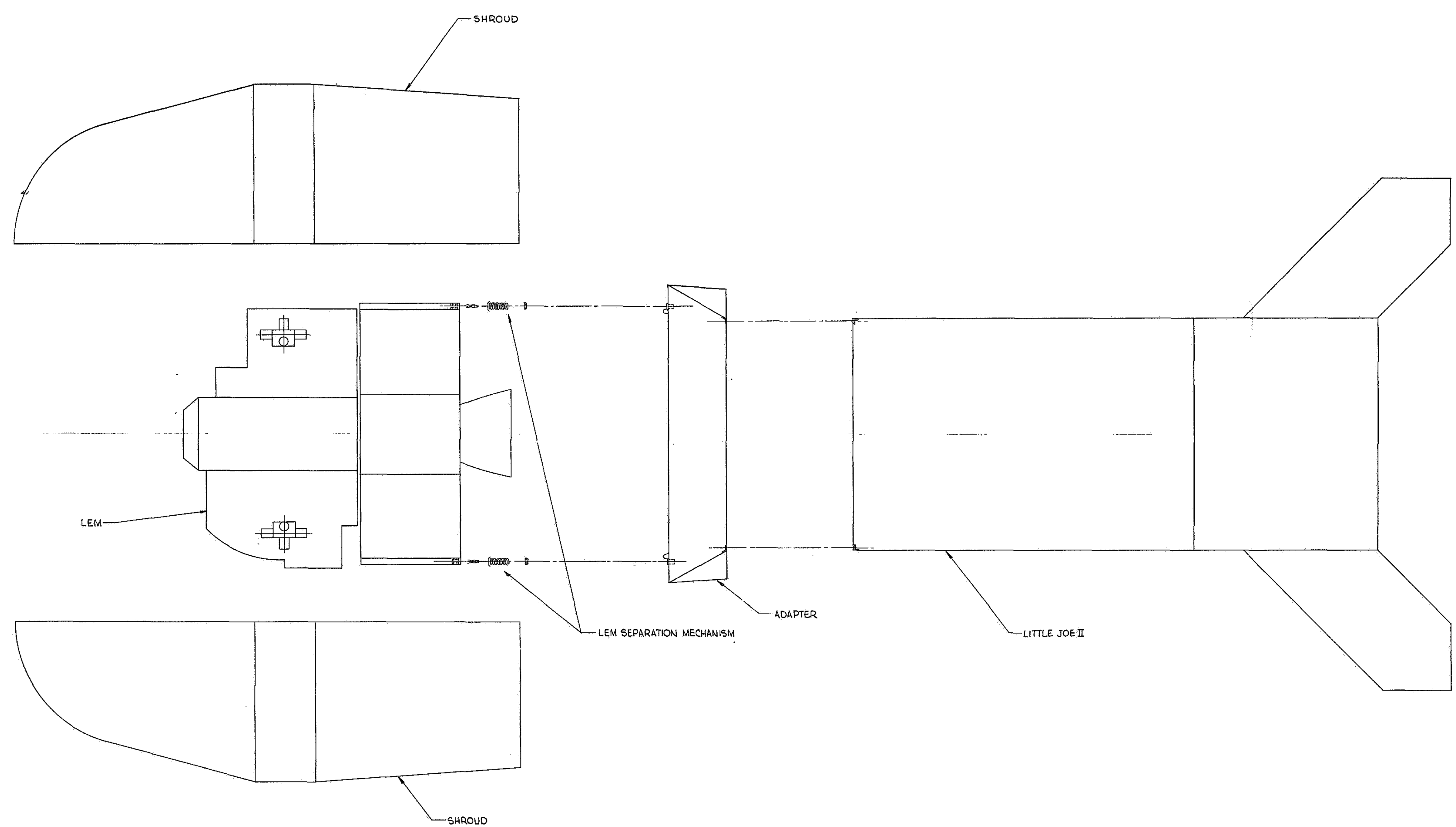

SHROUD
LEM
LEM SEPARATION MECHANISM
ADAPTER
LITTLE JOE II
SHROUD

FIG. B 4-1

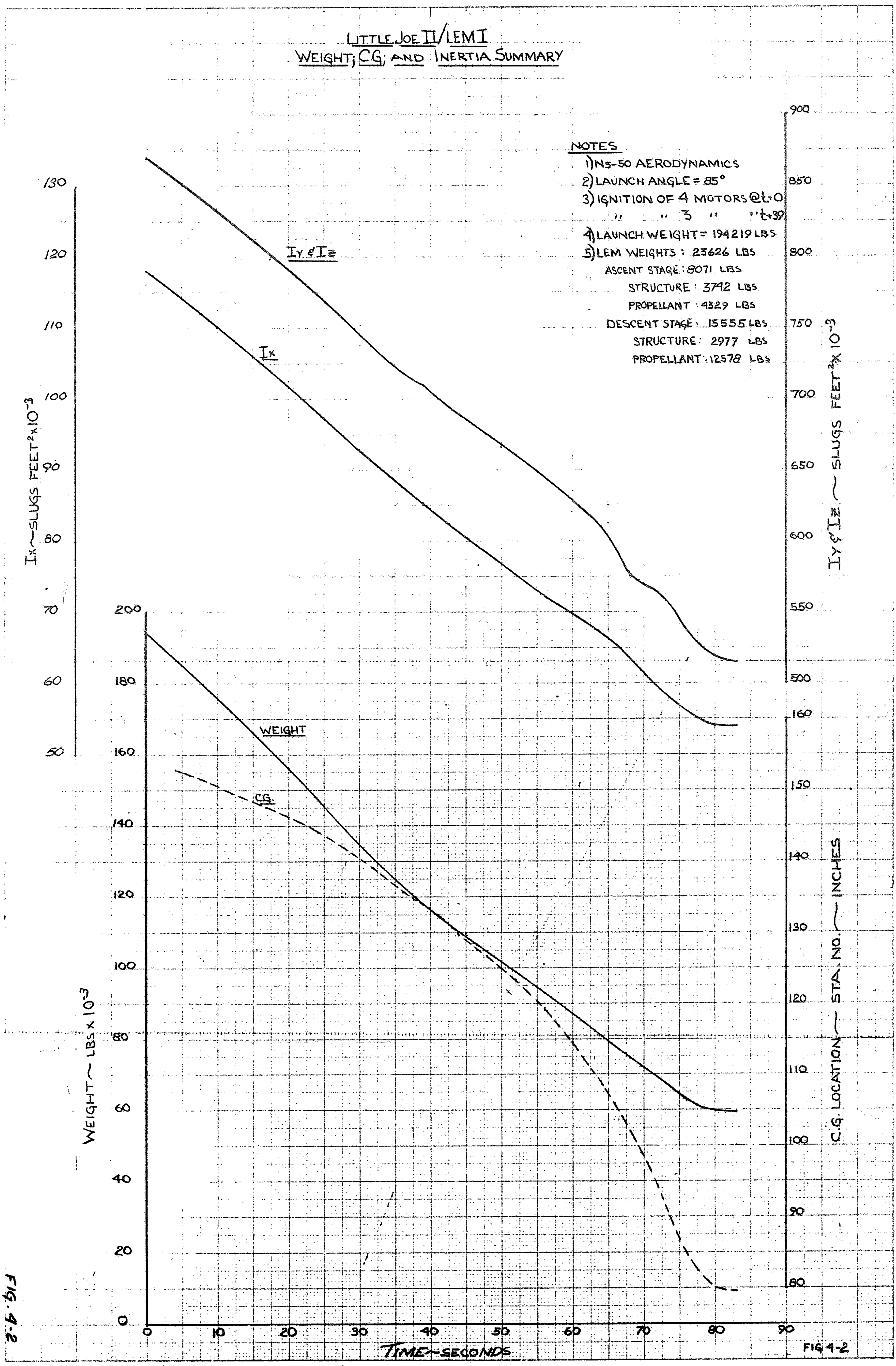

LITTLE JOE II/LEM I
WEIGHT; C.G.; AND INERTIA SUMMARY
NOTES
1) NS-50 AERODYNAMICS
2) LAUNCH ANGLE = 85°
3) IGNITION OF 4 MOTORS @ t=0
" " 3 " " t+39
4) LAUNCH WEIGHT = 194219 LBS
5) LEM WEIGHTS : 23626 LBS
ASCENT STAGE : 8071 LBS
STRUCTURE : 3742 LBS
PROPELLANT : 4329 LBS
DESCENT STAGE : 15555 LBS
STRUCTURE : 2977 LBS
PROPELLANT : 12578 LBS
Iy & Iz
Ix
WEIGHT
C.G.
Ix ~ SLUGS FEET² x 10⁻³
Iy & Iz ~ SLUGS FEET² x 10⁻³
WEIGHT ~ LBS x 10⁻³
C.G. LOCATION ~ STA. NO. ~ INCHES
TIME ~ SECONDS

FIG. 4-2

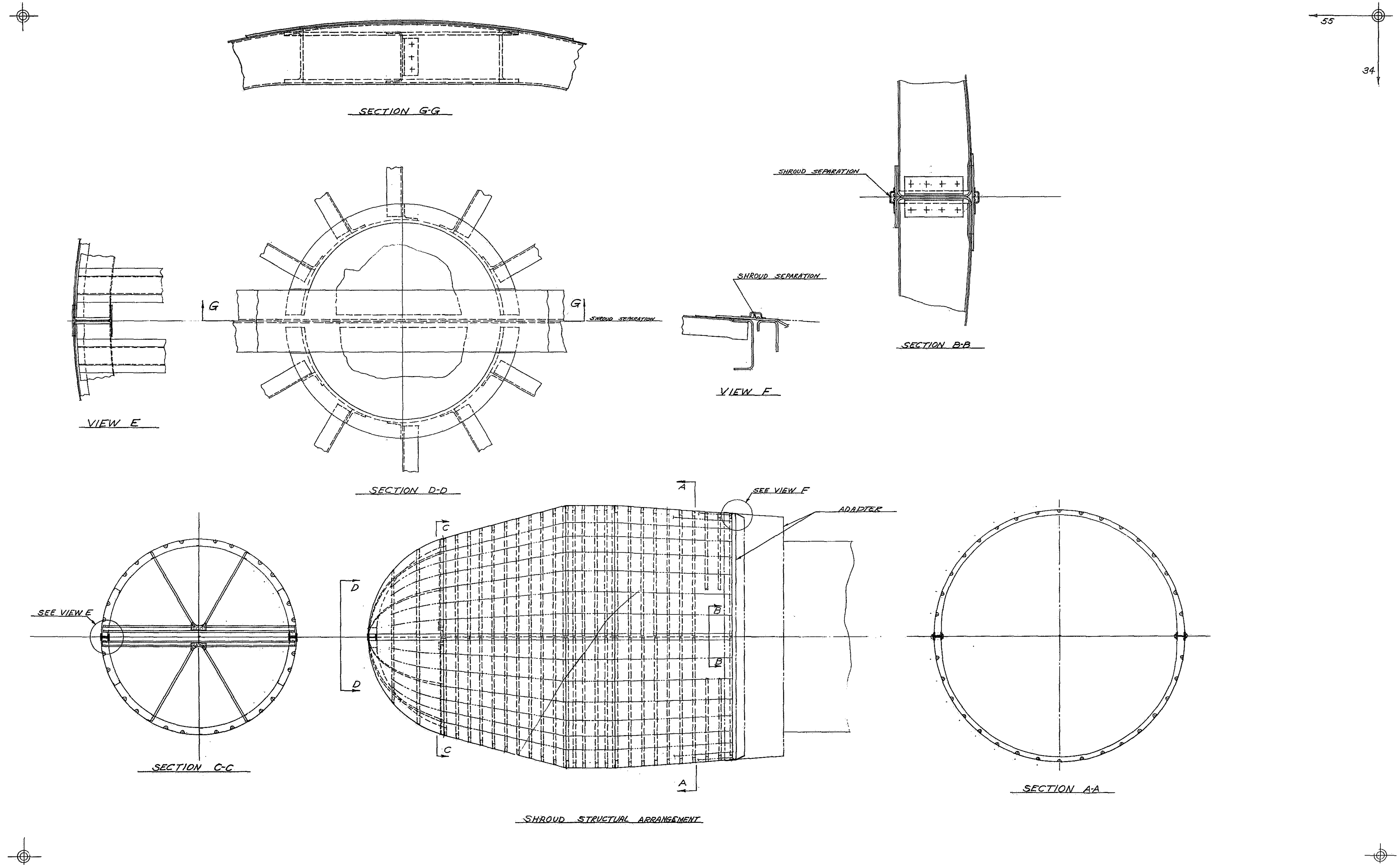

55
34
SECTION G-G
SHROUD SEPARATION
G
G
SHROUD SEPARATION
VIEW E
SECTION D-D
SHROUD SEPARATION
VIEW F
SECTION B-B
SEE VIEW F
ADAPTER
A
A
C
C
D
D
B
B
SEE VIEW E
SECTION C-C
SECTION A-A
SHROUD STRUCTURAL ARRANGEMENT

FIG. B 4-3

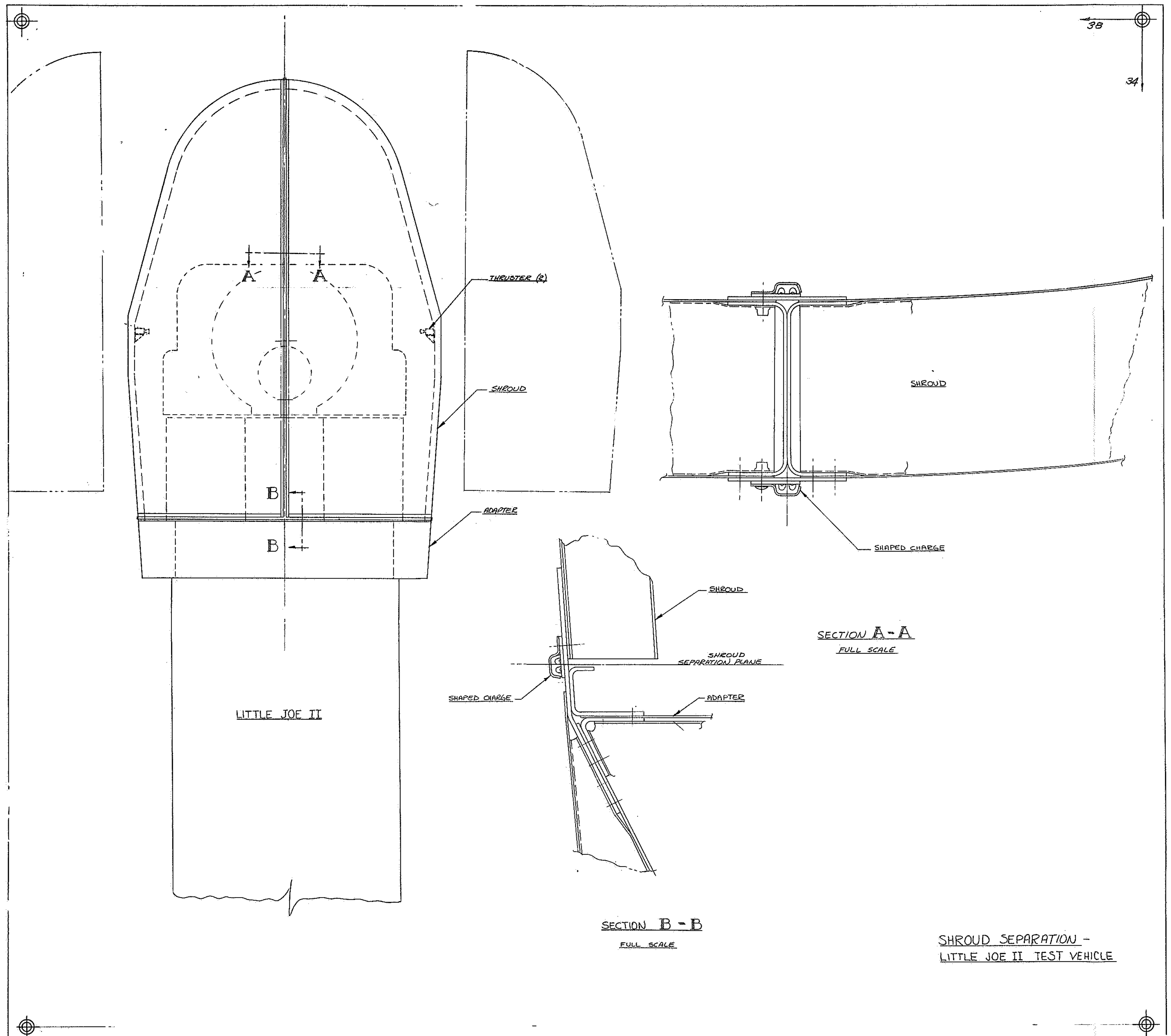
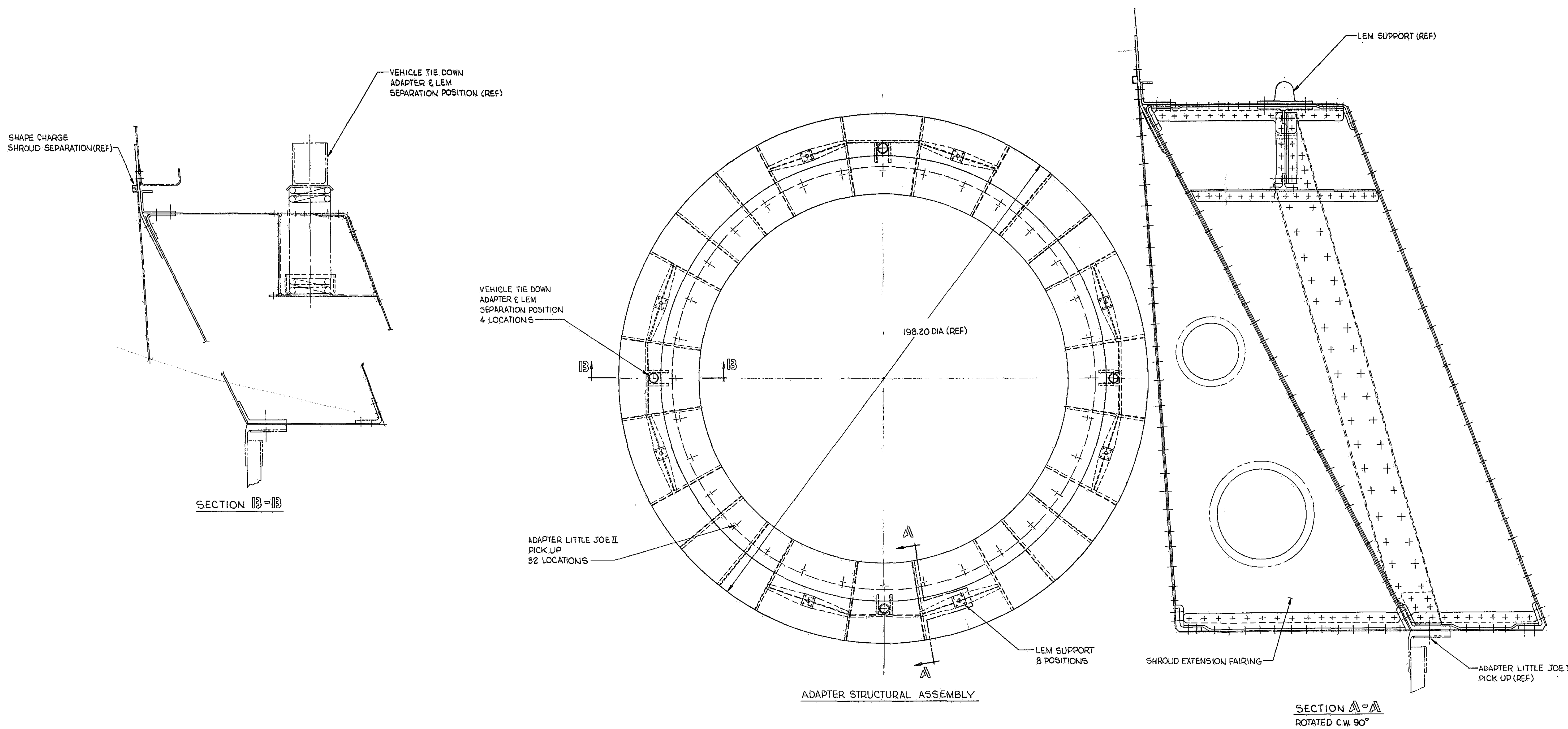

38
34
THRUSTER (2)
SHROUD
ADAPTER
A
A
B
B
LITTLE JOE II
SHROUD
SHAPED CHARGE
SHROUD
SHROUD SEPARATION PLANE
SHAPED CHARGE
ADAPTER
SECTION A-A
FULL SCALE
SECTION B-B
FULL SCALE
SHROUD SEPARATION -
LITTLE JOE II TEST VEHICLE

FIG. B4-4



SHAPE CHARGE
SHROUD SEPARATION (REF)
VEHICLE TIE DOWN
ADAPTER & LEM
SEPARATION POSITION (REF)
SECTION B-B
VEHICLE TIE DOWN
ADAPTER & LEM
SEPARATION POSITION
4 LOCATIONS
198.20 DIA (REF)
ADAPTER LITTLE JOE II
PICK UP
32 LOCATIONS
LEM SUPPORT
8 POSITIONS
ADAPTER STRUCTURAL ASSEMBLY
LEM SUPPORT (REF)
SHROUD EXTENSION FAIRING
ADAPTER LITTLE JOE II
PICK UP (REF)
SECTION A-A
ROTATED C.W. 90°

FIG. B 4-5

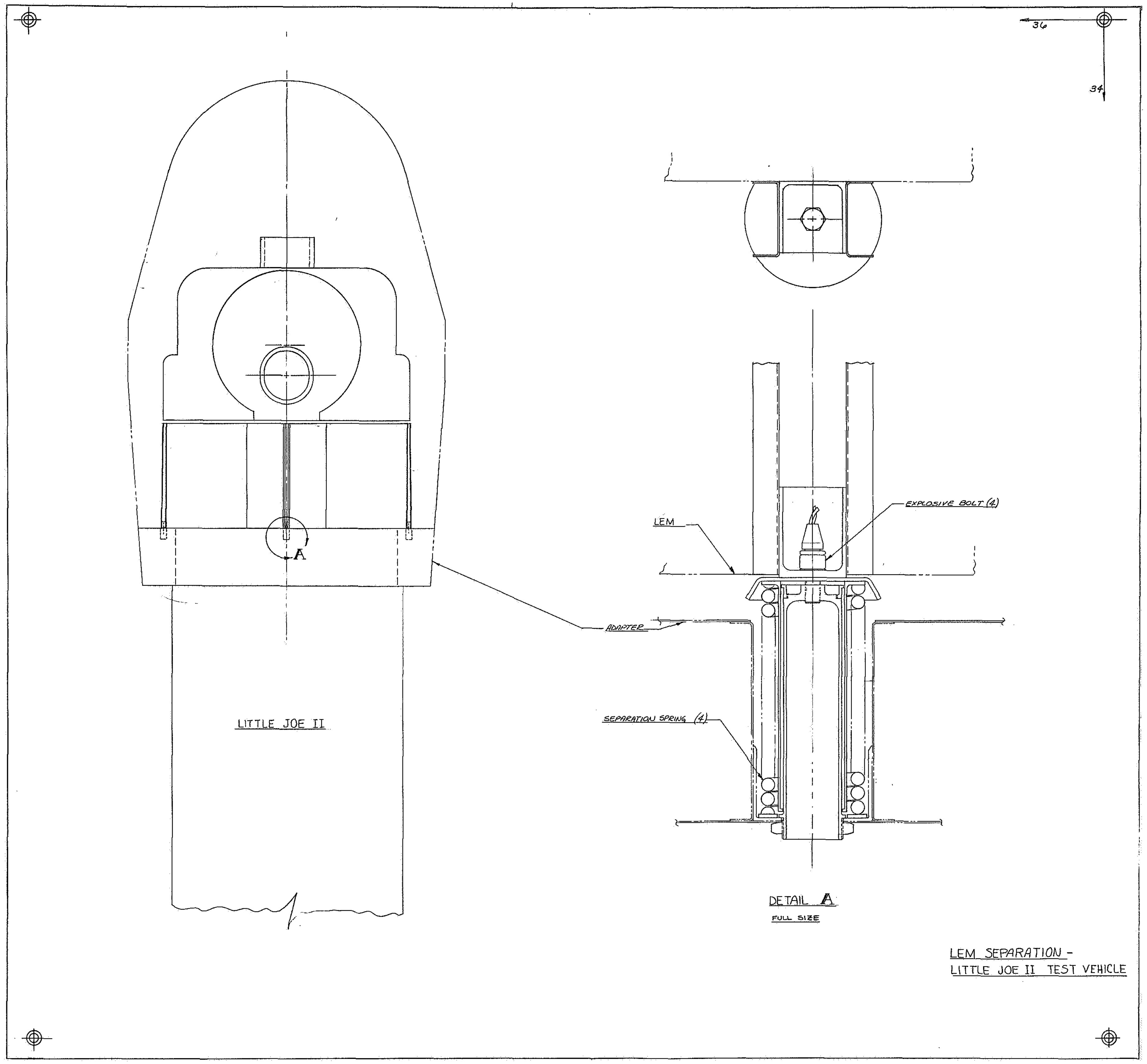

36
34
LITTLE JOE II
A
LEM
EXPLOSIVE BOLT (4)
ADAPTER
SEPARATION SPRING (4)
DETAIL A
FULL SIZE
LEM SEPARATION -
LITTLE JOE II TEST VEHICLE

FIG. B 4-6

LITTLE JOE II / LEM-I
DRAG DURING LEM - BOOSTER SEPARATION
CONFIGURATION 4-1

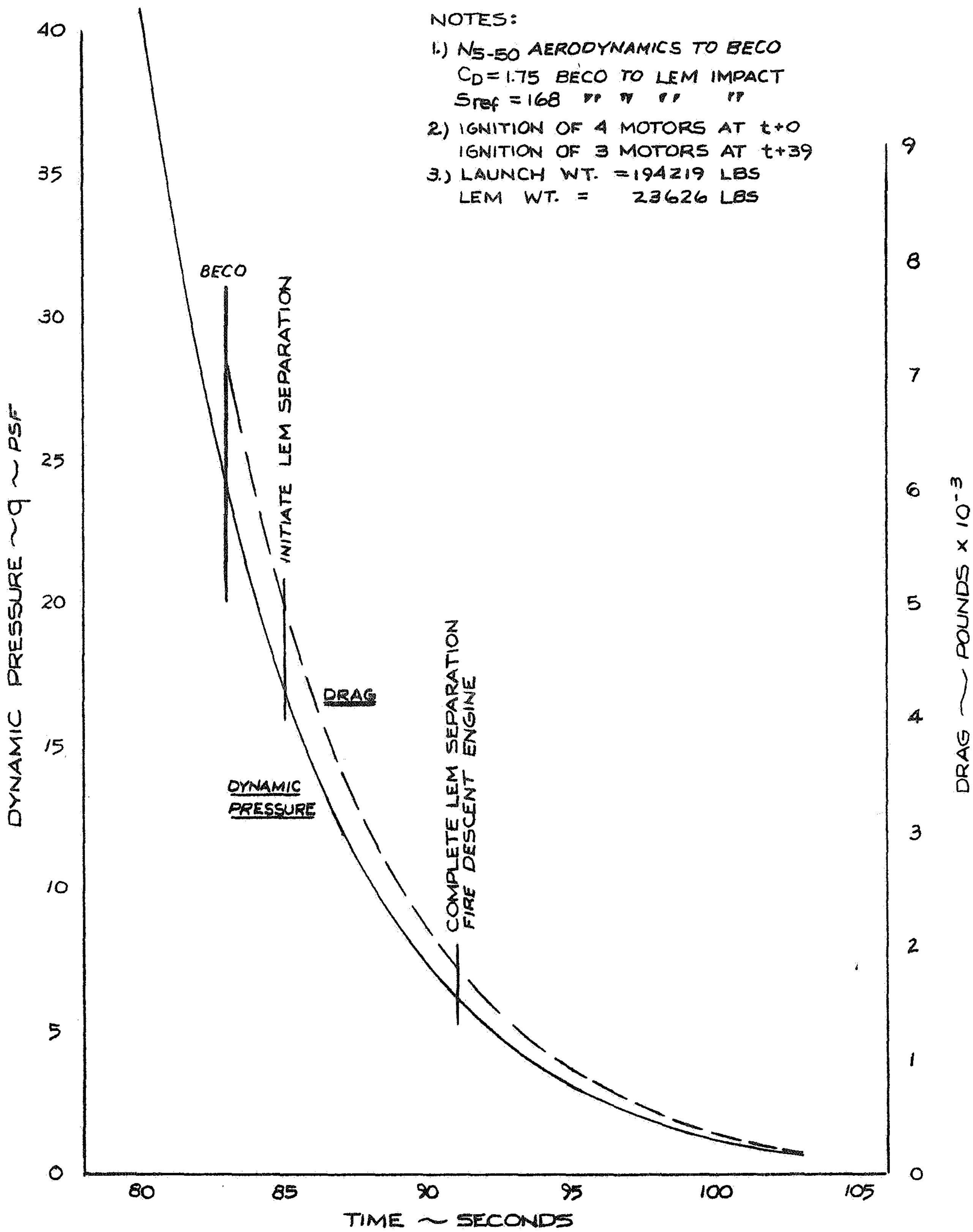


FIG. 4-7

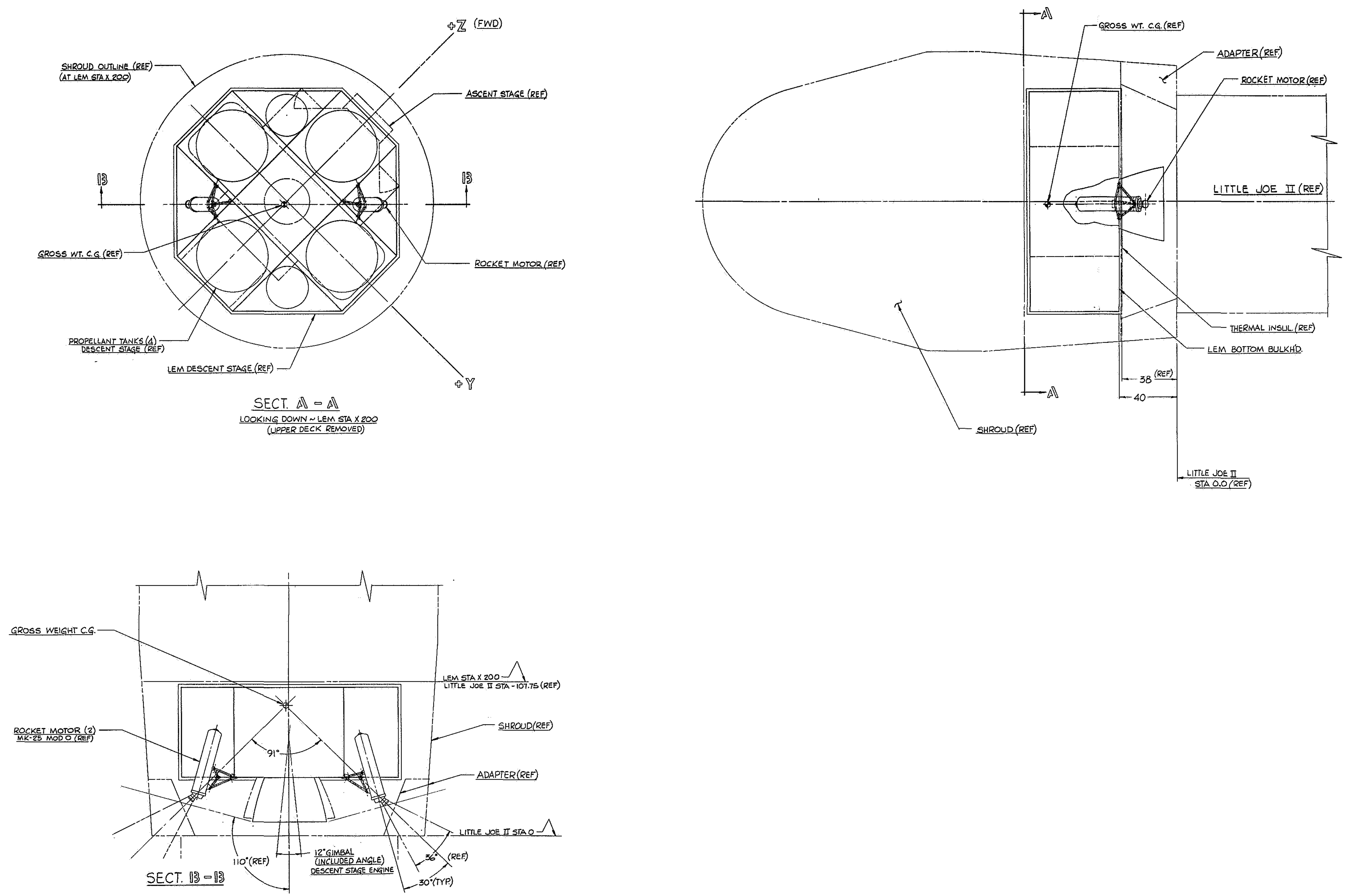

+Z (FWD)
SHROUD OUTLINE (REF) (AT LEM STA X 200)
ASCENT STAGE (REF)
GROSS WT. C.G. (REF)
ROCKET MOTOR (REF)
PROPELLANT TANKS (4) DESCENT STAGE (REF)
LEM DESCENT STAGE (REF)
+Y
SECT. A - A
LOOKING DOWN ~ LEM STA X 200
(UPPER DECK REMOVED)
GROSS WT. C.G. (REF)
ADAPTER (REF)
ROCKET MOTOR (REF)
LITTLE JOE II (REF)
THERMAL INSUL. (REF)
LEM BOTTOM BULKH'D.
38 (REF)
40
SHROUD (REF)
LITTLE JOE II STA 0.0 (REF)
GROSS WEIGHT C.G.
LEM STA X 200
LITTLE JOE II STA - 107.75 (REF)
ROCKET MOTOR (2) MK-25 MOD 0 (REF)
SHROUD (REF)
91°
ADAPTER (REF)
LITTLE JOE II STA 0
110° (REF)
12° GIMBAL (INCLUDED ANGLE) DESCENT STAGE ENGINE
36° (REF)
30° (TYP)
SECT. B - B

FIG. B 4-8

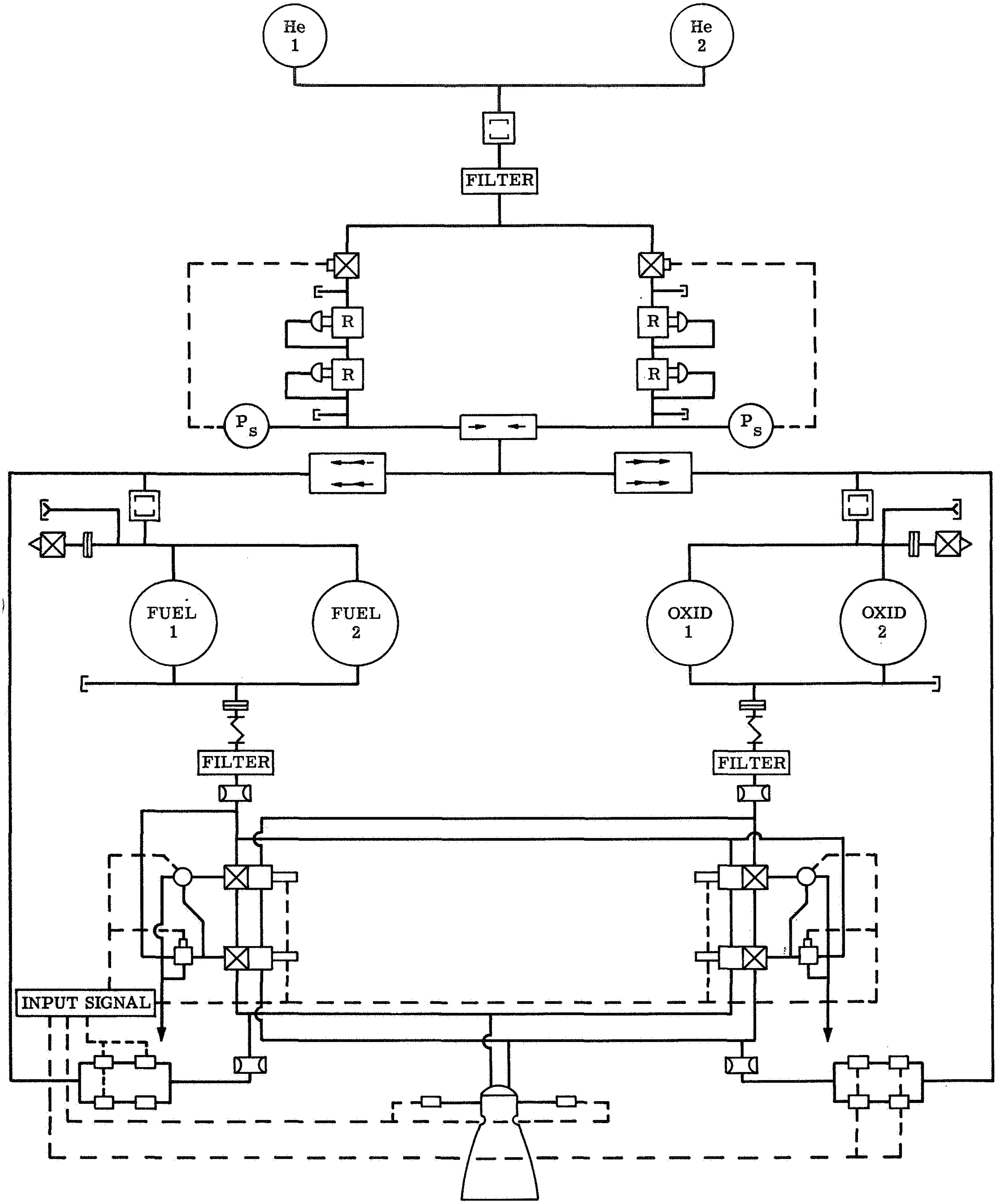

He 1
He 2
FILTER
R
R
R
R
P_s
P_s
FUEL 1
FUEL 2
OXID 1
OXID 2
FILTER
FILTER
INPUT SIGNAL

FIG. B 4-9

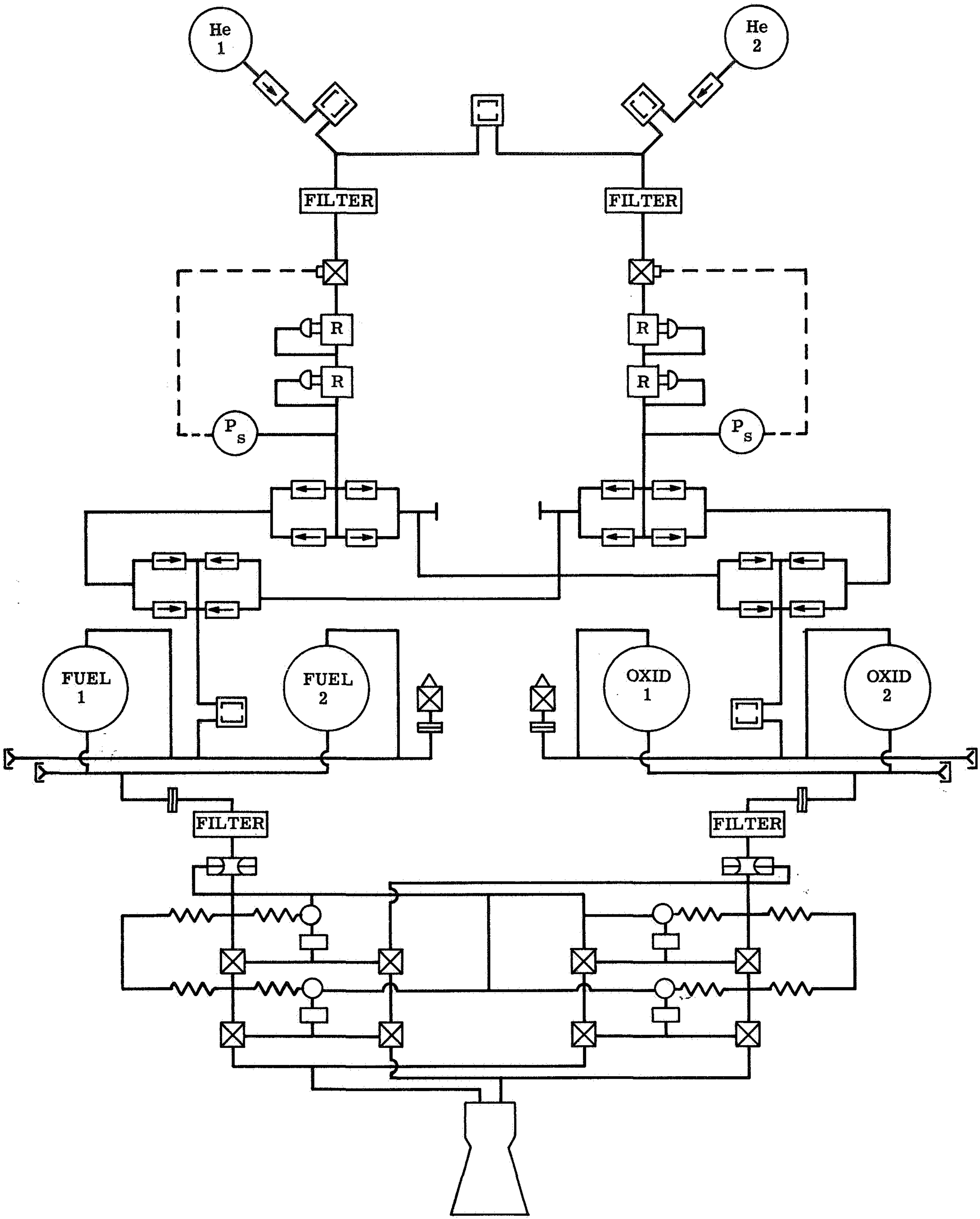

He 1
He 2
FILTER
FILTER
R
R
R
R
Ps
Ps
FUEL 1
FUEL 2
OXID 1
OXID 2
FILTER
FILTER

FIG. B 4-10

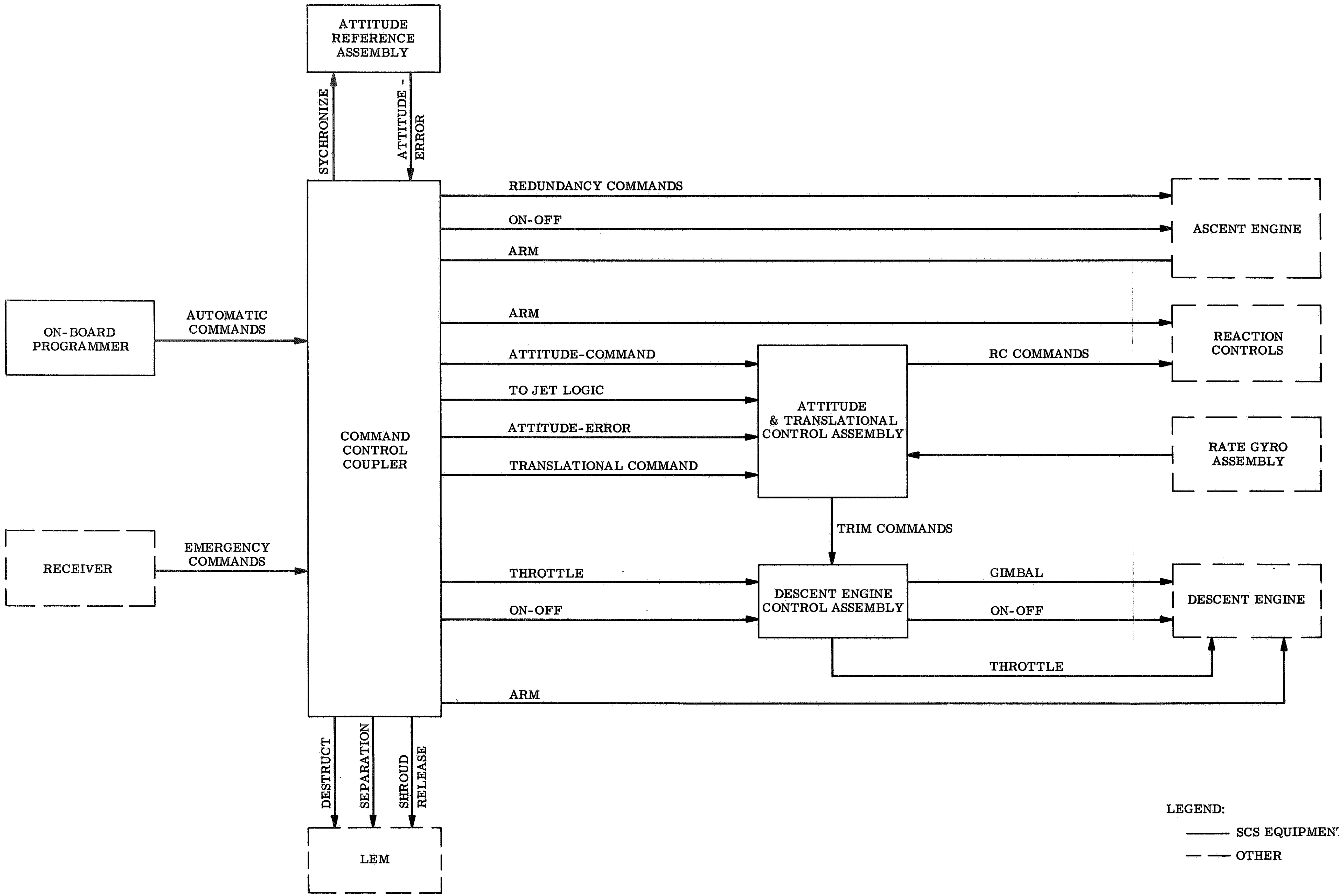

ATTITUDE
REFERENCE
ASSEMBLY
SYCHRONIZE
ATTITUDE - ERROR
ON-BOARD
PROGRAMMER
AUTOMATIC
COMMANDS
RECEIVER
EMERGENCY
COMMANDS
COMMAND
CONTROL
COUPLER
REDUNDANCY COMMANDS
ON-OFF
ARM
ASCENT ENGINE
ARM
REACTION
CONTROLS
ATTITUDE-COMMAND
TO JET LOGIC
ATTITUDE-ERROR
TRANSLATIONAL COMMAND
ATTITUDE
& TRANSLATIONAL
CONTROL ASSEMBLY
RC COMMANDS
RATE GYRO
ASSEMBLY
TRIM COMMANDS
THROTTLE
ON-OFF
DESCENT ENGINE
CONTROL ASSEMBLY
GIMBAL
ON-OFF
DESCENT ENGINE
THROTTLE
ARM
DESTRUCT
SEPARATION
SHROUD
RELEASE
LEM
LEGEND:
SCS EQUIPMENT
OTHER

FIG. B 4-11

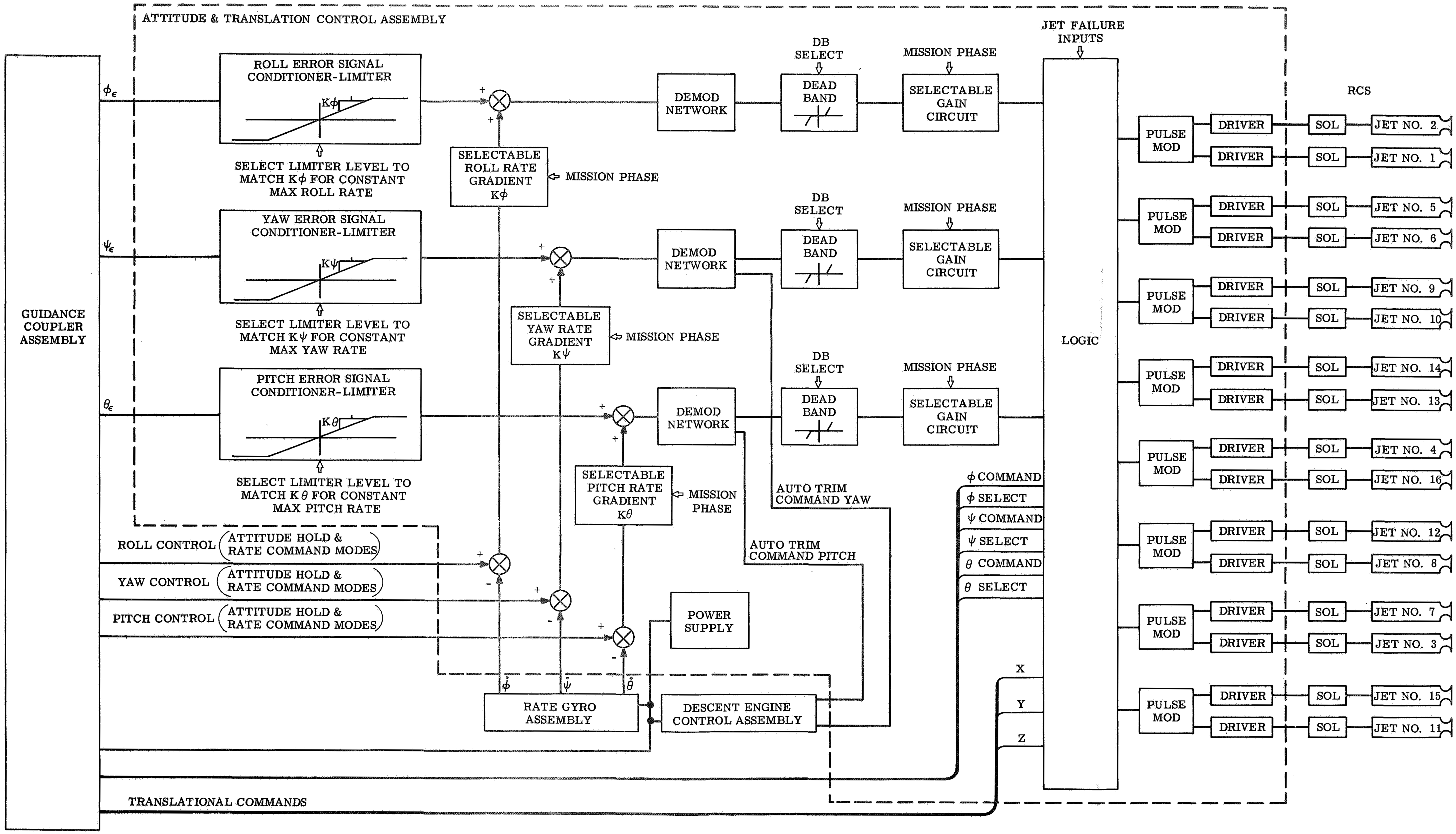
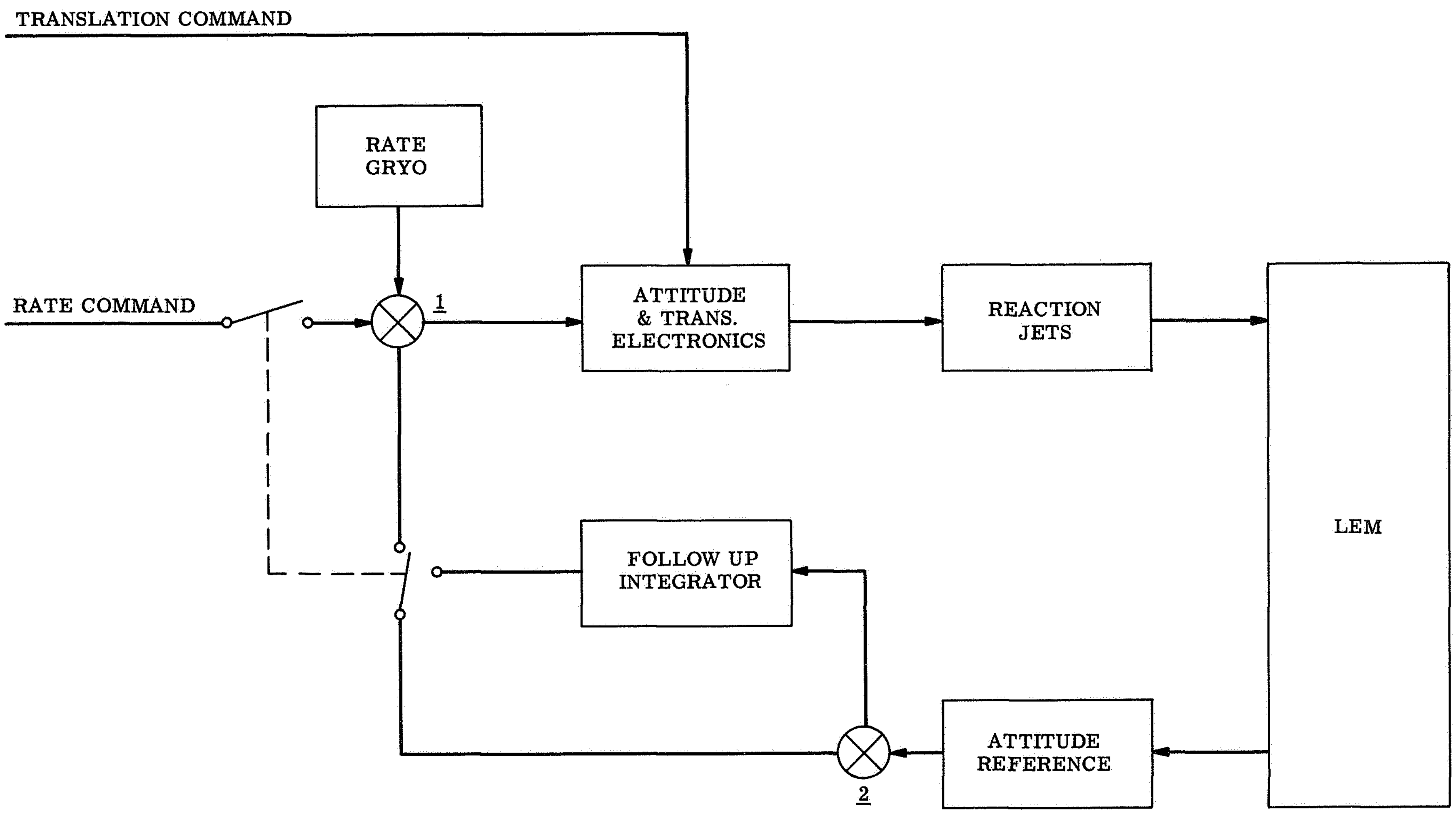
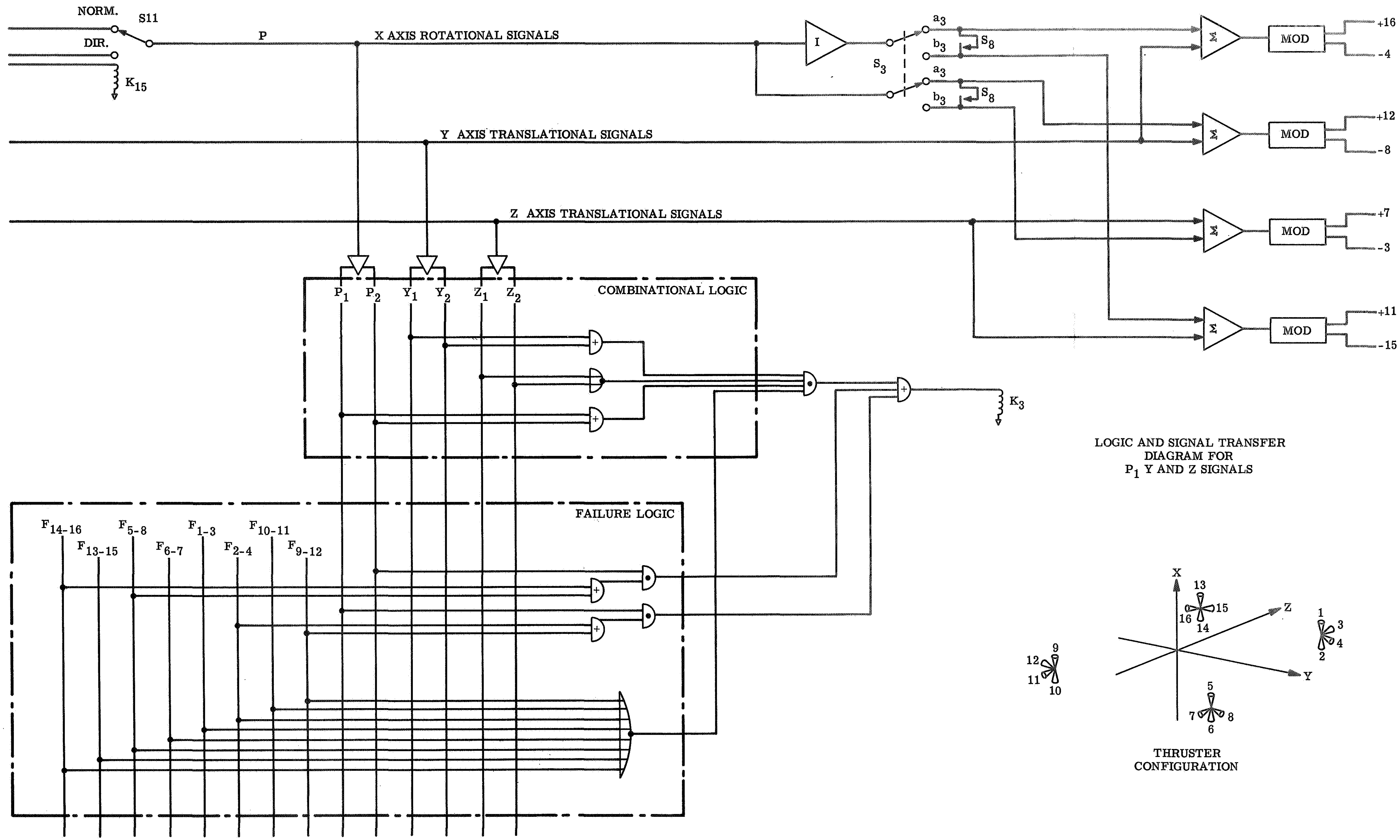

ATTITUDE & TRANSLATION CONTROL ASSEMBLY
GUIDANCE COUPLER ASSEMBLY
ROLL ERROR SIGNAL CONDITIONER-LIMITER
SELECT LIMITER LEVEL TO MATCH Kϕ FOR CONSTANT MAX ROLL RATE
YAW ERROR SIGNAL CONDITIONER-LIMITER
SELECT LIMITER LEVEL TO MATCH Kψ FOR CONSTANT MAX YAW RATE
PITCH ERROR SIGNAL CONDITIONER-LIMITER
SELECT LIMITER LEVEL TO MATCH Kθ FOR CONSTANT MAX PITCH RATE
SELECTABLE ROLL RATE GRADIENT Kϕ
SELECTABLE YAW RATE GRADIENT Kψ
SELECTABLE PITCH RATE GRADIENT Kθ
MISSION PHASE
DEMOD NETWORK
DB SELECT
DEAD BAND
SELECTABLE GAIN CIRCUIT
JET FAILURE INPUTS
LOGIC
RCS
PULSE MOD
DRIVER
SOL
JET NO. 2
JET NO. 1
JET NO. 5
JET NO. 6
JET NO. 9
JET NO. 10
JET NO. 14
JET NO. 13
JET NO. 4
JET NO. 16
JET NO. 12
JET NO. 8
JET NO. 7
JET NO. 3
JET NO. 15
JET NO. 11
AUTO TRIM COMMAND YAW
AUTO TRIM COMMAND PITCH
ϕ COMMAND
ϕ SELECT
ψ COMMAND
ψ SELECT
θ COMMAND
θ SELECT
X
Y
Z
ROLL CONTROL (ATTITUDE HOLD & RATE COMMAND MODES)
YAW CONTROL (ATTITUDE HOLD & RATE COMMAND MODES)
PITCH CONTROL (ATTITUDE HOLD & RATE COMMAND MODES)
POWER SUPPLY
RATE GYRO ASSEMBLY
DESCENT ENGINE CONTROL ASSEMBLY
TRANSLATIONAL COMMANDS

FIG. B 4.12



NOTE: SWITCH SHOWN IN ATTITUDE HOLD POSITION

FIG. B 4-13



LOGIC AND SIGNAL TRANSFER
DIAGRAM FOR
P_1 Y AND Z SIGNALS

FIG. B 4-14A

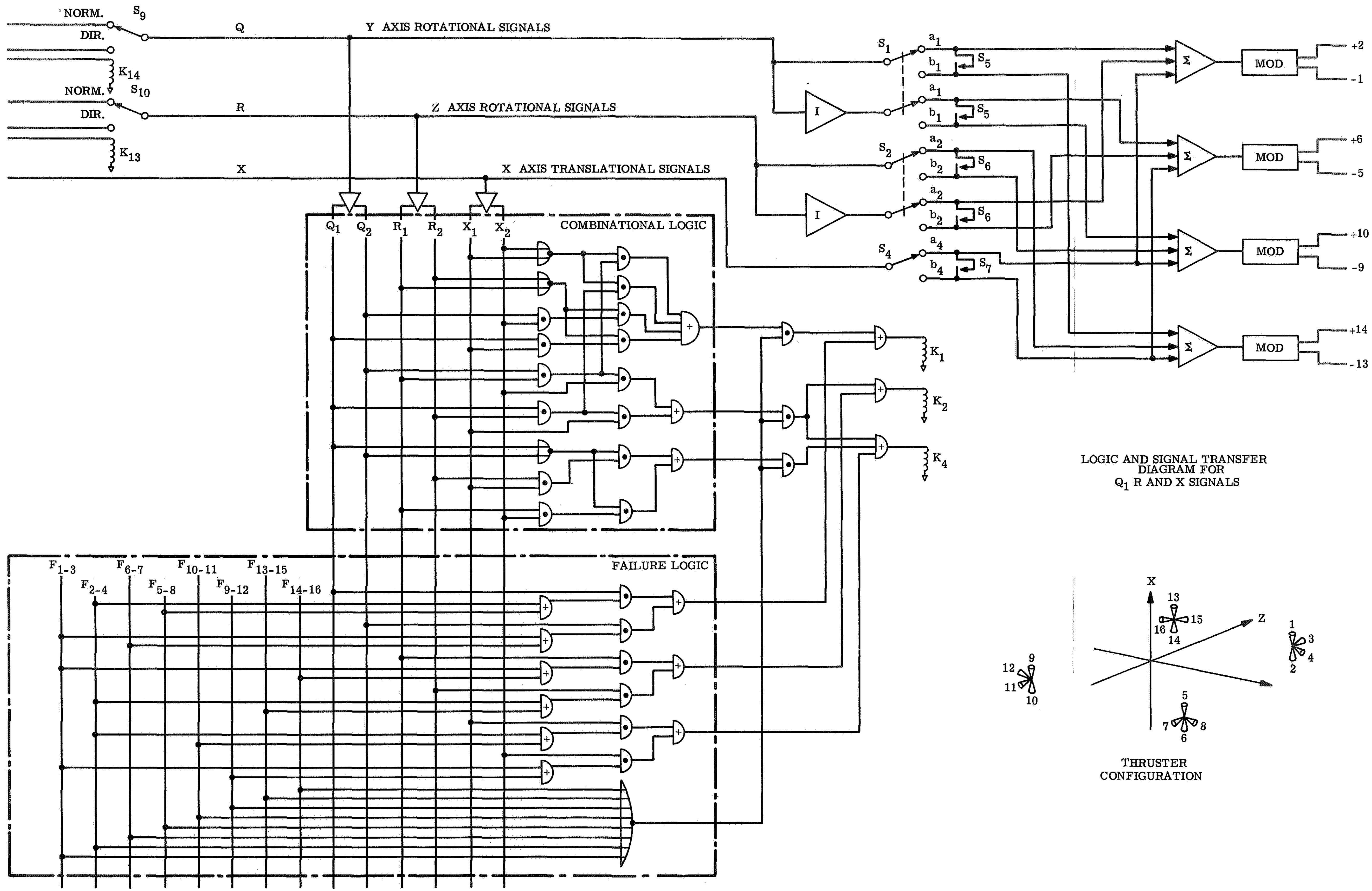

NORM.
DIR.
S_9
K_{14}
NORM.
DIR.
S_{10}
K_{13}
Q
R
X
Y AXIS ROTATIONAL SIGNALS
Z AXIS ROTATIONAL SIGNALS
X AXIS TRANSLATIONAL SIGNALS
COMBINATIONAL LOGIC
Q_1 Q_2 R_1 R_2 X_1 X_2
FAILURE LOGIC
F_{1-3} F_{2-4} F_{6-7} F_{5-8} F_{10-11} F_{9-12} F_{13-15} F_{14-16}
S_1 S_2 S_4
a_1 b_1 a_2 b_2 a_4 b_4
S_5 S_6 S_7
I
Σ
MOD
+2 -1 +6 -5 +10 -9 +14 -13
K_1 K_2 K_4
LOGIC AND SIGNAL TRANSFER DIAGRAM FOR Q_1 R AND X SIGNALS
X Z
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16
THRUSTER CONFIGURATION

FIG. B 4-14B

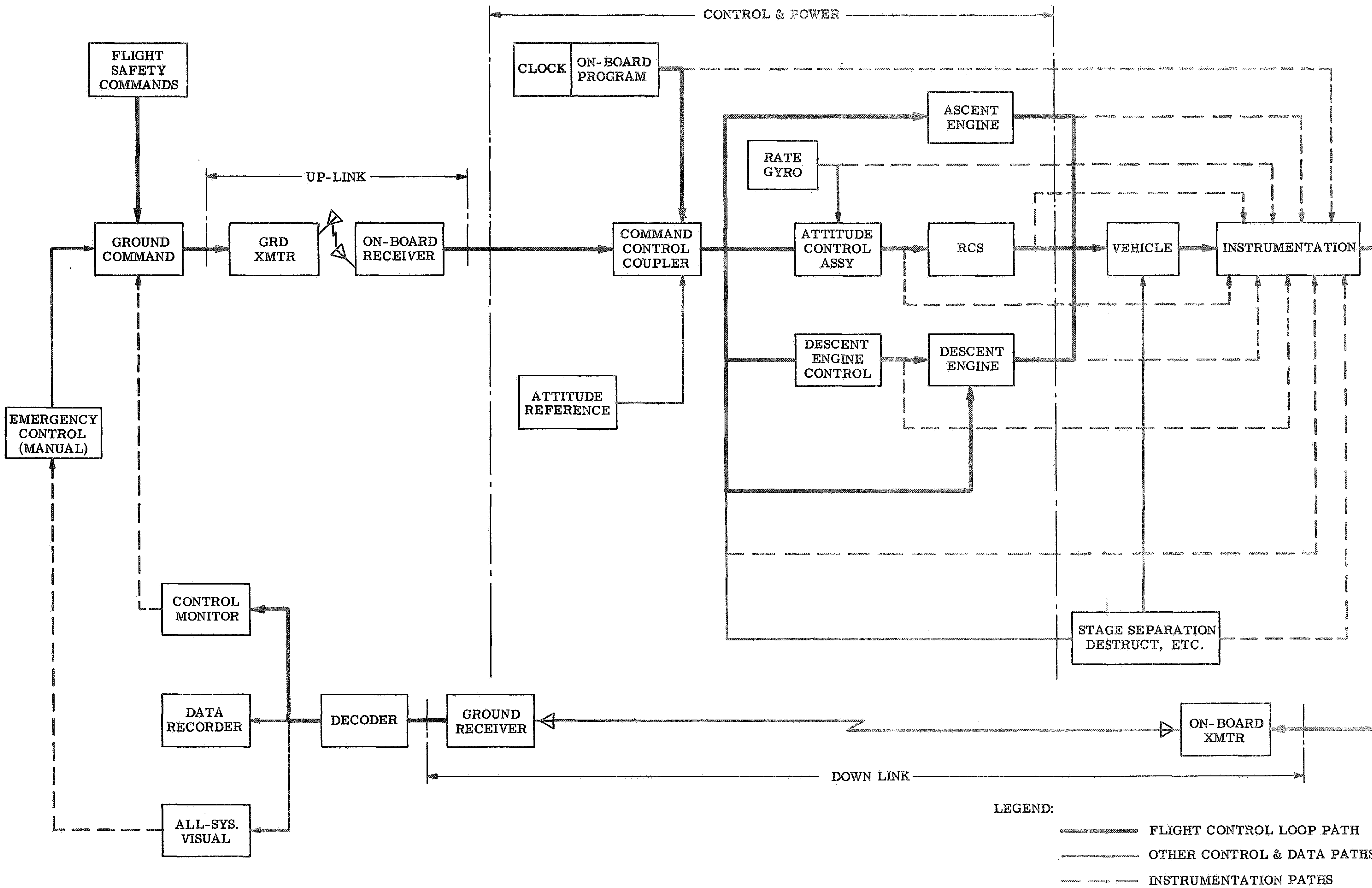
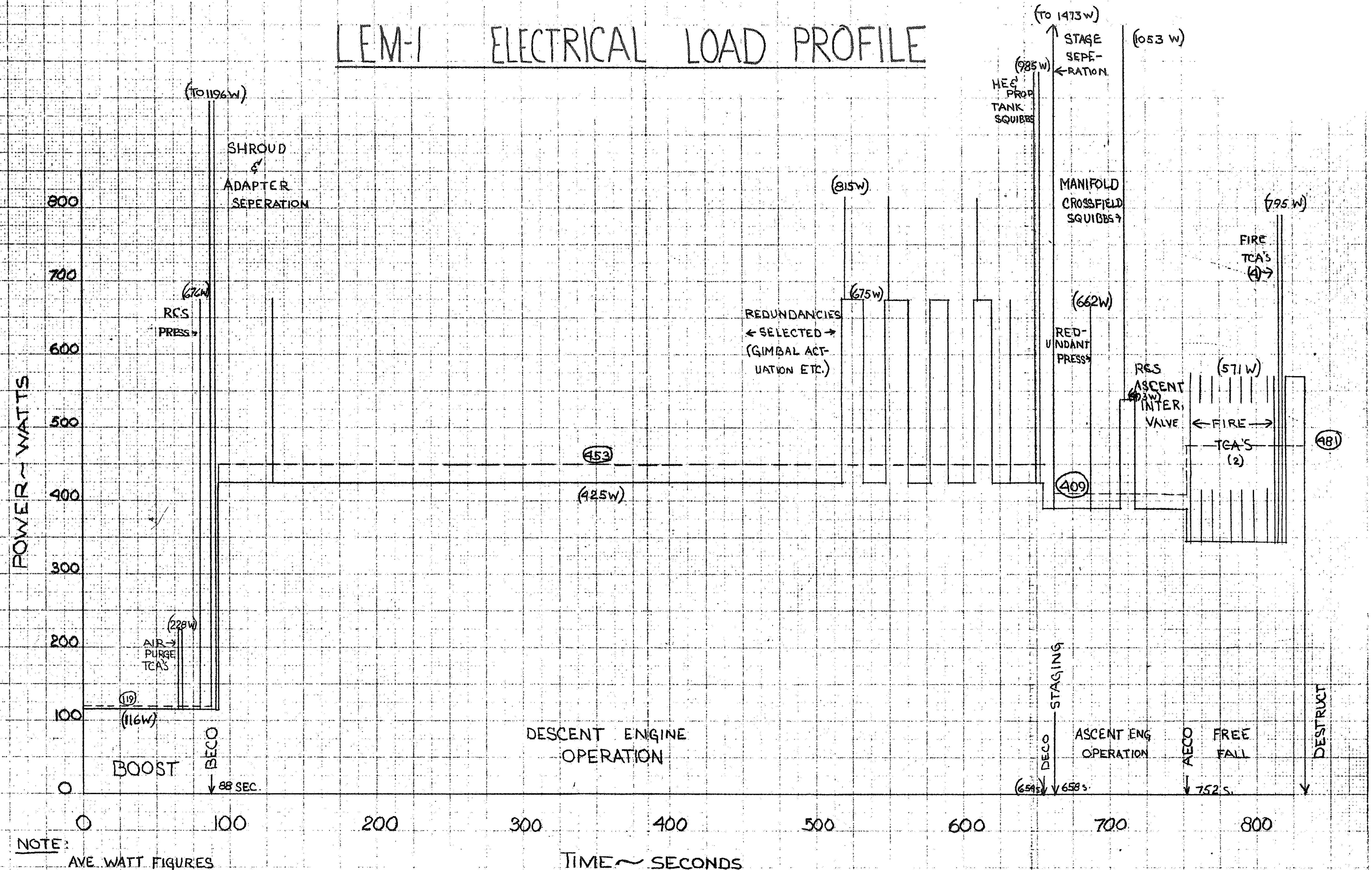

CONTROL & POWER
FLIGHT SAFETY COMMANDS
CLOCK
ON-BOARD PROGRAM
ASCENT ENGINE
RATE GYRO
UP-LINK
GROUND COMMAND
GRD XMTR
ON-BOARD RECEIVER
COMMAND CONTROL COUPLER
ATTITUDE CONTROL ASSY
RCS
VEHICLE
INSTRUMENTATION
DESCENT ENGINE CONTROL
DESCENT ENGINE
ATTITUDE REFERENCE
EMERGENCY CONTROL (MANUAL)
CONTROL MONITOR
STAGE SEPARATION DESTRUCT, ETC.
DATA RECORDER
DECODER
GROUND RECEIVER
ON-BOARD XMTR
DOWN LINK
ALL-SYS. VISUAL
LEGEND:
FLIGHT CONTROL LOOP PATH
OTHER CONTROL & DATA PATHS
INSTRUMENTATION PATHS

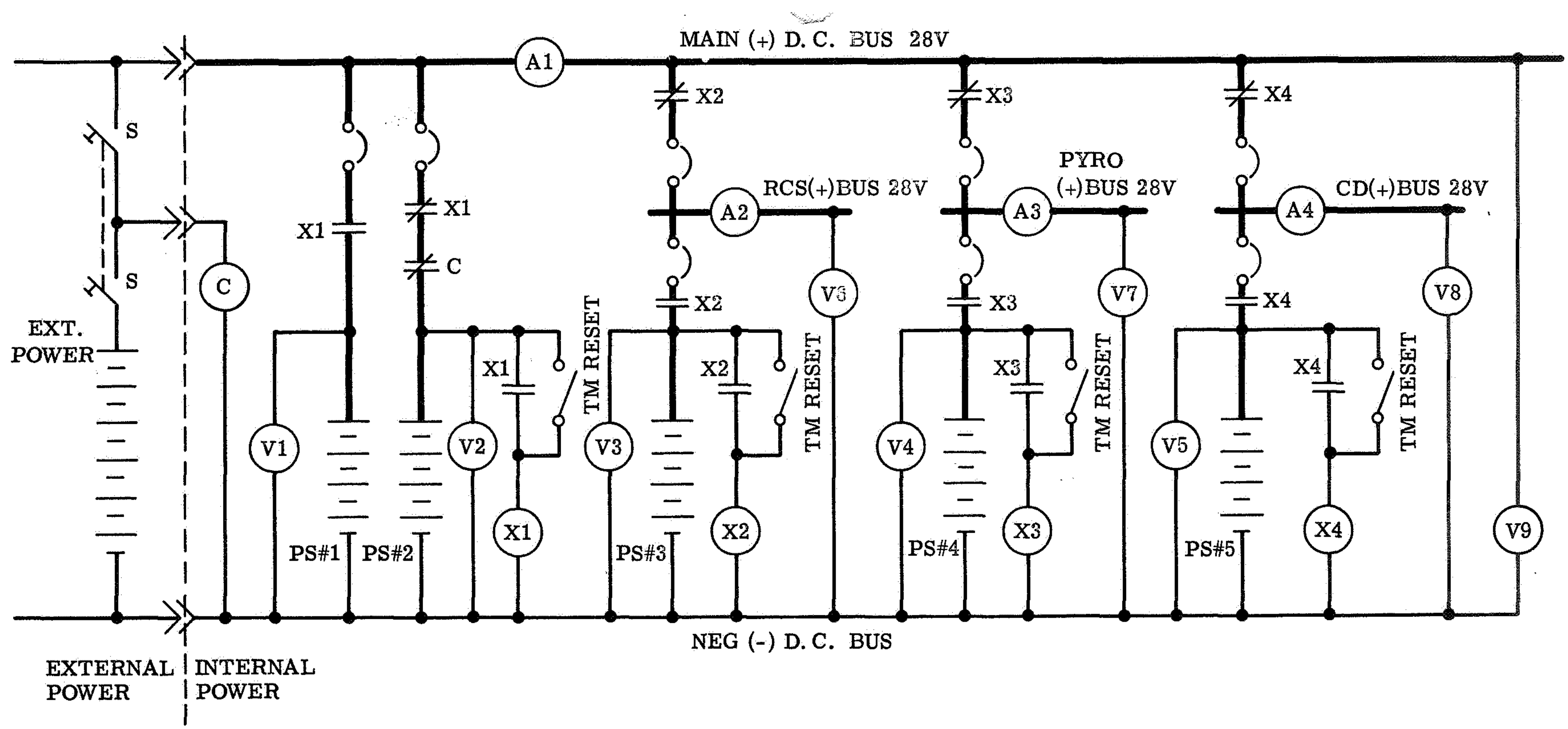
FIG. B 4-15

LEM-I ELECTRICAL LOAD PROFILE



NOTE: AVE WATT FIGURES ARE CIRCLED

FIG 4-16

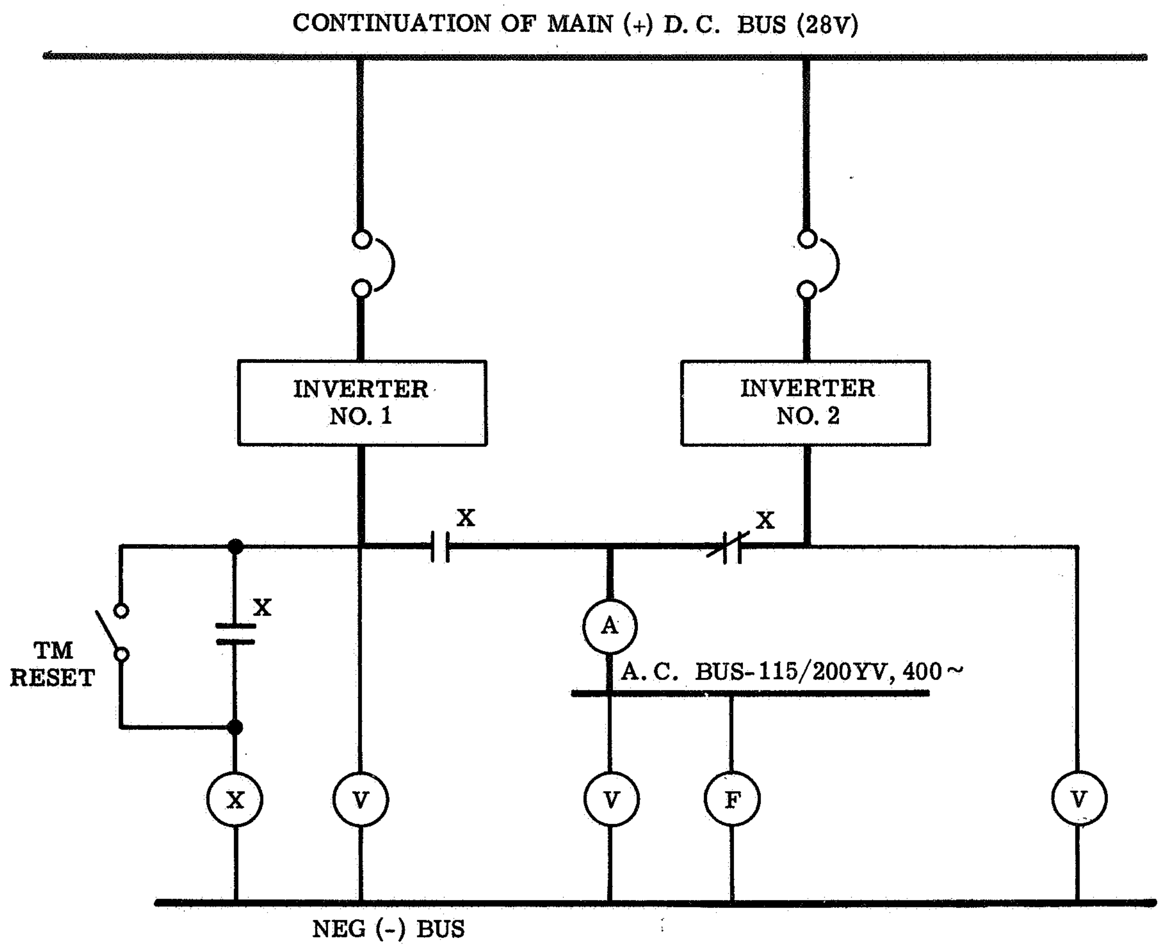


LEGEND:

- PS – POWER SUPPLY
- X – LOCKOUT CONTACTOR, RESET BY TELEMETER PULSE
- V – VOLTAGE SENSOR
- A – CURRENT SENSOR
- C – EXTERNAL POWER CONTACTOR
- (symbol) – CIRCUIT PROTECTION DEVICE
- S – SWITCH

PRELIMINARY SCHEME FOR LEM #1 & #2
D. C. DISTRIBUTION SYSTEM

FIG. B 4-17



LEGEND:

INV - 3 ϕ INVERTER 28V. D. C. TO 120/208YV, 400 ~

V - 3 ϕ VOLTAGE SENSOR

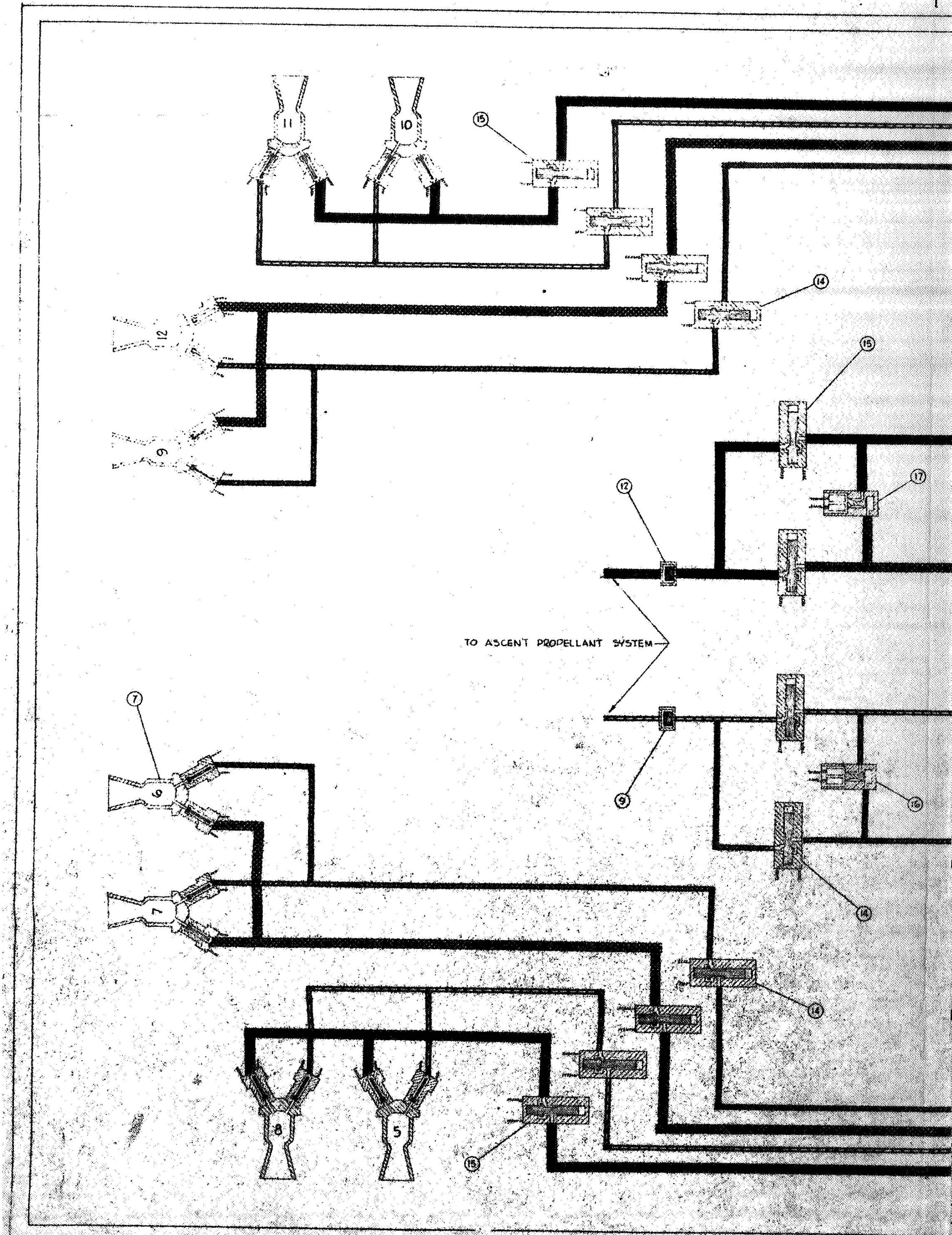
A - 3 ϕ CURRENT SENSOR

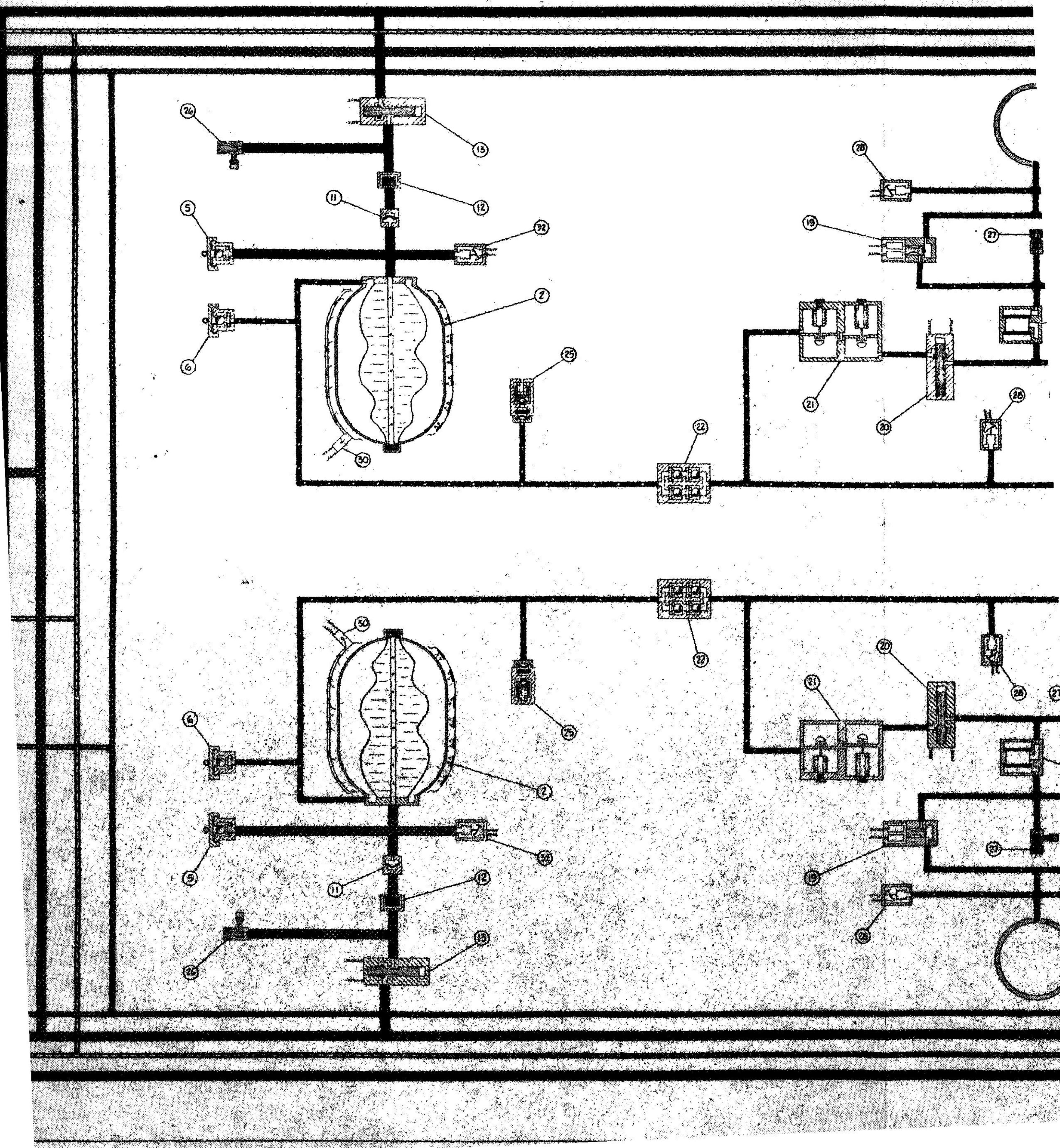
F - FREQUENCY SENSOR

X - LOCKOUT CONTACTOR, RESET BY TM PULSE

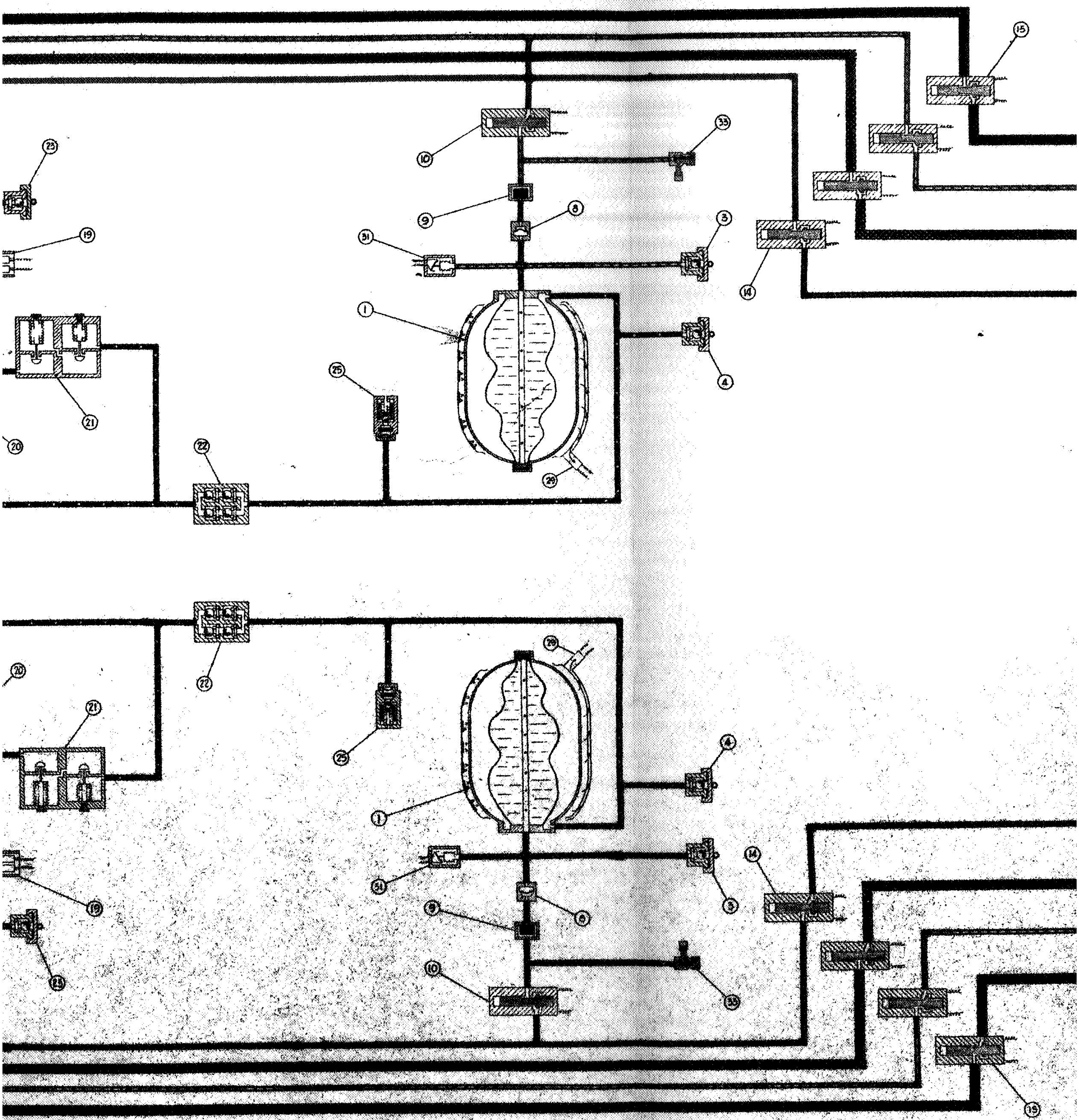
PRELIMINARY SCHEME FOR LEM 1 AND 2 A. C. DISTRIB SYSTEM

FIG. B 4-18

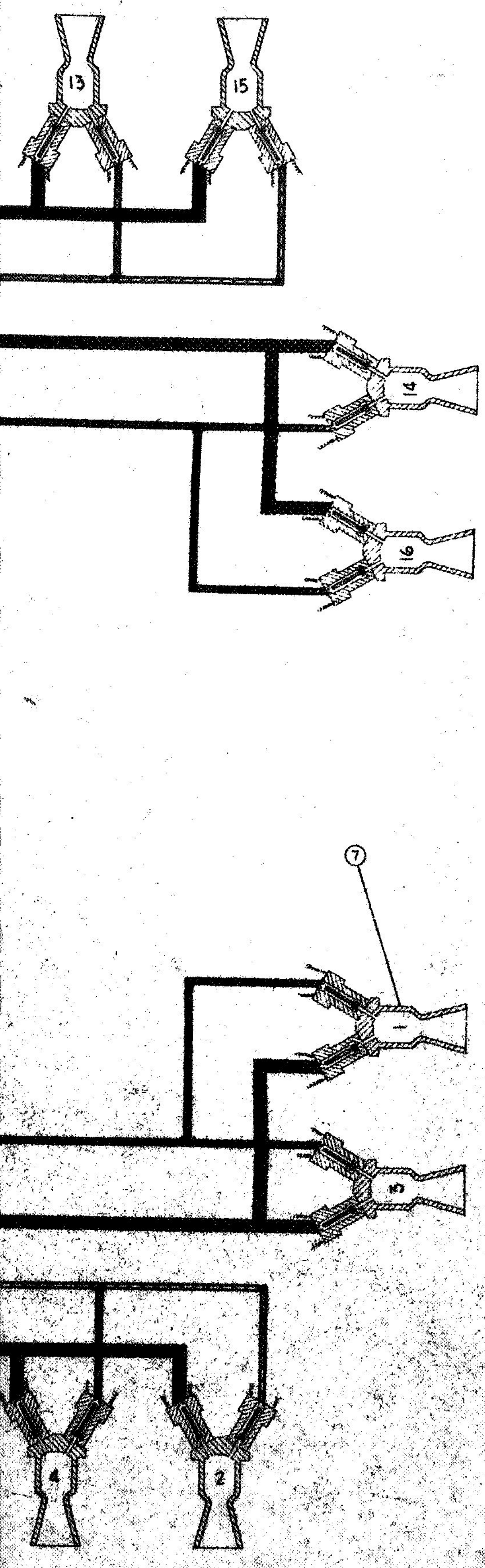

TO ASCENT PROPELLANT SYSTEM



26
13
12
11
5
32
2
6
25
30
22
28
19
27
21
20
28
30
22
6
25
20
21
28
2
5
32
11
12
19
26
13
28


15
10
33
23
9
8
3
31
19
14
1
4
21
25
20
22
29
22
20
21
25
28
4
1
14
31
3
19
8
9
23
10
33
15

CHANGE LETTER	CHANGE NO.	DESCRIPTION OF CHANGE AND REASON	CHANGED BY AND DATE	ENGRG APPD



ITEM NO.	DRAWING NO.	NAME OF ITEM
1	LSC 310-111	POSITIVE EXPULSION OXIDIZER TANK
2	-112	POSITIVE EXPULSION FUEL TANK
3	-117	OXIDIZER TANK FILL & DRAIN DISCONNECT COUPLING
4	-118	OXIDIZER TANK FILL & VENT DISCONNECT COUPLING
5	-119	FUEL TANK FILL & DRAIN DISCONNECT COUPLING
6	-120	FUEL TANK FILL & VENT DISCONNECT COUPLING
7	-150	THRUST CHAMBER ASSEMBLY
8	-151	OXIDIZER BURST DISC ASSEMBLY
9	-152	OXIDIZER FILTER ASSEMBLY
10	-153	OXIDIZER MAIN SHUTOFF VALVE
11	-154	FUEL BURST DISC ASSEMBLY
12	-155	FUEL FILTER ASSEMBLY
13	-156	FUEL MAIN SHUTOFF VALVE
14	-157	OXIDIZER VALVE, THRUST CHAMBER ISOLATION
15	-158	FUEL VALVE, THRUST CHAMBER ISOLATION
16	-159	OXIDIZER CROSSFEED VALVE
17	LSC 310-160	FUEL CROSSFEED VALVE
18		RCS HELIUM TANK
19	LSC 310-302	EXPLOSIVE ACTUATED HELIUM INITIATING VALVE
20	-303	HELIUM REGULATOR SHUTOFF VALVE
21	-304	HELIUM DUAL STAGE PRESSURE REGULATOR
22	-306	HELIUM QUAD ASSEMBLY CHECK VALVE
23	-307	HELIUM TANK FILL DISCONNECT COUPLING
24	-308	HELIUM FILTER
25	LSC 310-510	LOW PRESSURE HELIUM RELIEF VALVE
26		FUEL GROUND TEST DISCONNECT
27		HELIUM GROUND TEST DISCONNECT
28		HELIUM PRESSURE TRANSDUCER
29		OXIDIZER QUANTITY GAUGE
30		FUEL QUANTITY GAUGE
31		OXIDIZER PRESSURE TRANSDUCER
32		FUEL PRESSURE TRANSDUCER
33		OXIDIZER GROUND TEST DISCONNECT

LINE CODE:

- 'A' SYSTEM FUEL LINE
- 'B' SYSTEM FUEL LINE
- 'A' SYSTEM OXIDIZER LINE
- 'B' SYSTEM OXIDIZER LINE
- HELIUM LINE

PRIMARY NO. 500
TYPE II

Fig B4-19

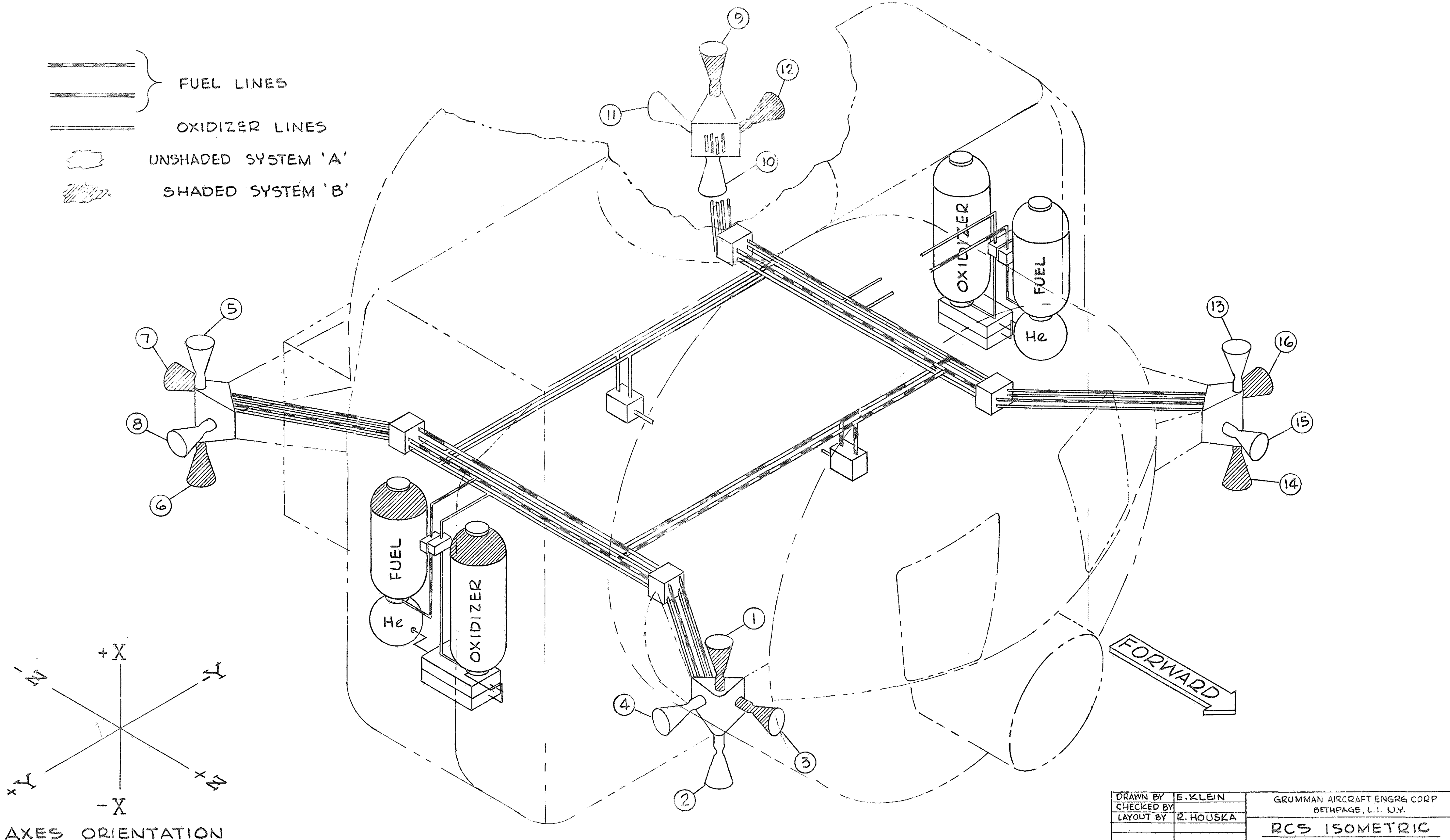
GRUMMAN AIRCRAFT ENGRG CORP
BETHPAGE, L. I., N. Y.

FLOW DIAGRAM

REACTION CONTROL SUBSYSTEM

CONTRACT NO. NAS-9-1100

26512 J LDW-310-10100



DRAWN BY	E. KLEIN	GRUMMAN AIRCRAFT ENGRG CORP BETHPAGE, L.I. N.Y.	
CHECKED BY			
LAYOUT BY	R. HOUSKA	RCS ISOMETRIC INSTALLATION	
		CONTRACT NO.	
		CODE IDENT NO. 26512	FIG. B 4-20 O A
		SCALE NONE	7-31-63 SHEET

LEM-1 COMMAND DESTRUCT

(PRELIMINARY)

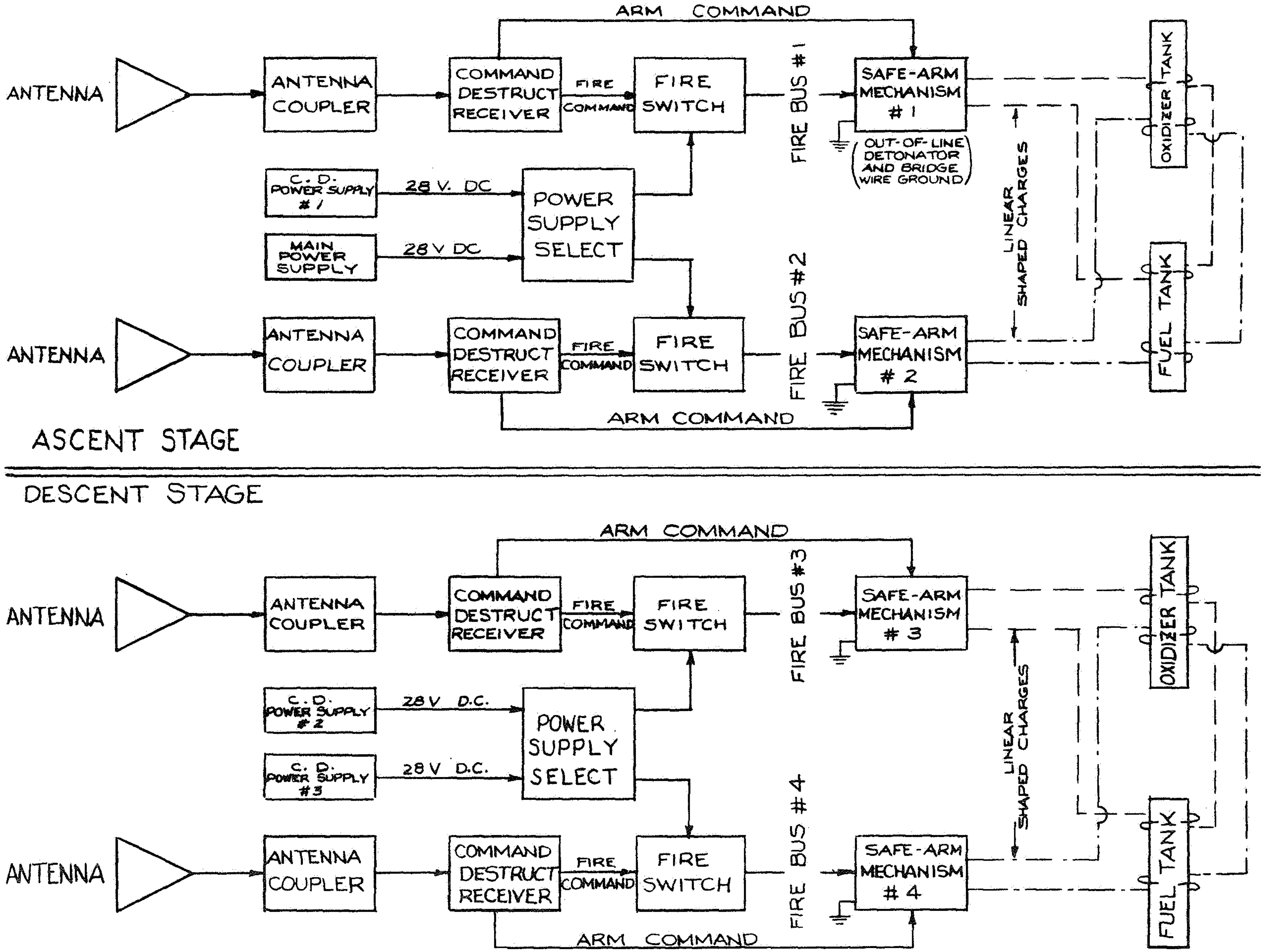


FIG 4-21

5.0 Flight Test Constraints.-

5.1 LEM System Prerequisites.-

The prerequisites for this test are:

a. The rocket motors (ascent and descent) shall have completed qualification tests. P-5 rig testing shall have advanced to the point where prototypes of the ascent and descent propulsion subsystems have been defined.

b. Satisfactory demonstration of "fire-in-the-hole" on the P-5 rig shall have been completed.

c. Completion of one full duration descent engine firing on LTA-5. During the firing, the ascent and descent stages mated. The engine will be controlled by a stabilization and control subsystem configured as in LEM-1 and installed in the test vehicle.

d. Completion of one full duration ascent engine firing on LTA-5. The engine shall be controlled by a stabilization and control subsystem configured as in LEM-1 and installed in the test vehicle.

e. The following partial subsystems shall have been qualified as systems to LEM-1 mission environments:
   - Environmental Control
   - Electric power supply
   - Stabilization and control

f. The LEM-1 command subsystem shall have been qualified to mission level.

g. The R & D Instrumentation shall have been qualified.

h. Satisfactory EMI check of the LEM-1 configuration shall have been completed.

i. The Command Destruct subsystem shall have been qualified and checked out in an EMI environment.

j. GSE verification with LSTU for WSMR shall have been completed on LTA-1.

k. Structural qualification (pressure, static, and vibration) tests on LTA-3 shall have been completed except for landing gear and landing loads.

l. Demonstration of LEM staging shall have been completed.

m. RCS design verification at Marguardt and altitude cluster tests at AEDC shall have been completed.

5.1 (con't)

n. RCS and ascent propulsion subsystem interface tests shall have been completed.

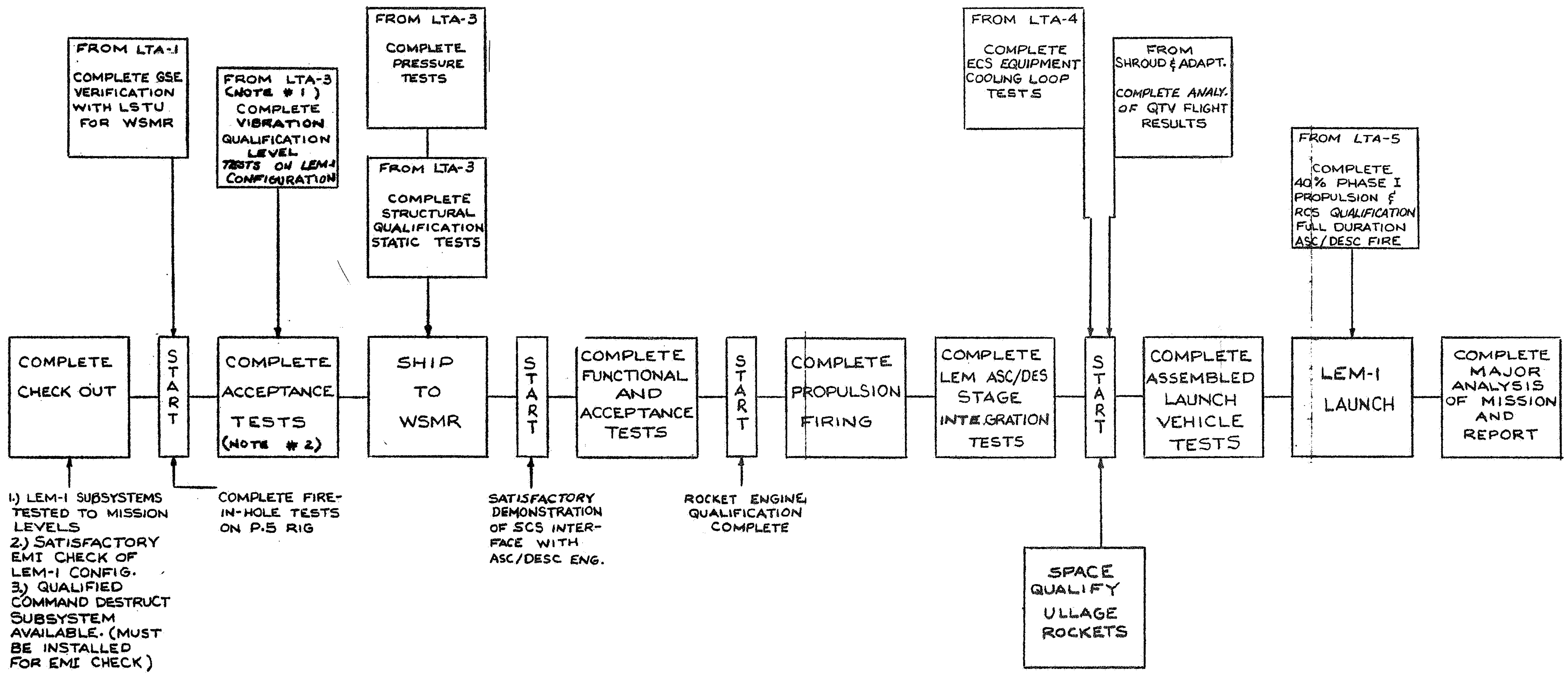
o. If included, the qualification of the LEM-1 descent stage ullage rockets shall be completed.

p. Analysis of the booster QTV flight data shall have been completed.

q. Prelaunch checkout shall have been completed.

5.2 P.E.R.T.-

The LEM-1 test is presently being PERTed. An early indication of the PERT flow is given in Figure 5-1.



Note I The LEM-1 configuration will be simulated with dummy masses on LTA-3. After a modal survey, vibration to qualification level will be imposed and the response at equipment locations will be measured. The qualification levels to which LEM-1 equipment will be subjected on a component level will then be reviewed to insure that these levels encompass the levels obtained on LTA-3.

Note 2 Vibration tests during the checkout and acceptance period at Grumman will determine that the vehicle will operate while vibrations not exceeding flight levels are applied at the engine mounts. These vibration inputs will be obtained from rocket engine firings at the engine manufacturer.

FIG. 5-1

6.0 Aerodynamic and Structural Design Criteria.-

6.1 Aerodynamic Design Criteria.-

6.1.1 Description.- (Shroud)

The configuration studies at Grumman and NASA MSC together with support from Ames Laboratory personnel have evolved two LEM/Little Joe shroud designs. The two configurations which were designated by Grumman as N_5-50 and N_6-50 are shown in figure 6-1. The two differ only in the nose shape detail forward of the 212 inch maximum diameter.

The generally squat appearance of the LEM shrouds is a direct result of minimizing the forebody destabilizing moments. The entrance angle at the nose-major diameter juncture is held at 15^{o} to minimize the intense buffet loads encountered in the shoulder region at larger angles. The boattail angle (4^{o}) was selected from Ames experience which indicated that excessive boattail convergence on hammer-head configurations can lead to unsteady air loads which correlate with structural loads and contribute strongly to vehicle dynamic instability.

6.1.2 Tail Effectiveness Considerations.-

Pending wind tunnel test, the best available evidence indicates only a minor loss of tail effectiveness is to be expected in the presence of the forebody wake at subsonic speeds. This opinion is expressed with knowledge that during earlier wind tunnel tests of a Little Joe I/Apollo configuration resulted in completely disruptive wake effects. In this latter instance more than five times the LEM base area was involved and the forebody and boattail angles were extremely aggravating.

6.1.3 Aerodynamic Performance Parameters.-

Figures 6-2 through 6-5 contain the estimated aerodynamic characteristics of the Little Joe II/LEM N_5-50 configuration. For all aerodynamic coefficients, other than elevon hinge moments, the booster cross sectional area (129 ft^2), and the Little Joe diameter (12.8 ft.), are used for the reference area and reference length, respectively.

The elevon hinge moment about a center of gravity near Xcp/D = 1.5 is very closely approximated by the C_{n_δ} of figure 6-4, with the assumption that the tail load acts at the booster base. This curve includes aeroelastic effects and an estimated 7% loss in subsonic-transonic fin effectiveness.

6.1.4 Longitudinal Stability.-

6.1.4.1 General.-

Figure 6-6 gives an over-all appreciation of the longitudinal

6.1.4.1 General.- continued

stability. The figure was generated with the data from the curves of the previous sections and a reference center-of-gravity variation with Mach number that approximates the LEM-1 mission. The center-of-gravity position is indicated in booster diameters on the top of the figure. In this figure the ordinate indicates the ratio of elevon angle to the angle of attack required to balance out the unstable pitch-up moment contributed by the forebody.

6.1.4.2 Aerodynamic Static Longitudinal Stability.-

Figure 6-6 shows the N_5-50 boost vehicle has a mild transonic static instability which grows with increasing Mach number. This instability requires elevon deflection to statically balance the vehicle for quasi steady-state angle of attack and thereby reduces the elevon travel available for dynamic α excursions (wind shear, etc.).

6.1.4.3 Longitudinal Dynamic Stability.-

6.1.4.3.1 General.-

Although a rigorous study is required to determine dynamic stability, an appreciation can be obtained by the addition of the Little Joe II autopilot "attitude error gain" to figure 6-6. Assuming that the attitude error is the result of a pitch excursion and thus equivalent to an angle of attack, the 8.4 degrees of elevon per degree of error is seen to render the vehicle aerodynamically stable (i.e. a one-degree error will command 8.4 degrees deflection and less than 8 are required). The margin of aerodynamic stability, however, decreases with Mach number as the increasing static instability results in less and less control surface deflection for dynamic excursions. The over-all control margin, however, is increased at a Mach number above 2.0 (actual number to be determined) when the reaction control subsystem is activated. The RCS acting as a bang-bang control (with a dead band of 0.8 degree) provides sizable restoring moments over and beyond the elevon contribution. The combined RCS and aerodynamic control is maintained through Mach 4 above which the low dynamic pressure renders the RCS to be dominant.

6.1.4.3.2 Control Effectiveness and Dynamic Response of Little Joe II Booster Performance.-

6.1.4.3.2.1 Programmed Booster Description.-

Trajectories were computed using a three-degree-of-freedom digital program (translation in a plane and pitching dynamics) to examine the dynamic stability of the Little Joe II booster/LEM ascent through the earth's atmosphere. Since the launch configuration is unstable during the entire boost phase, a

6.1.4.3.2.1 Programmed Booster Description.- (con't)

prime objective was to determine the effectiveness of the booster control system to overcome disturbing moments and maintain the body attitude at commanded, inertially-fixed directions. The booster control system used for this analysis was the latest control system of the Little Joe II and is comprised of two parts: namely, elevons and reaction jets. The elevons were assumed mounted at 45° angles to the body pitching plane and were capable of deflecting to a maximum angle of 29°. However, for this analysis the maximum deflection was limited to the stall limit of the actuators (assumed equal to 7800 ft/lbs of torque) or 22 degrees, whichever was less. A maximum deflection of 22 degrees was based upon the assumption that the aerodynamic data was non-linear above the deflection. Elevon deflections were commanded by position and rate feedback of error in pitch angle and pitch velocities using control system gains of 8.4 degree/degree error and 7.0 degree/degree/second error, respectively. Elevon response was assumed to react according to a first order system and was computed by the equation:

$$\dot{\delta} = \frac{\delta \text{ command} - \delta \text{ actual}}{\tau}$$

where τ = over-all time lag of the system = .05 sec.

The reaction control subsystem was assumed comprised of jets mounted on the fixed portion of the fin and producing a thrust force of 450# per jet normal to the fin in the direction in which the deflected elevons were producing a force. The reaction jets were energized during the first 8 seconds of flight and the last 32 seconds of the boosted flight, and produced forces only upon commanding an elevon deflection greater than 0.8 degrees. During the mid 43 seconds of boost flight, the reaction jet circuits were de-energized, thereby not producing any effective forces during this time. (The mid 43 seconds of boost flight consists of the high dynamic pressure region where the elevons are sufficiently effective to control the disturbing moments generated by angle of attack).

6.1.4.3.2.2 Results.-

Presented in Figures 6-7 through 6-10 are a complete time history of the boost portion of the ascent trajectory. For this trajectory, it was assumed that launch occurred during a 99% head wind environment. From these curves, one can readily note the small displacement of the boost inertial flight path angle from the commanded angle. (Maximum position error of 2.5°.) During this time, the elevon deflection was saturated for approximately 2 seconds. This is directly attributed to the 99% wind conditions that were assumed during launch. In practice launch should occur under more desirable wind conditions and this error in vehicle

6.1.4.3.2.2 Results.- continued

attitude would be significantly decreased. During the remaining portion of flight, booster attitude error is confined to a maximum of 0.6° and commanded elevon deflections will remain within the limit deflection. From these curves, one can readily conclude that the Little Joe II booster/LEM configuration is dynamically stable during booster burn and contains sufficient control system effectiveness to overcome disturbances during adverse wind conditions at launch.

6.1.5 Alternate Shroud Configuration.-

Up to this point all reference has been to the N_5 shroud. Both it and the N_6 shroud (figure 6-1) conform to the basic stability considerations which dictate a short or blunt nose and the buffet requirement calling for moderate entrance angles into the shoulder at the major diameter. Although N_6 is physically shorter than N_5 (two feet), it approaches the more idealistic parabolic nose shape. This advantage of decreased nose length together with the smoother axial variation of cross sectional area should lend the N_6 nose more regular characteristics through the transonic Mach range.

6.1.6 Little Joe II/LEM Launch Vehicle.-

Wind Tunnel Test Program.- *

The wind tunnel test program for the Little Joe II/LEM launch vehicle has two objectives: (1) provide an early and firm basis for vehicle detail design and as a contingency item; (2) yield sufficient related data for assessing the effects of configuration modifications.

In the design of "Hammerhead" payloads particular attention must be given to the details of forebody shape. For the Little Joe II/LEM vehicle stability considerations dictate a blunt payload envelope. Thus restricted, the only remaining design freedom revolves around the choice of head shapes which may be typified as possessing either "massive bluntness", such as N_5, or "progressive bluntness", as does N_6. The wind tunnel program recommended would test both N_5 and N_6 in the Langley 8-foot transonic pressure tunnel throughout the $M = 0.3$ to $M = 1.2$ Mach range with $\delta_f = 0$. A comparison of data obtained in this critical area would then be the basis for selecting the one payload configuration to be carried through the remaining test program outlined below. All tests are to be conducted with the standard Little Joe II 50-foot^2 fins. The data rate and early data availability from the 8-foot TPT should permit the required configuration selection to be made within one and a half days from commencing test.

* Grumman wishes to acknowledge the generous assistance of NASA personnel of the "Experimental Aerodynamics Section", Manned Spacecraft Center, in the Test Plan formulation.

6.1.6.1 Tests in the Langley 7' x 10' Wind Tunnel.-

Mach Number: M = 0.1 and 0.2

$\alpha = \pm 0^\circ, 1^\circ, 2^\circ, 3^\circ, 5^\circ, 8^\circ, 11^\circ, 15^\circ$

$\delta_f = 0^\circ, 5^\circ, 10^\circ, 20^\circ, 30^\circ$, Fins Off

For $\delta_f = 0^\circ, +5^\circ$ Alpha range taken out to $+90^\circ$ at 10° increments

6.1.6.2 Tests in the Langley 8' Transonic Pressure Tunnel.-

$\alpha = -11^\circ, -5^\circ, -3^\circ, -2^\circ, -1^\circ, 0^\circ, 1^\circ, 2^\circ, 3^\circ, 5^\circ, 8^\circ, 11^\circ, 15^\circ$

$\delta_f = 0^\circ, 5^\circ, 10^\circ, 20^\circ, 30^\circ$, Fins Off

NOTE: Flow visualization data (Schlieren) are particularly important in this range. Also, all traverses in angle of attack should commence at -11° increasing to $+15^\circ$ (taking data) and then conclude by running (taking data) at $\alpha = +5^\circ, 3^\circ, 2^\circ, 1^\circ, 0^\circ, -1^\circ$. The object here is to detect hysteresis near = 0°. Hysteresis or major non-linearities near $\alpha = 0^\circ$ are generally indicative of poor dynamic stability characteristics (the correlated, unsteady aerodynamic loads cited in the "Aerodynamics" section of the present report).

6.1.6.3 Test In the Unitary Plan Wind Tunnel (Section #1).-

Mach number: M = 1.57, 1.80, 2.16, 2.80

$\alpha = \pm 0^\circ, 1^\circ, 2^\circ, 3^\circ, 5^\circ, 8^\circ, 11^\circ, 15^\circ$

$\delta_f = 0^\circ, 5^\circ, 10^\circ, 20^\circ, 30^\circ$

6.1.6.4 Tests In the Unitary Plan Wind Tunnel (Section #2).-

Mach number: M = 3.86, 4.65

α, δ_f, same as Section #1

6.1.6.5 Tests In the Ames 14-Foot Transonic Wind Tunnel.-

After the final shroud configuration has been selected, a model should be built to obtain the fluctuating pressures over critical areas of the shroud from Mach 0.60 to 1.20. This is recommended because of the uncertainty in applying available model data or analysis. These pressures will be used to conduct a buffeting load analysis on the shroud panels.

The Ames 14-Foot Transonic Wind Tunnel is recommended because of the experience that Ames personnel have in this field.

6.2 Structural Design Criteria.-

6.2.1 Design Loads.-

The most significant parameters affecting the design loads for this test shot are the wind shear profile, which produces a resultant angle of attack, and the dynamic pressure, which determines the load produced by a given angle of attack. The maximum wind velocity and the Mach number have second-order effects.

A number of performance trajectories were run for the LEM/ Little Joe II configuration on a digital computer in order to establish the most satisfactory firing order for the test shot. For the firing sequence selected, it was noted that the maximum dynamic pressure was attained at an altitude of about 27,000 feet and that 90% of maximum dynamic pressure was still encountered at 34,500 feet. The minimum altitude at which maximum wind shear values are obtained, according to NASA/Huntsville memo M-AERO-6-33-62, is 32,800 feet.

The maximum design loads should be attained at some altitude intermediate between that where maximum dynamic pressure occurs and that where maximum wind shear occurs. Since, for this feasibility study, there was not sufficient time to check each altitude, runs were made through a synthetic wind profile based on an altitude of 32,800 feet. The resultant load calculations are within 5% of the theoretical maximum. Wind tunnel pressure distribution measurements are required to justify any further refinement of the design loads.

Three-degree-of-freedom trajectories for a 99% probability wind velocity at 32,800 feet and for an 84% probability wind shear build-up to an 84% probability wind velocity at the same altitude are calculated. In both cases, a 29.5 ft./sec. imbedded trapezoidal jet having a 300 meter wave length was superimposed on the synthetic wind profile just above 32,800 feet, as recommended in M-AERO-6-33-62, NASA/Huntsville memo, see figure 6-11. Maximum angles of attack were obtained during the traverse of this imbedded jet in each case. The fins are oriented at 45° to the wind velocity, since this is the most favorable attitude in regard to vehicle controllability.

Axial and circumferential pressure distributions are presented in figures 6-12 through 6-14.

The resultant axial shear and bending moments are presented in figures 6-15, 6-16, and 6-17.

Although these calculations were performed by treating the vehicle as a rigid body, a rough check shows that if the control system has the required ability to overcome the small degradation in stability due to aeroelastic effects, the increase in loads due to aeroelastic effects will be less than 5%.

6.2.1 Design Loads.- continued

A summary of the significant parameters at the critical design point is as follows:

						Airloads		
Wind Profile	q psf	α deg.	δ deg.	n_x	n_z	Shear @ 0 Kips	B.M. @ 0 Inch Kips	One Fin Kips
84%	957.0	6.02	11.8	1.28	.936	51.9	11,900	30.1
99%	967.5	7.15	13.7	1.30	1.12	62.0	14,300	36.0

6.2.2 Structural Dynamics Requirements.-

6.2.2.1 Shroud Panel Flutter.-

The Little Joe II shroud panels were investigated for panel flutter, using the method of NASA TN D-451. The analysis was performed from Mach 1.17 to Mach 1.53. Mach 1.53 corresponds to the maximum dynamic pressure (1020 psf). The panels are 10 by 17.5 inches (10 inches along the stream-wise direction) and 0.050 inches thick. All our results fall in the "no flutter" region of figure 2 of the above reference; therefore, preliminary results indicate that panel flutter is not anticipated on the LEM shroud of the Little Joe II/LEM-1 configuration. More refined methods of panel flutter will be employed in the future, and it is anticipated that model tests of panels will be performed to substantiate the theoretical results.

6.2.2.2 Booster Flutter Analysis.-

Flutter analysis on the booster for the present fin configuration was conducted by General Dynamics, San Diego. The analysis consisted of 10 modes. These modes included cantilevered fin, symmetrical fin on body, fin bending, fin torsion, elevon rotation, and the first four body bending modes. According to General Dynamics, the configuration was flutter-free up to q = 1580 psf and Mach 5.0.

6.2.2.3 Buffeting.-

The buffeting analysis was based on model test results of NASA TM-779. The C_p (rms) used in the analysis were obtained from figure 9c, which is for a model with closest correspondance to the LEM shroud. The fluctuating pressure of figure 9c corresponds to an angle of attack of 8 degrees. Therefore, the results should be conservative, since the Little Joe II/LEM-1 trajectory is not expected to exceed 5 degrees during the transonic part of its flight.

Figure 9c shows that fluctuating pressures built up from Mach 0.70 to 0.92. This corresponds to approximately 5 seconds of flight.

6.2.2.3 Buffeting.- continued

Figure 9c also shows that the pressure build-up is not stationary but moves in the aft direction as the Mach number increases.

The model C_p (rms) was multiplied by the full scale vehicle dynamic pressure to obtain an over-all rms pressure level. Figure 6-18 shows this over-all rms pressure level as a function of flight time and shroud position. Figure 6-18 shows that pressure higher than 0.2 psi occurs for about one second at the highest loaded section of the shroud. These over-all rms pressure levels were then converted to power spectral density by assuming a flat spectrum from 0 to 500 cps, which is considered conservative.

The maximum rms stress, which occurs at the center of each panel, was then calculated by obtaining the forced response of a flat plate of 10 by 17.5 inches and 0.050 inches thick, assuming the panel to be simply supported and vibrating in a fundamental half sine mode. The maximum rms stress as a function of flight time and shroud position is shown in figure 6-19.

Stress levels were also calculated at the highest loaded shroud position for a skin thickness of 0.060 inches. From figure 6-19, it can be seen that if 0.060 inch skin thickness is used in critical areas and 0.050 inch skin thickness is used elsewhere on the shroud, an rms stress level of less than 20,000 psi due to buffeting can be maintained.

A more refined analysis will be performed in the future by obtaining fluctuating pressures on models that will be an exact geometrical representation of the Little Joe II/LEM-1 configuration. This analysis will also consider other possible mode shapes and restrain conditions. However, preliminary calculations indicate that no extensive design problems due to buffeting are anticipated.

6.2.2.4 Vibration Requirements.-

Vibration envelopes representative of Little Joe II anticipated environment were calculated as described in Grumman Report No. LED-520-3, Appendix C "Proposed Qualification Test Levels for LEM Components and Equipment". A comparison of the Little Joe II estimates and the C-5 estimates is shown on Table 6-1. Although Little Joe II requirements are higher, the difference is not believed large enough to significantly effect equipment or structure design or weight.

The estimates for equipment vibration levels were based on a maximum rms pressure of 0.1 psi outside the Little Joe II shroud. Local vibration levels on the LEM near the region of the separated flow might be higher because of the buffeting, but this would be limited to about 2 seconds at about 20 seconds after launch. These local buffeting pressures (estimated at 0.2 psi) are not

TABLE 6-1

LEM EQUIPMENT

QUALIFICATION VIBRATION TEST LEVELS

CONDITION	RANDOM VIBRATION		SINUSOIDAL VIBRATION	
Launch & Boost - Saturn C-5 (S-IVB) Equipment mounted on shelves, racks, displays, etc.	10-23 cps	12 db/oct rise to .025 g^2/cps	5-18.5 cps	0.20" D.A.
	23-80 cps	.025 g^2/cps	18.5-250 cps	3.5 g
	80-110 cps	12 db/oct rise to .075 g^2/cps	250-380 cps	0.0011" D.A.
	110-950 cps	.075 g^2/cps	380-2000 cps	8.0 g
	950-1200 cps	12 db/oct roll-off to .025 g^2/cps		
	1200-2000 cps	.025 g^2/cps		
Launch & Boost - Little Joe II Equipment mounted on shelves, racks, displays, etc.	10-23 cps	12 db/oct rise to .025 g^2/cps	5-18.5 cps	0.20" D.A.
	23-80 cps	.025 g^2/cps	18.5-100 cps	3.5 g
	80-120 cps	12 db/oct rise to .115 g^2/cps	100-160 cps	.007" D.A.
	120-850 cps	.115 g^2/cps	160-2000 cps	9.0 g
	850-1200 cps	12 db/oct roll-off to .025 g^2/cps		
	1200-2000 cps	.025 g^2/cps		
Launch & Boost - Saturn C-5 (S-IVB) Equipment Mounted on Interior Primary Structure	20-2000 cps	.06 g^2/cps	5-16 cps	0.20" D.A.
			16-100 cps	2.5 g
			100-140 cps	.005" D.A.
			140-500 cps	5.0 g
			500-700 cps	.0004" D.A.
			700-2000 cps	10.0 g
Launch & Boost - Little Joe II Equipment mounted on Interior Primary Structure	20-2000 cps	.073 g^2/cps	5-16 cps	0.20" D.A.
			16-78 cps	2.5 g
			78-120 cps	.008" D.A.
			120-420 cps	6.0 g
			420-600 cps	.00065" D.A.
			600-2000 cps	12.0 g

6.2.2.4 Vibration Requirements.- continued

expected to materially increase the vibration environment over the entire LEM vehicle because of the limited time duration and small area exposed to the peak pressure. This problem is not unique with the Little Joe II, since a similar local buffeting on vibration levels will be investigated during the Little Joe II wind tunnel tests and design phase.

6.2.2.5 Acoustical Levels.-

The aerodynamically induced acoustical levels for the shroud outside the LEM on Little Joe II were calculated, using the procedures in WADD TR 61-62 and WADC TR 58-343 Volume II. The external overall sound pressure level during maximum 'q' condition was estimated to be .05 psi. These levels were compared with calculated engine exhaust noise during launch, again, using WADC TR 58-343 information. Calculated engine exhaust noise over-all pressure level was 0.1 psi. The octave band distribution is shown on Table 6-2. The noise levels inside the shroud are expected to be 5 to 10 decibels below these values. These levels are expected to be 3 to 5 decibels higher than corresponding C-5 launch levels.

Higher oscillating pressures than those estimated above are expected to occur for about 2 seconds at about 20 seconds after launch due to buffeting. These pressures will be limited to the area aft of the separated flow region, as shown in figure 6-18. They are expected to affect the acoustical environment over only a small portion of the LEM, causing an estimated local 12 decibel increase. Over-all noise levels outside the LEM in this region would be 152 decibels. No difficulty is anticipated with these pressures at present because of the short time and the relatively limited area, and because this pressure value is not particularly high.

TABLE 6-2

ESTIMATED LJ-II - LEM-1 ACOUSTICAL LEVELS

	OCTAVE BAND SOUND PRESSURE LEVELS	
FREQUENCY	Decibels, Referred to 0.0002 dynes/cm^2	
RANGE - CPS	LAUNCH	MAXIMUM Q
37.5-75	143	131
75-150	145	134
150-300	145	134
300-600	143	138
600-1200	139	139
1200-2400	136	137

6.2.3 Structural Temperatures.-

Transient temperature responses are presented for 3 points on the Little Joe II shroud, the stagnation point on the spherical nose (point 1), the conical transition behind the spherical nose (point 2), and the cylindrical center portion (point 3), for various thicknesses of aluminum sheet in figures 6-20 and 6-21.

6.2.3.1 Aerodynamic Heat Inputs.-

The convective heat transfer coefficients for the stagnation point were computed, using the methods of Reshotko and Cohen (reference NACA Report 1294, 1956). The temperature response was calculated using these coefficients in conjunction with the total temperatures as the driving potential.

The heat transfer coefficients for the conical transition and cylindrical sections were based on the Van Driest equations for turbulent convective heat transfer on a cone. Boundary layer build-up was accounted for by using the developed distance along the spherical nose and cone. For the cylindrical portion half, the distance from stagnation point to the cone-cylinder transition was used. Free stream flow conditions and the recovery temperature as the driving temperature were used in these sections.

This procedure yields approximate coefficients. A more detailed analysis will be performed as required.

6.2.3.2 Structural Temperature Response.-

The structural temperature response is computed by using finite difference methods programmed for the digital computer. The input data is based on the Mach number and altitude time histories shown in Section 3.4, figure 3-1. Because of the relatively thin aluminum skins being considered, one-dimensional "thin skin" assumptions are made. Convective heat input, re-radiation from the outer surface and heat storage in the aluminum skin are considered in the heat balance. The outer surface is assumed to be painted resulting in an infra-red emissivity of 0.8. In a more detailed analysis, the effects of two-dimensional, conduction and joint resistances for riveted built-up sections will be included to obtain an accurate picture of the temperature distribution through the structural cross sections.

The temperature responses shown in figures 6-20 and 6-21 indicate the approximate operating regime for the vehicle. Temperatures on the spherical nose will decrease as distance from the stagnation point increases until the sonic point is reached; at this point, a temperature rise will occur which might result in temperatures slightly above those at the stagnation point. From this point, back temperature will decrease toward the values shown for the transition cone. At the transition cone cylinder joint an over expansion will occur which could cause a local drop in temperatures on the forward part of the cylinder section. Temperatures are not shown for the tapered after-portions; these temperatures will be somewhat lower than those on the cylindrical section.

6.2.4 Structural Analysis.-

6.2.4.1 General.-

The LEM shroud is an aluminum alloy semi-monocoque shell structure housing the LEM on the Little Joe II flight which will be separated into two halves by a shaped charge and be jettisoned at a predetermined time after launcy. The shroud is designed to resist aerodynamic and inertia loads and associated environmental conditions. The adapter provides the structural support of the LEM and also serves as transition structure between the shroud and the Little Joe II interface. A prime requirement of the adapter is to provide a sufficiently uniform distribution of the LEM inertia loads and shroud aerodynamic loads to the booster interface attachment structure so as to remain within its strength capabilities.

6.2.4.2 Design Loads.-

The aerodynamic and inertia loads from the 99% wind profile trajectory are critical for design. The maximum aerodynamic loads occur at T = 37.7 seconds at maximum ultimate dynamic pressure of 1450 psf, and the maximum inertia loading occurs at T = 72.0 seconds. The shear, axial load and bending moment distribution from the aerodynamic loads are shown in Figure 6-17. The ultimate inertia load factors are shown in Table 6-3. The rotational accelerations are extremely small and have been neglected for this preliminary study. The air loads combined with the concurrent inertia loads result in the net design loads which are tabulated in Table 6-4 at station 0.

TABLE 6-3

ULTIMATE LOAD FACTORS

Time Seconds	n_x	n_z
37.7	1.95	1.68
72.0	8.25	.165

TABLE 6-4

NET ULTIMATE LOADS AT STATION 0

	Shear (lbs)	Axial Load (lbs)	Moment (in-lbs)
Air Load	93,000	176,800	21.0×10^6
Inertia Load	-46,200	55,000	-5.6×10^6
Net Load	46,800	231,800	15.4×10^6

6.2.4.3 Environmental Requirements.-

Structural temperature analysis is presented in Section 6.2.3. The plots of skin temperature as a function of time in Figures 6-20 and 6-21 indicate that they are minimum at time of critical aerodynamic loading, and increase to significant values only when the air loads are considerably diminished. The dynamic requirements presented in Section 6.2.2 are also considered in the

6.2.4.3 Environmental Requirements.- continued

selection of the skin thicknesses and structural configuration.

6.2.4.4 Structural Design of Shroud.-

The shroud structure consists of major ring bulkheads at stations -137, -177, and -275.9, with intermediate ring frames spaced at approximately 10-inch increments and a series of 36 axial longerons terminating at the adapter splice. The intermediate ring frames are required to provide stability of the shell structure, support the longerons against column buckling, and to adequately support the skin under dynamic loading environments. The frame axial load and bending continuity will be maintained across the axial separation joint by virtue of doubler plates, as shown in View B.B of the shroud drawing, figure 4-3. This joint will also provide skin shear continuity of the external skin.

The longerons are axial load members and will be designed to resist the applied axial loads and bending moments. They are stabilized by the frames and bulkheads; and will be continuous through the frames, spliced through the bulkheads, and spliced to the adapter skin at station -40. The longerons at the axial separation joint are channels of the same depth as the frames, joined on each side by a doubler plate containing the separation shaped charge. The longerons between station -275.9 and -361.901 are arches terminating at bulkhead station -275.9 and are critical for the dynamic pressure loading. It is expected that the shroud skin be .071 inch thick from the nose to station -275.9, .063 inch thick to station -137 and .050 inch thick aft to the adapter splice. The continuous skin shell structure procides capability of resisting the shear and torsion loads; however, skin thickness is predicated by the flutter and buffeting requirements. Skins are to be riveted to the frames and longerons.

6.2.4.5 Structural Design of Adapter.-

The adapter structure consists of two ring bulkheads at station 0 and at station -40 which will be designed to resist in plane moments, axial loads and shears resulting from skin axial load lateral components due to contour change. In addition, normal are applied by the LEM attachment to bulkhead compression pads and the LEM separation system. The ultimate LEM axial inertia load is 195,000#, for a 23,626# LEM weight, at 72 seconds after launch and is considered distributed over the eight compression pads. The bulkhead at station 0 provides interface attachment to the Little Joe II booster ring frame at station 0 with (32) 5/16 inch diameter bolts in the axial direction. This bulkhead will be designed to balance all lateral components of the skin loads and inertia loads so that only axial loads and shears are applied to the booster structure.

The adapter skins will be designed as stiffened monocoque structures to carry the applied bending moment, axial load and

6.2.4.5 Structural Design of Adapter.- continued

shears. The shroud longerons terminate at the upper edge of the adapter skins, which will serve to distribute the longeron axial loads to the booster structure. Separation of the shroud will be effected by a circumferential shaped change on the adapter skin immediately forward of the bulkhead at station -40.

6.2.4.6 Payload Booster Interface Joint.-

The axial loads applied to the periphery of the Little Joe II structure at Little Joe station 0 are shown in Table 6-5 for the most critical 99% wind profile flight condition.

TABLE 6-5

ULTIMATE APPLIED LOADS TO BOOSTER AT STATION 0

Time Seconds	Peripheral Axial Load
37.7	1,306 lbs/in Compression
37.7	350 lbs/in Tension
72.0	1,342 lbs/in Compression (23,626-lb LEM)
72.0	1,492 lbs/in Compression (26,522-lb LEM)

The load capability of the Little Joe II booster structure is obtained from General Dynamics/Convair Report GD/C 62-278A "Interim Structural Design and Loads Criteria for Test Launch Vehicle and Launcher - Little Joe II Booster" 25 Sept. 1962, and from Report GD/C 63-039 "Little Joe II Vehicle Stress Analysis". Significant General Dynamics booster design loads are tabulated in Table 6.4.4.

TABLE 6-6

BOOSTER ULTIMATE DESIGN LOADS AT STATION 0

Load	Conditions Cond. C28G (15°)	Cond. C18
Moment (in. lbs.)	13.5×10^6	2.0×10^6
Axial Load (lbs.)	160,000	610,000
Shear (lbs.)	57,000	--

The booster shell structure is of constant section and strength above station 227 and has the capability of a 27×10^6 inch lb. bending moment. The corrugated skin shell has the compression strength of 2000 lbs/in. ultimate. It is evident that the basic booster structure is well within the strength required for the LEM payload. The Little Joe II booster interface attachment ring was designed for a 300 lb/in. tensile load (cond. C28G (15°)) with a 10% margin of safety on the (32) MS 20005 attachment bolts for an allowable strength of 330 lbs/in. tension. The GD/C Report 63-039 indicates that provisions have been made to allow doubling the number of attachment bolts to obtain an allowable

6.2.4.6 Payload Booster Interface Joint.- continued

strength of 660 lbs./in., which would be adequate to sustain the 350 lb./in. LEM load shown in Table 6-5. The same joint has been designed for the -1350 lb./in. compression load with a +.32 margin on the attachment rivets. This indicates an allowable strength of -1780 lbs./in., which is also well above the applied compression load of 1492 lbs./in. shown in Table 6-5.

6.2.4.7 Fin Loads.-

The LEM Little Joe II booster flight in the 99% wind profile results in an ultimate fin load of 54,000 lbs. The General Dynamics Report GD/C 62-278A indicates that the fin and support structure has been designed for a 51,020 lb. ultimate applied load. The stress report GD/C 63-039 indicates a minimum margin of safety of +.10 or a strength for a 56,200 lb. fin load.

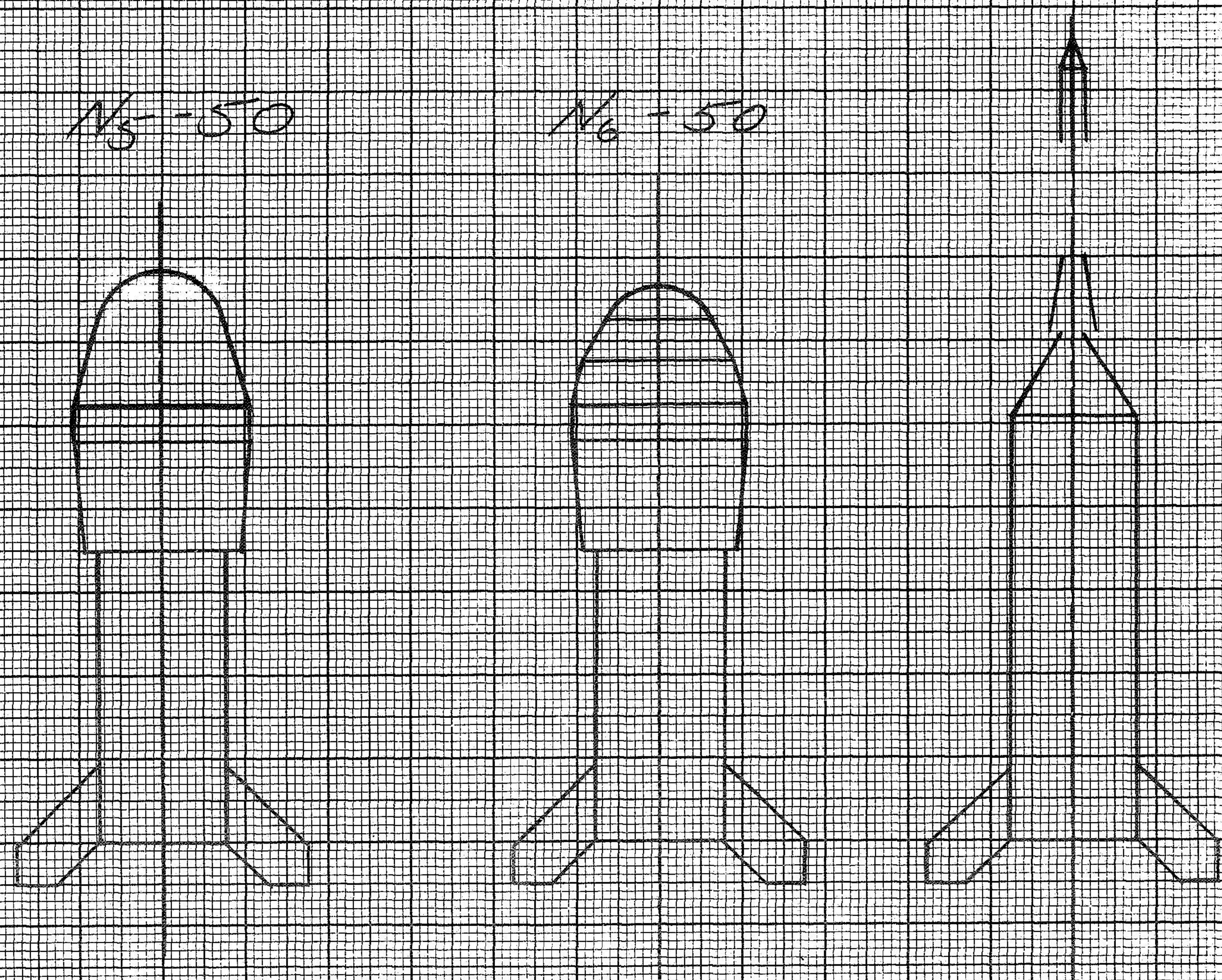


FIG. 6-1

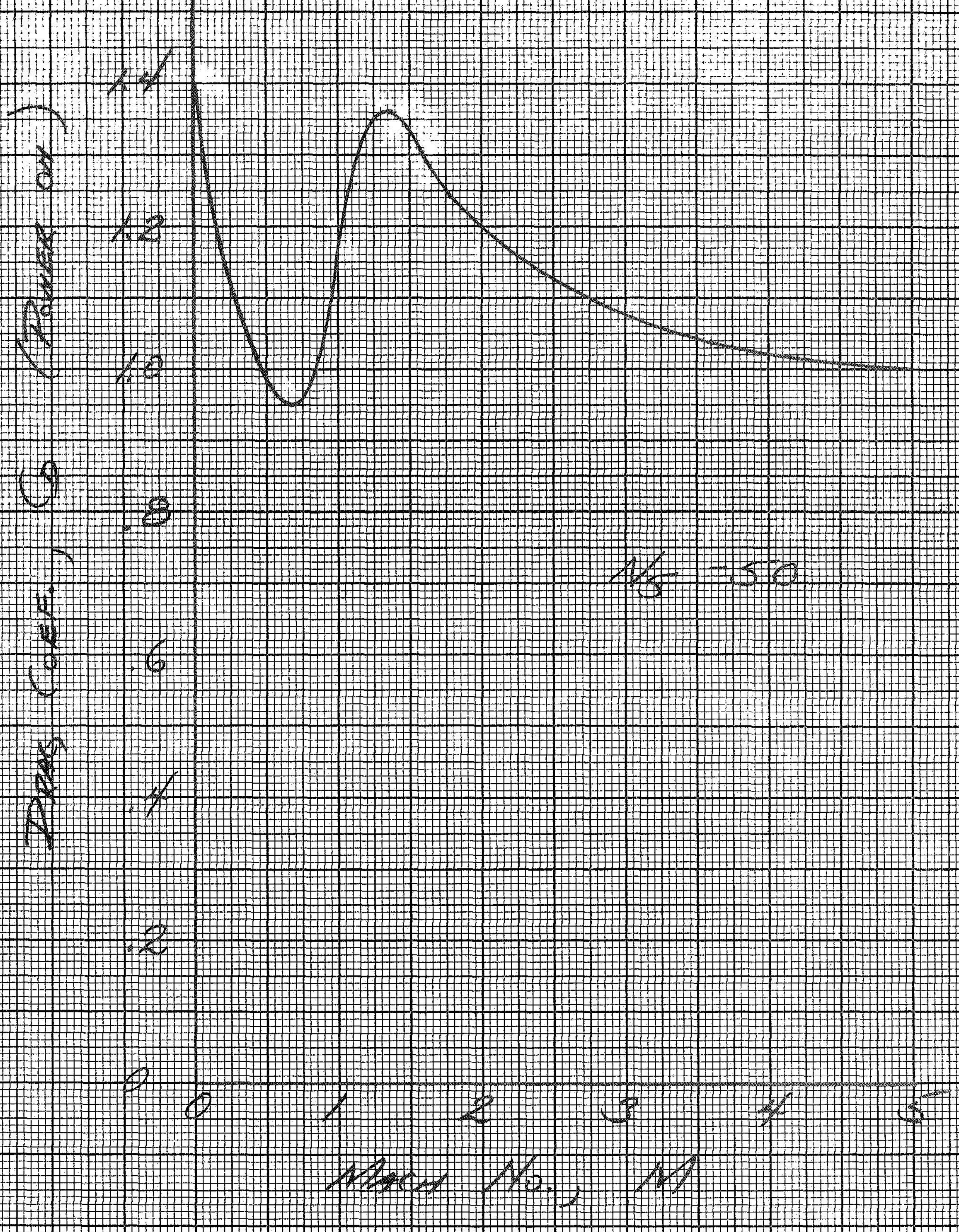


FIG. 6-2

GRUMMAN
ESTIMATED AERO CHARACTERISTICS
NS - 50

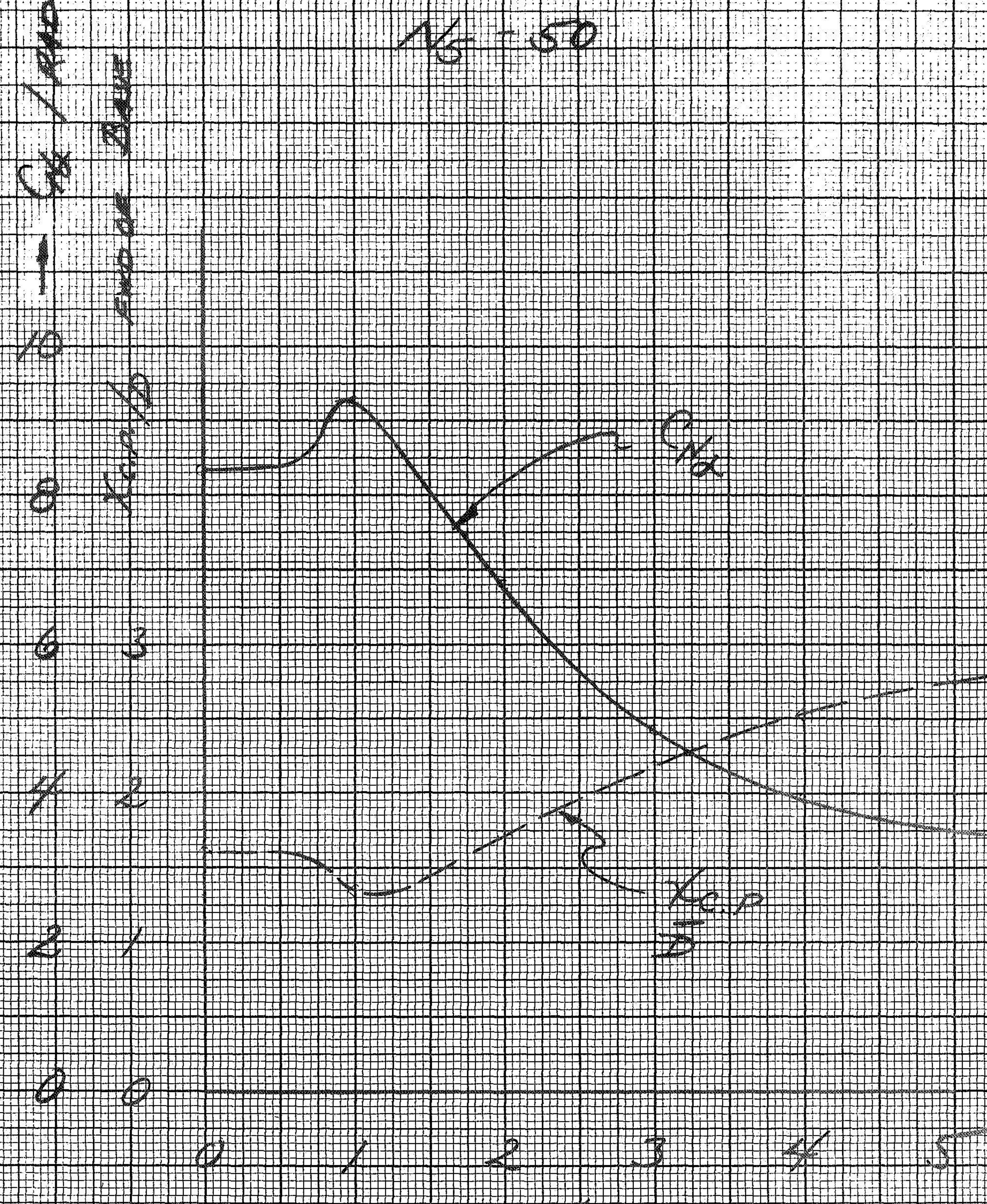


FIG. 6-B

Grumman

FLEVON EFFECTIVENESS

M5-50

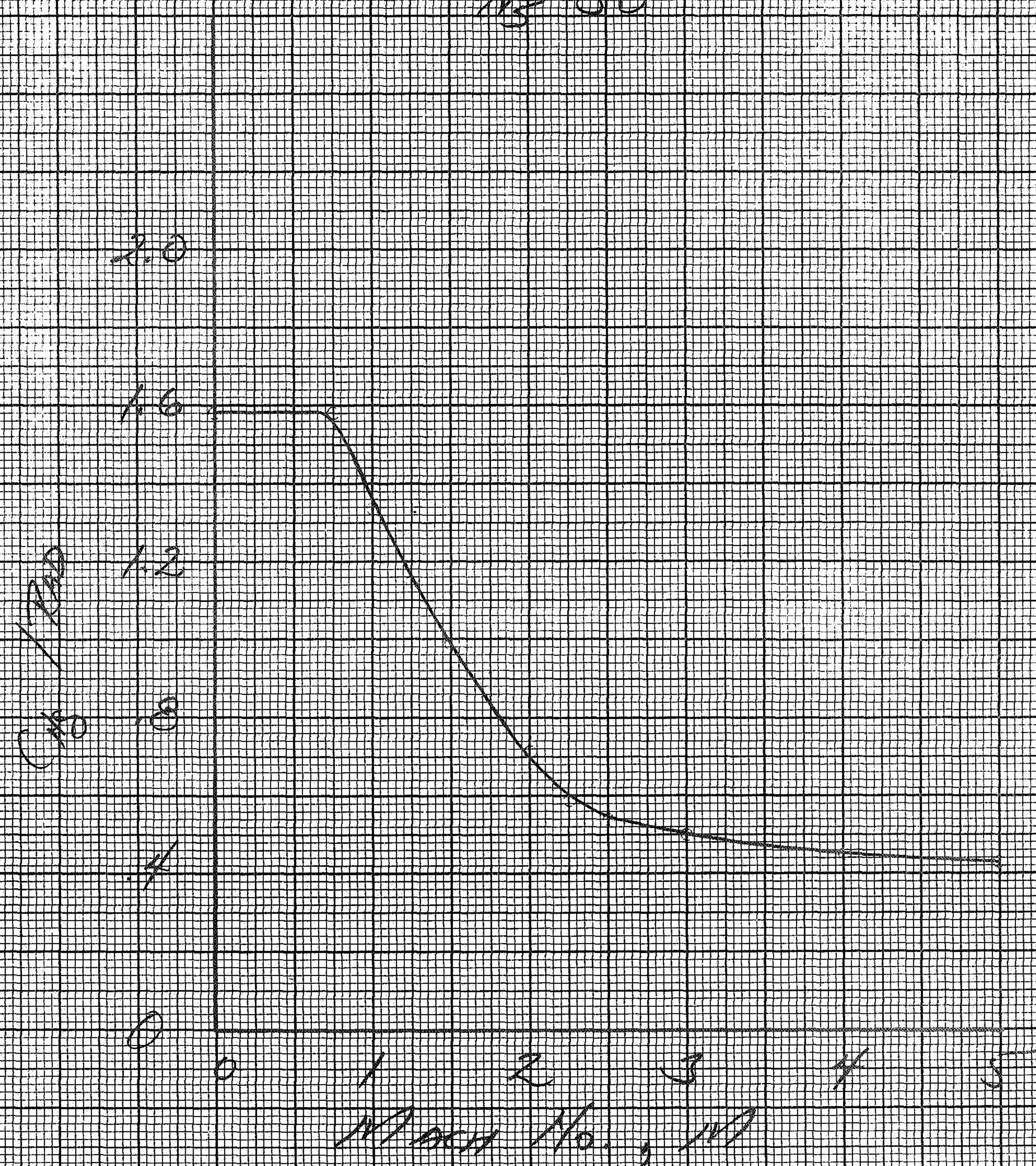


FIG. 6-1

$$C_{m_q} = \partial C_m / \partial \left(\frac{q D}{2V} \right)$$

q = Pitch Rate

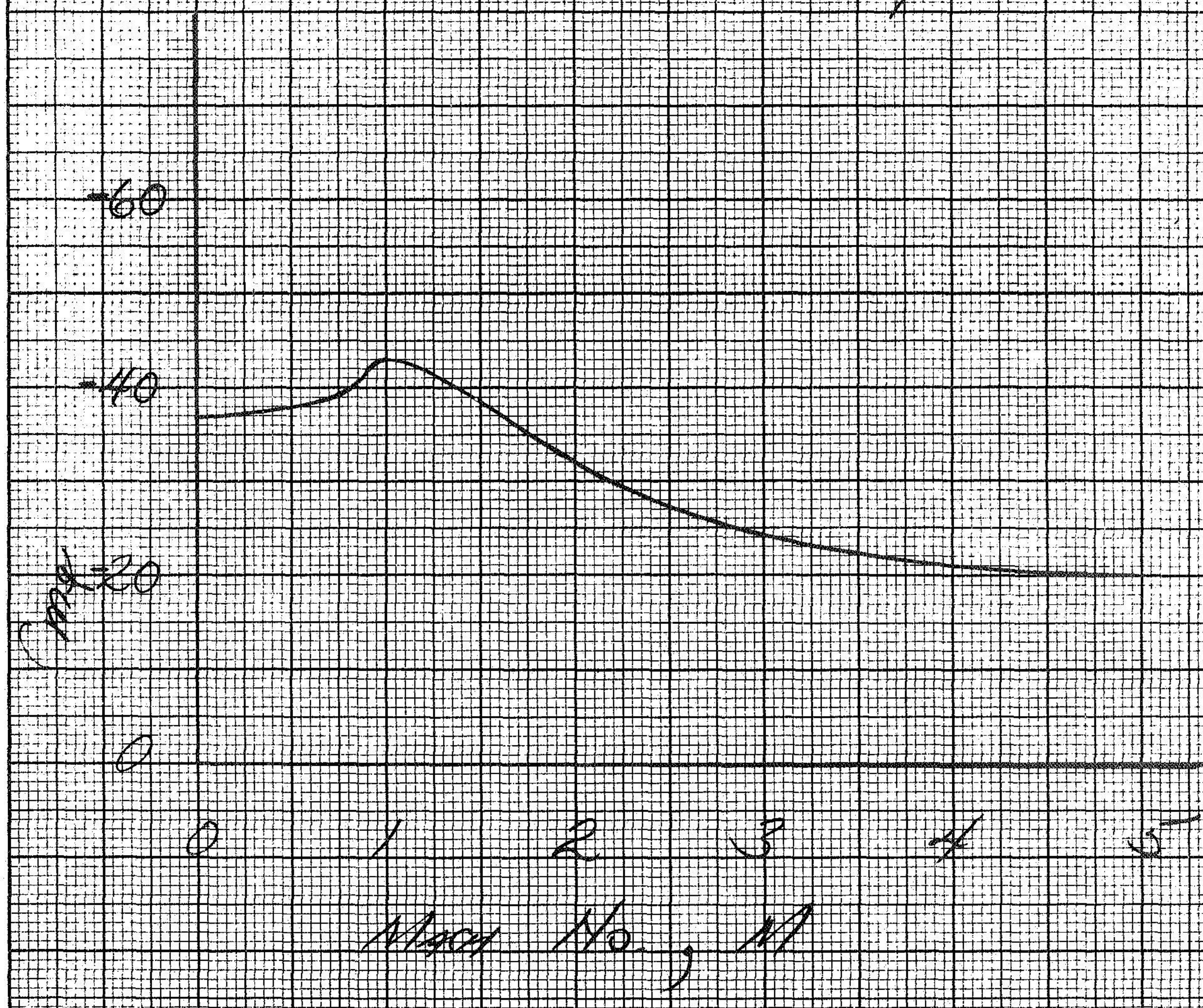


FIG 6-5

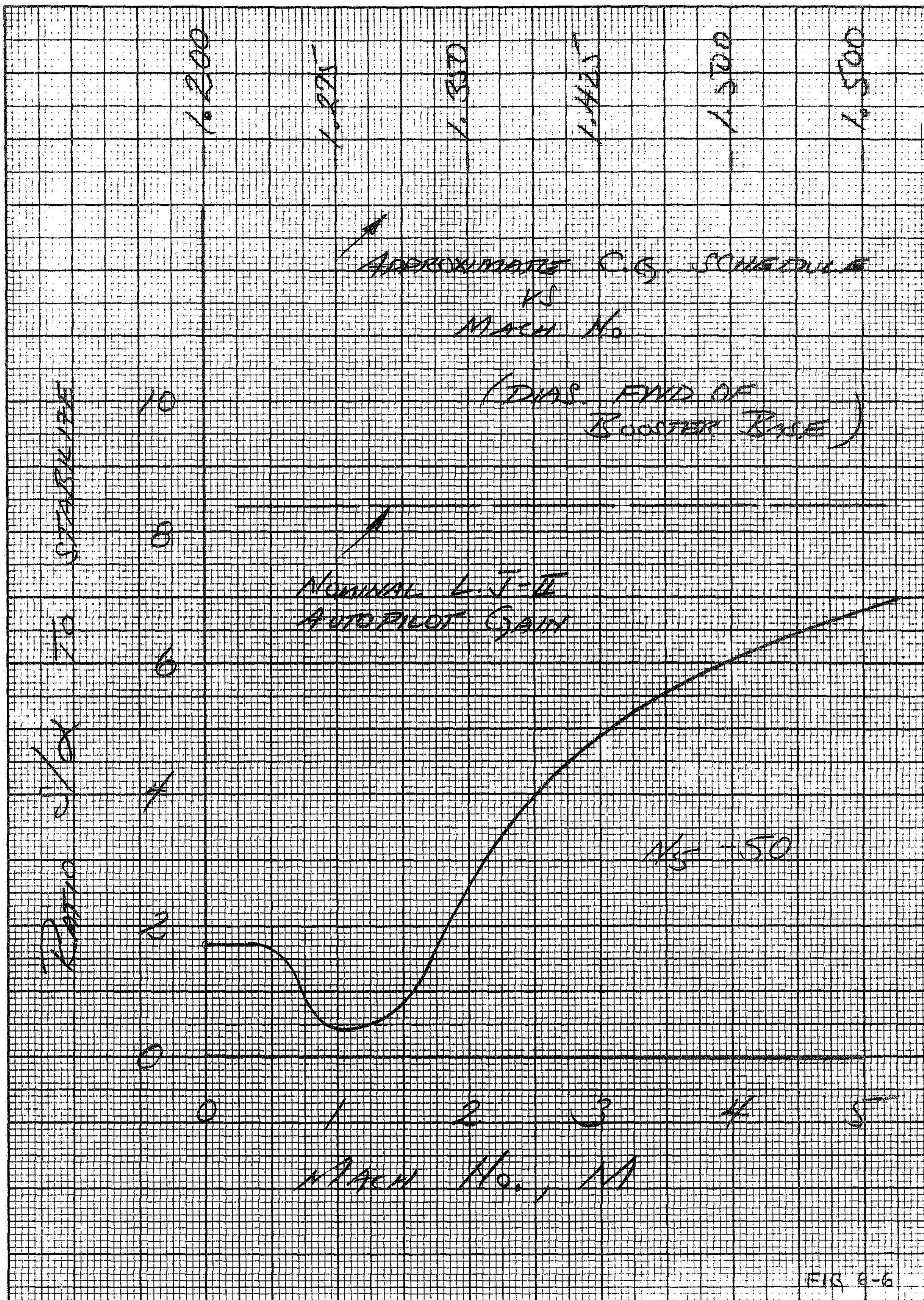

1.200
1.271
1.330
1.425
1.500
1.500
APPROXIMATE C.G. SCHEDULE
VS
MACH No.
(DIAS. FWD OF
BOOSTER BASE)
NOMINAL L.J-II
AUTOPILOT GAIN
RATIO δ/α TO STABILIZE
10
8
6
4
2
0
Kδ = 50
0
1
2
3
4
5
MACH No., M

FIG 6-6

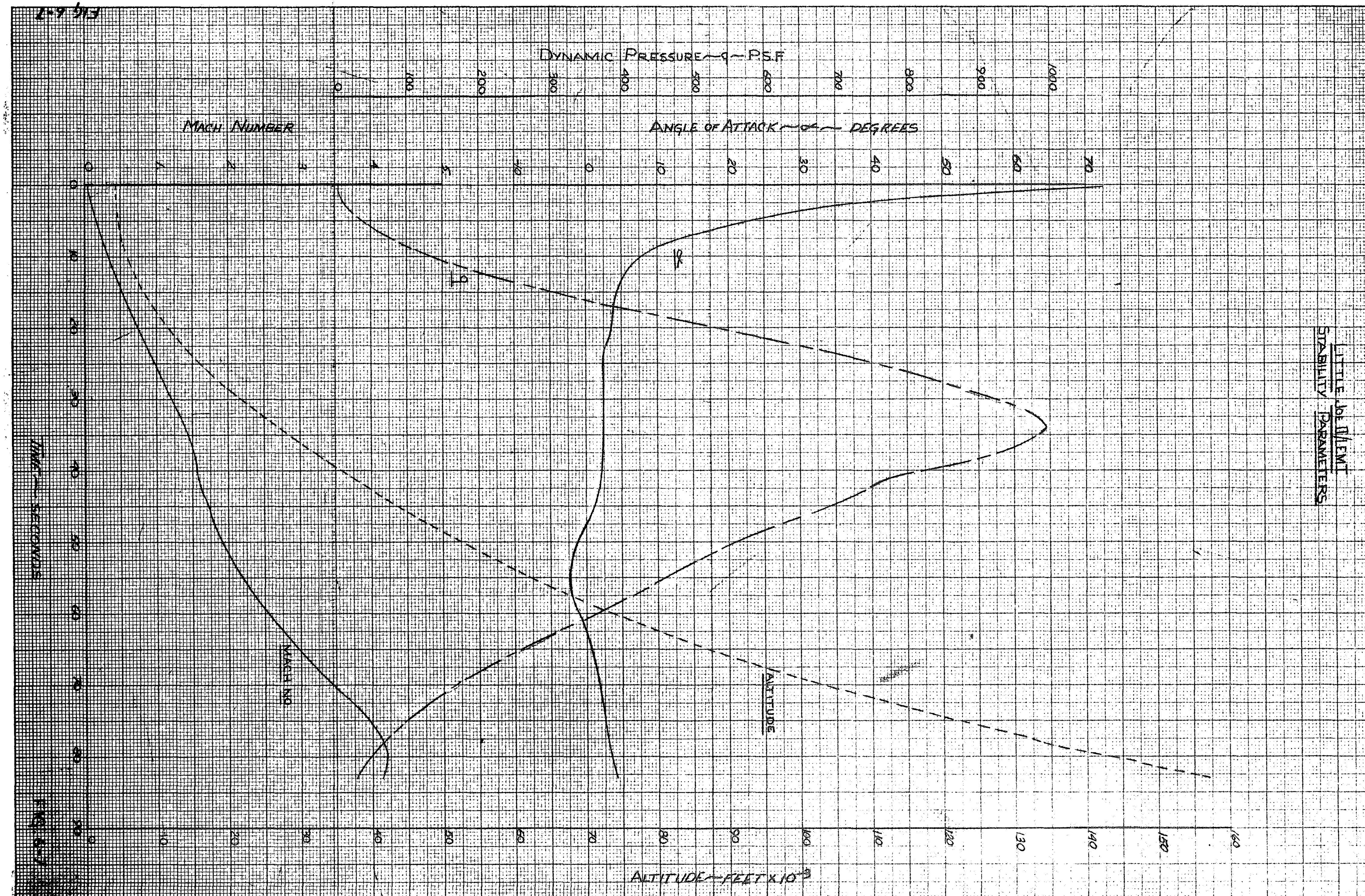
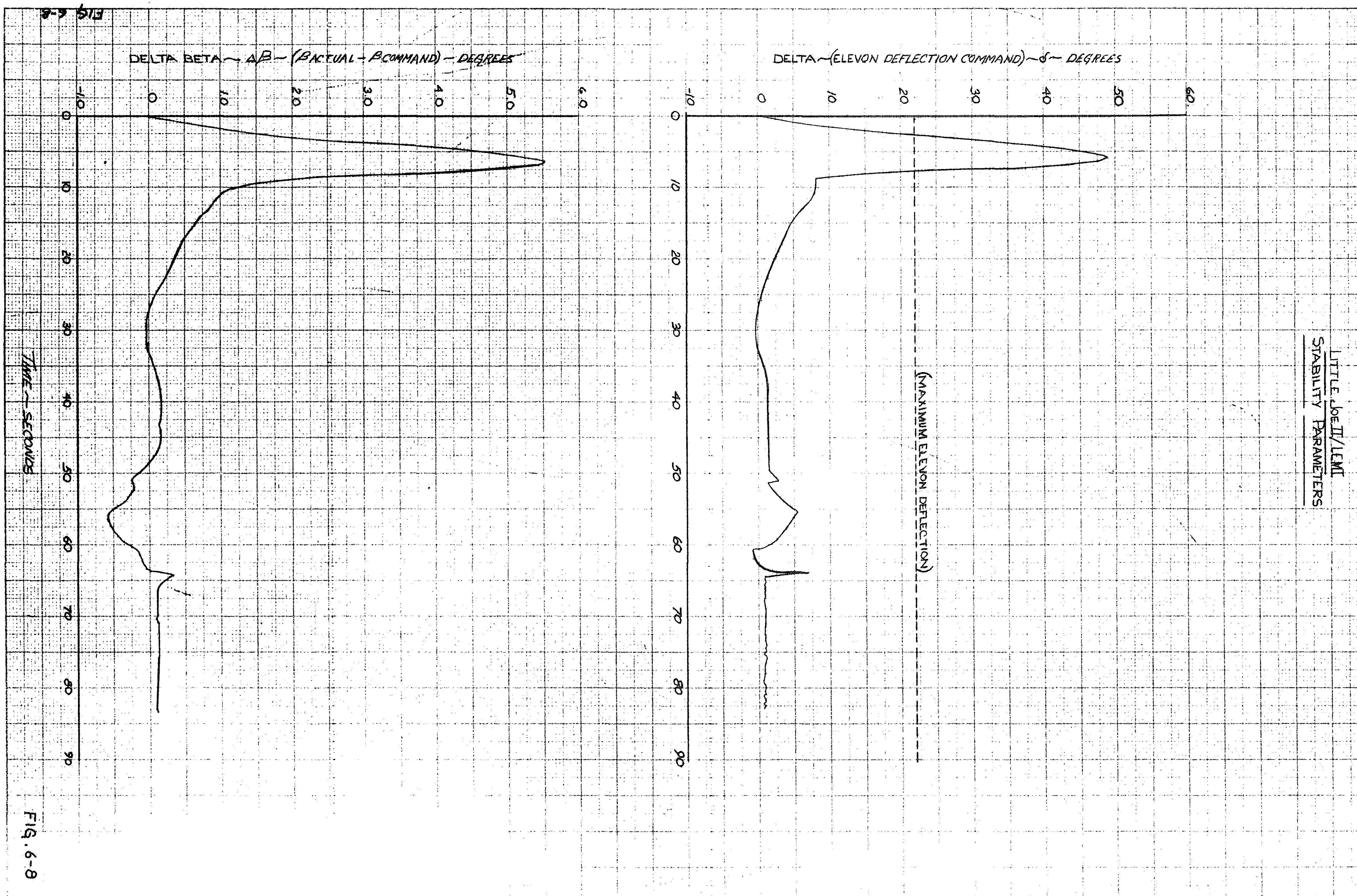

DYNAMIC PRESSURE ~ q ~ P.S.F
0
100
200
300
400
500
600
700
800
900
1000
MACH NUMBER
0
1
2
3
4
5
ANGLE OF ATTACK ~ α ~ DEGREES
-10
0
10
20
30
40
50
60
70
q
α
MACH NO
ALTITUDE
TIME ~ SECONDS
0
10
20
30
40
50
60
70
80
90
ALTITUDE ~ FEET X 10^3
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160
LITTLE JOE II/LEM
STABILITY PARAMETERS

FIG 6-7



LITTLE JOE II/LEM
STABILITY PARAMETERS

FIG. 6-8

LITTLE JOE II/LEM I
STABILITY PARAMETERS

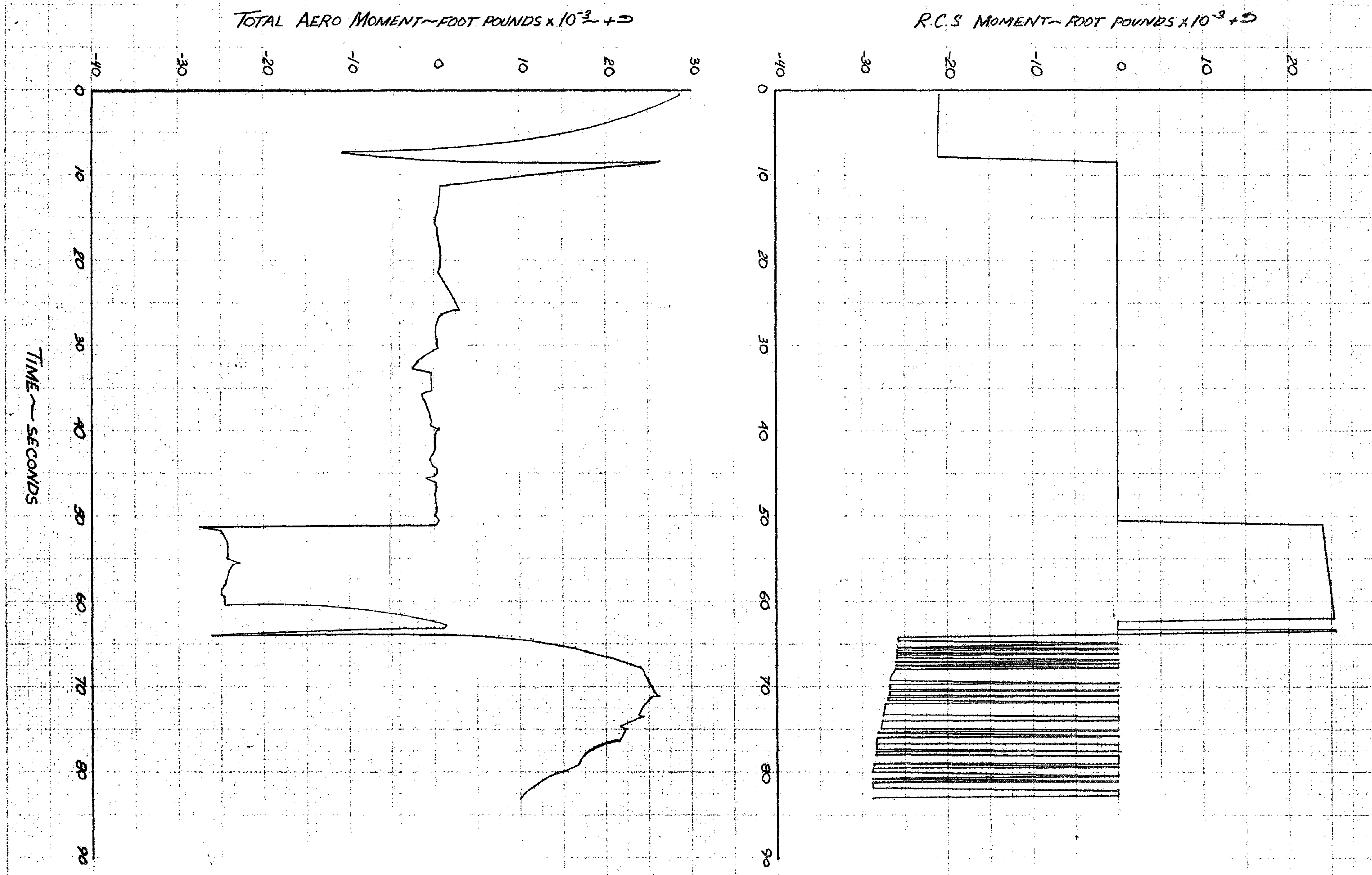
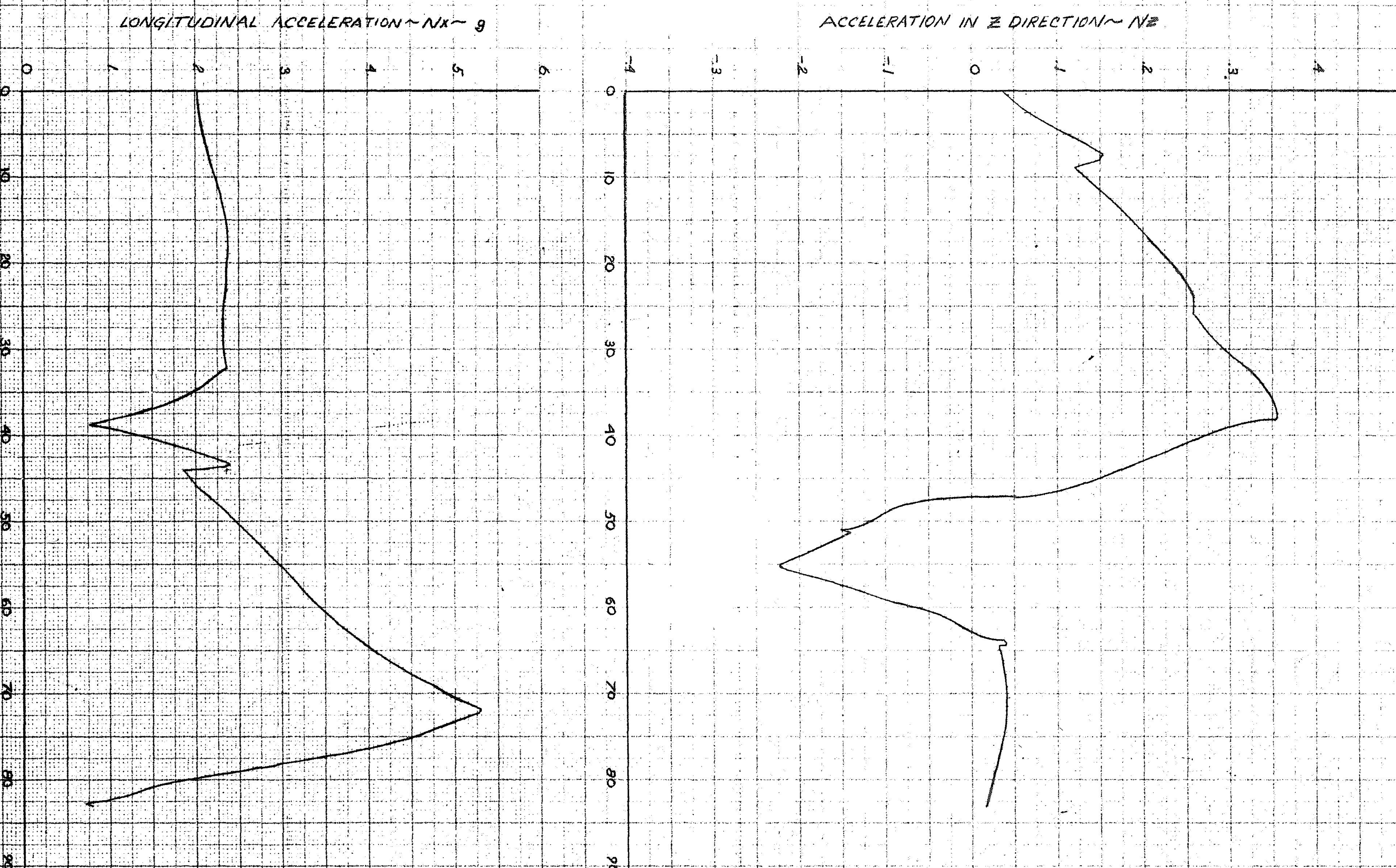


FIG. 6-9

FIG 6-10



LITTLE JOE II/LEM I
STABILITY PARAMETERS

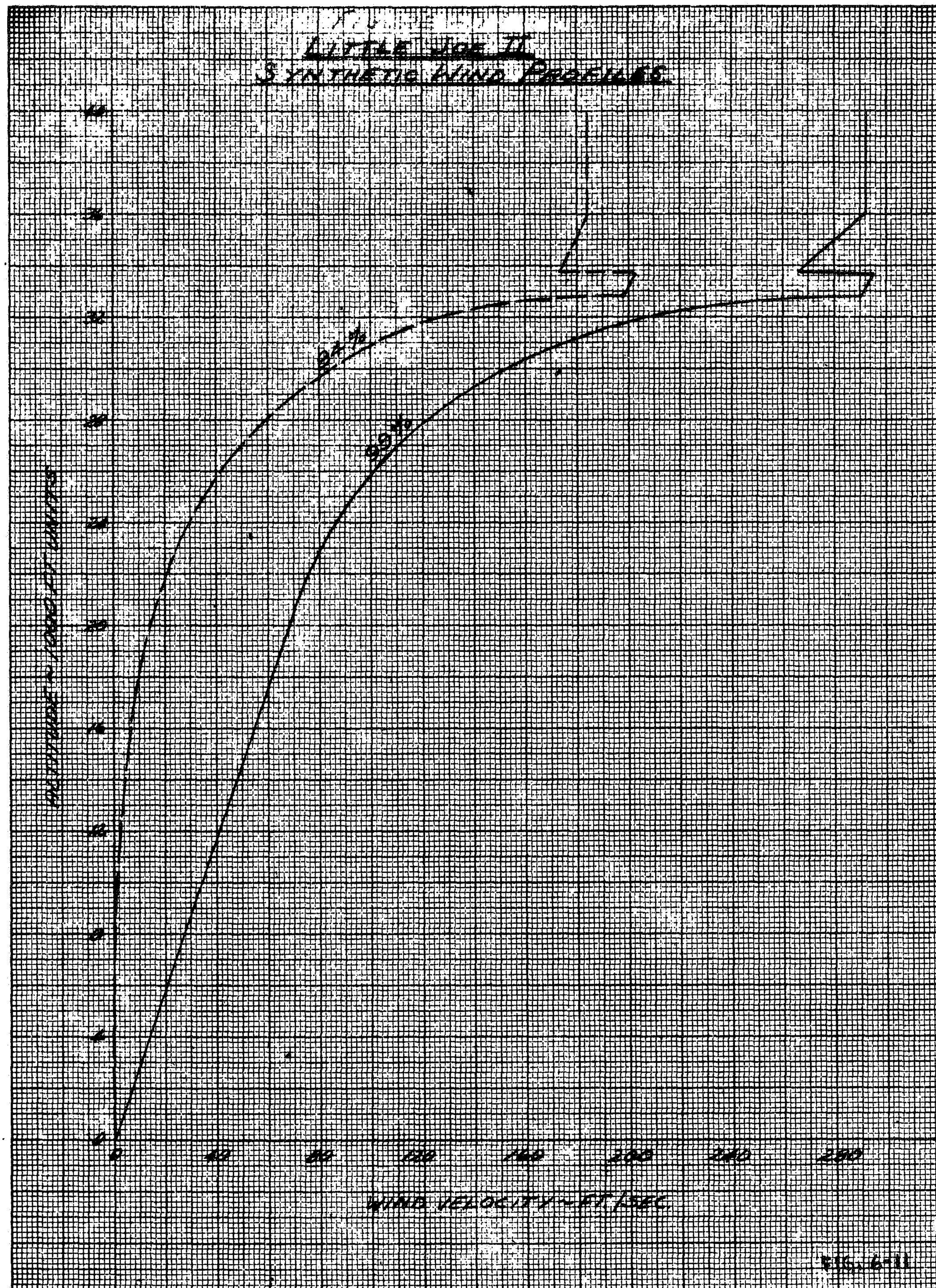


FIG. 6-11

7/31/63

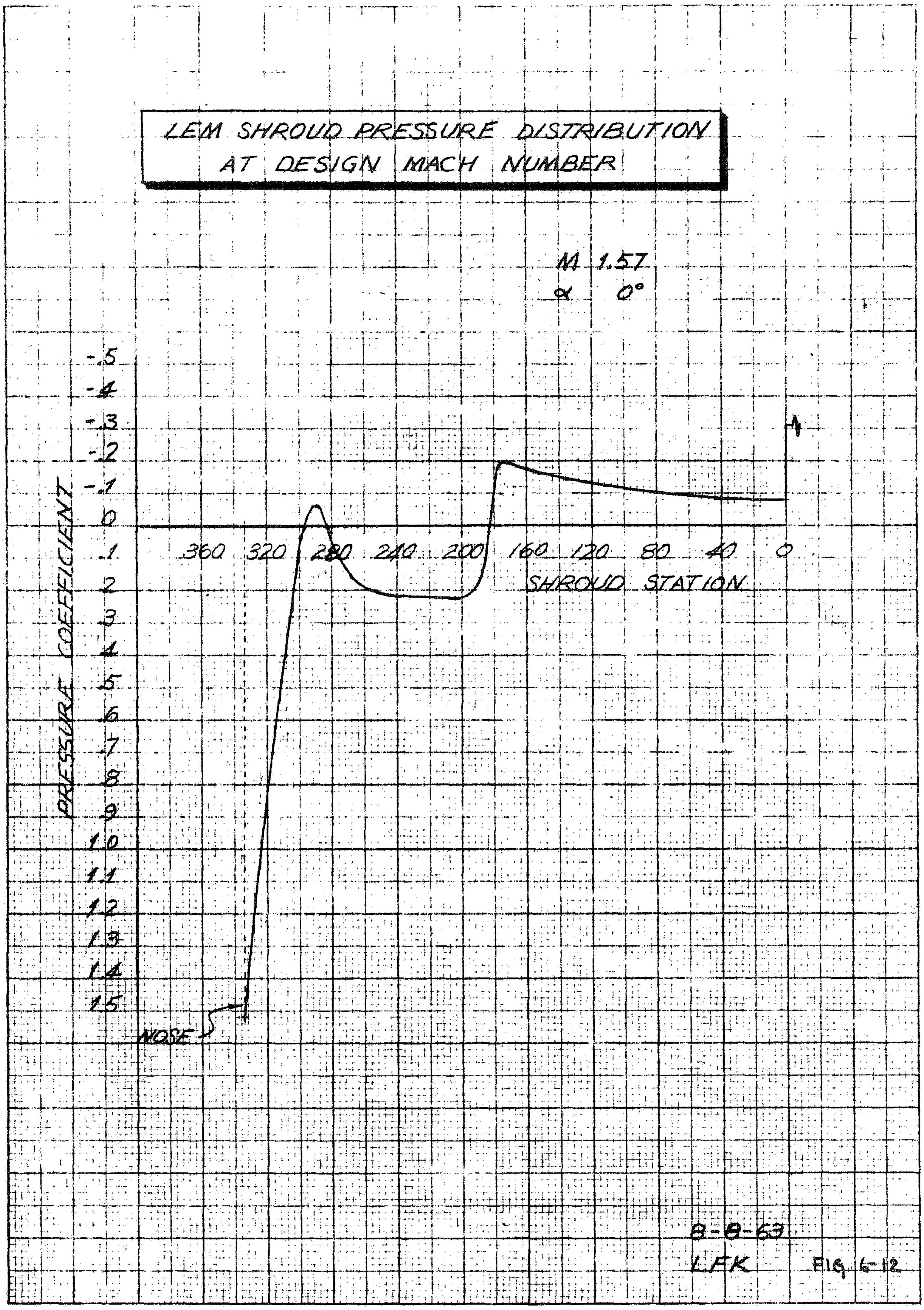

LEM SHROUD PRESSURE DISTRIBUTION
AT DESIGN MACH NUMBER
M 1.57
α 0°
PRESSURE COEFFICIENT
-.5
-.4
-.3
-.2
-.1
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
360 320 280 240 200 160 120 80 40 0
SHROUD STATION
NOSE
8-8-63
LFK

FIG. 6-12

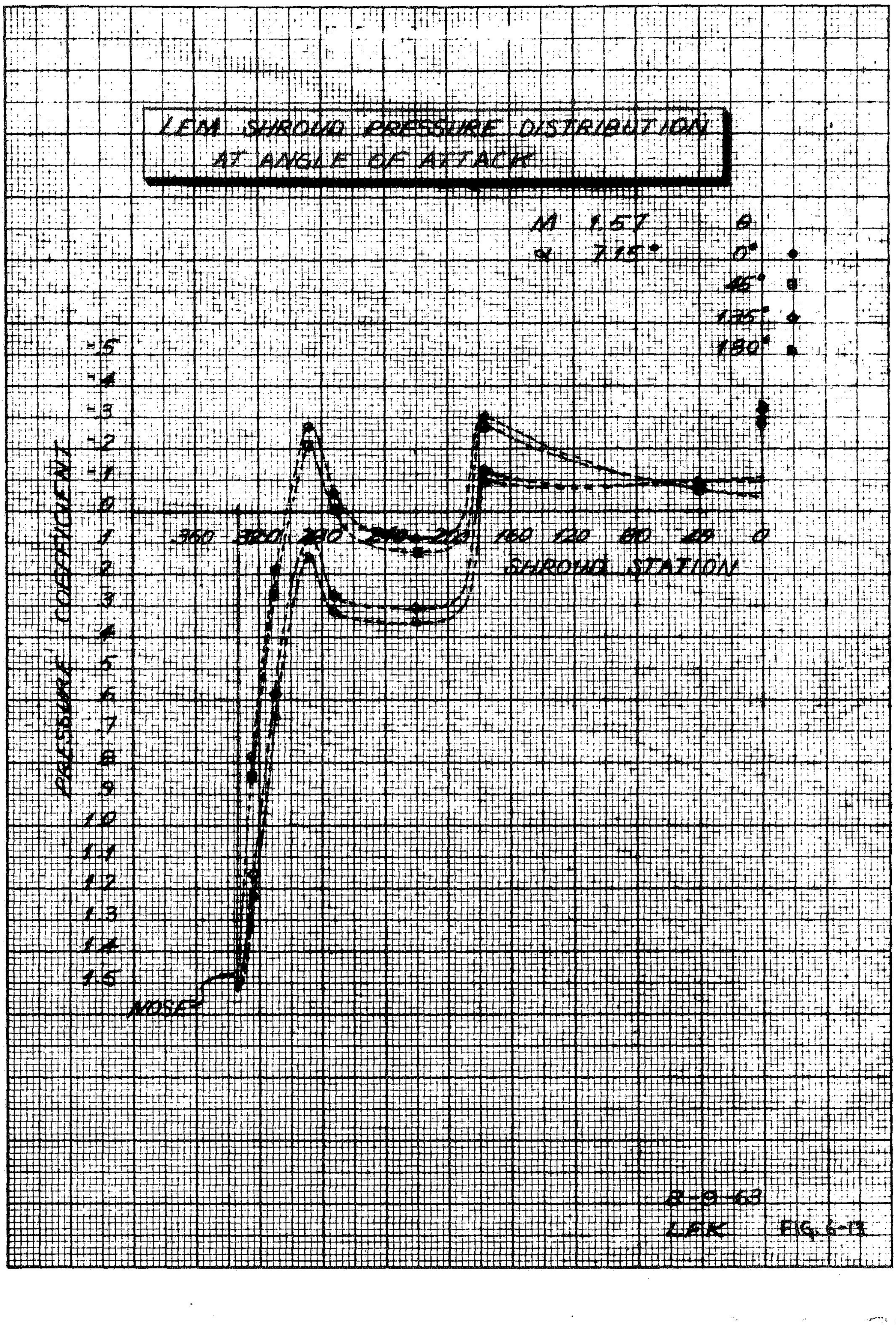

LEM SHROUD PRESSURE DISTRIBUTION
AT ANGLE OF ATTACK
M 1.57
α 7.15°
θ
0°
45°
135°
180°
PRESSURE COEFFICIENT
.5
.4
.3
.2
.1
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
360
320
280
240
200
160
120
80
40
0
SHROUD STATION
NOSE
8-9-63
LEK

FIG. 6-13

FIGURE 6.1.5

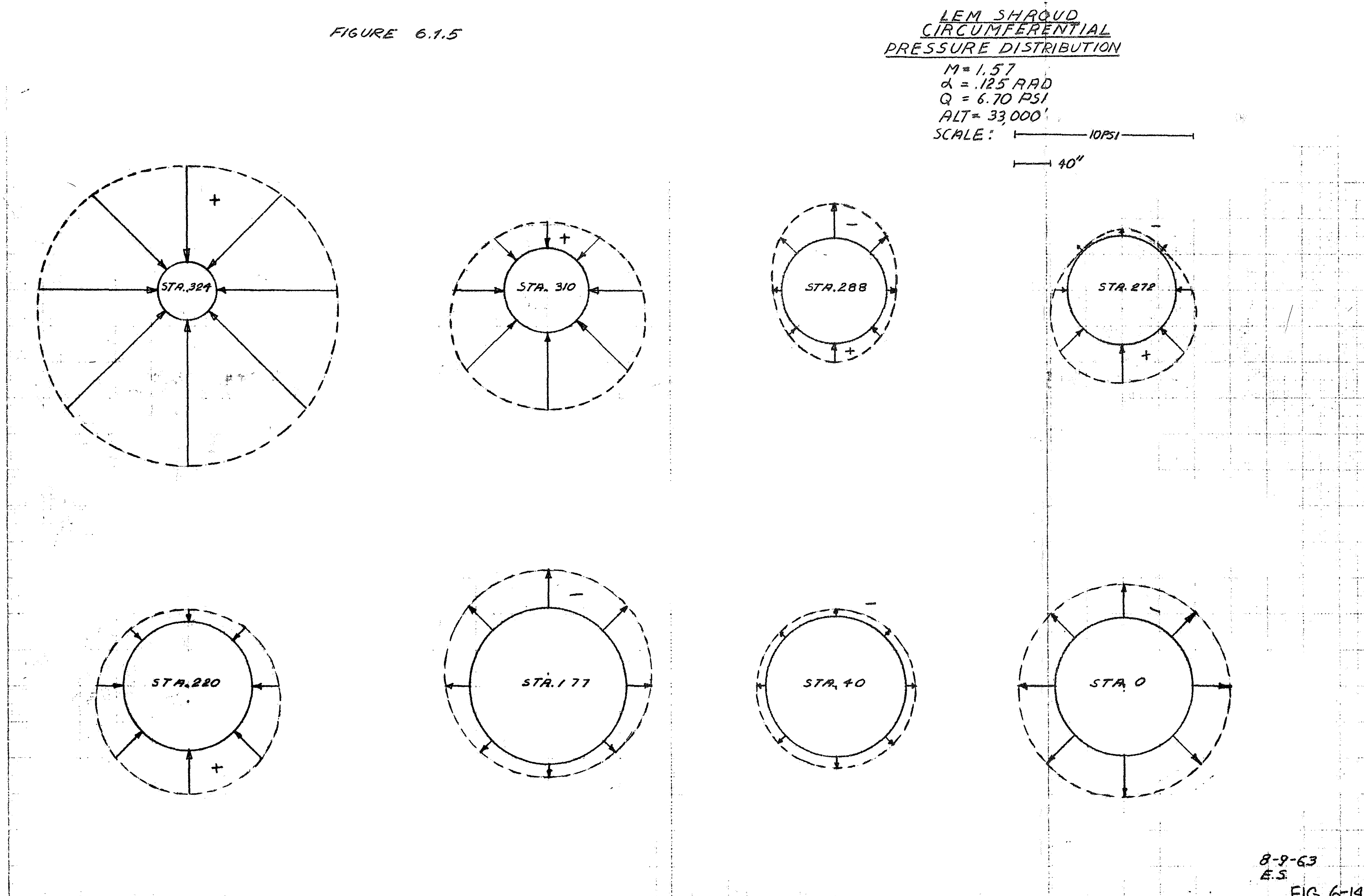


FIG. 6-14

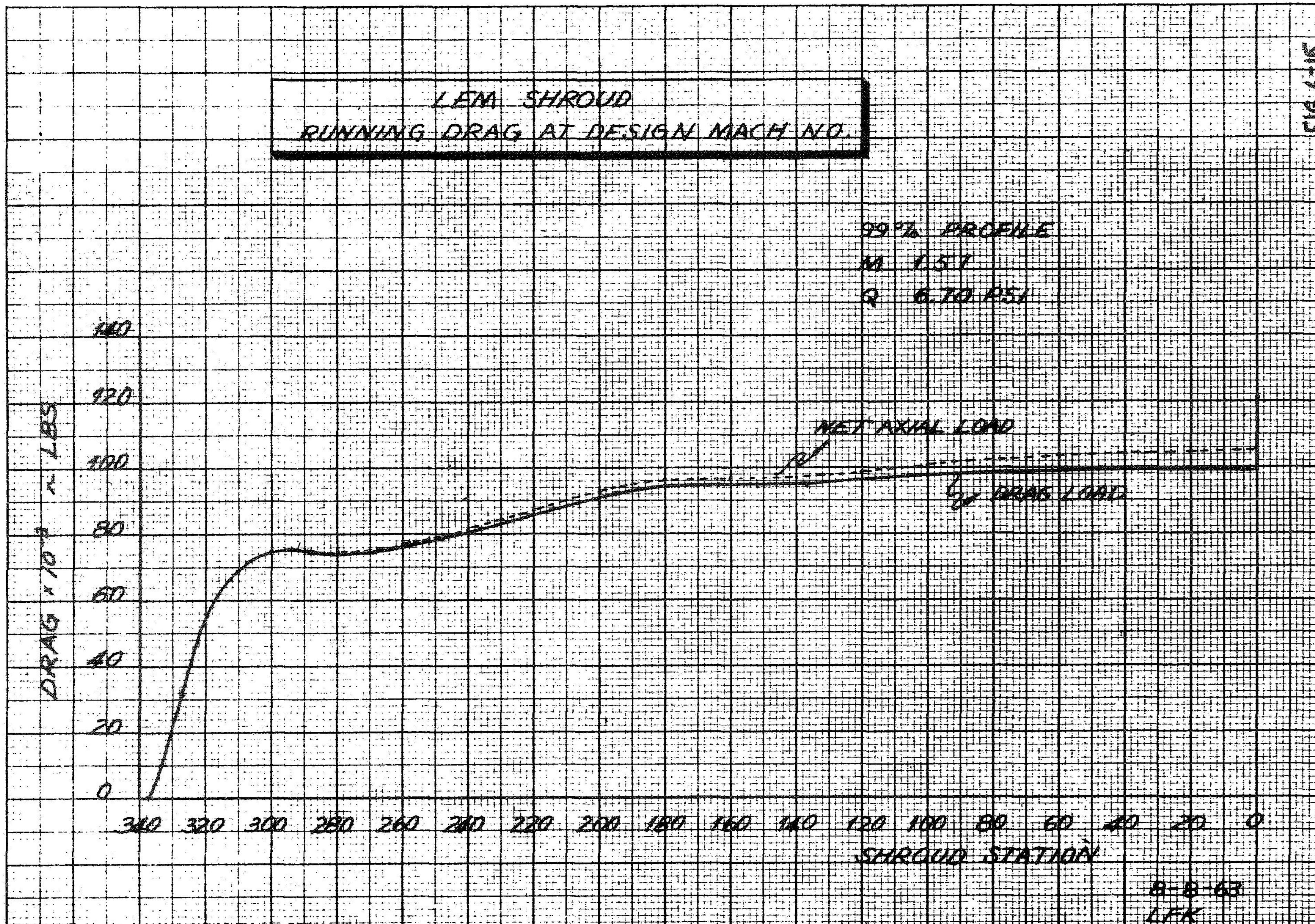

LEM SHROUD
RUNNING DRAG AT DESIGN MACH NO.
99% PROFILE
M 1.57
Q 6.70 PSI
DRAG × 10⁻³ ~ LBS
0
20
40
60
80
100
120
140
NET AXIAL LOAD
DRAG LOAD
340 320 300 280 260 240 220 200 180 160 140 120 100 80 60 40 20 0
SHROUD STATION
8-8-63
LFK

FIG 6-15

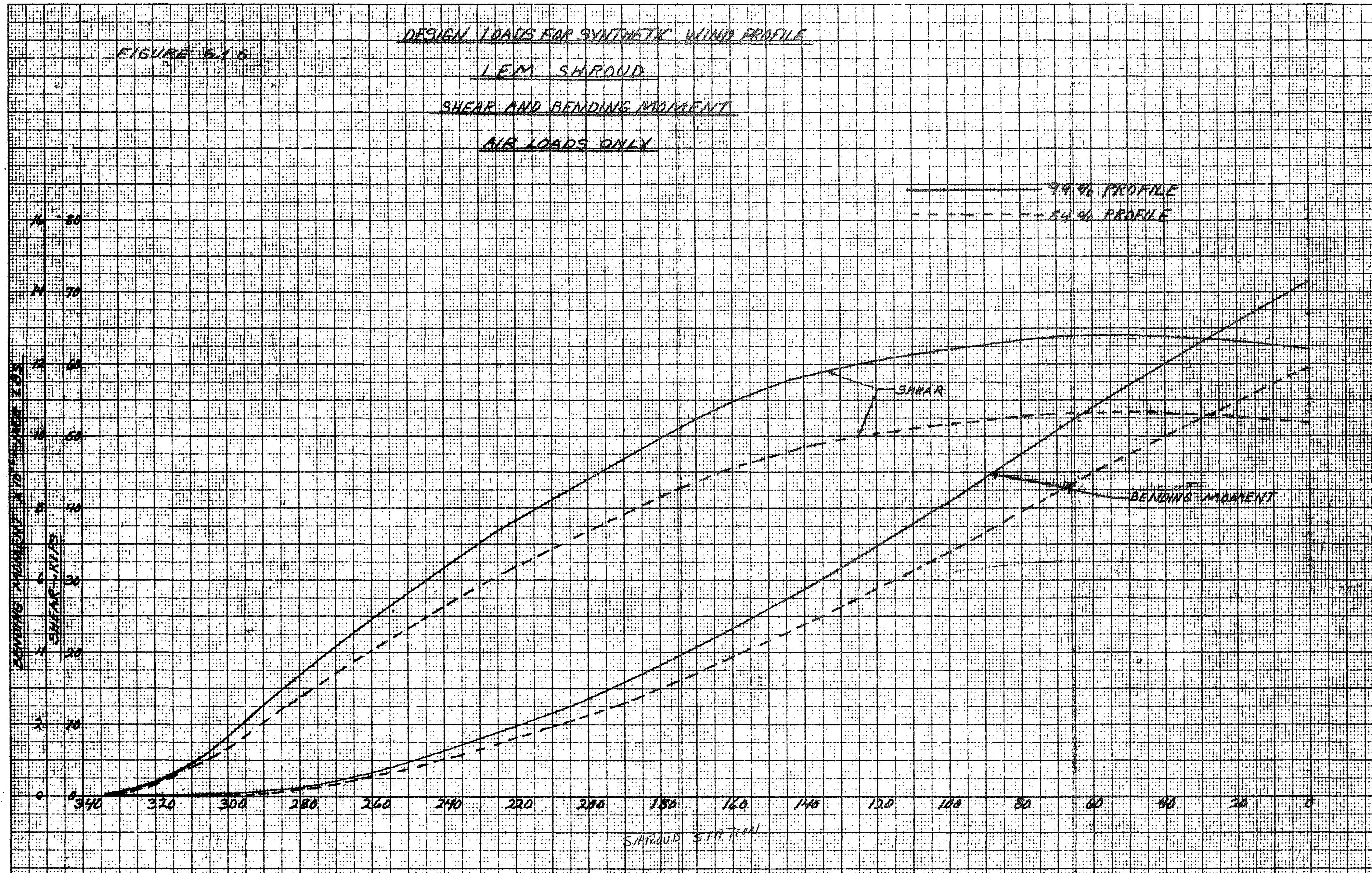


FIGURE 6.16
DESIGN LOADS FOR SYNTHETIC WIND PROFILE
LEM SHROUD
SHEAR AND BENDING MOMENT
AIR LOADS ONLY
99% PROFILE
84% PROFILE
SHEAR
BENDING MOMENT
SHEAR - KIPS
SHROUD STATION
340
320
300
280
260
240
220
200
180
160
140
120
100
80
60
40
20
0
8-6-63
FIG. 6-16

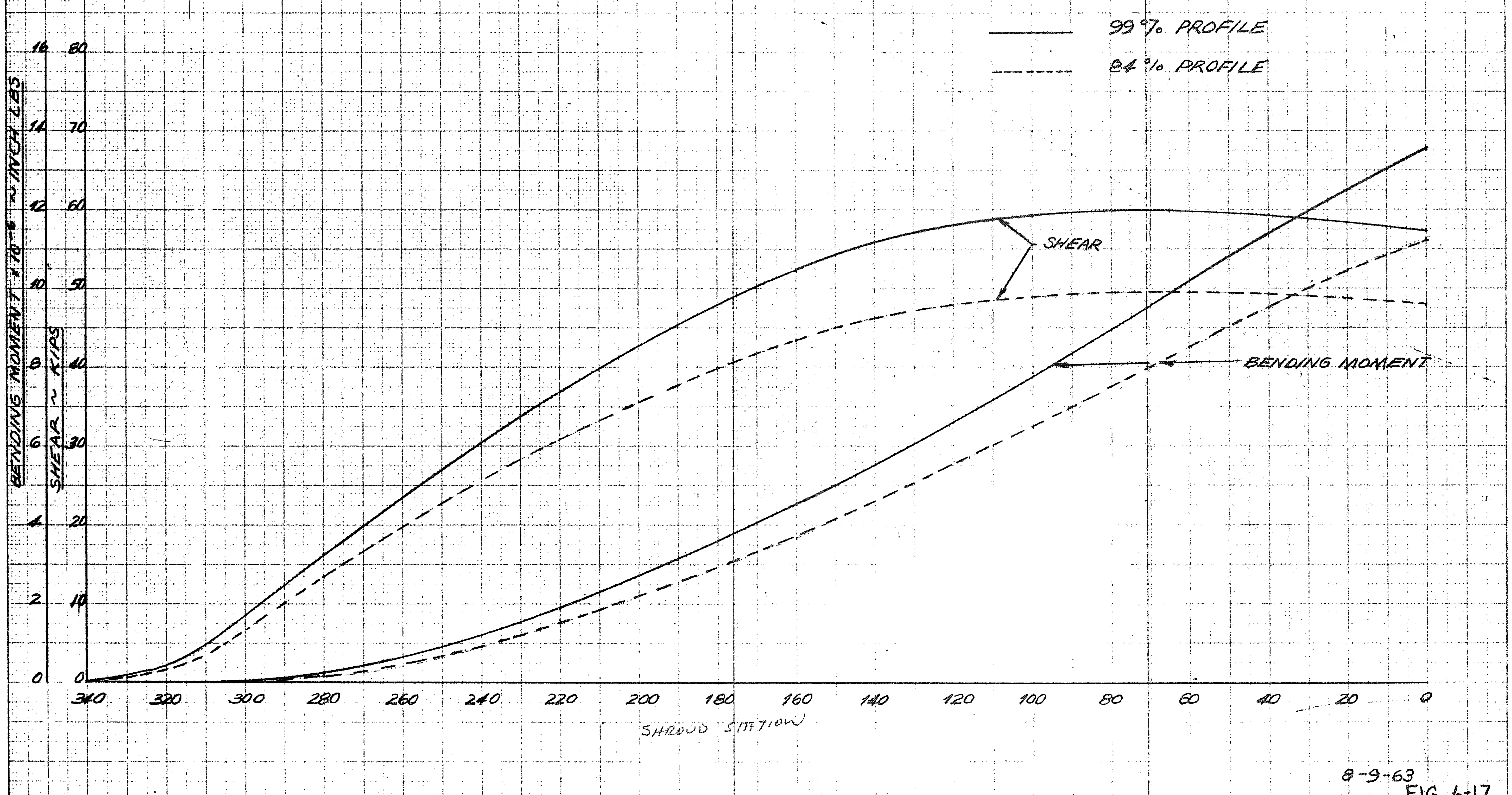
FIGURE 6.1.7

DESIGN LOADS FOR SYNTHETIC WIND PROFILE

LEM SHROUD

SHEAR AND BENDING MOMENT

NET LOADS



8-9-63

FIG 6-17

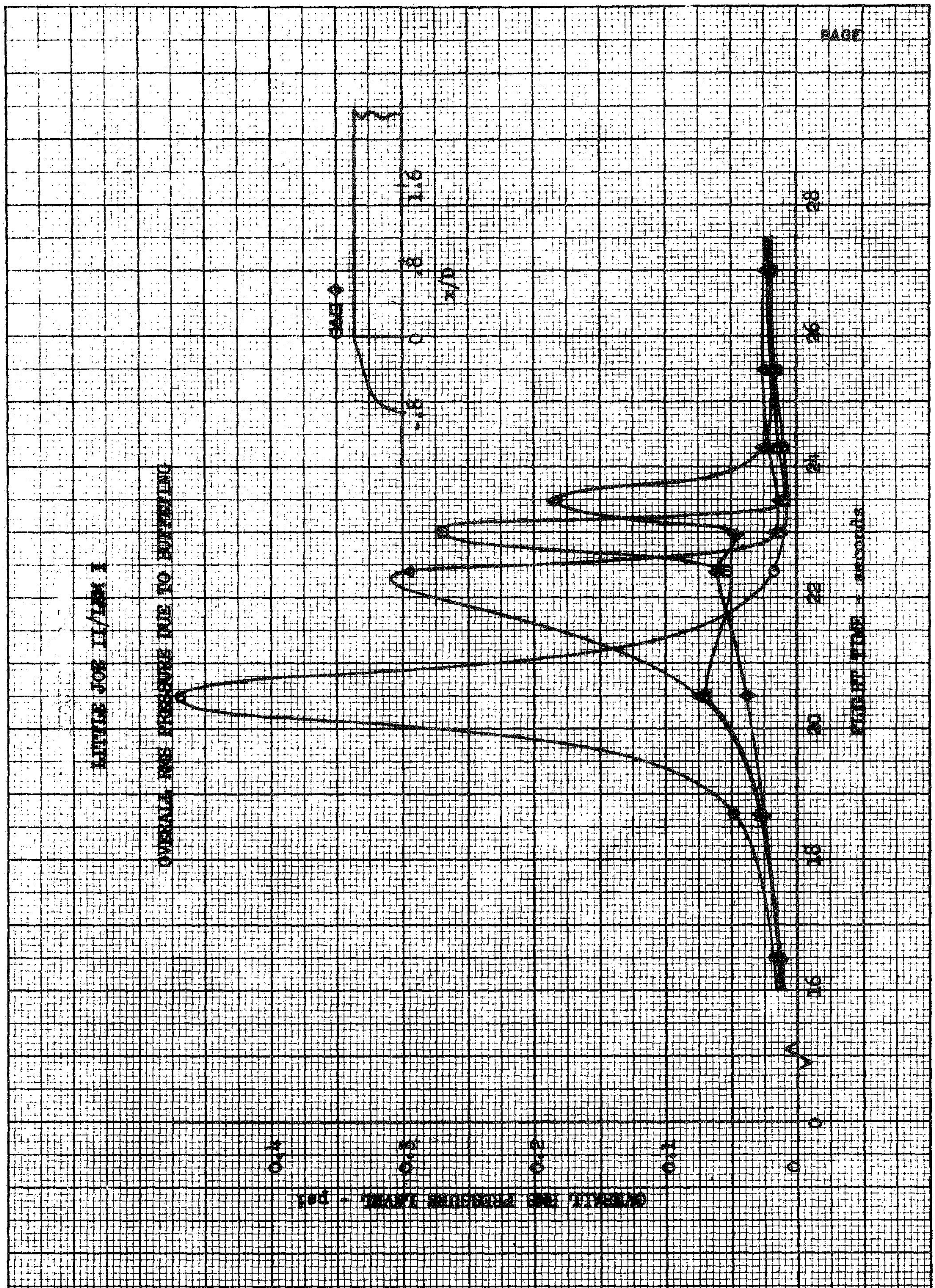


FIG. 6-18

REPORT NO.

LITTLE JOE II/LEM I

MAXIMUM RMS STRESS DUE TO BUFFETING

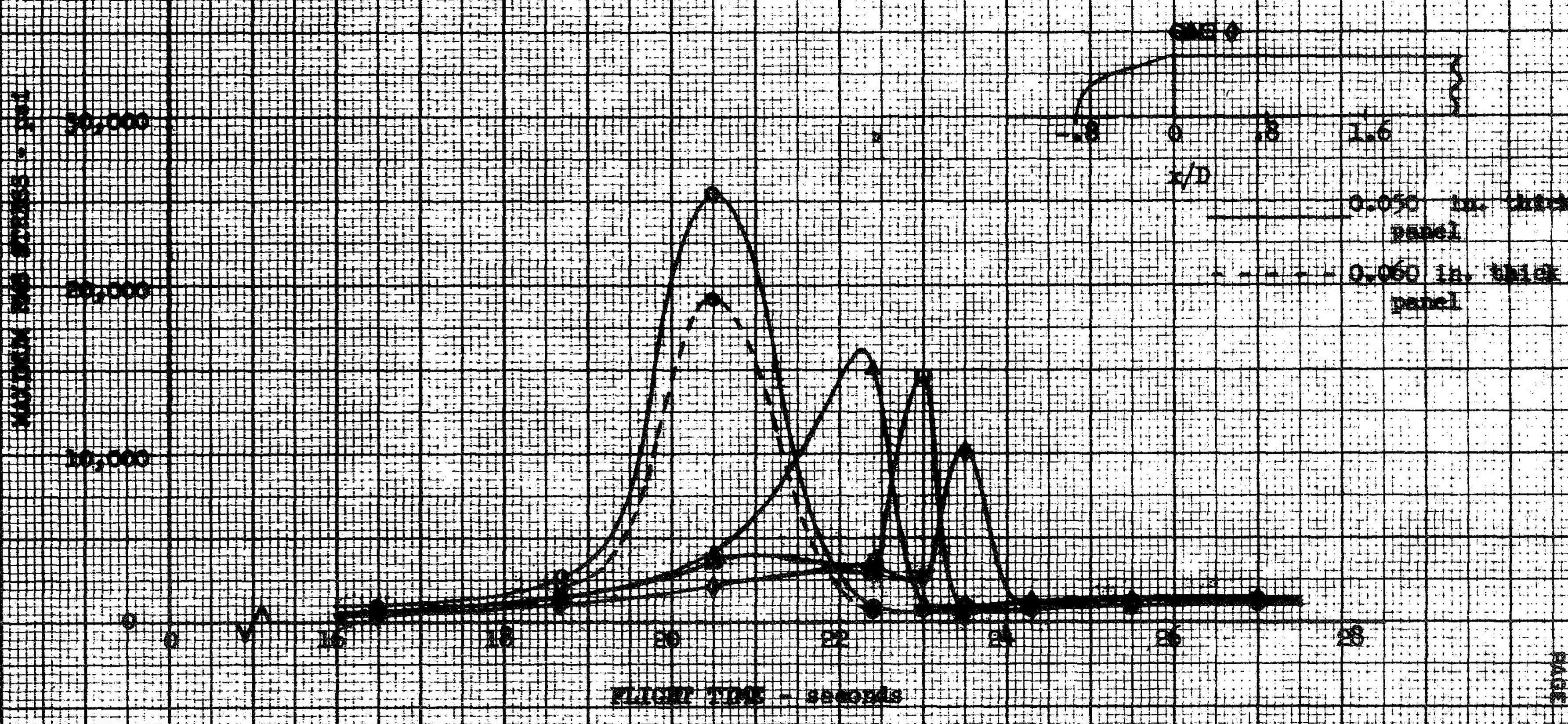


FIG. 6-19

LITTLE JOE II LEM I
NS-50 SHROUD CONFIG

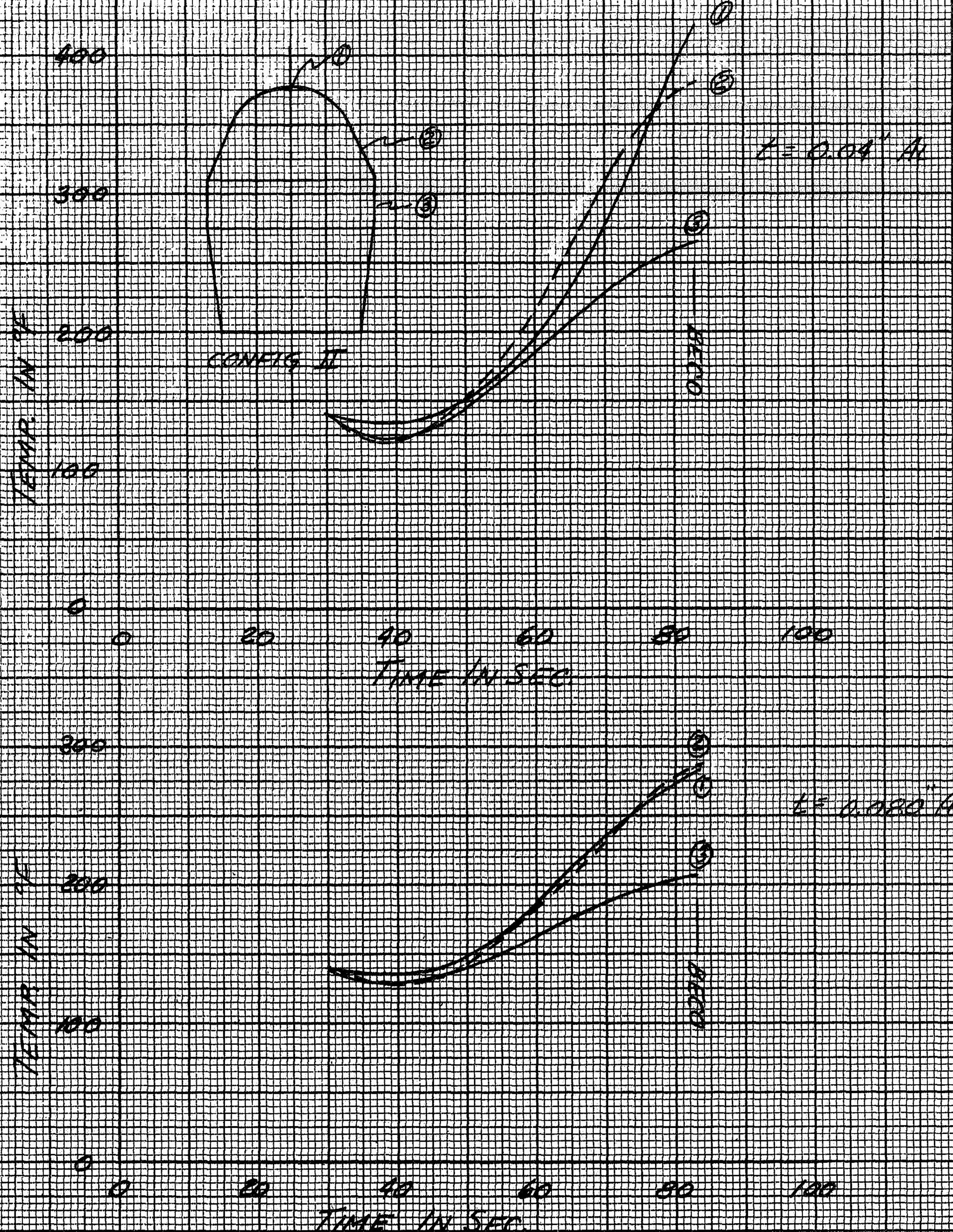


FIG 6-20

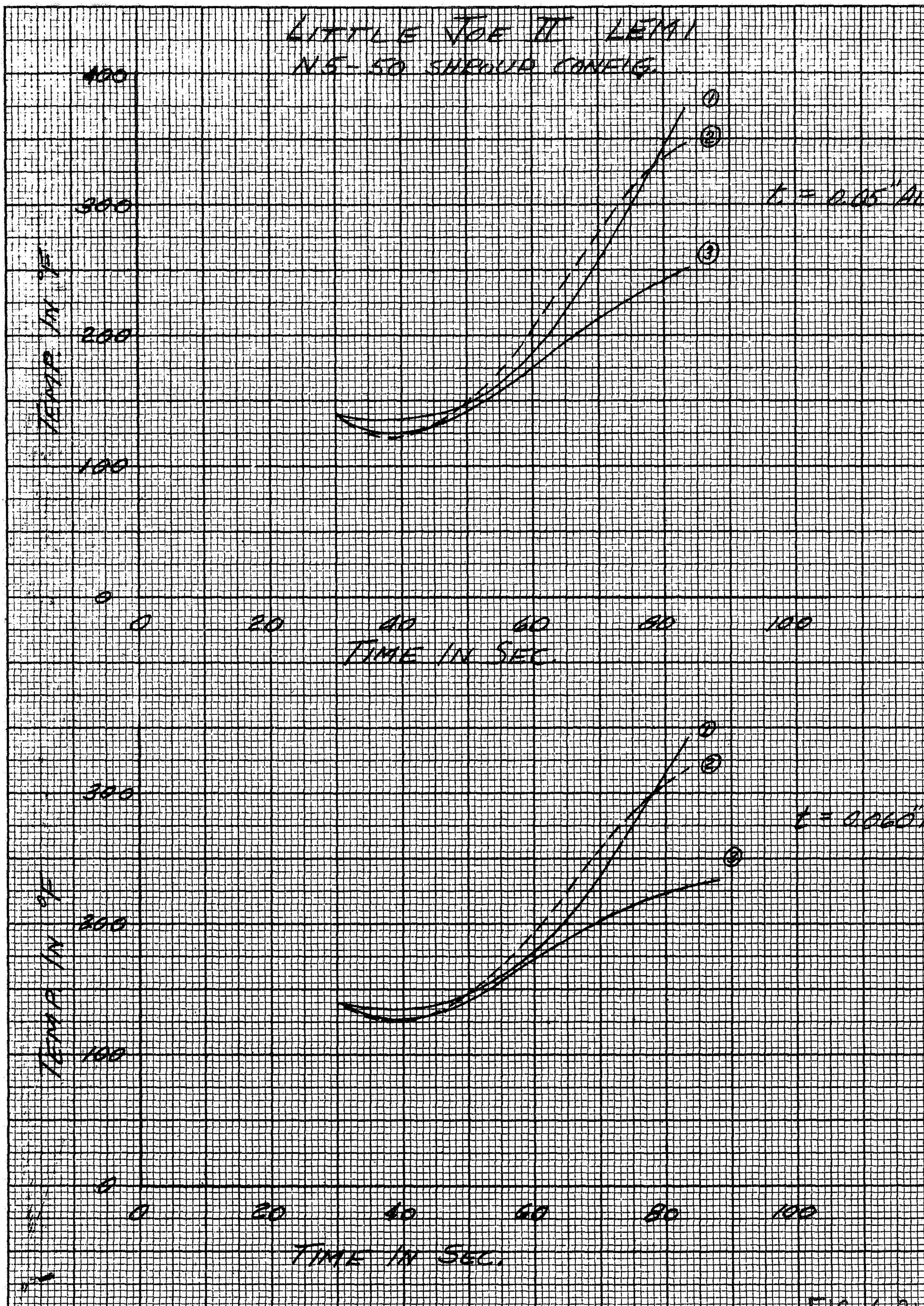

LITTLE JOE II LEM 1
NS-50 SHROUD CONFIG.
t = 0.05" AL
t = 0.060" AL
TEMP. IN °F
TIME IN SEC.
0
20
40
60
80
100
200
300
400
①
②
③

FIG. 6-21

7.0 Instrumentation Requirements.-

7.1 Onboard Instrumentation.-

7.1.1 Philosophy.-

The instrumentation requirements will be determined by using the following philosophies:

a) Relatively simple tests shall be scheduled for LEM-1

b) Test emphasis for LEM-1 will be on ascent stage testing

c) The instrumentation requested shall:

1. insofar as possible, not require modification of LEM components;

2. be concentrated in the areas concerned with the prime objective of the test;

3. be of minimum complexity;

4. permit determination of component or assembly failure, but not analysis of the cause of the failure.

7.1.2 Listing.-

Instrumentation shall be provided, consistent with the definition of terminology prescribed by NASA, to accomplish the first and second order objectives presented in this report. It is presently felt that, in order to attain these objectives, the following major areas will be instrumented. The instrumentation will, in general, be of the type indicated for each area:

a) Propulsion Subsystems (Ascent & Descent)

1. pressure
2. temperature
3. quantity
4. valve action

b) Reaction Control Subsystem

1. pressure
2. temperature
3. quantity
4. valve action

c) Stabilization & Control Subsystem

1. subsystem voltage inputs and output
2. gimbal position

7.1.2 Listing.- continued

d) Environmental Control Subsystem

1. pressure
2. temperature
3. fluid level

e) Electrical Power Subsystem

1. voltage
2. current
3. switch positions

f) Structure

1. pressure
2. temperature
3. vibration
4. acceleration
5. motion picture camera

g) Mechanical Systems

1. electrical signals

The detailed listing of instrumentation requirements will be formulated after the LEM-1 mission has been resolved and will be presented in the Measurements List for LEM-1.

7.2 Data Acquisition Systems.-

7.2.1 Telemetry and Antenna System.-

NASA-supplied FM/FM/PAM telemetry systems will be utilized for data acquisition and transmission. The details of the system (i.e. commutation, etc.) are to be determined when the instrumentation list is completed.

7.2.2 Camera.-

A motion picture camera will record the LEM ascent-descent stage separation. This camera will be installed in a recoverable package.

7.2.3 Tape Recorder.-

Telemetry acquisition of the fire-in-the-hole data could be hampered by the venting of the ascent engine exhaust from between the LEM stages. In order to assure acquisition of this data, a tape recorder will be included in the recoverable package provided for the camera.

7.3 Real-Time Readouts.-

A study of the real time visual displays required for selection of ground commands is in progress. The results of this study will be indicated in a revision to this report.

8.0 Tracking And Support Data Requirements.-

8.1 General.-

Specific requirements for tracking will be covered in the forthcoming issue of "Requirements of Work and Resources" (RFWAR). Preliminary requirements will be outlined in this section.

8.2 Phototheodolite Data.-

Phototheodolite data will be required for position, velocity, acceleration and attitude data. It is anticipated that the Contraves, Askania 53, and Gorid Cinetheodolite Systems will be utilized to obtain this data. Coverage from launch to LEM ascent engine cutoff is desired.

8.3 Radar Tracking Data.-

Radar tracking data will be required for determination of range, azimuth, and elevation. It is anticipated that the FPS-16 radar system will acquire this data with support from the Chain Radar System. Data will be required for real time data reduction facility for vehicle programming flight safety, and inflight analysis.

8.4 Photographic Data.-

Engineering sequential and documentary films will be included in this section.

8.4.1 Engineering Sequential Films.-

For recording launch conditions, two fixed high speed 16 or 35 mm cameras will be required. Location of the cameras should be 90 degrees apart on a circle with the launch pad as center.

High speed telescopic photography will be required to provide data for technical review of the test. The standard tracking telescope system of the range operating at 120 frame/sec will satisfy this requirement.

8.4.2 Documentary Films.-

A record shall be maintained by the use of documentary motion picture and still photography of all major events associated with the test; e.g. vehicle transporation, launch preparation and checkout, flight operations, and post flight activities.

8.5 Meteorological Data.-

Data consisting of wind direction and velocity, air density and temperature, and relative humidity at various altitudes will be required from the range. Data to be available from the launch site to 100,000 feet above ground level.

9.0 Prelaunch Operations.-

9.1 Test Preparation - Bethpage.-

Each system will be excercised and checked out in all respects practical at the Grumman plant in Bethpage. Such checkout efforts, aside from proving the configuration and integrity of the vehicle proper, appropriately prove or refine the field procedures, demonstrate the utility of the support equipment, and develop proficiency in the operating personnel. The detailed requirements for this check out are presently being formulated.

9.2 Test Preparation - WSMR.-

A checkout flow diagram for the prelaunch tests is given in Figure 9-1. This flow will be further detailed as the LEM development progresses.

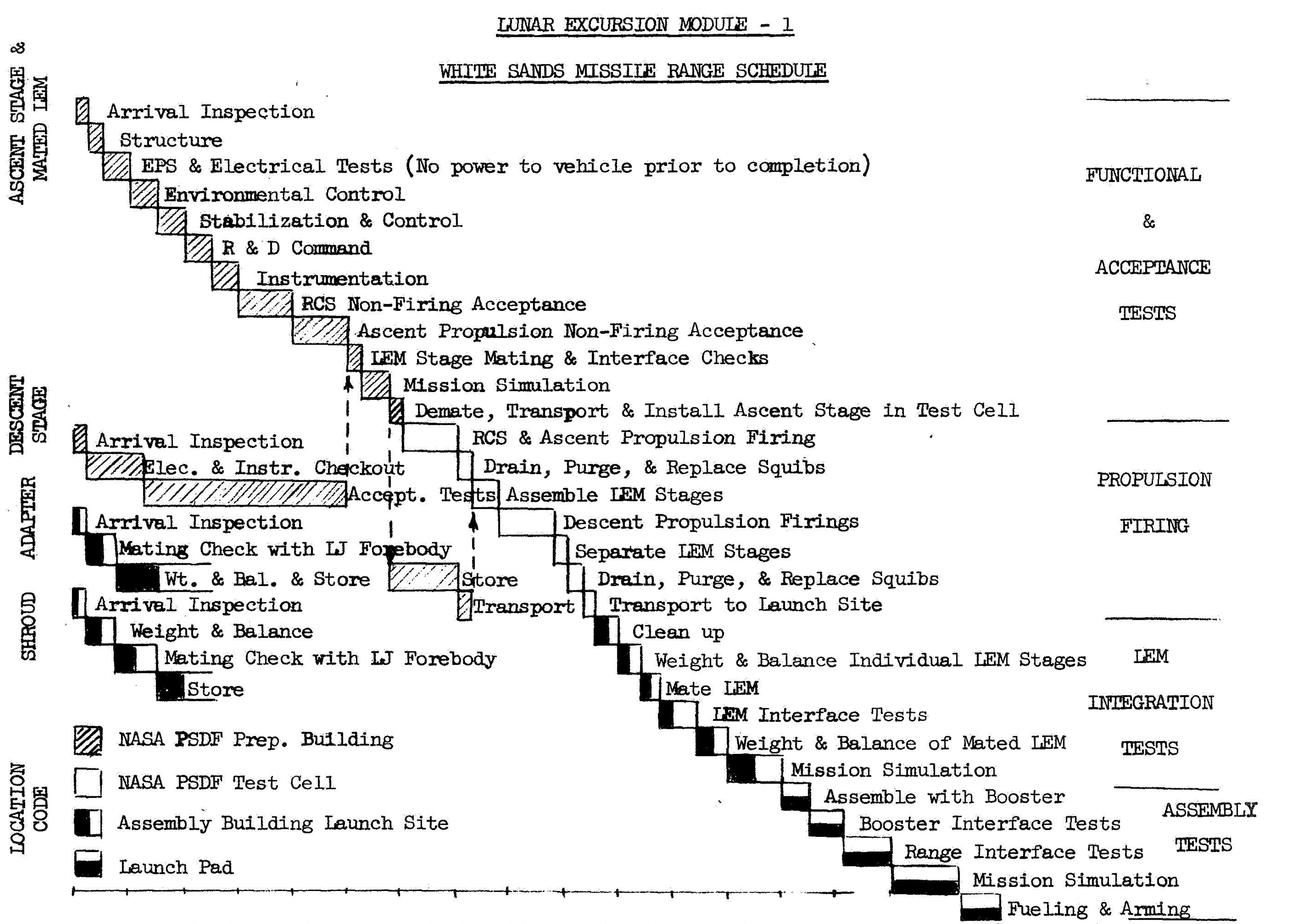

LUNAR EXCURSION MODULE - 1
WHITE SANDS MISSILE RANGE SCHEDULE
ASCENT STAGE & MATED LEM
Arrival Inspection
Structure
EPS & Electrical Tests (No power to vehicle prior to completion)
Environmental Control
Stabilization & Control
R & D Command
Instrumentation
RCS Non-Firing Acceptance
Ascent Propulsion Non-Firing Acceptance
LEM Stage Mating & Interface Checks
Mission Simulation
Demate, Transport & Install Ascent Stage in Test Cell
RCS & Ascent Propulsion Firing
Drain, Purge, & Replace Squibs
Assemble LEM Stages
Descent Propulsion Firings
Separate LEM Stages
Drain, Purge, & Replace Squibs
Transport to Launch Site
Clean up
Weight & Balance Individual LEM Stages
Mate LEM
LEM Interface Tests
Weight & Balance of Mated LEM
Mission Simulation
Assemble with Booster
Booster Interface Tests
Range Interface Tests
Mission Simulation
Fueling & Arming
DESCENT STAGE
Arrival Inspection
Elec. & Instr. Checkout
Accept. Tests
Store
Transport
ADAPTER
Arrival Inspection
Mating Check with LJ Forebody
Wt. & Bal. & Store
SHROUD
Arrival Inspection
Weight & Balance
Mating Check with LJ Forebody
Store
LOCATION CODE
NASA PSDF Prep. Building
NASA PSDF Test Cell
Assembly Building Launch Site
Launch Pad
FUNCTIONAL & ACCEPTANCE TESTS
PROPULSION FIRING
LEM INTEGRATION TESTS
ASSEMBLY TESTS

FIG. 9-1

10.0 Test Management & Organization.-

10.1 GAEC - LEM Organization.-

An organization chart of the Grumman - LEM project is given in Figure 10-1.

10.2 LEM-1 Test Organization.-

This information will be provided at a later date.

GAEC-LEM ORGANIZATION CHART

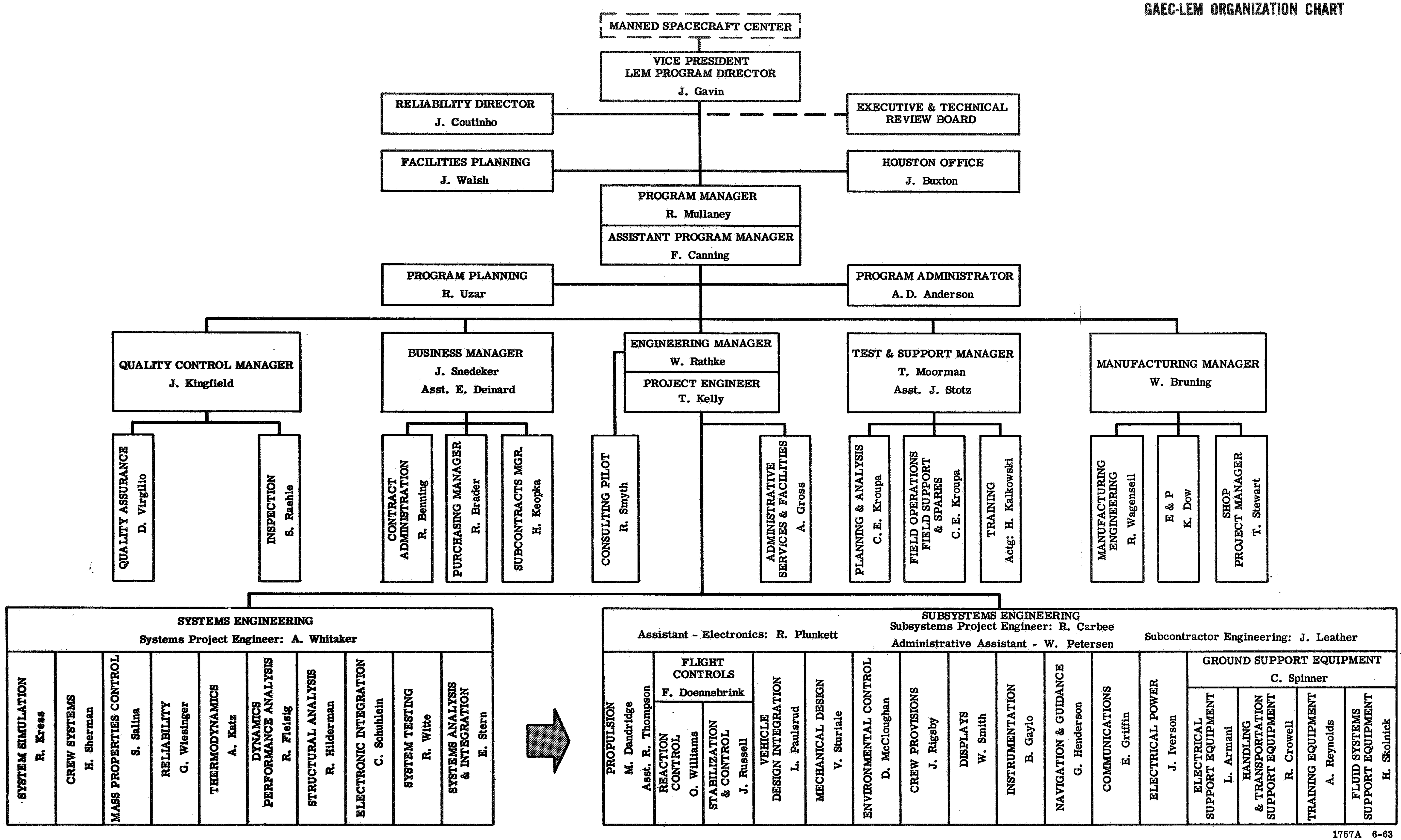


FIG. B 10-1

11.0 Recovery Requirements.-

11.1 Test Vehicle.-

At present the Grumman Aircraft Engineering Corporation has no requirement for recovery of the test vehicle. The requirements in this area shall be supplied by the White Sands Missile Range and MSC.

11.2 Data Capsule.-

A motion picture camera and a tape recorder will be enclosed in a recoverable package. The details of the recovery requirements will be supplied when the method has been finalized (i.e. a package within vehicle which is able to take reentry heat and impact shock or a parachuted capsule.)

12.0 Launch Day Requirements.-

12.1 Weather requirements for Launch.-

12.1.1 Visibility.-

The visibility requirements for optical and camera tracking will be supplied at a later date.

12.1.2 Meteorological Support.-

Meteorological data will be required for GO-NO-GO decisions, for launcher azimuth corrections, and for data reduction requirements. The specific data and the accurancies required will be included in a revision to this report.

13.0 Pad and Range Safety Requirements.-

13.1 Pad.-

13.1.1 Personnel.-

The launch pad, blockhouse, and rocket-motor assembly areas will be designated as limited access areas, and only those personnel required to perform the assigned tasks will be near the dangerous areas.

Personnel admitted to the limited access area will be required to wear safety helmets, safety shoes or let straps and safety glasses as necessary. Hospital and medical care is to be provided by the WSMR Medical Staff. An ambulance and crew and a fire truck will be available when needed.

13.1.2 Explosive and Pyrotechnic Control.-

All ordinance items will be delivered to the solid propellant storage area. This includes:

a. solid motors

b. squibs

c. igniters

d. primacord and destruct charges

13.1.3 Test Vehicle Build-up.-

This information will be supplied at a later date.

13.1.4 Standard Operating Procedures.-

Standard operating procedures will be prepared. These SOP's will define safety measures which will be taken during the handling of the rocket motors, installation of igniters, testing of ignition circuits, installation of destruct charges and associated circuitry, utilization of shorting plugs and final arming of the rocket motors and other pyrotechnic devices.

13.1.5 Launch Vehicle Tie-Down Requirements.-

It is anticipated that the requirements in this area will be similar to those of earlier Little Joe launches. These launches require an immediate alert on a 7 day-per-week basis, 24 hours prior to a time when winds in excess of 30 miles per hour are predicted. After such an alert, reports should be furnished for each incremental rise of 5 miles per hour.

13.2 Range Dispersion.-

An analysis has been conducted to determine the impact dispersion of the LEM Flight test vehicles. For purposes of clarity, impact dispersion is defined as the probable boundary of the area in which the LEM will impact as a result of uncertainties in Little Joe II booster performance and variations in launch conditions. For this analysis, the factors that were considered in defining the impact dispersion were:

a. Misalignment of the effective thrust vector during Little Joe II boost.

b. Magnitude variations of the effective thrust vector during Little Joe II boost.

c. Gyro drift of booster guidance system.

d. Variation in local wind magnitude and direction at White Sands Missile Range during the entire flight.

Presented in Figure 13-1 is a plot showing the probable boundary of LEM impact as a result of any one or combined uncertainties described above. The plot was constructed by a root sum square of the probable impact boundaries defined by each of the uncertainties, computed independently of each other. All data presented herein were computed using a three dimensional point mass digital program and a thrusting law consisting of aligning the booster thrust axis with commanded inertial pitch and azimuth angles. The nominal impact point was determined by the ballistic trajectory resulting from commanding the booster thrust axis to remain aligned with an 85° pitch angle and 10° west of North heading angle at launch. LEM descent and ascent engine thrusts were always assumed to remain aligned outward along the instantaneous local vertical.

From Figure 13-1, it can readily be seen that the probable boundary of LEM impact is not within the geographical confines of White Sands Missile Range. Four small area of range overshoot occurs over the northern boundary of the test range. This is primarily attributed to trajectories resulting from a +7000 pound thrust error simultaneously occurring in each Algol rocket motor.

Presented in Figure 13-2 is a plot showing the impact dispersion resulting from thrust misalignment of the Little Joe Booster. In accordance with the QTV Mission Directive, the maximum thrust misalignment was assumed to be a be a $\pm\frac{1}{2}$ degree error in pitch and azimuth angles at launch. One can readily note the small crossrange dispersion resulting from this uncertainty.

Figure 13-3 is a plot showing impact dispersion resulting from thrust magnitude variation in each of the algol rockets. For maximum dispersion, it was assumed that each Algol rocket produced an error of ±7000 lbs. of thrust simultaneously. This condition resulted in off range impact.

Presented in Figure 13-4 is a plot showing the impact dispersion resulting from gyro in the booster guidance system. Various combinations of linear drift rates were assumed in the body mounted pitch and yaw gyros to produce a total of two degree error in pitch and yaw at the time of booster burnout.

Presented in Figure 13-5 is a plot showing a series impact dispersions caused by a variation in magnitude and direction of wind during flight. The nominal impact point was defined by assuming 84% winds from the West. Dispersions are defined about this point as a result of ±5%, ±10%, and ±15% variation in wind magnitude from all directions. Since wind data is obtained at frequent intervals prior to launch, it was assumed that the maximum level of uncertainty would not be greater than 15%.

From this analysis it can be concluded that, if the above mentioned uncertainties are realistic, LEM impact dispersion could result outside the boundry of the White Sands Missile Range. In order to confine LEM impact dispersion within the boundaries of White Sands Missile range, it is suggested that the commanded inertial flight path angle during boost be increased to 85.5 degrees or greater. (A commanded flight path angle of 85.0 was used in this analysis). This modification would decrease the total downrange distance traversed by LEM without altering the overall trajectory characteristics significantly.

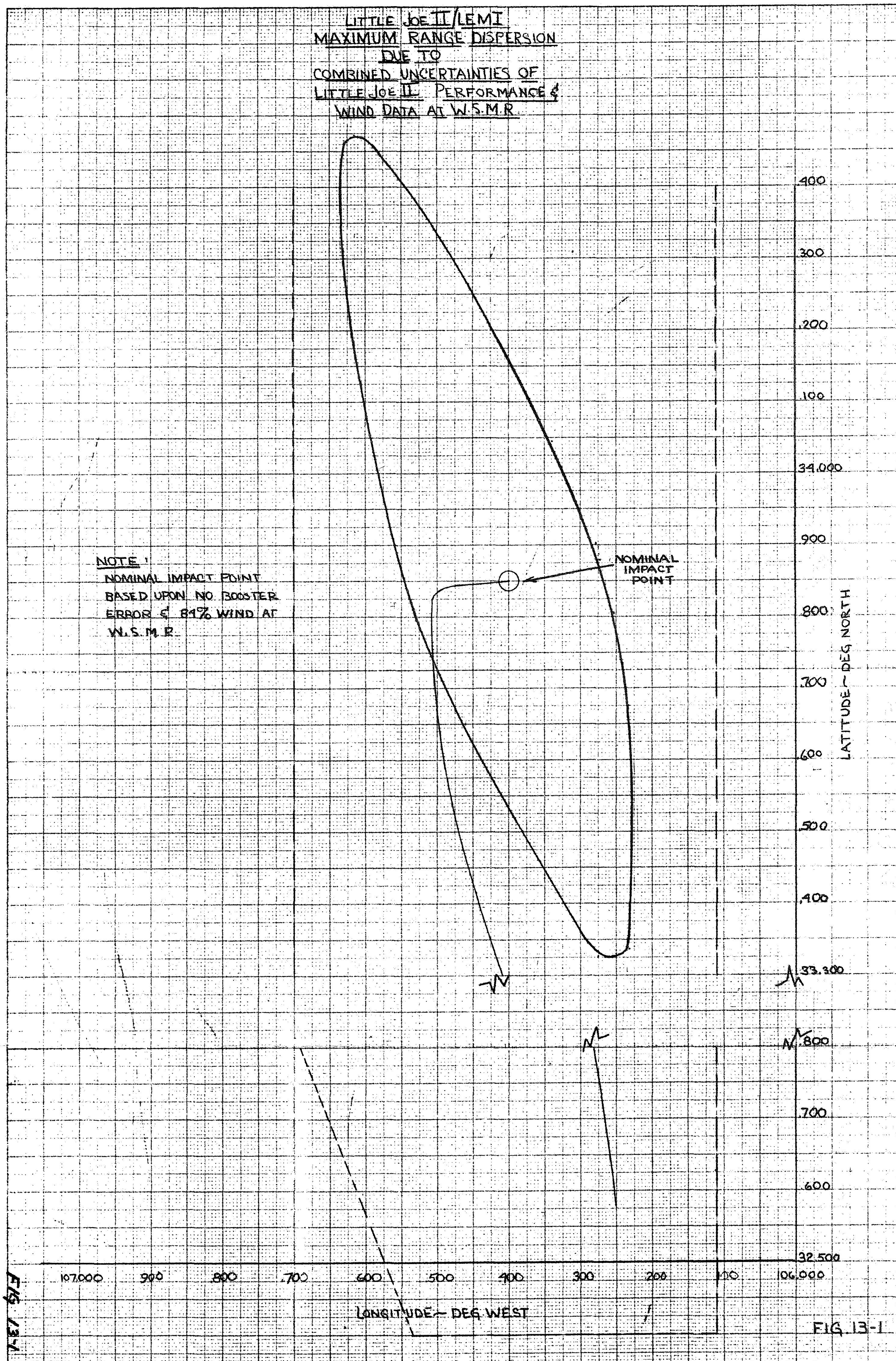

LITTLE JOE II/LEMI
MAXIMUM RANGE DISPERSION
DUE TO
COMBINED UNCERTAINTIES OF
LITTLE JOE II PERFORMANCE &
WIND DATA AT W.S.M.R.
NOTE:
NOMINAL IMPACT POINT
BASED UPON NO BOOSTER
ERROR & 89% WIND AT
W.S.M.R.
NOMINAL
IMPACT
POINT
LATITUDE ~ DEG NORTH
.400
.300
.200
.100
34.000
.900
.800
.700
.600
.500
.400
33.300
.800
.700
.600
32.500
107.000
900
800
.700
600
500
400
300
200
100
106.000
LONGITUDE ~ DEG WEST

FIG. 13-1

LITTLE JOE II LEM I
RANGE DISPERSION DUE TO-
THRUST MISALIGNMENT OF SEVEN (7) ALGOL MOTORS
(±.5 DEGREE/MOTOR)

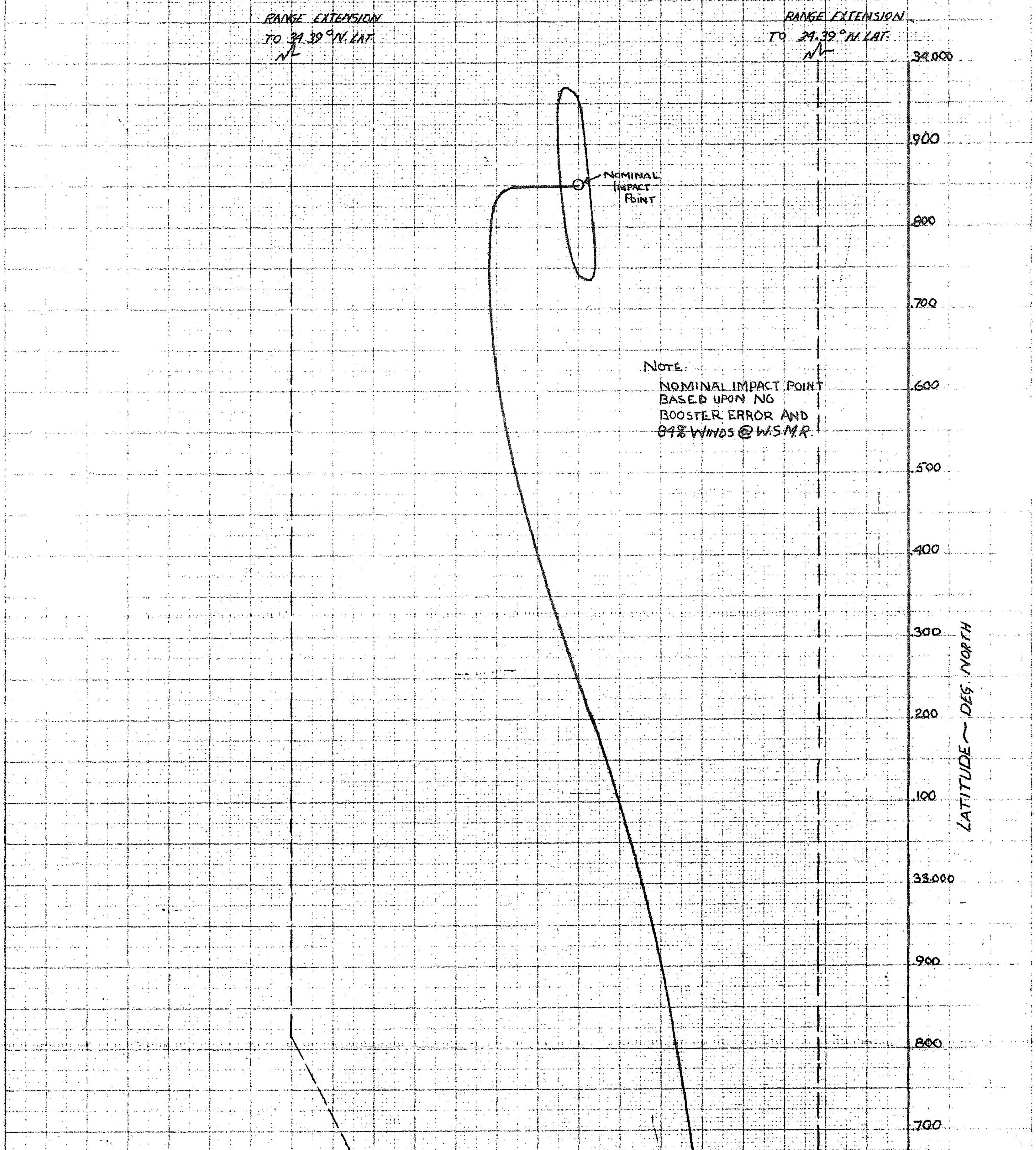
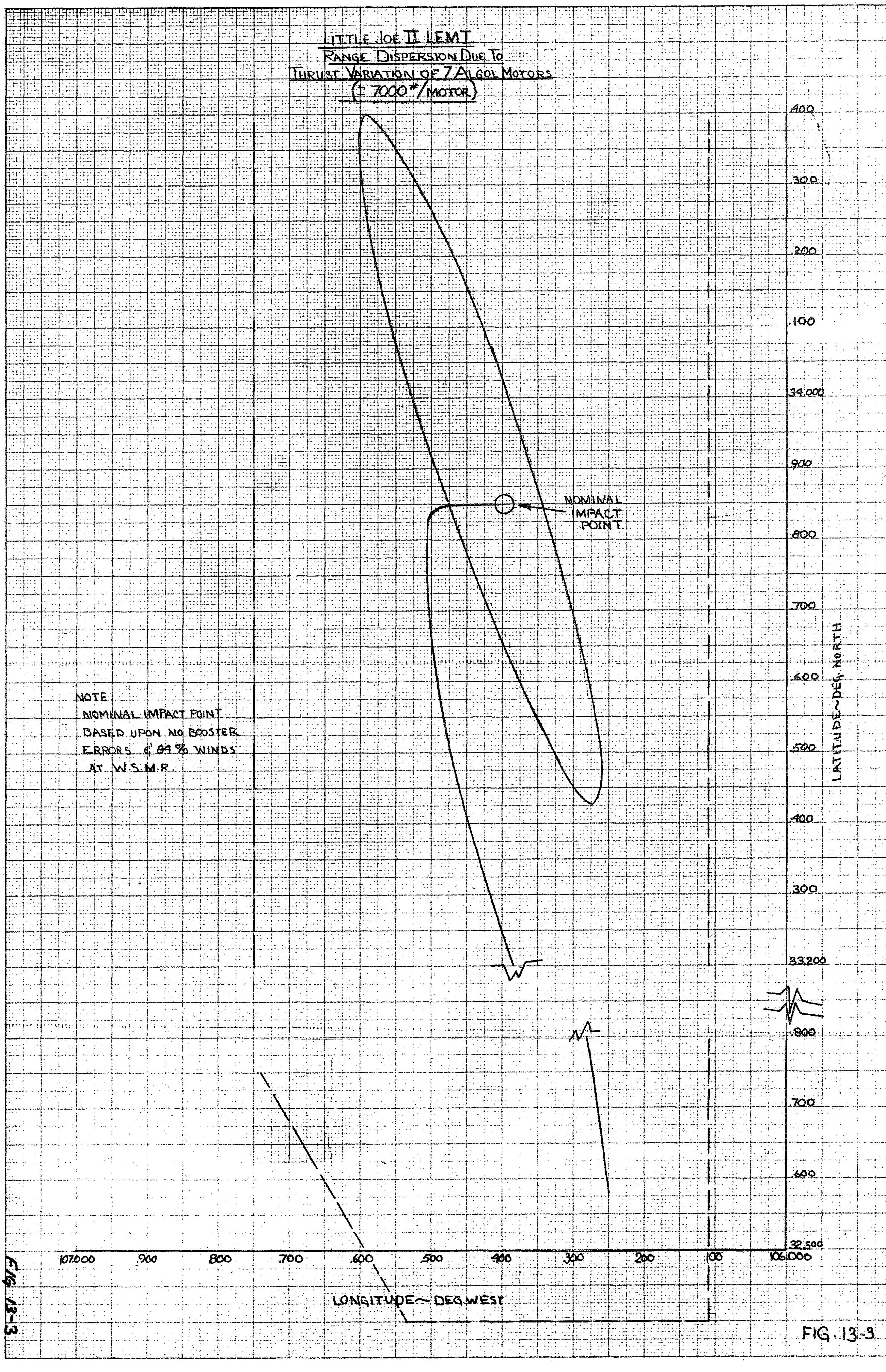


FIG 13-2



LITTLE JOE II LEMT
RANGE DISPERSION DUE TO
THRUST VARIATION OF 7 ALGOL MOTORS
(± 7000#/MOTOR)
NOMINAL IMPACT POINT
NOTE
NOMINAL IMPACT POINT
BASED UPON NO BOOSTER
ERRORS & 99% WINDS
AT W.S.M.R.
400
300
200
.100
34.000
900
800
700
600
500
400
300
33.200
.800
.700
.600
32.500
LATITUDE~DEG. NORTH
107.000
.900
800
700
.600
.500
400
300
200
.100
106.000
LONGITUDE~DEG.WEST
FIG. 13-3

FIG. 13-3

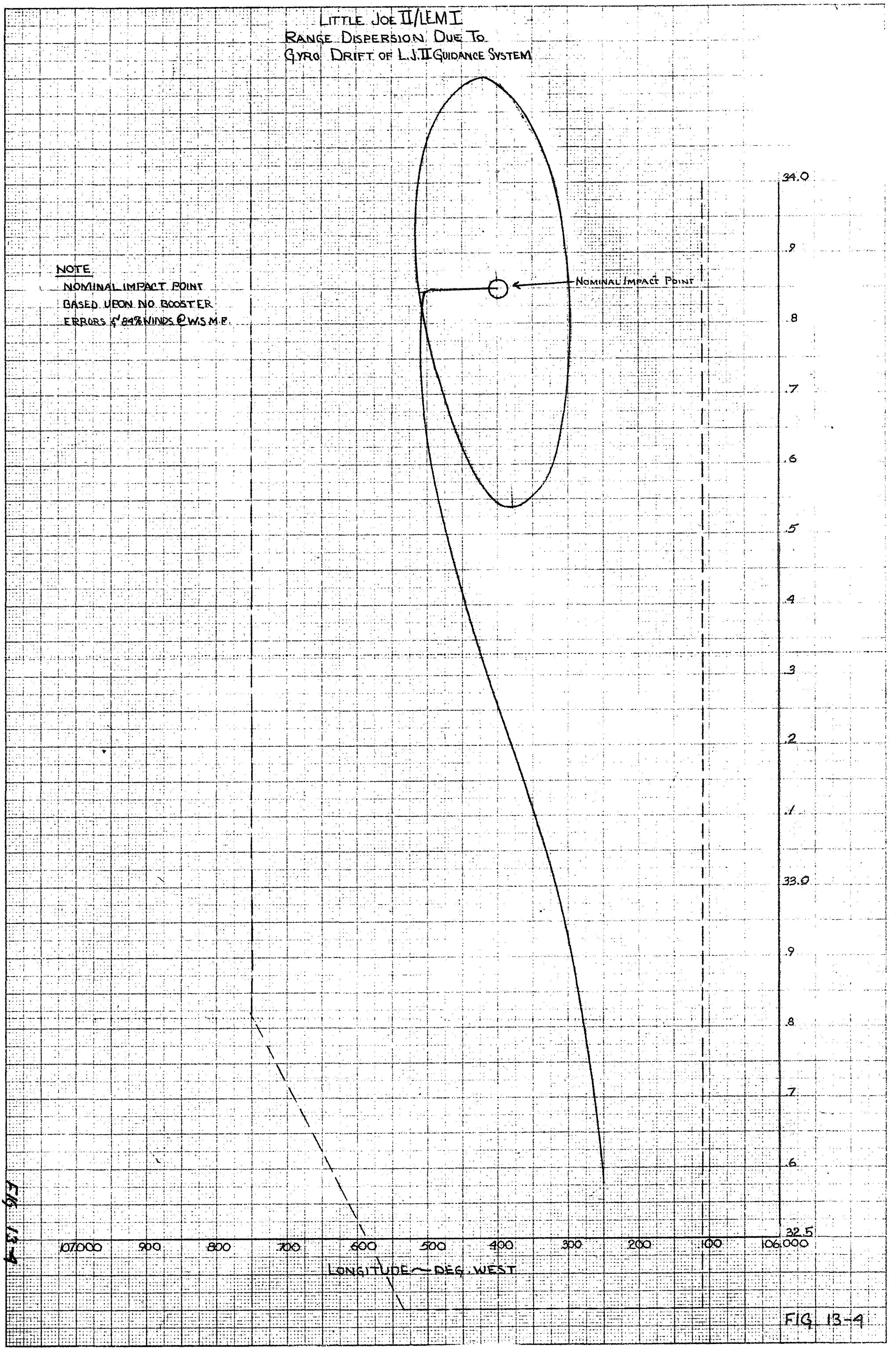
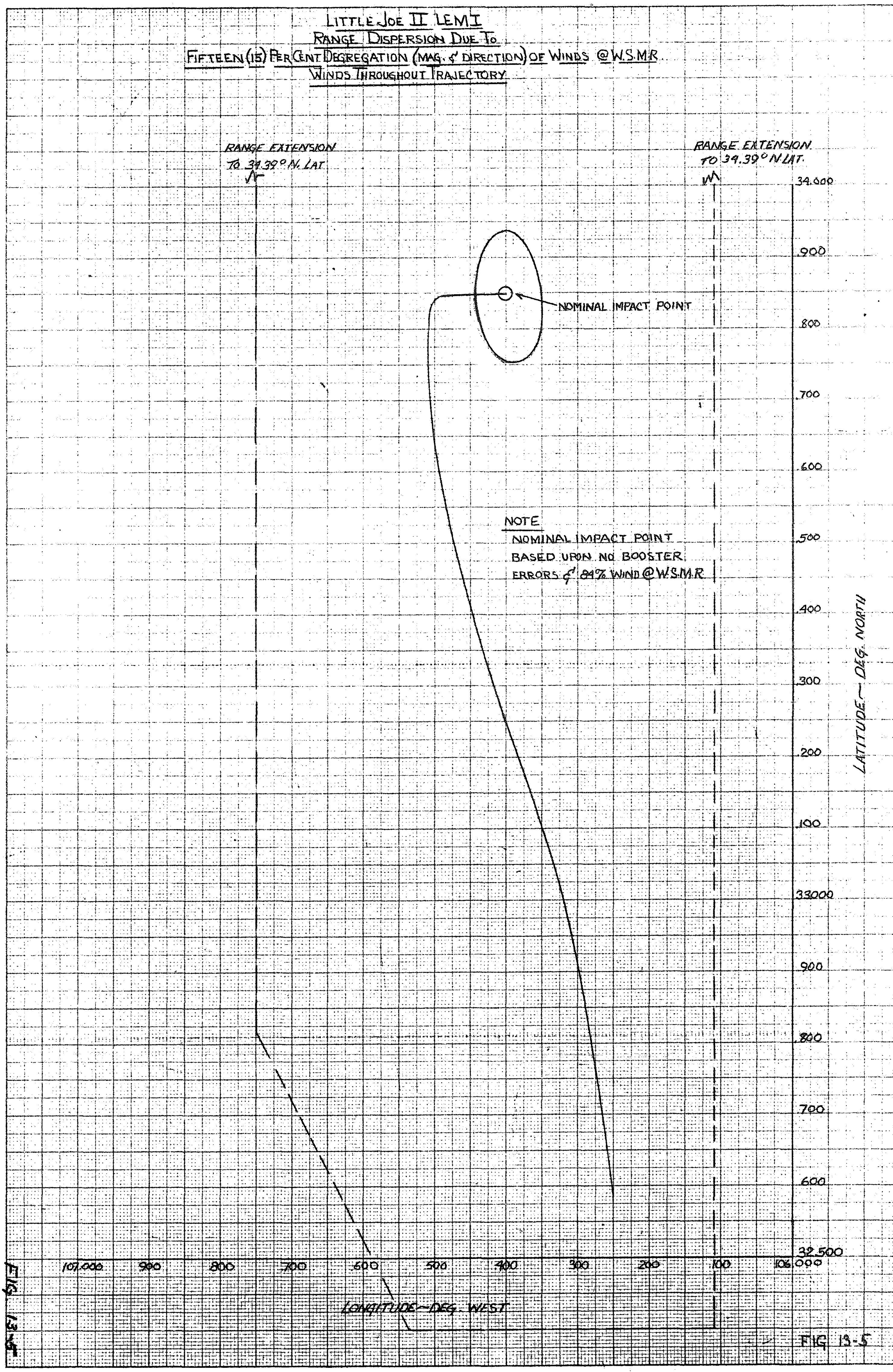

LITTLE JOE II/LEM I
RANGE DISPERSION DUE TO
GYRO DRIFT OF L.J.II GUIDANCE SYSTEM
NOTE
NOMINAL IMPACT POINT
BASED UPON NO BOOSTER
ERRORS & 84% WINDS @ W.S.M.R.
NOMINAL IMPACT POINT
34.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
33.0
.9
.8
.7
.6
32.5
107.000
900
800
700
600
500
400
300
200
100
106.000
LONGITUDE ~ DEG. WEST

FIG. 13-4



LITTLE JOE II LEM I
RANGE DISPERSION DUE TO
FIFTEEN (15) PER CENT DEGREGATION (MAG. & DIRECTION) OF WINDS @ W.S.M.R.
WINDS THROUGHOUT TRAJECTORY
RANGE EXTENSION TO 34.39° N. LAT.
RANGE EXTENSION TO 34.39° N LAT.
NOMINAL IMPACT POINT
NOTE
NOMINAL IMPACT POINT BASED UPON NO BOOSTER ERRORS & 84% WIND @ W.S.M.R
LATITUDE ~ DEG. NORTH
LONGITUDE ~ DEG. WEST
34.000
900
800
700
600
500
400
300
200
100
33000
900
800
700
600
32.500
107.000
900
800
700
600
500
400
300
200
100
106.000

FIG 13-5

14.0 Data Handling, Processing, and Reporting.-

14.1 Data Handling and Processing.-

Coordination of data handling and processing requirements with range personnel will be accomplished through NASA-MSC.

14.1.1 Prelaunch and Checkout Data.-

A checkout trailer shall be provided at the launch site for recording prelaunch and checkout data. Oscillographs, pen recorders, and visual displays will be provided for monitoring during checkout. Recorded magnetic tapes will be stored at the base facility for detailed playback if required. Copies of selected checkout tapes will be made for transmittal to GAEC/BPA after preliminary analysis at the test site is complete.

14.1.2 On-Board Tape Recorder.-

Following recovery of the on-board magnetic tape recorder, a quick look playback shall be provided for instrumentation and recording verification. Copies of the tape shall be made with disposition as follows: original and copy to MSC/Houston, copy to GAEC personnel, and one copy to remain at WSMR for further on-site analysis, if required.

14.1.3 Flight Telemetry Data.-

Flight telemetry data will be recorded at selected telemeter acquisition sites along the trajectory path. In addition, the mobile telemeter trailer will be utilized to record flight telemetry information. Real time data display of selected parameters at the control center is required for flight assessment to support the ground command system.

Quick-look playback of recorded tapes shall be made immediately after flight and primary data tapes are designated. Three copies shall be made of these master tapes and distributed as follows: original to MSC/Houston, copy to GAEC personnel, one copy to remain at WSMR for further analysis.

14.1.4 Range Data.-

Range data shall be processed to required form by the range contractor. Data shall be made available to GAEC through the NASA/MSC office at WSMR.

14.1.4.1 Meteorlogical Data.-

Meteorlogical data of prelaunch and post-launch conditions shall be provided by the range in tabulated form.

14.1.4.2 Optical Data.-

Range optical data shall be made available to GAEC through NASA/MSC office at WSMR. Finalized optical data shall be made available to GAEC within ten days following the test.

14.1.4.3 Radar Data.-

Radar plots containing real time, azimuth, elevation, and range to the cehicle, shall be provided from the range stations. Finalized data in a common coordinate system shall be provided to GAEC within five days following the test.

14.1.4.4 Engineering Sequential Film and Documentary Film Data.-

Engineering sequential films and documentary film will be processed in range facilities. Copies of these films will be made available to GAEC through the NASA/MSC office at WSMR.

14.2 Calibration Data.-

Procedures for calibration data handling will be issued in forthcoming "GAEC Data Plan for White Sands Missile Range Tests".

14.3 Data Analysis.-

14.3.1 On-Site Analysis.-

Quick-look test analysis shall be performed at the range facilities for assessment of the test.

14.3.2 Final Detailed Analysis.-

Procedures for final data analysis will be defined in the forthcoming "GAEC Data Plan for White Sands Missile Range Tests".

14.4 Test Reporting.-

The requirements for reporting the test results will be established at a future date.

15.0 Ground Support Equipment.-

15.1 Test Vehicle Ground Support Equipment.-

The test vehicle ground support equipment is listed below in the three categories of (1) Handling and Transportation Equipment; (2) Fluid System Support Equipment and; (3) Electrical Support Equipment.

15.1.1 Handling and Transportation Equipment.-

Platform Set, Altitude Chamber
Adapter Set, D/S Hard Mount
Adapter Hard Mount A/S
Vertical Drive Support Frame
Sling, LEM Hoisting
Work Stand LEM
Transporter A/S
Clean Room Dolly A/S
Sling A/S Hoist
A/S Work Stand
A/S Support Stand
Transporter D/S
Clean Room Dolly D/S
Sling D/S Hoist
D/S Work Stand
D/S Support Stand
Engine Installation Fixture
Dolly A/S Engine Handling
Sling A/S Engine Hoist
Sling D/S Engine Hoist
Sling RCS
Dolly RCS Engine
Sling A/S Fuel Tank
Dolly A/S Fuel Tank
Sling D/S Fuel Tank
Dolly A/S Oxidizer Tank*
Sling A/S Oxidizer Tank*
Dolly D/S Oxidizer Tank**
Sling D/S Oxidizer Tank**
Ground Cooling Cart (GFE)
Optical Alignment Equipment (GFE)
Little Joe II Adapter Dolly
Little Joe II Adapter Sling
Little Joe II Shroud Transporter
Little Joe II Shroud Sling
Little Joe II Access Kit
Sling-Transporter Cover
Cover: Little Joe II Shroud
Cover: Little Joe II Adapter
Little Joe II Shroud Weight and Balance Equipment (Undefined)

* May be same as A/S Fuel Tank
** May be same as D/S Fuel Tank

15.1.2 Fluid System Support Equipment.-

15.1.2.1 Propulsion and RCS Subsystems.-

Helium Transfer Cart
Helium Booster Cart
Helium Transfer Module
Helium Leak Detector

15.1.2.1.1 Propulsion Subsystem.-

Fluid Temperature Conditioning Unit
Oxidizer Temperature Conditioning Unit
Fuel Flush and Purge Unit
Oxidizer Flush and Purge Unit
Fuel Vapor Disposal Unit
Oxidizer Vapor Disposal Unit
Fuel Storage and Transfer Stand
Oxidizer Storage and Transfer Stand
Propulsion Subsystem Checkout Cart

15.1.2.1.2 RCS Subsystem.-

RCS Subsystem Checkout Cart
Fuel Flush and Purge Cart
Oxidizer Flush and Purge Cart
Fuel Transfer Module
Oxidizer Transfer Module

15.1.2.2 Environmental Control Subsystem.-

Water Transfer Unit
Water Glycol Service Unit
Oxygen Transfer Unit
Cabin Leak Test Unit

15.1.3 Electrical Support Equipment.-

15.1.3.1 Bench Maintenance Equipment.-

The BME is used to support the subsystem black boxes after removal or prior to installation in the LEM. The areas involved are a parital stabilization and control subsystem, a modified electrical power subsyste, and displays and controls. The BME is used to fault isolate a black box to the smallest replaceable module.

The BME is estimated as including the following:

Test Station, Maintenance, Platform Electronics and 4 Gimbal Platforms.
Test Station, Maintenance, Rate Gyro
Test Station, Maintenance, Acclerometers
Test Station, Maintenance, Power Supply
Test Station, Maintenance, Descent Engine Control and Gimbal Drive Actuator Assemblies

15.1.3.1 (con't)

Test Station, Attitude and Translation Control Assembly
Test Station, Maintenance, Control Panel
Test Station, Maintenance, Guidance Coupler Assembly
Test Station, Maintenance, Distribution
Test Station, Maintenance, Inverter
Test Station, Maintenance, Battery

15.1.3.2 Prelaunch Checkout.-

As a result of meetings between NASA and GAEC Representives, it was decided that PACE or LSTU were not required for LEM-1. It was felt that a GFE LEM test set, as yet undefinged, could adequately perform necessary prelaunch checks.

TABLE OF CONTENTS, SECTION C

TABLE OF CONTENTS, SECTION C (con't)

1.0 Introduction.-

The information presented in the Introduction for LEM-1 (Page B1-1) is directly applicable to LEM-2.

2.0 Test Objectives and System Priorities.-

2.1 General.-

The LEM-2 test is the second of the unmanned flight tests which are required to man-rate the LEM vehicle. The test emphasis will be concentrated to the descent propulsion subsystem and the effects associated with it.

2.2 Test Objectives.-

2.2.1 First Order Test Objectives.-

a. Evaluate and certify the safe operation of the descent engine propulsion subsystem.

b. Evaluate the fire-in-the-hole condition during an abort.

c. Demonstrate the safe operation of the ascent propulsion subsystem.

d. Evaluate the thermal distribution of the LEM in the space environment with a fully expanded plume.

e. Evaluate the in-flight dynamics of the LEM system throughout a wide range of propellant loadings.

2.2.2 Second Order Test Objectives.-

a. Evaluate the LEM engine induced environments.

b. Demonstrate the compatability of the applicable GSE.

2.3 System Priorities.-

The system priorities are unchanged from the priorities presented for LEM-1 (page B2-1).

3.0 Mission Description.-

3.1 General.-

The launch is planned to take place from the Little Joe facility of the White Sands Missile Range. The vehicle will be launched at an elevation of 85° and on an azimuth which will depend on the prevailing wind. Four Algol motors will be ignited simultaneously at launch. The launch attitude will be maintained by the booster aerodynamic and reaction controls. At launch plus 39 seconds, the remaining three Algols will be ignited. At BECO the LEM shroud will be separated followed by LEM separation and booster retrograde. The LEM will be rotated and stabilized with the X-X axis parallel to the weight vector and Z-Z axis in the plane of the planned trajectory. The test ullage rockets will be ignited, the LEM descent engine started, and the descent engine tests will commence. These tests will include steady-state performance at various thrust levels, throttling transients and response, multiple starts including zero 'g', operation in the redundant modes, gimbal monitoring, and possibly malfunction tests by simulated failure. Upon completion of the descent propulsion subsystem evaluation, the descent stage will be separated in a simulated abort sequence. Ascent engine and RCS tests will then be conducted until impact. The nature of these tests will be determined with inputs from the ground test program and possibly the LEM-1 results.

3.2 Detailed Test Sequence.-

Time	Event
t--	Electrical power on
t--	Activate environmental control
t--	Erect gyros (or align platform)
t+0	First stage booster ignition 4 motors
t+39	Second stage booster ignition 3 motors
t+44	First stage booster cut-off
t+62	Fire T.C.A's 1 thru 16 in 0.15 second intervals (air purge)
t+74	Shut down T.C.A.'s 1 thru 16
t+75	Pressurize RCS (fire squibs)
t+77	Select descent engine on engine mode select
t+81	Actuate descent helium tank squibs
t+82	Actuate descent propellant tank squibs
t+83	Second stage booster cut-off (BECO determined by acceleration sense)
	Initiate shroud separation
t+85	Separate LEM from adapter
	Initiate booster retrograde
t+86	Rotate LEM X-X axis to earth radial and stabilize
t+87	Select maximum descent engine thrust
	Initiate LEM ullage firing
t+88	Ignite LEM ullage rockets
t+89	Fire descent engine

3.2 Detailed Test Sequence.- continued

Time	Event
t+90.25	Ullage rocket burnout
t+214	Select redundancies
t+486	Reduce thrust to 5250 lbs.
t+501	Select redundancies
t+516	Reduce thrust to 3675 lbs.
t+531	Select redundancies
t+546	Reduce thrust to 2100 lbs.
t+561	Select redundancies
t+576	Reduce thrust to 1050 lbs.
t+591	Select redundancies
t+603	Open helium injection override
t+605	Select auto on helium injection override
t+606	Increase to maximum thrust in 0.5 seconds
t+609	Decrease to minimum thrust in 0.5 seconds
t+612	Increase to maximum thrust in 0.5 seconds
t+612.5	Decrease to minimum thrust in 0.5 seconds
t+613	Increase to maximum thrust in 0.5 seconds
t+617	Select redundancies
t+620	Decrease to minimum thrust in 0.5 seconds
t+620.5	Increase to maximum thrust in 0.5 seconds
t+621	Decrease to minimum thrust in 0.5 seconds
t+624	Shutdown
t+626	Start
t+629	Shutdown
t+631	Start
t+634	Shutdown
	Stabilization at zero (0) gravity
t+651	Start
t+652	Actuate ascent helium tank squibs
t+653	Actuate ascent propellant tank squibs
t+654	Initiate abort
t+786	Ascent engine cut-off
	RCS test procedure to be supplied at a later date

3.3 Flight Parameters.-

The following parameters are given as descriptive information and are not requirements of the test. The test requirement is simply that the test altitude during the flight be sufficient to provide a space environment.

3.3.1 Launch (t+0).-

Flight path angle - 85°
Azimuth - dependent on the wind

3.3.2 Booster Engine Cut-Off (t+83).-

Mach number - 4.3
Altitude (above MSL) - 163,500 feet
Dynamic Pressure - 24 psf
Range - 3.2 NM

3.3.3 LEM Descent Engine Cut-Off (t+654).-

Mach number - .63
Altitude - 999,500 feet
Dynamic Pressure - 0
Range - 50.6 NM

3.3.4 LEM Ascent Engine Cut-Off (t+786).-

Mach number - .93
Altitude - 720,800 feet
Dynamic Pressure - 0
Range - 70.8 NM

3.4 Trajectory.-

The nominal, no-perturbation trajectory is shown in figure 3-1. Since the Little Joe is stabilized only in an attitude hold mode, there will be disturbances from this nominal. The range dispersion due to disturbances and/or unknowns in prediction is discussed in Section 13.0.

3.5 Orientation.-

Vehicle orientation will be as prescribed for LEM-1 (page B3-4.).

3.6 Flight Path Corrections.-

Flight path corrections will be applied as planned for LEM-1 (page B3-4).

LITTLE JOE II/LEM II FLIGHT PARAMETERS

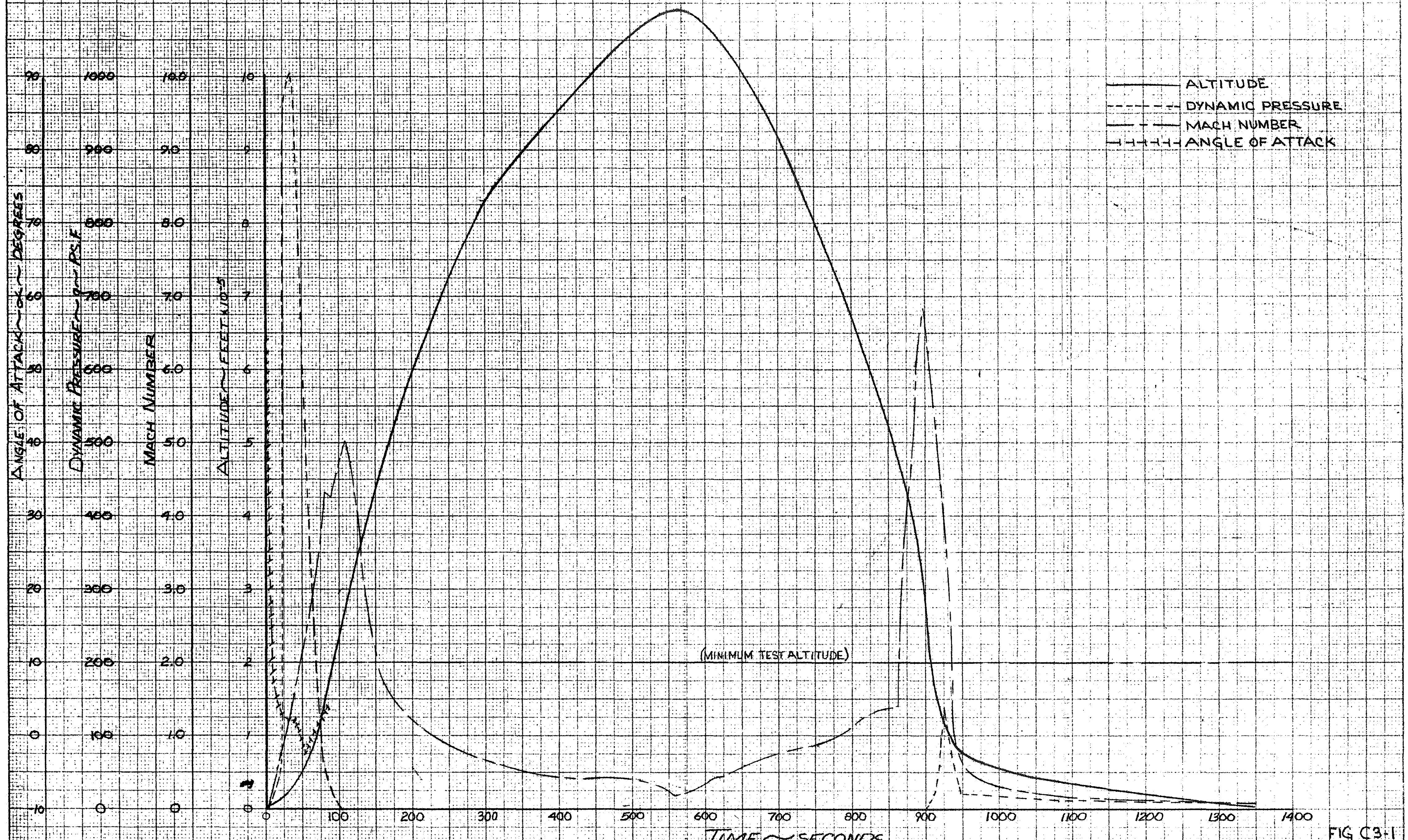


FIG C3-1

4.0 Description of Test Vehicle.-

4.1 General.-

The test vehicle comprises the LEM, shroud, and an adapter, supported on the Little Joe II booster. The LEM is supported by the adapter which is attached to the booster. The shroud is supported by the adapter which is attached to the booster. The configuration is illustrated in Figure B4-1.

4.2 Vehicle Weight.-

The mass characteristics of the test configuration prior to launch are shown in the following table.

Table 4-1

Item	Weight Lbs. Ascent	Weight Lbs. Descent	C.G. Arm - inches * X	Y	Z	Moments of Inertia, slug - ft^2 Ix	Iy	Iz
			* Station 0 taken at LEM Adapter/Little Joe II Separation Line					
LEM Subsystems:								
Structure	1219	806						
Stab. & Control	88							
Nav. & Guid.								
Crew Systems								
Environmental Control	380							
Landing Gear								
Instrumentation								
Electrical Power Supply	321							
Propulsion - inert	671	2171						
- usable	Below	Below						
Reaction Control	633							
Communications								
Spares								
Instrumentation	300							
R & D Command	100							
Command Destruct	30							
LEM-1 Totals								
Ascent - inert	3742		-150.1	-1.1	-17.9	1013	623	844
- usable prop.	1433		-155.7	0.0	0.0	747	242	945
Descent- inert	2977		- 84.7	0.0	0.8	719	380	504
- usable prop.	15474		- 73.6	0.0	0.0	10826	7819	5570
Total LEM -	23626		- 92.1	-0.2	- 2.7	13528	14433	13012
Shroud - N_5	4744		-161.4	0.0	0.0	9026	13962	13962
Adapter	300		- 15.0	0.0	0.0	364	182	182

Table 4-1

Item	Weight Lbs.		C.G. Arm - inches *			Moments of Inertia, slug - ft^2		
	Ascent	Descent	X	Y	Z	Ix	Iy	Iz
			* Station 0 taken at LEM Adapter/Little Joe II Separation Line					
Little Joe II:								
Forebody	1919		98.5					
Afterbody	4238		321.5					
Fins - 50 sq. ft.	3029		331.7					
Electrical	134		95.3					
Measurement & Destruct	258		112.7					
Surface Contr. & Guid.	529		278.8					
Reaction Controls	711		318.3					
Algol Instl. prov.	91		348.0					
R.C. Propellant	640		318.3					
Algol cases	20846		243.0					
Algol Propellant	133154		186.6	0.0	0.0	62851	233135	233135
LEM-1/Little Joe II Booster B.O.	61065		87.2	-0.07	-1.06	55194	522548	521103
Lift-off	194219		155.4	-0.02	-0.3	118055	844927	843472

4.2 (con't)

The mass characteristics of the vehicle are presented in Figure C4-1 as a function of time.

4.3 Shroud.-

The LEM-1 shroud configuration will be used.

4.4 Adapter.-

The LEM-1 adapter configuration will be used.

4.5 LEM.-

LEM-2 will have subsystems which are identical to LEM-1 with the exception of the programmed command sequence. The description presented for LEM-1 on pages B4-3 thru B4-21 is therefore directly applicable to LEM-2.

LITTLE JOE II / LEM-II

WEIGHT, BALANCE, AND INERTIA SUMMARY

(CONFIGURATION 4-1)

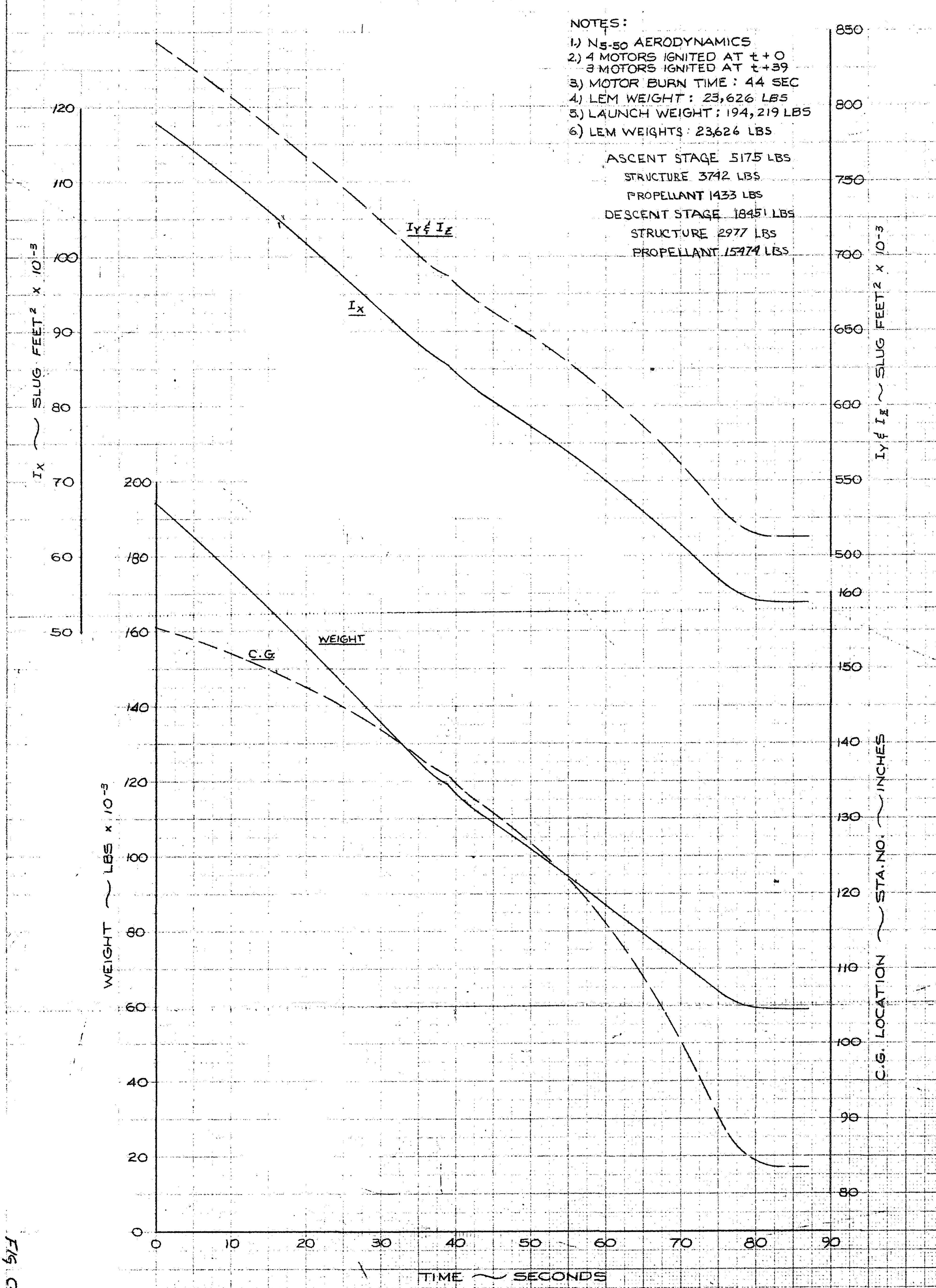


FIG. C4-1

PAGE 1 OF 2
LEM-FT-30

5.0 Flight Test Constraints.-

5.1 The prerequisites for this test are:

a. Analysis of LEM-1 data shall have been completed.

b. Prelaunch checkout shall have been completed.

5.2 P.E.R.T..-

The LEM-2 test is presently being PERTed. The basic flow from checkout to launch is the same as LEM-1 (page B5-1).

6.0 Aerodynamic & Structural Design Criteria.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

7.0 Instrumentation Requirements.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

8.0 Tracking and Support Data Requirements.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

9.0 Prelaunch Operations.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

10.0 Test Management & Organization.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

11.0 Recovery Requirements.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

12.0 Launch Day Requirements.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

13.0 Pad and Range Safety Requirements.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

14.0 Data Handling, Processing Analysis and Reporting.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

15.0 Ground Support Equipment.-

The information presented in this area for LEM-1 is directly applicable to LEM-2.

TABLE OF CONTENTS, SECTION D

TABLE OF CONTENTS, SECTION D (con't)

1.0 Introduction.-

The information presented in the introduction for LEM-1 (page B1-1) is directly applicable to the booster Qualification Test Vehicle.

2.0 Test Objectives & System Priorities.-

2.1 General.-

The purpose of the QTV test is to demonstrate the capability of the launch vehicle, with seven algols and a stabalization and control system to perform adequately the launch phase of the LEM-1 and LEM-2 missions.

2.2 Test Objectives.-

2.2.1 First-Order Test Objectives.-

a. Aerodynamic and control demonstration of the booster plus payload and shroud.

b. Demonstrate the structural integrity of the booster, payload, and shroud.

c. Demonstrate the separation of the shroud from the adapter.

d. Demonstrate the separation of LEM from the adapter and booster retrograde.

e. Demonstrate the operation of the LEM-1 & 2 posigrade motors (if required).

2.2.2 Second-Order Test Objectives.-

a. Evaluate the boost environments.

2.3 System Priorities.-

The booster system priorities are unchanged from the priorities presented for the booster in the LEM-1 test (page B2-1).

3.0 Mission Description.-

3.1 General Flight Plan.-

The flight plan for the QTV will duplicate the LEM-1 flight to the time of booster retrograde.

3.2 Detailed Test Sequence.-

Item	Event
t+0	Ignite 4 algols
t+39	Ignite 3 remaining algols
t+83	BECO
	Initiate shroud separation
t+85	Separate LEM simulator
	Initiate booster retrograde

3.3 Flight Parameters.-

The pertinent flight parameters for the test are given below:

a. Launch

Flight path angle 85°
Azimuth (dependent on the wind)

b. Second Stage Ignition

Mach Number
Altitude (above MSL)
Dynamic Pressure
Range

c. BECO

Mach Number
Altitude (above MSL)
Dynamic Pressure
Range

d. Booster Retrograde

Mach Number
Altitude (above MSL)
Dynamic Pressure
Range

4.0 Description of Test Vehicle.-

4.1 General.-

The test vehicle will be configured as for the LEM-1 test (page B4-) with the LEM test article replaced by mass and inertia simulation.

5.0 Flight Test Constraints.-

5.1 Prerequisites.-

a. Booster stability & control with the LEM-1/boost configuration shall have been confirmed by wind tunnel tests.

b. Structural qualification of the booster adapter and shroud shall have been completed.

c. Booster adapter and shroud separation tests shall have been completed.

d. Prelaunch checkout of the QTV shall have been completed.

5.2 Other Requirements.-

a. The "Requirements for Work and Resources (RFWAR) at White Sands Missile Range" shall have been published. This document shall include the requirements not only for the QTV flight but also for subsequent LEM flights.

6.0 Aerodynamic and Design Criteria.-

This information is given in Section B6.0.

7.0 Instrumentation Requirements.-

7.1 General.-

The instrumentation requirements for the QTV flight will be determined subsequent to NASA approval of the flight. Grumman philosophy is that this flight should be instrumented to demonstrate clear separation of the shroud and the LEM from the booster (a camera) and to evaluate the booster environments preparatory to the LEM-1 test.

8.0 Tracking and Support Data Requirements.-

The requirements in this area will be established subsequent to NASA approval of the flight.

9.0 Prelaunch Operations.-

The requirements in this area will be established subsequent to NASA approval of the flight.

10.0 Test Management & Organization.-

10.1 LEM Project.-

An organization chart of the Grumman - LEM project is given on page B10-2.

10.2 QTV Test Organization.-

This information will be supplied at a later date.

11.0 Recovery Requirements.-

11.1 Test Vehicle.-

At present Grumman Aircraft Engineering Corporation has no requirements for recovery of the test vehicle. The requirements in this area shall be supplied by WSMR and MSC.

11.2 Data Capsule.-

A motion picture camera will be enclosed in a recoverable package. The details are to be determined at a later date.

12.0 Launch Day Requirements.-

The launch day requirements supplied for LEM-1 are applicable to the QTV launch.

13.0 Pad and Range Safety Requirements.-

The LEM-1 pad and range safety requirements are applicable to QTV.

14.0 Data Handling, Processing, and Reporting.—

Data handling, processing, and reporting for the QTV will follow the same procedure presented for LEM-1 (except for the onboard tape recorder).

15.0 Ground Support Equipment.-

The test vehicle ground support equipment requirements have not been established at this time. This information will be determined subsequent to NASA approval of the flight.