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THE APOLLO "A"/SATURN C-1 LAUNCH VEHICLE SYSTEM (U)

by

Saturn Systems Office

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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GEORGE C. MARSHALL SPACE FLIGHT CENTER

MPR-M-SAT-61-5

THE APOLLO "A"/SATURN C-1 LAUNCH VEHICLE SYSTEM (U)

CLASSIFICATION CHANGE

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SATURN SYSTEMS OFFICE

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FOREWORD

This is a compilation of technical information on the Apollo "A"/Saturn C-1 two-stage launch vehicle system. The compilation was made by the Saturn Systems Office in cooperation with the divisions of the George C. Marshall Space Flight Center.

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SECTION 1. (C) INTRODUCTION

1.1. PURPOSE OF THE REPORT

The primary purpose of this report is to provide the prospective bidders for the Apollo spacecraft contract with information on the Saturn C-1/Apollo "A" launch vehicle system. The report is intended for use by the Apollo bidders at a briefing which is planned for August 1, 1961. This report may also be used to acquaint NASA personnel with the Saturn C-1 as applied to project Apollo "A".

1.2. SCOPE AND LIMITATIONS

An attempt has been made to emphasize those areas of the Saturn C-1 launch vehicle system that are closely related to the Apollo "A" spacecraft. These include particularly the design and performance of the two-stage Saturn C-1 launch vehicle as applied to the Apollo "A" low earth satellite mission. Information of a more descriptive nature on some of the many ground support systems of the Saturn is also given.

Recently, re-entry tests of the Apollo "A" command module have been incorporated in the flight test program of the two-stage Saturn C-1 vehicle. Design and performance data for these re-entry tests have not been included in this report.

The report gives selected data on the design of the two-stage Saturn C-1 Block II launch vehicle as of July 1, 1961. It must be realized that the design effort for this Saturn C-1 version was in full progress during the compilation of this report. This design effort, made by NASA/MSFC and its contractors, is planned to lead to a design freeze on September 1, 1961. An improved set of design criteria and data will become available within NASA after that date.

Furthermore, many tests are continuously being performed by NASA and industry in the Saturn R&D program. These tests, which are presently ranging from component tests to the captive test firings of the first stage (S-I), will soon proceed to captive test firings of the second stage (S-IV), to full-scale dynamic tests of the Block I space vehicle, and to the first flight test of the Stage S-I. The evaluation of these tests, and the expected requirements of the Apollo spacecraft will result in a continuous process of improvement of the Saturn C-1/Apollo "A" space vehicle system. This process of improvement will require the coordinated efforts of NASA and industry for a long period of time. The information offered in this report can only serve as a point of departure for this process.

1.3. THE NASA MANNED SPACE FLIGHT PROGRAM

The NASA is engaged in a program of manned exploration of space. This program was initiated with the projects Mercury-Redstone and Mercury-Atlas. The initial objective of the Mercury-Redstone project was accomplished with Cdr. Shepard's flight of May 5, 1961. The objective of the Mercury-Atlas project, that is, orbital flight and recovery of one man, is planned to be reached in 1961.

After completion of the Mercury projects the manned space flight program will continue with the development of both the Apollo spacecraft and of launch vehicles and ground facilities that are capable of supporting multiman crews during scientific and technological missions in earth-moon space. This program will begin with earth satellite flights of extended lifetime, proceed to the lunar circumnavigation, to lunar satellite flights, and ultimately to the lunar landing. The return to earth and the recovery of the manned spacecraft are important parts of these missions. The development of the Apollo spacecraft will proceed in several phases. Each phase, to the extent possible, will serve as qualification for the subsequent phase. The first phase of the spacecraft development is called Apollo Phase "A", or briefly Apollo "A". This phase is described below in more detail.

During Phase "A" the subsequent phases of the project will evolve. The exact number of phases and the scope of each phase will depend primarily on the availability of suitable launch vehicle systems. In consequence, the planning, design, and development of the Apollo spacecraft will be closely related to the planning, design, and development of the launch vehicles and ground facilities.

1.4. OBJECTIVES OF APOLLO "A"

The NASA Space Task Group (STG) has announced the specific objectives of the development of the Apollo "A" spacecraft as:

(1) Qualification of systems and features for the lunar missions within the constraints or the environment of earth satellite orbits of up to 300 nautical miles altitude by manned flights

(2) Qualification of the re-entry heat protection for the lunar missions in unmanned re-entry tests from supercircular and near-parabolic velocities

(3) Study of physiological and psychological reactions and capabilities of a multiman crew in the space environment during extended periods of time (at least two weeks)

(4) Development of flight and ground operational techniques and equipment for the support of space flights of extended periods

(5) Conduct of experimental investigations as needed to acquire information for the subsequent lunar missions.

1.5. LAUNCH VEHICLE FOR APOLLO "A"

The two-stage Saturn C-1 launch vehicle system will be utilized to carry the Apollo "A" spacecraft into the low earth-satellite orbits and re-entry trajectories.

1.6. INITIAL APOLLO "A" PLANNING

It is intended to direct the first year's (1962) effort of the development of the Apollo "A" spacecraft primarily toward an engineering study and design of the prototype. However substantially before completing the prototype design, the design of early so-called "boilerplate" Apollo "A" spacecraft and command modules will be completed, and the fabrication of boilerplate versions will begin. These boilerplate spacecraft will be utilized as described below.

1.7. FLIGHT TESTING OF BOILERPLATE APOLLO "A" SPACECRAFT

Experience has shown that the solution of foreseeable and unforeseeable problems in the development of space vehicles may be accelerated considerably by performing suitable flight tests early in the program. The two-stage Saturn C-1 test flights (Block II) offer an opportunity of conducting such early flight tests of the full-scale Apollo "A" spacecraft in its intended environment.

In consequence, the NASA presently plans to perform test flights of boilerplate Apollo "A" spacecraft as secondary missions during the development test flights of the Saturn C-1 Block II launch vehicles. These flights are presently planned as shown in Table 1.1.

The objectives of these Saturn C-1/Boilerplate Apollo "A" test flights are:

(1) Flight test of the two-stage Saturn C-1 launch vehicle consisting of Stages S-I and S-IV (primary mission)

(2) Demonstration of the compatibility of the Apollo "A" spacecraft and the two-stage Saturn C-1 launch vehicle

(3) Measurement of spacecraft/launch vehicle environment

(4) Validation of systems and subsystems for Apollo "A" prototype and for manned flights.

Table 1.1

BOILERPLATE APOLLO "A"/SATURN C-1 FLIGHT TESTS

SATURN C-1 LAUNCH VEHICLE	LAUNCH DATE	SPACECRAFT		TRAJECTORY
		TYPE	APPROX. WT. (LBS)	
SA-7	Oct. 63	Apollo "A", Nonrecoverable, Boilerplate	20,000	Earth Satellite at 100 to 300 NMi Altitude
SA-8	Dec. 63	Same as Above	Same	Same as Above
SA-9	Feb. 64	Apollo "A", Command Module Only, Recoverable, Boilerplate	7,500	Ballistic Re-entry at Near Parabolic Velocity
SA-10	Apr. 64	Same as Above	Same	Same as Above

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1.8. APOLLO "A" SPACECRAFT ASSUMPTIONS

As previously stated, the design of the prototype Apollo "A" spacecraft will not be completed before the end of the first year's effort, which may occur in 1962. In order to proceed with the design of the launch vehicle and with studies of the performance of the entire space vehicle, it was necessary to make certain early assumptions for the spacecraft. These assumptions have been furnished by NASA/STG in the form of a sketched preliminary layout of the boilerplate Apollo "A" spacecraft (Fig. 1.1) and an associated table of the weights and centers of gravity of its main components (Table 1.2). It should be re-emphasized that the spacecraft assumed in this report can only be representative of some of the early boilerplate versions, whose purpose has already been described. The actual prototype of the Apollo "A" may have an entirely different configuration, which, however, has to be within the constraints of the Saturn C-1 launch vehicle system.

Table 1.2

WEIGHTS AND CG OF ASSUMED APOLLO "A" SPACECRAFT

ITEM	LAUNCH WEIGHT (lbs)	CG Location (x on Fig. 1.1) (in)
Command Module	7,900	+ 115
Equipment-Propulsion Module (Less Main Motors)	3,500	+ 185
Main Motors	2,900	+ 184
Space Laboratory	3,000	+298
Launch Escape Propulsion*	3,700	- 119
Total Launch Weight	21,000	+ 121

* Since the launch escape propulsion system is separated after S-I flight, the satellite payload penalty is only about 200 pounds. The spacecraft weight in orbit is only 17,300 pounds.

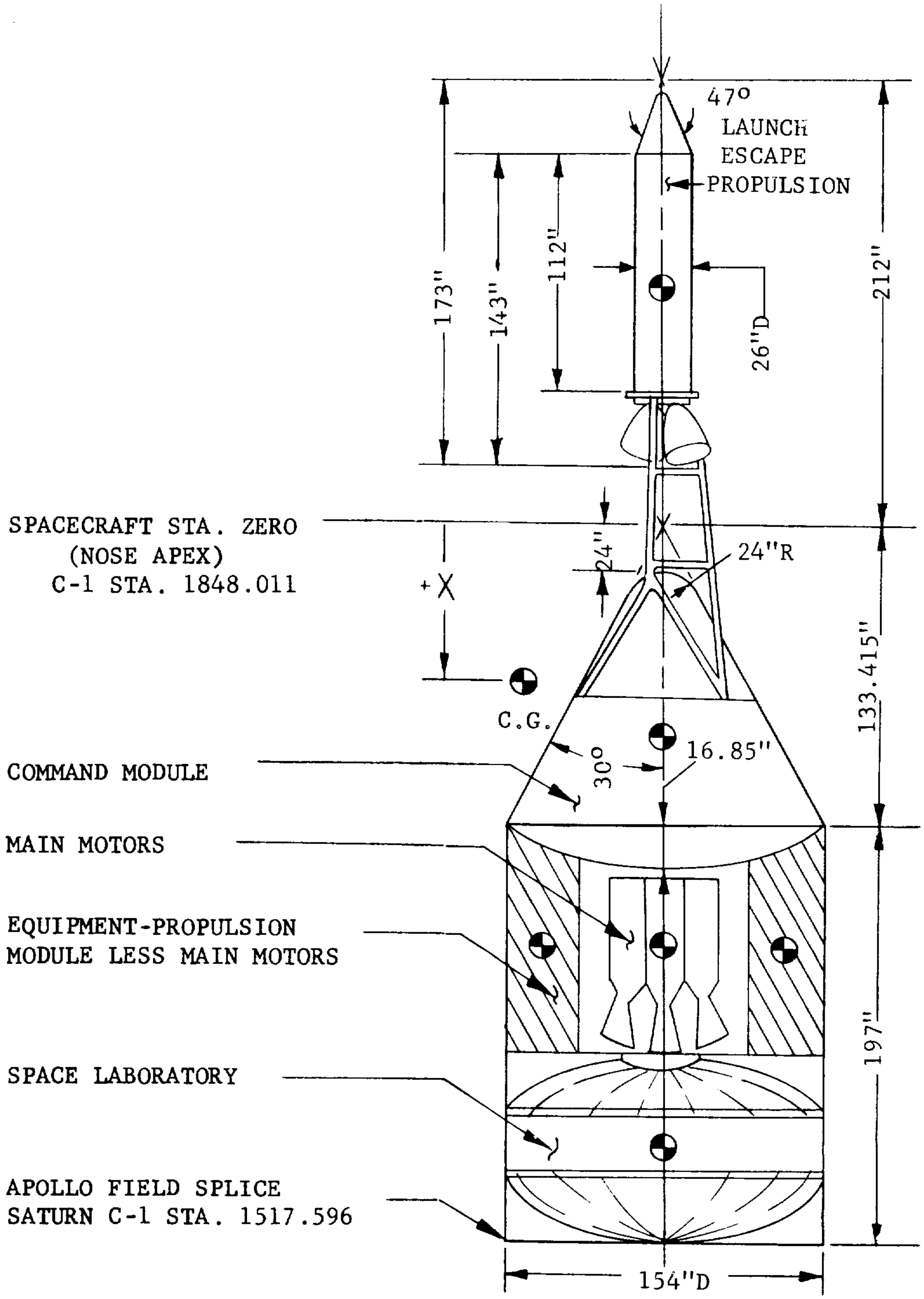


Fig. 1.1 ASSUMED APOLLO "A" SPACECRAFT CONFIGURATION

SECTION 2. (C) SATURN LAUNCH VEHICLE PROGRAM

2.1. BACKGROUND AND TECHNICAL ORIGINS OF THE SATURN PROGRAM

Studies were initiated by members of the Army Ballistic Missile Agency (ABMA) under Dr. von Braun in April 1957 to establish possible vehicle configurations to satisfy a payload capability of 20,000 to 40,000 pounds for orbital missions and 6,000 to 12,000 pounds for escape missions. These were urgent requirements for space missions of the near future. Immediate acceleration of the development of high thrust engines was also considered essential for future programs.

The initial studies, aiming at boosters in the 1.5 million-pound thrust class, placed special emphasis on arriving at an acceptable propulsion system. Clustering of four 380,000 pound thrust Rocketdyne E-1 engines, which were in early stages of development, was first considered.

In July 1958, representatives of the Advanced Research Projects Agency (ARPA) showed interest in a clustered booster with 1.5 million pounds thrust achievable with available engine hardware. ARPA formally initiated the program by issuing ARPA Order 14-59 on August 15, 1958. The immediate goal was to demonstrate the feasibility of the engine clustering concept with a full-scale captive test firing using Rocketdyne H-1 engines and available propellant containers. In September 1958 the program was extended to include four flight tests of the booster. A separate ARPA Order No. 47-59 dated December 11, 1958, requested that the Army Ordnance Missile Command (AOMC), on behalf of ARPA, perform the design, construction, and modification of the Army Ballistic Missile Agency Captive Test Tower and associated facilities required in this booster development and determine design criteria for suitable launch facilities.

In this context it may be interesting to note that project Mercury was initiated in October 1958.

In November 1958, development of a clustered booster to serve as the first stage of a multistage carrier vehicle capable of performing advanced space missions was approved.

In addition to the clustered booster concept with eight H-1 engines, feasibility studies and tests of high thrust single rocket engines were also made. A contract was let, in January 1959, to the Rocketdyne Division of North American Aviation, Inc., for the development of a regeneratively-cooled pump-fed 1.5 million-pound-thrust lox/RP-1 engine, called the F-1 engine.

The clustered booster project was unofficially known as Juno V until, on February 3, 1959, an ARPA memorandum made the name Saturn official for the project.

In March 1959, a Saturn system study, outlining various upper stage configurations and indicating that either an Atlas or Titan as a second stage would be possible, was submitted to ARPA. ABMA was informed by ARPA in May 1959 that a decision had been reached to use modified Titan hardware for the Saturn second stage and that the third stage should utilize the Centaur.

However, in July 1959, ARPA informed AOMC that all in-house and contract effort and other expenditures relating to the modified Titan second stage were to be suspended, unless the work was not directly connected with the stage diameter. This was necessitated by a new requirement to relate Saturn second stage planning to other National space programs.

On September 24, 1959, ARPA submitted guidelines and requested a study to determine the two best configurations for increasing Saturn capabilities for National Aeronautics and Space Administration (NASA) payloads.

As a result of the Presidential Order proposing transfer of the Development Operations Division of ABMA to NASA, an interim agreement was made on November 25, 1959, between ARPA-NASA and the Department of Defense. This agreement placed the technical direction of the Saturn program under NASA. ARPA was to retain administrative direction. Committees, composed of representatives from ARPA, NASA, ABMA, and the Air Force, were established to provide technical assistance to NASA.

On December 7, 1959, ARPA-NASA requested that AOMC prepare an engineering study for a Saturn configuration consisting of the clustered first stage with a second stage of four 20K pound-thrust hydrogen/oxygen engines, and a third stage of two 20K pound-thrust hydrogen/oxygen engines. Results of this study were formally transmitted on December 28, 1959.

A decision on the overall Saturn upper stage configuration was achieved with the report from the Saturn Vehicle Team (Silverstein Committee) to the Administrator, NASA, on December 15, 1959.

The Silverstein Committee Report outlined the general course and the time schedule to be followed for Saturn in the future. Primary in its recommendations was the utilization of liquid hydrogen/liquid oxygen upper stages, and the development of the Saturn building block approach. (Leading from the C-1 to the C-3 Saturn configuration, this approach utilizes developed hardware for each succeeding step as far as possible.)

The report also recommended the immediate initiation of the development of a 150 to 200K hydrogen/oxygen rocket engine. The report, as approved by the Administrator, NASA, and with some adjustments made, was the plan subsequently followed in the Saturn program.

On July 28, 1960, the Douglas Aircraft Company was awarded the contract for the development and fabrication of the second stage (S-IV) of the recommended Saturn C-1 configuration. This stage was to be designed with four Pratt & Whitney 17.5K hydrogen/oxygen engines (LR-119), which were similar to the Centaur 15K engine (LR-115).

On September 1, 1960, a contract between NASA and the Rocketdyne Division of North American Aviation, Inc., was signed for the development of a 200K cryogenic hydrogen/oxygen engine, designated the J-2 engine. This engine will be used in later Saturn configurations.

A study contract for the third stage (S-V) of the Saturn C-1 configuration was awarded to Convair Astronautics on October 21, 1960. This stage was for the escape-type missions of the C-1 with a design very similar to the Centaur.

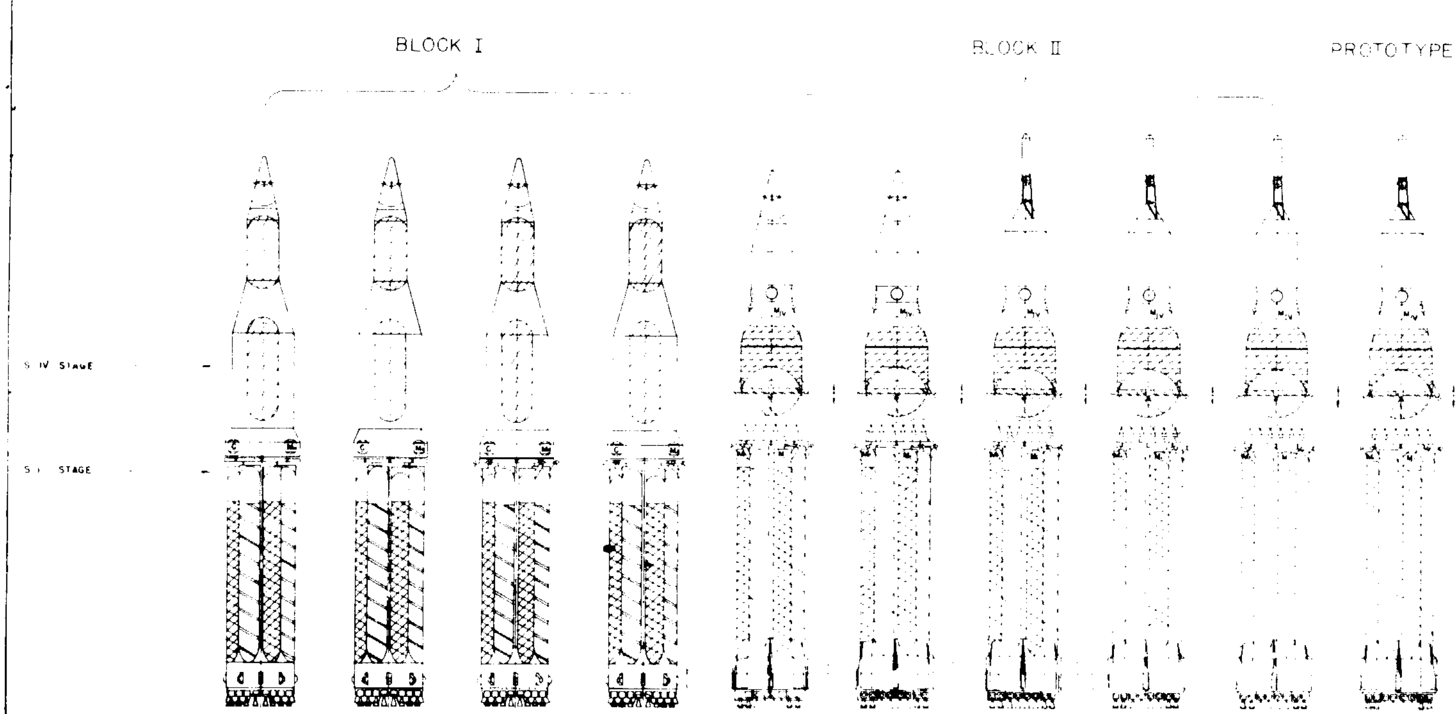
2.2. SATURN C-1 RESEARCH AND DEVELOPMENT PROJECT

In March 1961 a review of the mission requirements and of the development status of the entire Saturn program was made by NASA. A modification of the Saturn C-1 R&D project plan was one of the results of this review. Based upon the determination that the primary mission of the Saturn C-1 was to launch manned spacecraft into low earth satellite orbits, the development of the third stage (S-V) was postponed.

The design of the second stage (S-IV) was modified to utilize six Pratt & Whitney 15K hydrogen/oxygen engines (called RL-A3, or recently just A-3 engines). This modification increased the thrust of the C-1 second stage from 70K to 90K and allowed the combination of the S-IV and Centaur engine developments into one development of a common engine.

The design of the first stage (S-I) of the Saturn C-1 configuration was modified mainly by an increase of the propellant capacity and the addition of fixed tail fins, which improve the control stability of the space vehicle. The Saturn C-1 project development plan is now proceeding in two launch vehicle blocks, called Block I (launch vehicles SA-1 to SA-4) and Block II (launch vehicles SA-5 to SA-10) (Fig. 2.1). The Stage S-IV and the modified Stage S-I are introduced in Block II of the plan. The main differences between the characteristics of the Saturn C-1 Block I and Block II vehicles are listed in Table 2.1.

SATURN SER NO	SA-1	SA-2	SA-3	SA-4	SA-5	SA-6	SA-7	SA-8	SA-9	SA-10
LAUNCHING DATE	OCT 61	MAR 62	OCT 62	MAR 63	JUN 63	AUG 63	OCT 63	DEC 63	FEB 64	APR 64
PREPARE FOR LAUNCH	AUG 61	JAN 62	AUG 62	JAN 63	APR 63	JUN 63	AUG 63	OCT 63	DEC 63	FEB 64
SHIPPING DATE	AUG 61	JAN 62	JUL 62	DEC 62	APR 63	MAY 63	JUL 63	SEP 63	NOV 63	JAN 64
FINAL CHECKOUT	MAY 61	OCT 61	MAR 62	SEP 62	JAN 63	MAR 63	APR 63	AUG 63	OCT 63	DEC 63
S-I STATIC TEST	MAR 61	SEP 61	MAR 62	AUG 62	NOV 62	JAN 63	MAR 63	NONE	NONE	NONE
CHECKOUT	JAN 61	AUG 61	FEB 62	JUL 62	OCT 62	DEC 62	FEB 63	JUL 63	SEP 63	NOV 63
START ASSEMBLY	MAY 60	DEC 60	SEP 61	FEB 62	MAY 62	JUL 62	SEP 62	FEB 63	APR 63	JUL 63
S-IV STAGE DEL. DATE	——	——	——	——	MAR 63	MAY 63	JUL 63	SEP 63	NOV 63	JAN 64
LAUNCH COMPLEX	VLF 34	VLF 34	VLF 34	VLF 34	VLF 37B	VLF 37A	VLF 37B	VLF 34	OPTIONAL	OPTIONAL
MISSIONS	S-I PROPULSION, STRUCTURE, & CONTROL FLIGHT TEST WITH DUMMY UPPER STAGES				S-I PROPULSION, STRUCTURE & S-IV CONTROL FLIGHT TEST		S-I PROPULSION, STRUCTURE, & CONTROL FLIGHT TEST S-IV ORBITAL TEST OF APOLLO BOILERPLATE CAPSULE			



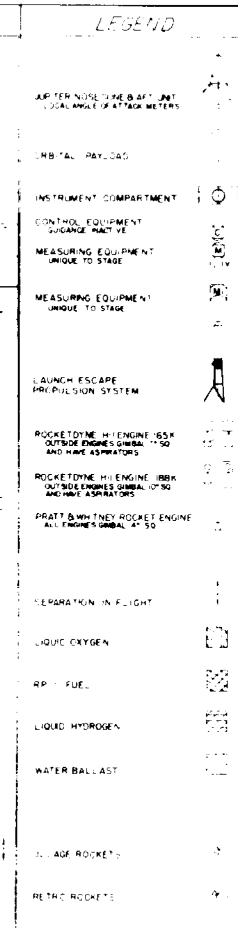


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Fig. 2.1 SATURN R&D PROGRAM, SA-1 THROUGH SA-10

Table 2.1

CHARACTERISTICS OF SATURN C-1 BLOCK I AND II VEHICLES

ITEM	BLOCK I	BLOCK II
<u>LAUNCH SYSTEM</u>		
Launch Support	Four Hold Down Arms and Four Retractable Arms	Eight Stationary Support and Hold Down Arms Accommodate Fins of S-I
Ground/Vehicle Connections	Partial Disconnect Prior to Liftoff. Final Disconnect at Liftoff	Total Disconnect at Liftoff
<u>SPACE VEHICLE</u>		
Design Bending Moment at Maximum Station	17×10^6 inch-lbs	55×10^6 inch-lbs

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Table 2.1 (Continued)

ITEM	BLOCK I	BLOCK II
STAGE S-I		
Thrust, Engines	8 x 165K lbs, H-1	8 x 188K lbs, H-1
Propellants	Lox/RP	Lox/RP
Propellant Capacity	750K lbs	850K lbs
Propellant Suction and Interconnect Lines	Fluid Transfer Type	Sump Design
Fuel Pressurization Gas (N_2)	In 48 Spheres of 1 cu ft Volume	In 2 spheres of 18 cu ft Volume
Gimbal Pattern	7° Square	10° Square
Tail Section	Octagonal, Outrigger Truss Construction	Cylindrical, Shear Panel Construction. Takes Structural Loads From Tail Fins.
Tail Fins	None	Four 182 sq ft Detachable Fins and Four 21 sq ft Launch Support Stubs to improve aerodynamic stability.
S-I/S-IV INTERSTAGE	Adapter Section of 90 in. Length	Adapter Section Eliminated. S-IV Mates to Top of S-I Spider Beam.

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Table 2.1 (Continued)

ITEM	BLOCK I	BLOCK II
STAGE S-IV	Dummy Stage	
Thrust, Engines		6 x 15K lbs, RL10-A-3
Propellants		Lox/Liquid Hydrogen
Propellant Capacity		100K lbs
Gimbal Pattern		4° Square
STAGE S-V	Dummy Stage	None
Telemetry, Tracking, Inertial Guidance, Control and Instrumentation	In Four Canisters Located in 90-in. S-I/S-IV Adapter Section	Stage Measuring and Tracking Equipment Located Separately in Stage S-I and S-IV. Vehicle Inertial G&C and Instrumentation Located in Instrument Compartment Above Stage S-IV.

Table 2.1 (Continued)

ITEM	BLOCK I	BLOCK II
PAYLOAD	Dummy Payload Consisting of: (a) Jupiter Nose Cone (b) Jupiter Aft Unit (c) Payload Adapter	Apollo "A" Spacecraft with Launch Escape Propulsion System, Boiler-Plate Version (Beginning with SA-7)
MISSIONS		
Primary:	Single Stage Launch Vehicle Test	Two-Stage Launch Vehicle Test
Secondary:	None	(a) Demonstration of Spacecraft Launch Vehicle Compatibility (b) Test of Apollo "A" System and Subsystems (c) Measurement of Spacecraft/ Launch Vehicle Environment
Trajectory:	Launch into Ballistic Trajectory of about 200 NMi	Launch into Earth Satellite Orbit of 100 to 300 NMi Altitude. Re-entry Tests of the Apollo Command Module are also Under Consideration)

2.3. SATURN C-2 AND C-3 CONFIGURATIONS

The Saturn C plan was conceived by the Saturn Vehicle Team (Silverstein Committee) in December 1959. As mentioned above, this committee recommended the initial development of the C-1 configuration to be followed by the C-2, using a new second stage. The Saturn C-3 vehicle, as originally envisioned, would have followed the C-2 and would have utilized the upper stages developed for C-2. More recent concepts envision the Saturn C-3 vehicle as a large step forward, bridging the gap between the Saturn class of vehicles and the Nova class, and at the same time providing a heavy weight lifting capability at an early date so that certain spacecraft missions can be met.

All three Saturn C configurations are illustrated in Figure 2.2. Only the Saturn C-1 configuration is described in this report.

2.4. SATURN PROGRAM IMPLEMENTATION

2.4.1. General. The Saturn vehicle and related facilities and Ground Support Equipment (GSE) are composed of a multitude of intricate parts - each dependent upon the functional ability of other parts and subsystems to perform properly. The complexity of the system is compounded by the fundamental necessity of having several large organizations (Government and Industry) combine and integrate their resources and effort toward the common goal. To perform this task effectively takes a considerable amount of effort in the field of technical and management liaison. This paragraph outlines the organizational structure, the review boards, and working groups which have been established for the liaison and technical coordination needed to pursue the Saturn objective. Bidders should prepare their management and cost proposals accordingly.

To aid all potential contractors in becoming familiar with the organization of management of the Saturn system development, a brief description of the plan follows.

2.4.2. Saturn Management Plan. The purpose of the Saturn Management Plan is to establish channels and methods to be used in the development of the Saturn Vehicle System.

Within the National Aeronautics and Space Administration the overall responsibility for program policy direction rests with the Director, Office of Launch Vehicle Programs (OLVP) at NASA Headquarters. Program management is the responsibility of the Director, Marshall Space Flight Center, and is carried out by the Director, Saturn Systems Office. An organization chart for MSFC is shown in Figure 2.3.

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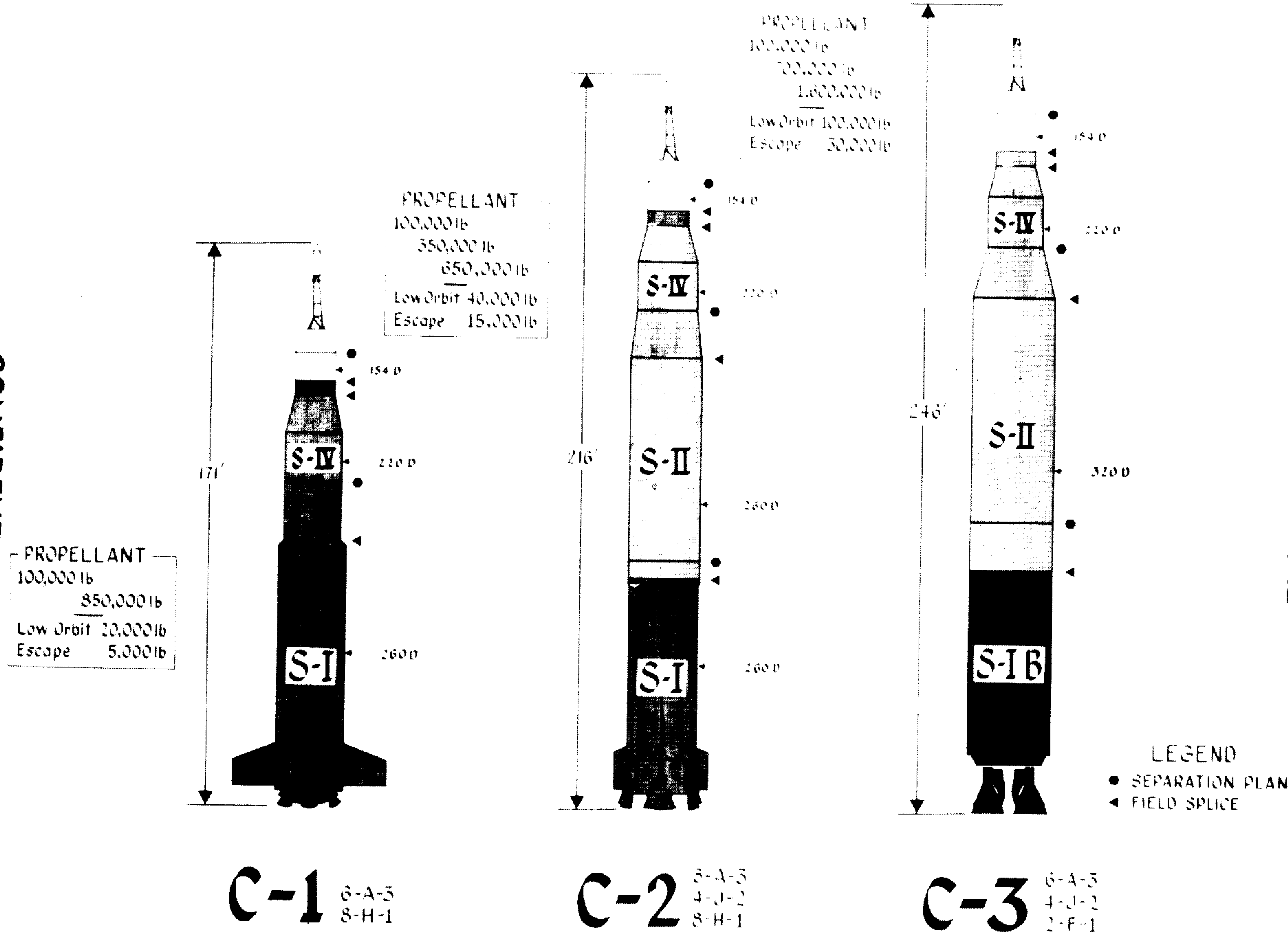

FIG. 2.2 SATURN C CONFIGURATIONS

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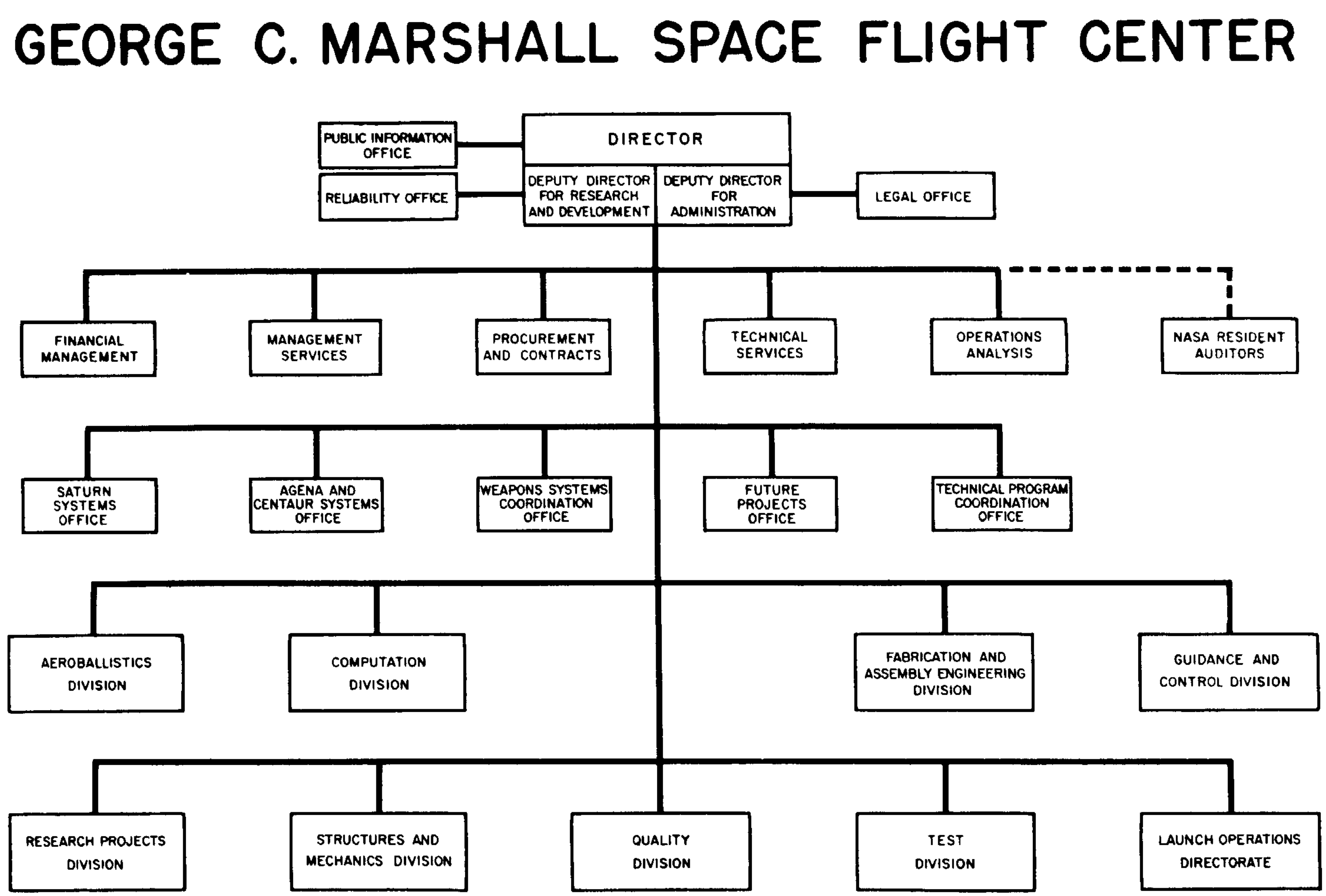

Fig. 2.3 GEORGE C. MARSHALL SPACE FLIGHT CENTER ORGANIZATION

A management system is pursued by the Director, Saturn Systems Office, through technically oriented review boards and active direction of in-house and contractor efforts. Fiscal activities are likewise conducted by the Director, Saturn Systems Office, within budgetary guidelines from OLVP. In addition to quarterly Program Reviews and Semi-Annual Technical Reviews at NASA Headquarters, the Saturn program is supported by two MSFC review boards and eight technical working groups. These policy and technical management tools supplement the more detailed analysis and direction performed by elements of the Saturn Systems Office. Frequent technical scheduling and programming meetings constitute the liaison line between the in-house effort and contractor effort. The Director is thus assured of accurate, up-to-date information for timely decision and dissemination of all essential program directive guidance.

The main tools within the MSFC Management Plan can be divided into "Action" functions and "Coordination, Review and Analysis" functions. Figure 2.4 illustrates graphically the organization used to implement these functions. The action functions are made up of the nine divisions and the Saturn Systems Office plus support from other MSFC offices. The coordination, review and analysis functions are accomplished by the Saturn Coordination Board on both program and vehicle integration.

The Saturn Systems Coordination Board meets bi-weekly at MSFC as required. The Director, MSFC, is the chairman with membership composed of directors or chiefs of MSFC organizational segments, working group chairmen, and others as required. The responsibility of the Board is to review and coordinate the composite vehicle system development and establish policies to aid its development.

The Saturn Program Coordination Board meets quarterly at MSFC. The Director, MSFC, is chairman of the board and membership is the same as that for the Systems Coordination Board. This group reviews and coordinates program aspects as well as prepares the subjects to be discussed in the NASA Headquarters Quarterly Program Review Meeting.

2.4.3. Technical Coordination and Liaison Meetings. To further implement the Saturn Management Plan and to assist the Coordination Boards in arriving at policy decisions on system development, the following technical meetings are held.

(1) S-I and S-IV Stage Meetings. Periodic meetings are held for the purpose of reviewing efforts of MSFC organizational segments and the individual stage contractors in the R&D phase of the Saturn program. Recommendations or direct solutions to current or anticipated problems are presented to the Director, Saturn Systems Office, by these groups after thorough review of all the facts.

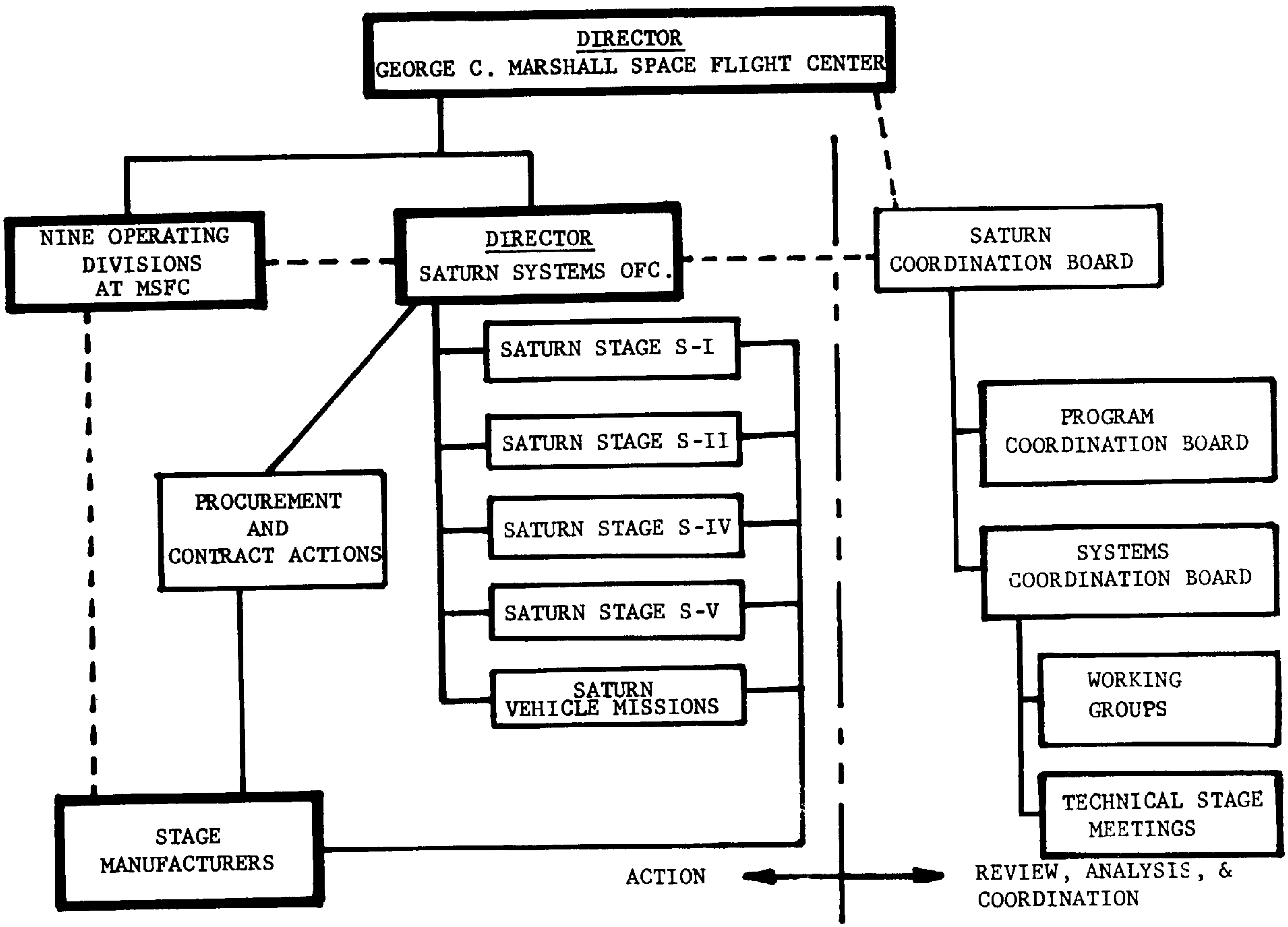

Fig. 2.4 SATURN VEHICLE SYSTEM MANAGEMENT PLAN

(2) Saturn Working Group Meetings. The following eight working groups have been established and hold periodic meetings in order to assure maximum integrated consideration of Saturn technical characteristics:

a. Vehicle Mechanical Design Integration

b. Vehicle Dynamics and Control

c. Vehicle Instrumentation

d. Electrical System Design Integration

e. Vehicle Assembly

f. System Checkout and Preflight Testing

g. Launch Operations

h. Flight Evaluation

These working groups integrate the available experience and efforts of MSFC and stage contractors toward the solution of individual stage interface, composite vehicle system, and launch facility technical problems. It could be stated that the "working groups" provide a means to pool the expert technical know-how of industry and government in specific functional areas so that technical problems arising during development can be analyzed by the best Saturn vehicle system talent in the country at the earliest possible time.

The authority of a working group is vested in the chairman whose function includes the initiation of approved actions (within the scope of his working group) to correct problems. The actions are then implemented either by employing contractors through respective project managers of the Saturn Systems Office or by other NASA organizational elements through their directors or chiefs in accordance with the assigned missions.

A description of working group responsibilities are listed below:

(1) Vehicle Mechanical Design Integration. Define and resolve those design problems, involving the overall operation philosophy, that relate to mechanical systems for stage mating, final checkout, and firing.

(2) Vehicle Dynamics and Control. Resolve the dynamics and control problems associated with flight conditions and stage separation. Recommend procedures and measurements necessary to define environmental conditions, and selection of trajectories.

(3) Vehicle Instrumentation. Investigate and resolve problems peculiar to the overall instrumentation system such as measuring, telemetry, tracking, and range safety, and insure mutual exchange of instrumentation information for the various stages.

(4) Electrical System Design Integration. Resolve problems peculiar to the overall electrical system such as defining, analyzing and implementing the requirements for system checkout during stage mating, final checkout and firing, and establishing overall electrical systems compatibility.

(5) Vehicle Assembly. Define and offer solutions for problems which may result from tooling, fabrication, and assembly concepts in connection with interfaces between stages.

(6) System Checkout and Preflight Testing. Coordinate methods and procedures for vehicle checkout to make them as uniform as possible. Investigate the possible application of multiple purpose checkout and automation for use at the stage contractor, MSFC, and the launching site.

(7) Launch Operations. Define, establish, and coordinate methods and procedures for implementation of individual stage checks and complete vehicle checks in the industrial area at Cape Canaveral and at the Launch Complex.

(8) Flight Evaluation. Prior to each flight, establish coordinated evaluation procedures for all stages. After each flight, conduct the early engineering evaluation and inform representatives of all design groups concerned with results of the flight.

2.4.4. NASA Sequenced Milestone System (SMS). The NASA and its contractors will implement the Saturn projects with the aid of the Sequenced Milestone System which is basically designed to serve the NASA Project Manager. At the same time it provides the selected information flow necessary for project monitoring and coordination within NASA and project coordination among all participating organizations.

SMS is an adaption of the well established Navy PERT (Air Force PEP) System. It includes utilization of the following basic PERT/PEP features:

(1) A network of jobs and milestones placed in dependent sequence

(2) Critical path and slack determination

(3) The use of data processing equipment.

The principal difference between SMS and PERT/PEP is that SMS uses a single time estimate for jobs in the network, whereas PERT/PEP uses a three time estimate system. SMS also provides for computer generation of top level milestone schedules, which are to serve as the master schedules for the project, and for communicating changes to those schedules.

The distinguishing features of SMS are as follows:

(1) Dependent Sequence Network. A flow plan in which the interrelationships of jobs and milestones are reflected in a dependent sequence order

(2) Critical Path. The sequence of interconnected jobs and milestones within the dependent sequence network which will require the greatest time to reach the end objective

(3) Slack Time. Positive slack is the extra time that can be taken in completing a sequence (or path) of jobs and milestones without jeopardizing accomplishment of the end objective on the date scheduled. Negative slack exists when the critical path is expected to require more time than is allowed by the scheduled date for the end objective.

It is intended that future improvements to the system will be made to provide the project manager with additional computer generated aids to assist in:

(1) Correlating financial data with technical progress

(2) Determining and allocating manpower resources

(3) Determining facility loading and requirements.

A manual further describing the SMS will be published by NASA in the near future.

SECTION 3. (C) THE APOLLO "A"/SATURN C-1 SPACE VEHICLE

3.1 DESCRIPTION

The typical two-stage Apollo "A"/Saturn C-1 space vehicle (Fig. 3.1) to be used for satellite missions (e.g. SA-7) consists of the following major components:

(1) Stage S-I. The first stage (S-I) is powered by eight Rocketdyne H-1 engines with a total thrust of 1.5 million pounds. Propellants for these engines consist of 850,000 pounds of lox and RP-1. Tail fins are provided to improve the stability of the first stage flight. The diameter of the propellant tanks is about 257 inches.

(2) S-I/S-IV Interstage Section. This is a cylindrical section of 220 inches diameter that remains with the Stage S-I after inflight separation of the second stage (S-IV).

(3) Stage S-IV. The second stage (S-IV) will be powered by six Pratt & Whitney RL10-A-3 engines with a total thrust of 90,000 pounds. Propellants for these engines consist of 100,000 pounds of lox and liquid hydrogen. The rear and forward interfaces of the S-IV have nominal diameters of 220 inches and 154 inches, respectively.

(4) Instrument Unit. The instrument unit (previously called instrument compartment) contains the Saturn C-1 launch vehicle guidance and instrumentation. It has a nominal diameter of 154 inches.

(5) Apollo "A" Spacecraft. The boilerplate Apollo "A" spacecraft will be the initial payload of the Saturn C-1 launch vehicle. The first flight of this configuration is planned for the C-1 test flight SA-7.

3.2. WEIGHT SUMMARY

The weight summary for the Apollo "A"/Saturn C-1 is given in Tables 3.1, 3.2, and 3.3 for a typical flight into a nominal 100 nautical mile earth satellite orbit.

3.3. MASS CHARACTERISTICS

The roll and pitch moments of inertia and the weight and center of gravity shifts throughout a flight are plotted in Figures 3.2 through 3.5.

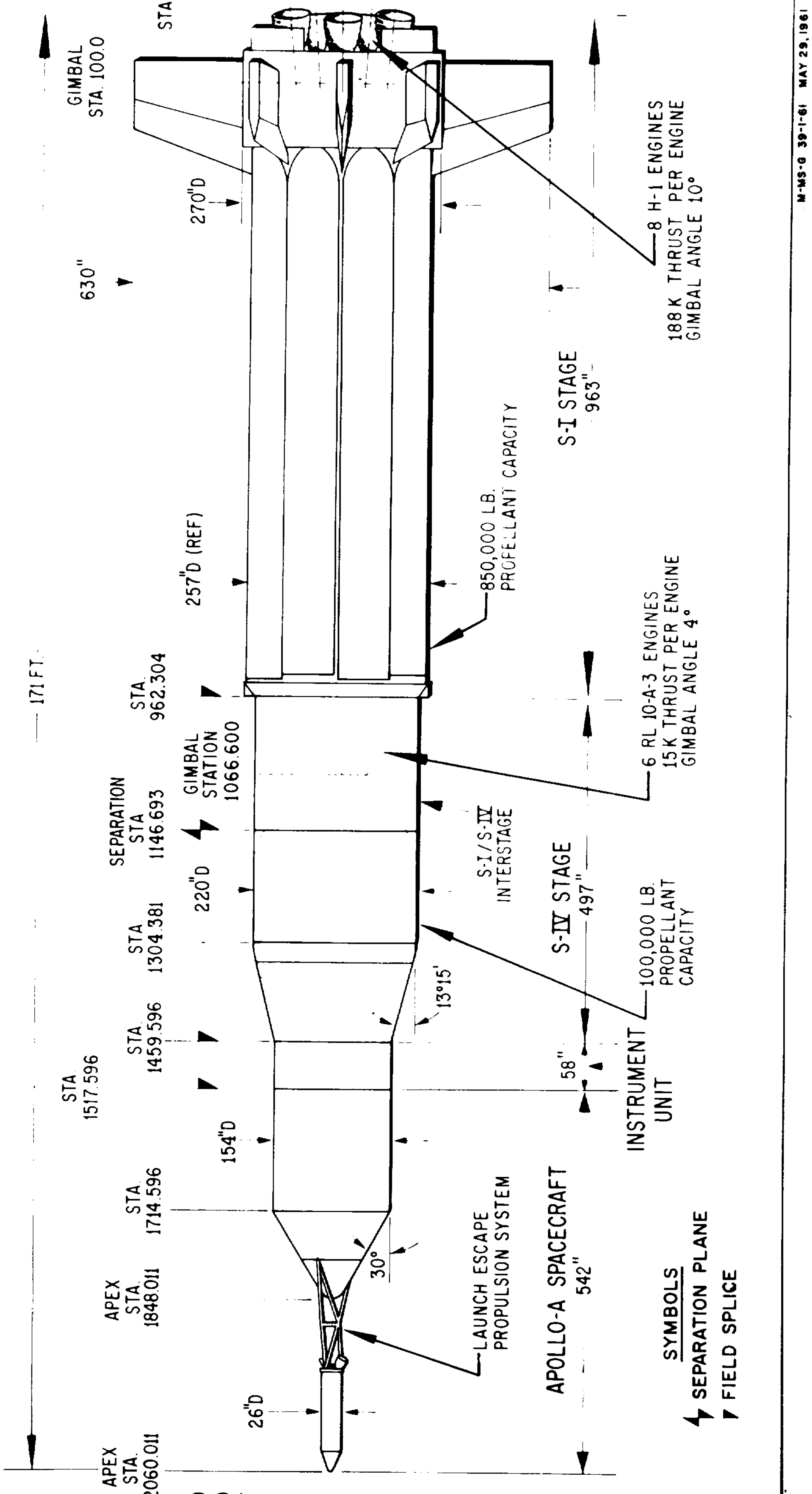

FIG. 3.1 APOLLO "A"/SATURN C-1 SPACE VEHICLE.

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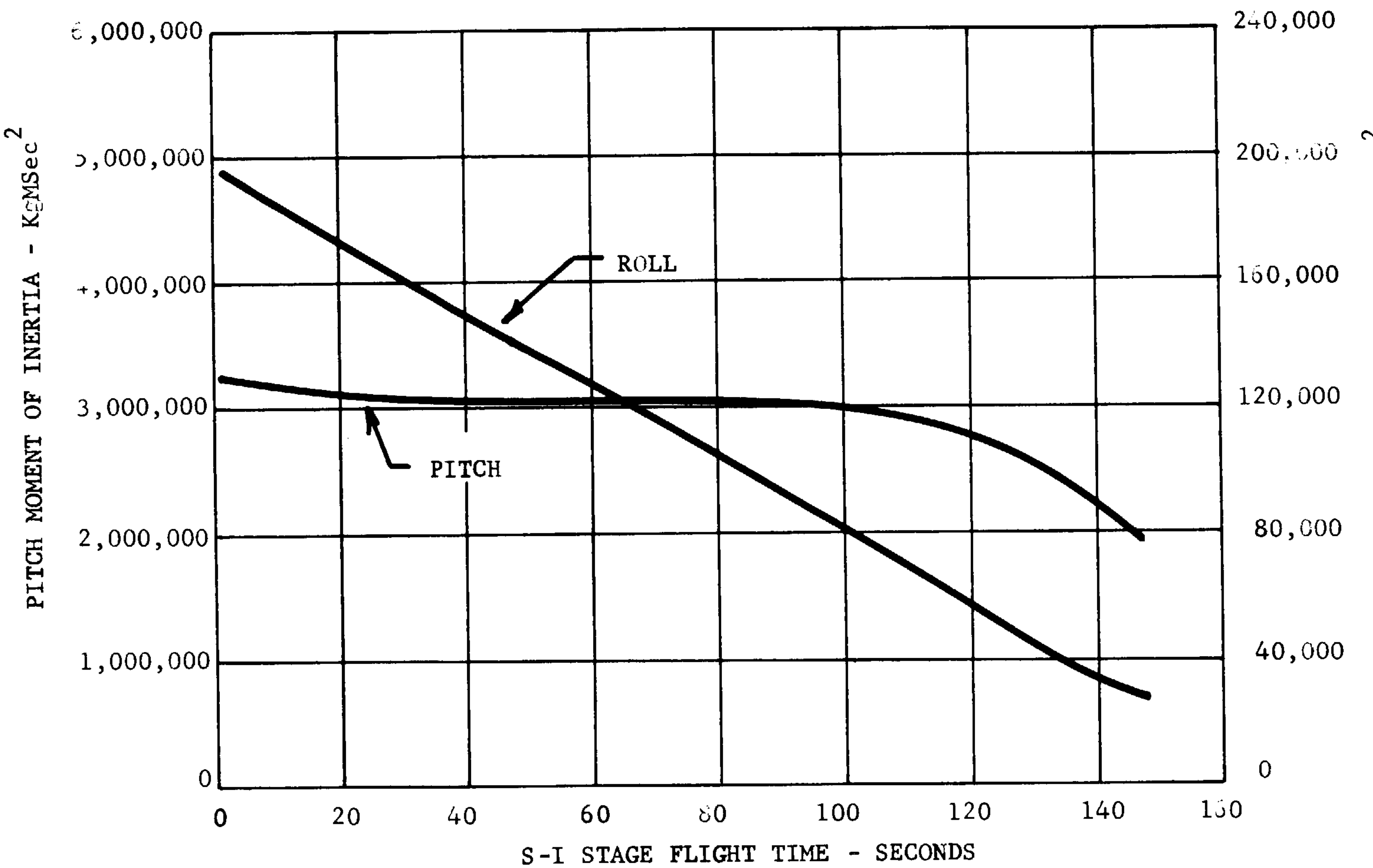

FIG. 3.2 MASS MOMENT OF INERTIA VERSUS BURNING TIME, SATURN C-1 VEHICLE FIRST STAGE FLIGHT

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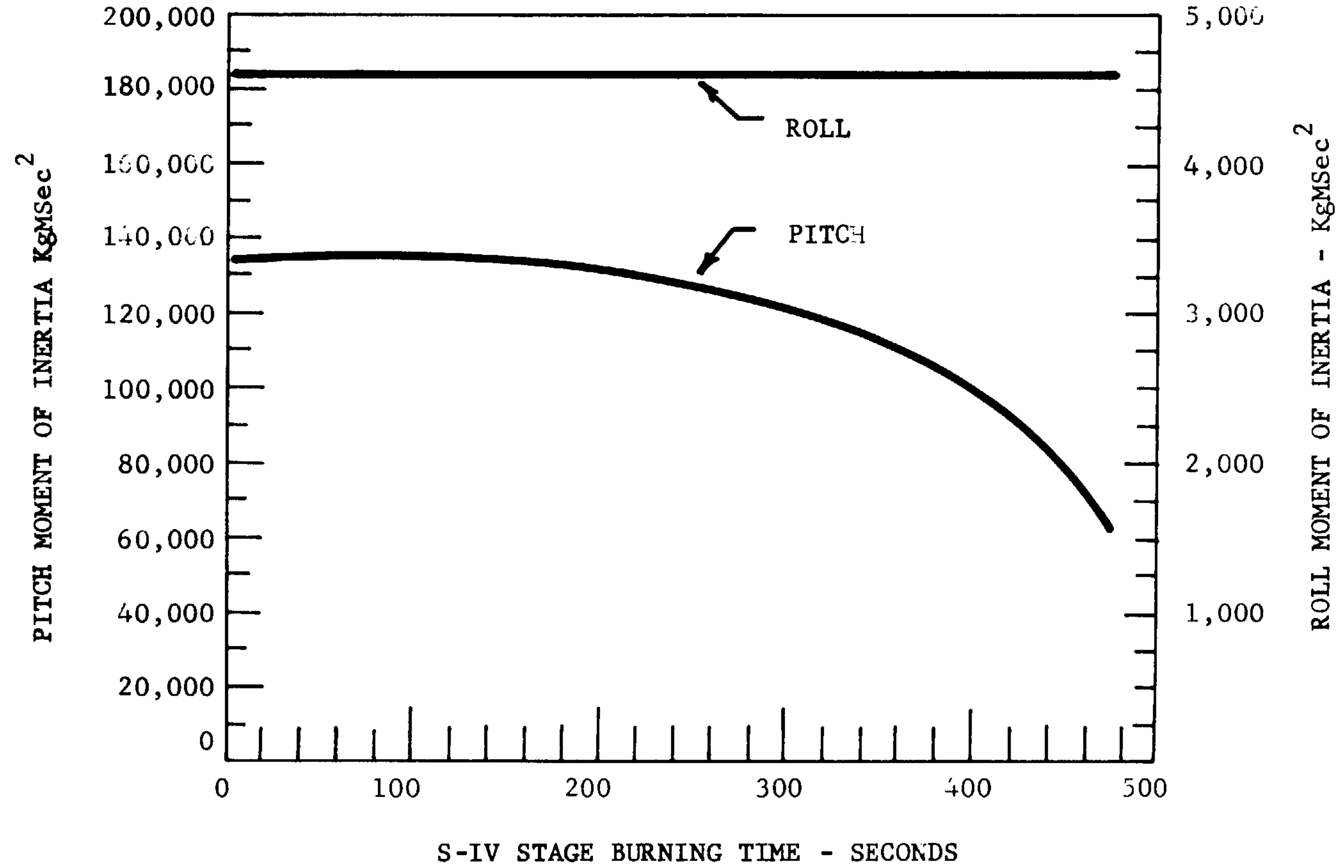

FIG. 3.3 MASS MOMENT OF INERTIA VERSUS BURNING TIME, SATURN C-1 VEHICLE SECOND STAGE FLIGHT

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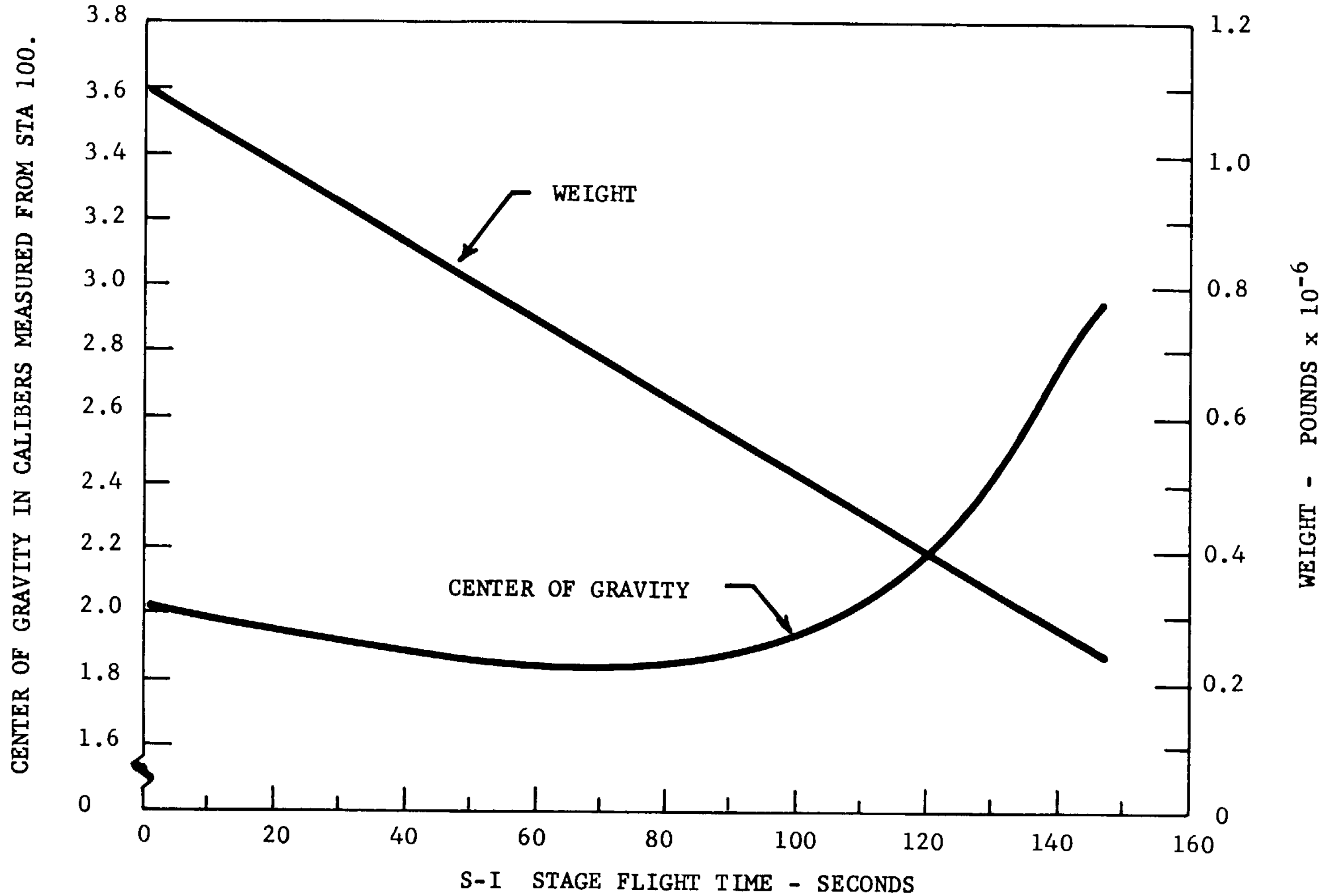

1 CAL = 257"

FIG. 3.4 CENTER OF GRAVITY SHIFT AND DECREASE IN WEIGHT, SATURN C-1 VEHICLE FIRST STAGE FLIGHT

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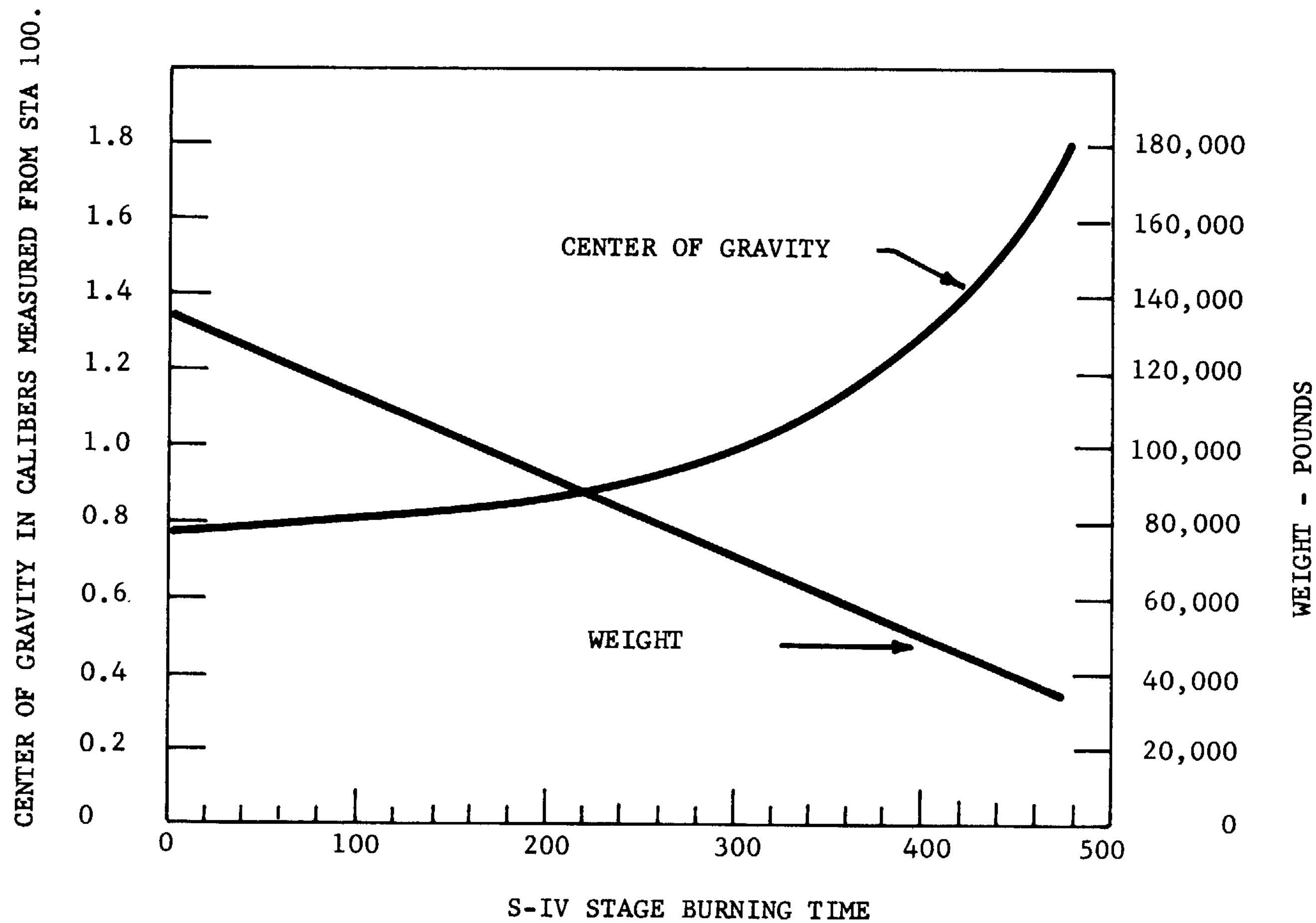


1 CAL = 220"

FIG. 3.5 CENTER OF GRAVITY SHIFT AND DECREASE IN WEIGHT, SATURN C-1 VEHICLE SECOND STAGE FLIGHT

3.4. INBOARD PROFILE

The inboard profile of the Apollo "A"/Saturn C-1 space vehicle and some sections of it are shown in Figures 3.6.A through 3.6.G. The call-outs for the inboard profile are given in Table 3.4.

(C) Table 3.1

APOLLO "A"/SATURN C-1 WEIGHT SUMMARY (FIRST STAGE FLIGHT)
FOR TYPICAL INJECTION INTO 100 NAUTICAL MILE ORBIT

ITEM	LIFTOFF WEIGHT (LBS)	CUTOFF WEIGHT (LBS)
Liftoff for Second Stage Flight	137,149	137,149
S-IV Chill-Down & Vented Gases	357	-
S-I/S-IV Interstage Section	1,700	1,700
Stage S-I Retrorockets	3,000	3,000
Stage S-I, Dry Total	90,000	90,000
Mainstage Propellants (Lox/RP-1)	844,900	-
Fuel for Lubrication	552	-
Thrust Decay Propellants	2,235	2,235
Reserve for M. R. Shift	5,100	5,100
Propellants Trapped in Engines	2,056	2,056
Propellants Trapped in Lines	8,962	8,962
Pressurizing Oxygen	5,109	3,426
Pressurizing Nitrogen	744	744
Apollo "A" Launch Escape Propulsion*	3,700	3,700
TOTALS	1,105,564	258,072

* Separated After S-I Cutoff

(C) Table 3.2

APOLLO "A"/SATURN C-1 WEIGHT SUMMARY (SECOND STAGE FLIGHT)
FOR TYPICAL INJECTION INTO 100 NAUTICAL MILE ORBIT

ITEM		LIFTOFF WEIGHT (LBS)	CUTOFF WEIGHT (LBS)
Apollo "A" Spacecraft		17,300	17,300
Payload Contingency		3,418	3,418
Instrument Unit		3,000	3,000
Stage S-IV, Dry Total		11,700	11,700
Main Stage Propellants (Lox/LH_2)		100,000	500*
Helium Heater Propellants		60	-
Reserve for M. R. Shift		500	500
Residuals		482	482
Oxidizer trapped in engines	212		
Fuel trapped in engines	16		
Gas residuals in lox tank	133		
Gas residuals in fuel tank	116		
Gas residuals in pneu. tank	5		
Lost Propellants During Separation/Start		689	
Propellant for start	216		
Chill-down	54		
Ullage rocket propellants	298		
Jettisoned rocket cases	121		
TOTALS		137,149	37,000

*Flight Performance Reserve

(C) Table 3.3

APOLLO "A"/SATURN C-1 DRY WEIGHT BREAKDOWN

ITEM	WEIGHT (LBS)		CG STA
STAGE S-I, DRY TOTAL		90,000	361
Structure, Including Tail Fins	59,000		387
Propulsion System	22,000		187
Instrumentation & El. Network	3,000		761
Contingency	6,000		550
STAGE S-I RETROROCKETS		3,000	922
S-I/S-IV INTERSTAGE SECTION		1,700	1,053
STAGE S-IV, DRY TOTAL		11,700	1,185
Fuselage & Equipment	8,008		1,227
Propulsion System & Accessories	3,494		1,093
Contingency	198		1,103
INSTRUMENT UNIT		3,000	1,485
Structure	1,000		1,485
Equipment	2,000		1,485
APOLLO "A" SPACECRAFT (AT LAUNCH)		21,000	1,727
Command Module	7,900		1,733
Equipment & Propulsion Module Less Main Motors	3,500		1,663
Main Motors	2,900		1,664
Space Laboratory	3,000		1,550
Launch Escape Propulsion*	3,700		1,967

*Separated After S-I Cutoff

(C) Table 3.4

APOLLO "A" - SATURN C-1 SPACE VEHICLE INBOARD PROFILE CALLOUTS FOR FIGURES 3.6A THROUGH G

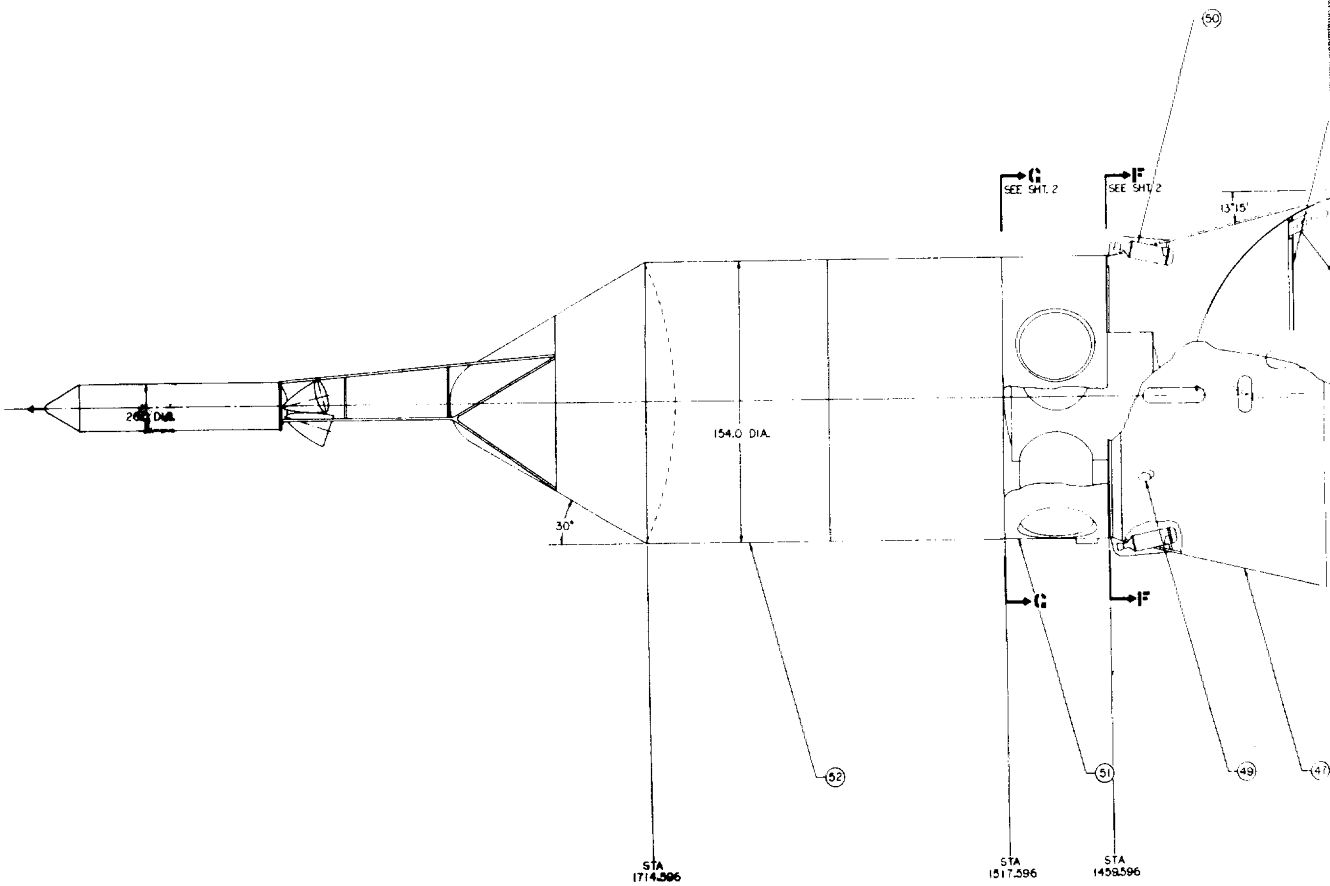
1. Fin (4)
2. Fin, Short (4)
3. Outboard Engine Shroud
4. Inboard Engine Exhaust Duct
5. H-1 Engine
6. Inboard Lox Suction Line
7. Outboard Lox Suction Line
8. Outboard Fuel Suction Line
9. Outboard Engine Heat Exchanger
10. Aspirator
11. Gox Line
12. Hydraulic Actuators
13. Heat Shield
14. Gox Line From Heat Exchanger
15. Inboard Fuel Suction Line
16. Firewall
17. Fuel Fill and Drain
18. Lox Fill and Drain
19. Erection Target
20. Antislosh Baffles (105-inch lox container)
21. Antislosh Baffles (Typical for eight 70-inch containers)
22. Gox Line
23. Pressurant Diffuser
24. Fuel Pressurization Manifold
25. Instrument Unit - Typical, F-1 and F-2
26. Cable Trunk (4)
27. Spider Beam Assembly
28. Pressurant Distributor Assembly
29. Lox Pressurization Manifold
30. First Stage Retrorocket (4)
31. Helium Heater
32. Heat Shield and Support
33. Control Pressure Helium Sphere
34. Antivortex and Filter Screen
35. Aft Interstage
36. Engine Cool-Down Line
37. Separation and Ullage Positioning Rocket (4)
38. Air Conditioning Disconnect
39. Oxidizer Feed Line (4)
40. Lox Container
41. Fuel Feed Lines
42. Oxygen Tank Vent Valve (2)

(C) Table 3.4 (Continued)

43. Oxygen Tank Slosh Baffle
44. Propellant Utilization Probe
45. Helium Sphere (cold)
46. LH_2 Container
47. Forward Interstage
48. Hydrogen Tank Slosh Baffle
49. Hydrogen Tank Vent
50. Retrorocket (2)
51. Instrument Unit
52. Apollo "A" Spacecraft
53. Access Chute Cover
54. Tail Shroud
55. Hold Down Points (8)
56. Water Quench System (4)
57. Actuator Connection
58. Inboard Heat Exchanger
59. Lox Vent Valve (3)
60. Fuel Relief Valve (2)
61. Lox Pressurization Manifold
62. T/M Antenna (4)
63. Long Cable Mast Connection
64. Antenna Panel (4)
65. Command Antenna (4)
66. Fuel Vent Valve (2)
67. GN_2 Sphere (2)
68. Lox Pressure Interconnect Prevalve (4)
69. Honeycomb Plate
70. Removeable Honeycomb Panel (8)
71. Spider Beam Shroud
72. Hydrogen Tank Manhole
73. Hydrogen Tank Vent Valve (2)
74. Tunnel
75. Access Door
76. T/M Package
77. Telemetry Antenna (4)
78. Range Safety Antenna (4)
79. Flight Control Switching Assembly
80. Flight Separation Disconnection Assembly
81. LO_2 Fill and Topping Line
82. Tank Pressurization Disconnect
83. Cold Helium Fill Disconnect
84. Control Helium Fill Disconnect
85. LH_2 Fill Drain and Topping Line
86. Active Guidance and Control
87. Passenger Guidance and Control

(C) Table 3.4 (Continued)

88. Electrical Power Equipment
89. Telemetry Equipment
90. C-Band Radar Antenna
91. Azusa Antenna
92. Radar Altimeter
93. UDOP Receiver Antenna (2)
94. UDOP Transmitter Antenna (2)
95. Telemetry Antenna (2)
96. Umbilibal Panel
97. Instrument Unit Connector Panel
98. Instrument Unit Access Door



FOLDOUT FRAME

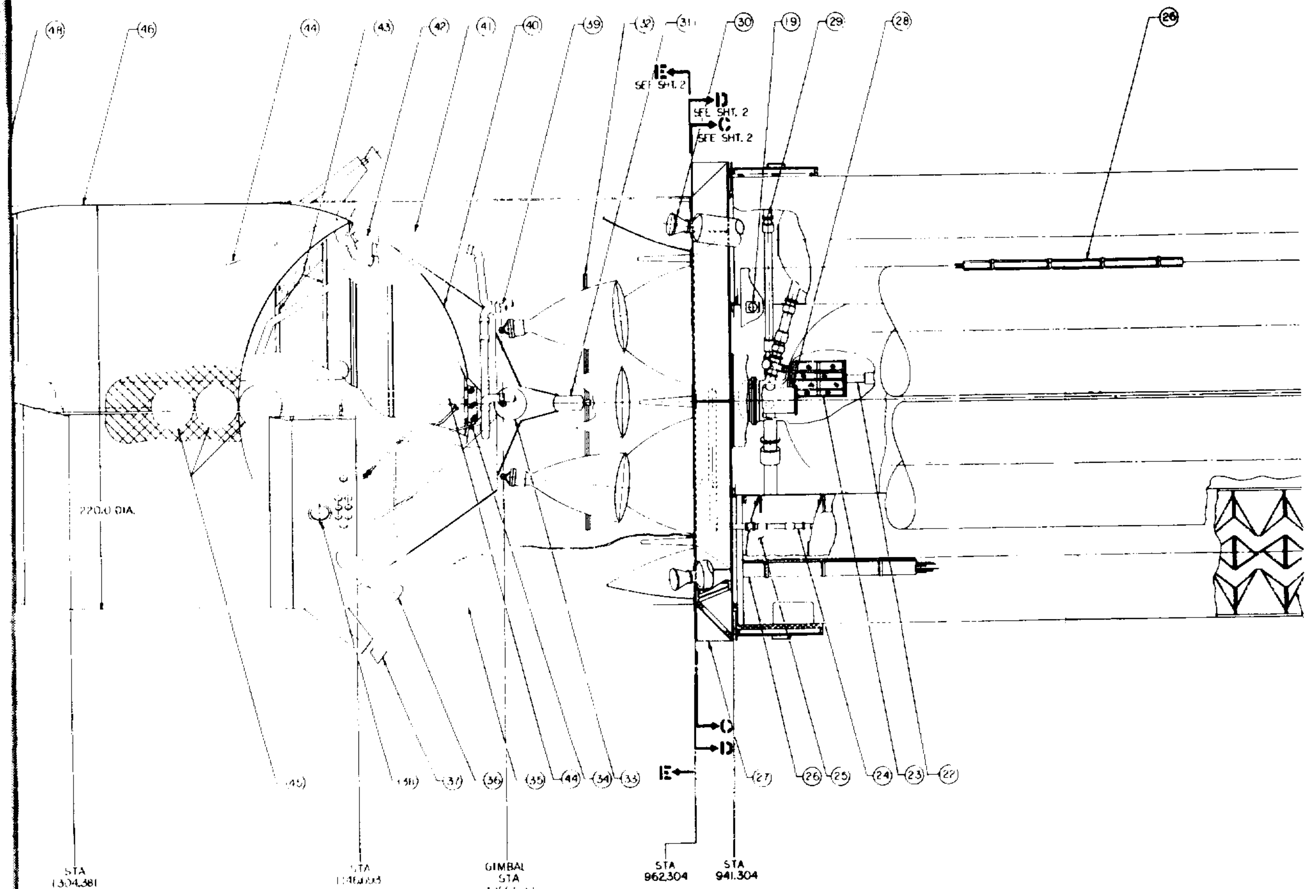


FIG. 3.6.A

FOLDOUT FRAME 2

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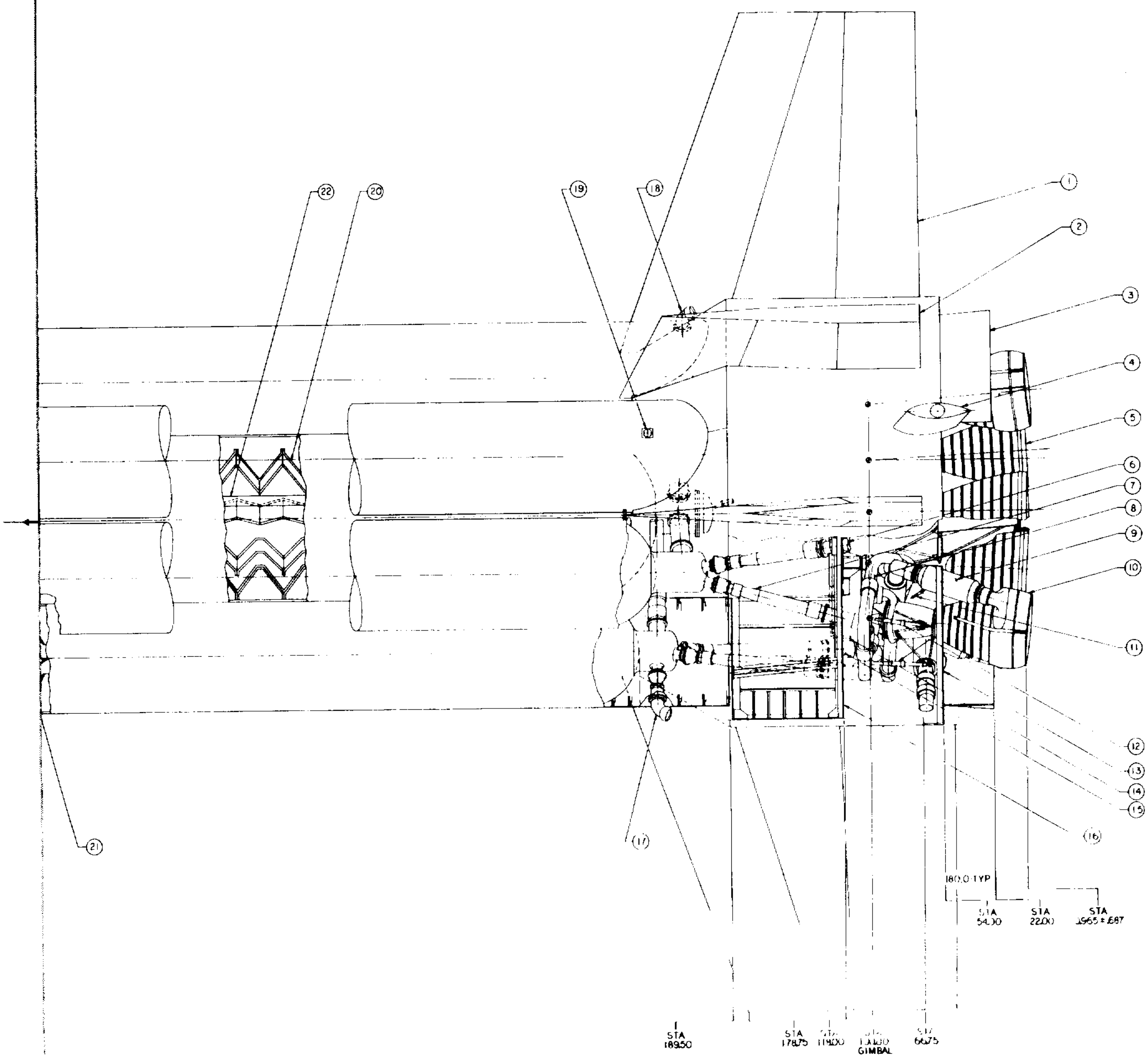

l. INBOARD PROFILE OF APOLLO "A"/SATURN C-1 SPACE VEHICLE
(See Table 3.4 for Callouts)

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~~FOLDOUT FRAME~~ 3

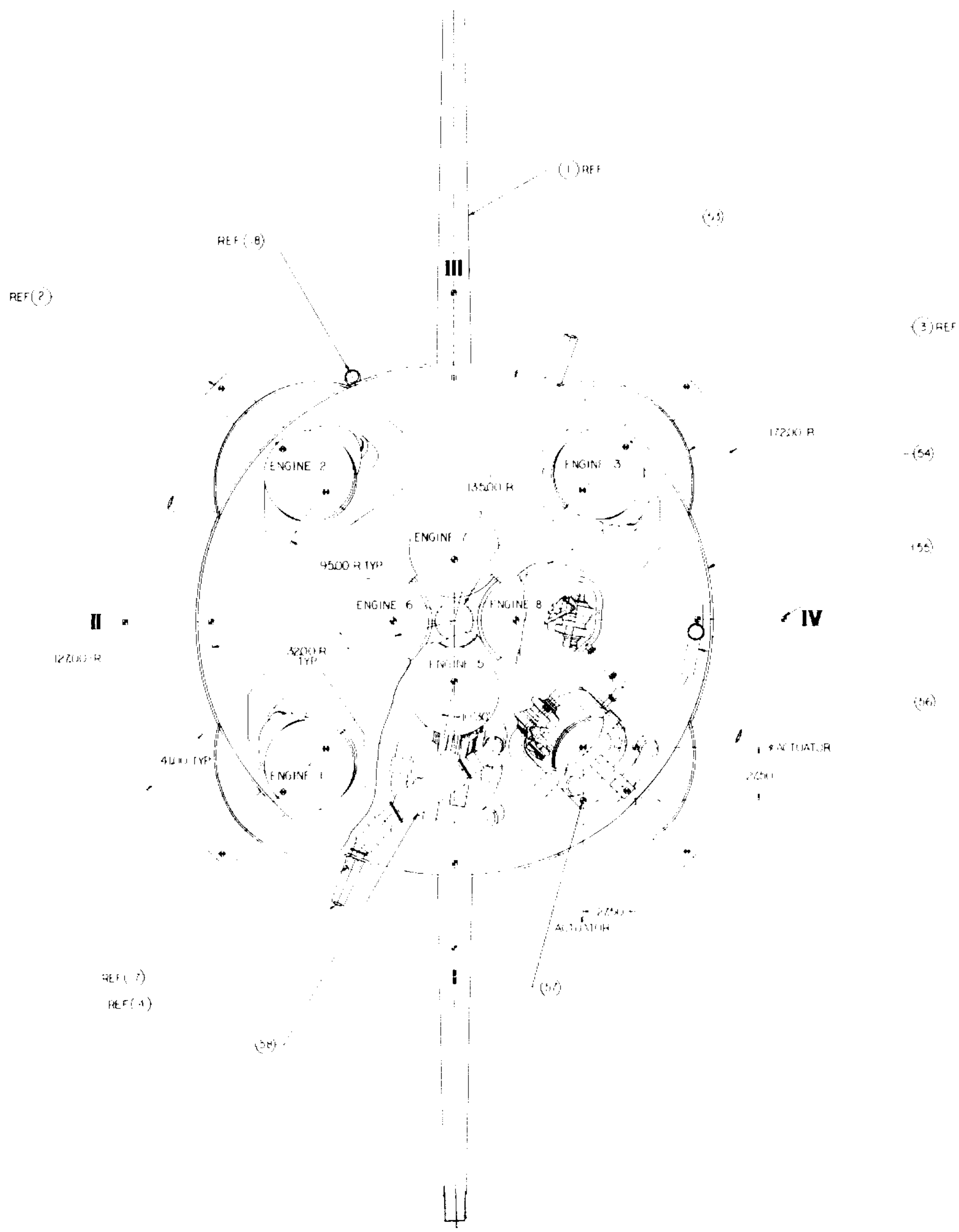

FIG. 3.6.B. END VIEW (See Table 3.4 for Callouts)

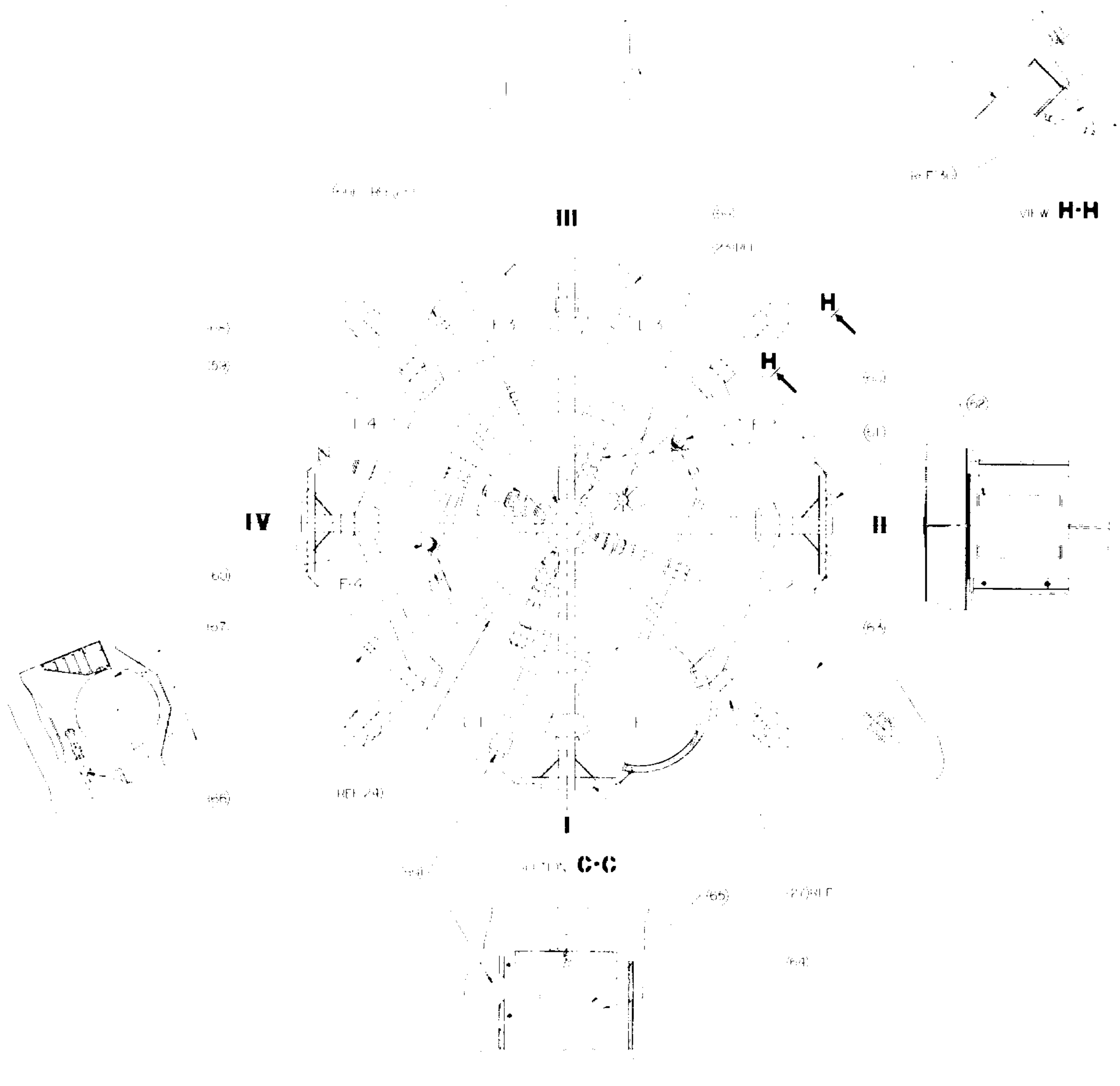

FIG. 3.6.C. SECTION C-C WITH VIEW H-H (See Table 3.4 for Callouts)

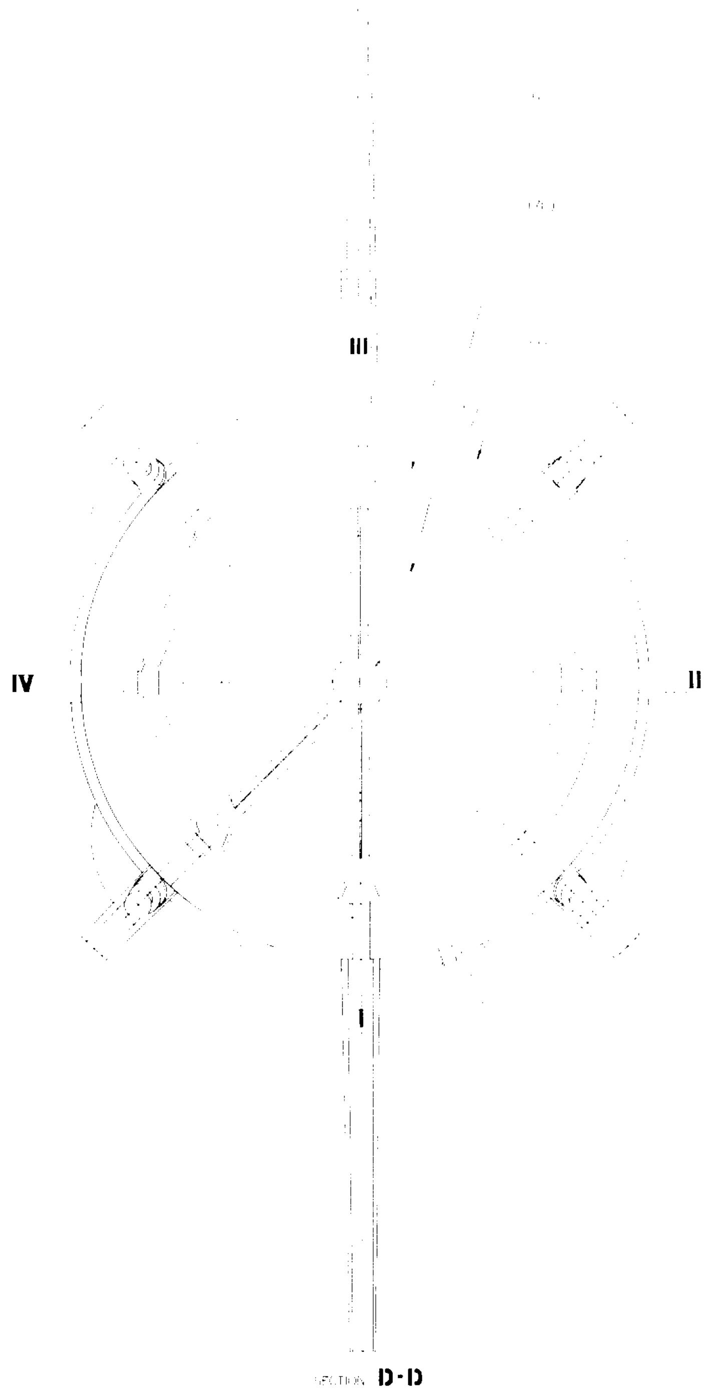

FIG. 3.6.D. SECTION D-D (See Table 3.4 for Callouts)

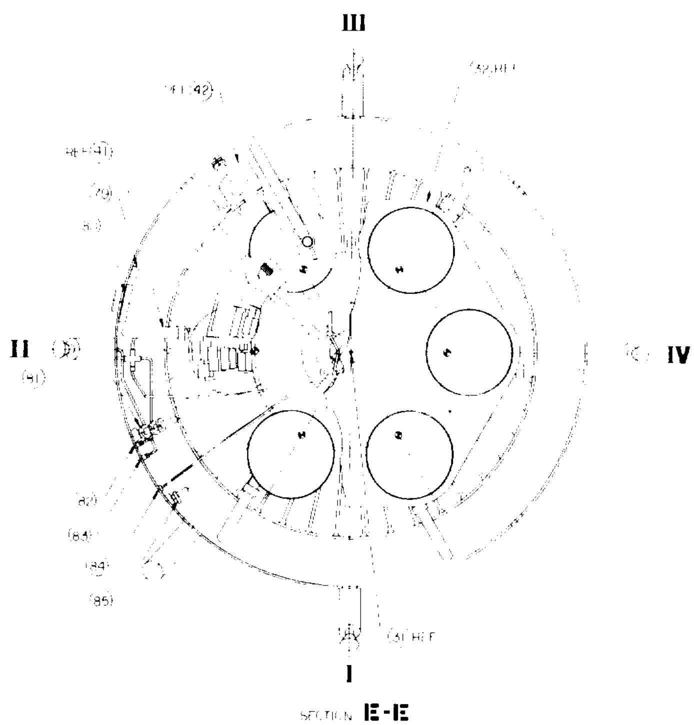

FIG. 3.6.E. SECTION E-E (See Table 3.4 for Callouts)

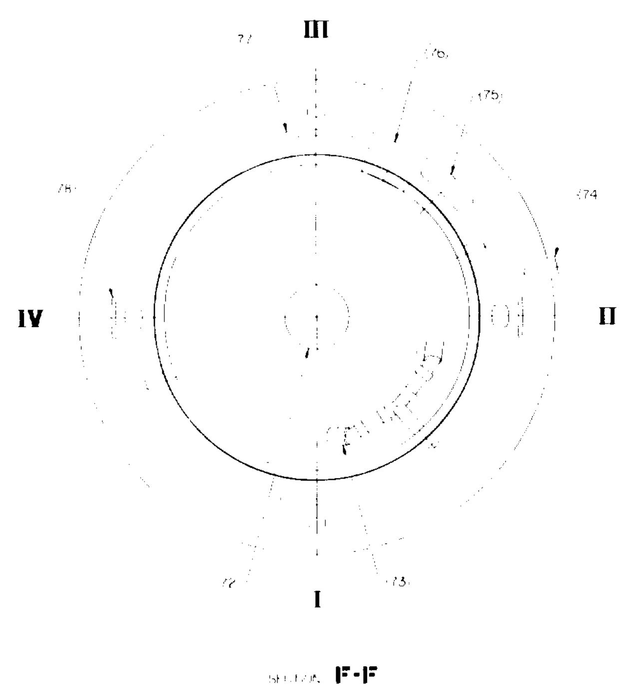

FIG. 3.6.F. SECTION F-F (See Table 3.4 for Callouts)

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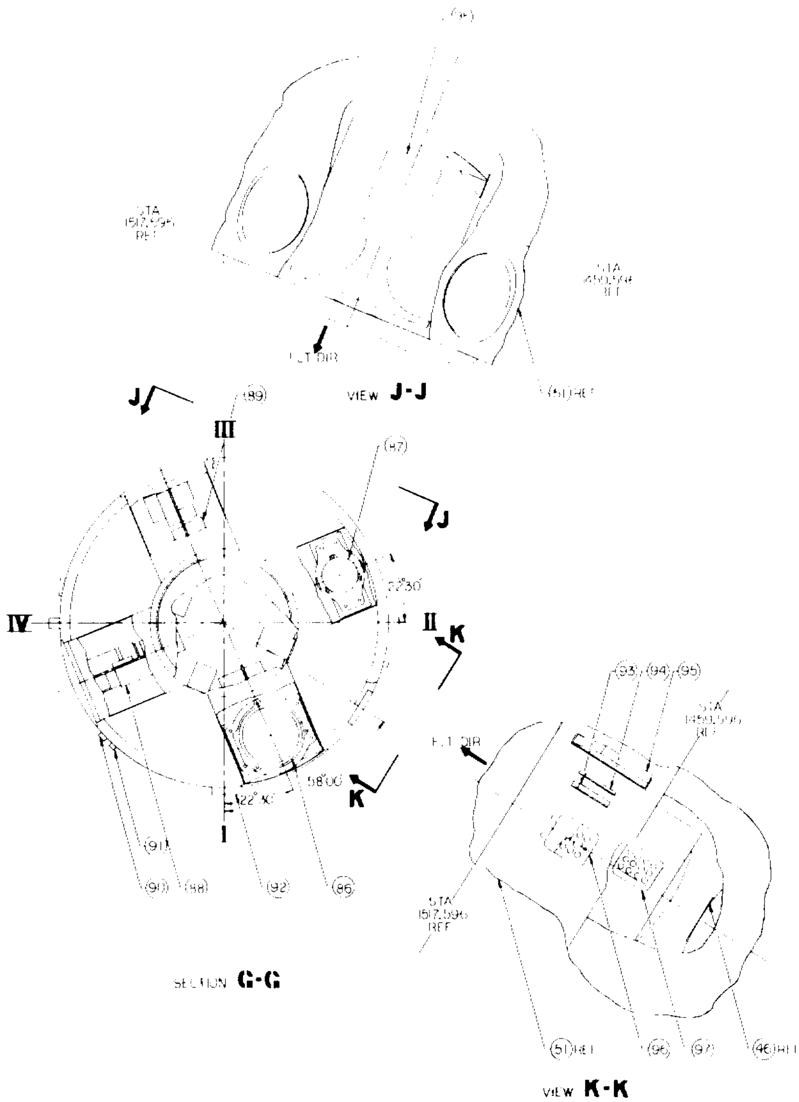

FIG. 3.6.G. SECTION G-G WITH VIEWS J-J AND K-K
(See Table 3.4 for Callouts)

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SECTION 4. (C) STAGE S-I

4.1. S-I DESCRIPTION

THE S-I stage (Fig. 4.1) is powered by eight 188,000-pound-thrust Rocketdyne H-1 engines. Propellants for these engines consist of 850 000 pounds of usable liquid oxygen (lox) and RP-1. Five lox tanks and four fuel tanks are the propellant containers. Fins are provided for flight stability (Figs. 4.2A and 4.2B). Propellant container and engine symbol designation is shown in Figure 4.3. S-I tail configuration and components are shown in Figure 4.4.

4.2. S-I STRUCTURES AND LOADS

Structural loads above the S-I stage are transmitted to the I-beam (Spider Beam) on top of the S-I stage through eight locating pins and eight explosive bolts. All loads introduced to the top of the S-I stage spider beam are reacted by the five oxidizer tanks. The four fuel tanks are positioned at the forward end by sliding pin connections which transmit shear loads only. This is necessary to permit thermal expansion or contraction of the lox tanks.

The loads at the aft end of the lox tanks are transmitted to the center barrel, shear beams. These fins provide eight support and hold down points for the vehicle.

Vehicle acceleration and atmospheric pressure versus flight time are shown in Figures 4.5 and 4.6, respectively.

4.2.1. Dynamic Load Factors (DLF's). The DLF for hold down is 1.4 and for rebound is 0.333. These load factors are considered adequate for the H-1 engine cluster with a firing sequence of 2-2-2-2, a shutdown sequence of 4-4 and time intervals of 100 milliseconds between the pairs of fours. An emergency shutdown sequence due to malfunction of one outboard engine will result in practically a 5-3 shutdown sequence with 100 millisecond intervals between the groups. This may result in a lower safety margin but is still considered to be an acceptable condition.

4.3. S-I WEIGHT SUMMARY

The Saturn C-1 S-I Stage Dry Weight Breakdown is shown in Table 4.1. Mass moment of inertia versus burning time is presented in Figure 4.7. Center of gravity shift and decrease in weight are shown in Figure 4.8. Data in both figures are for the S-I stage only.

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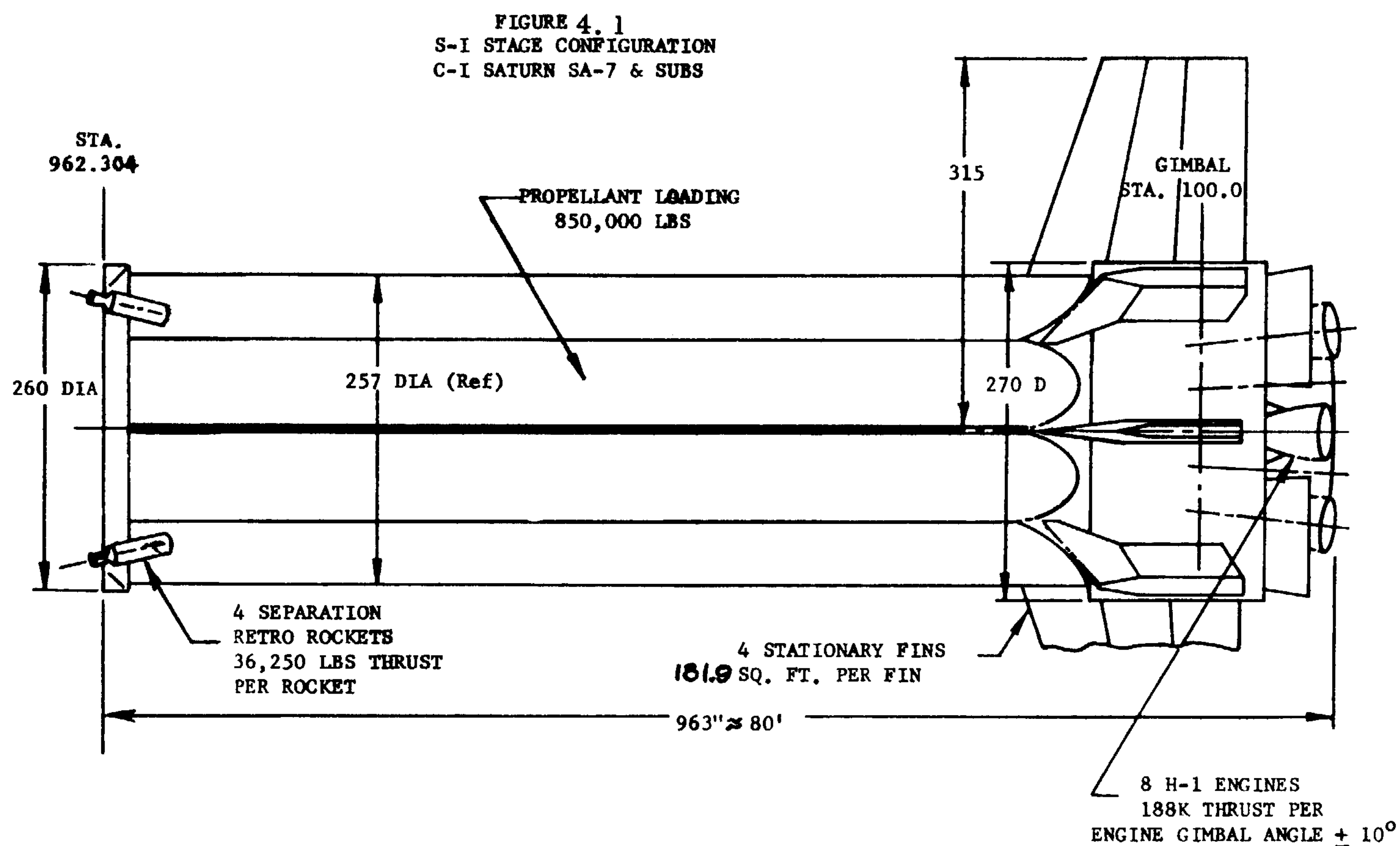

FIG. 4.1 S-I STAGE CONFIGURATION OF SATURN C-1

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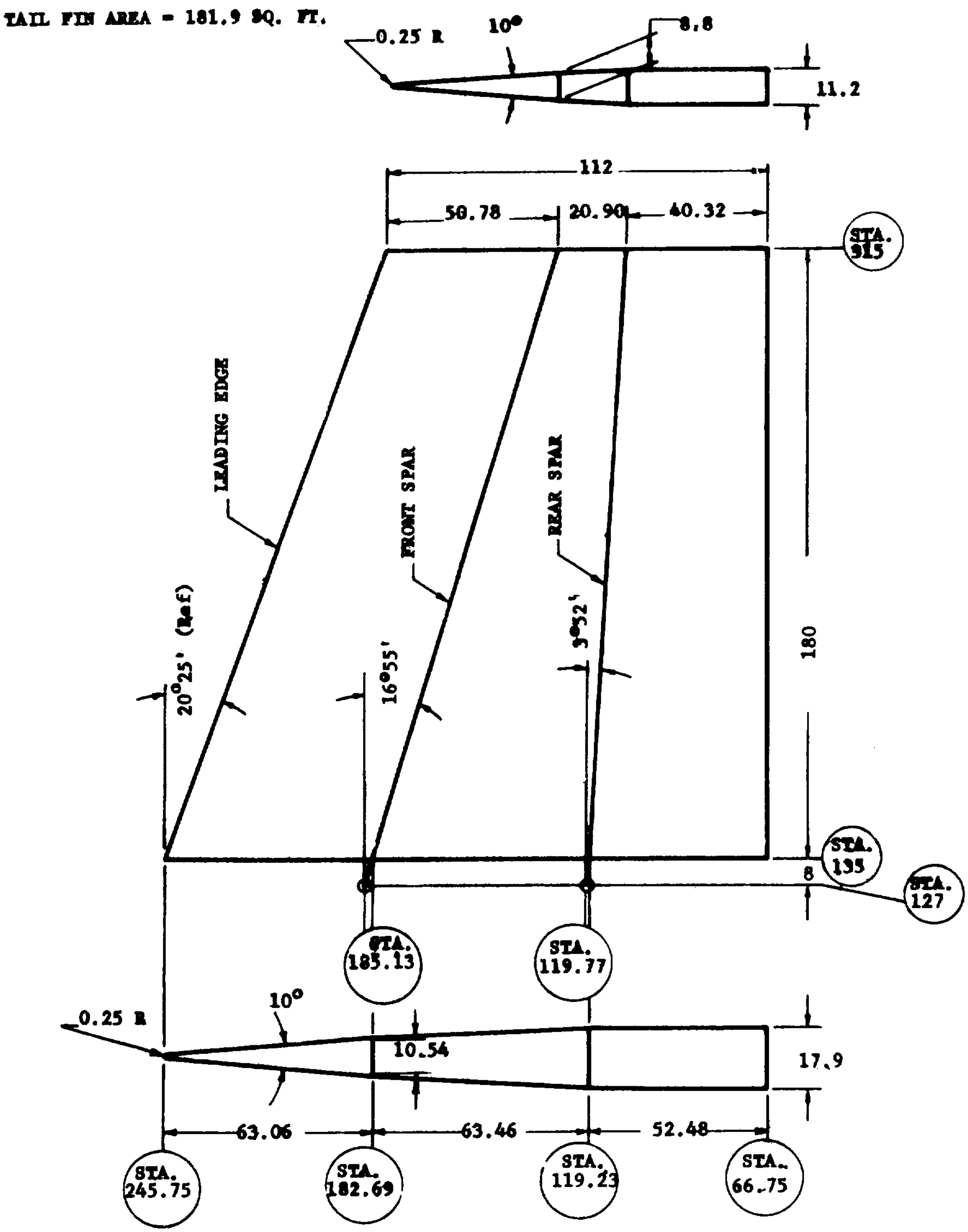
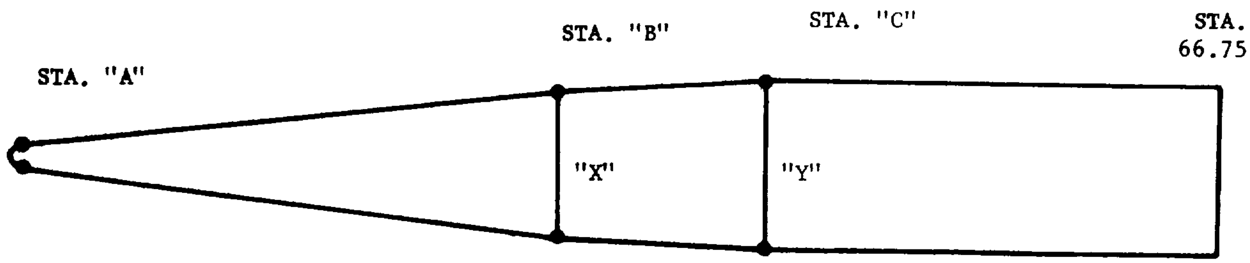
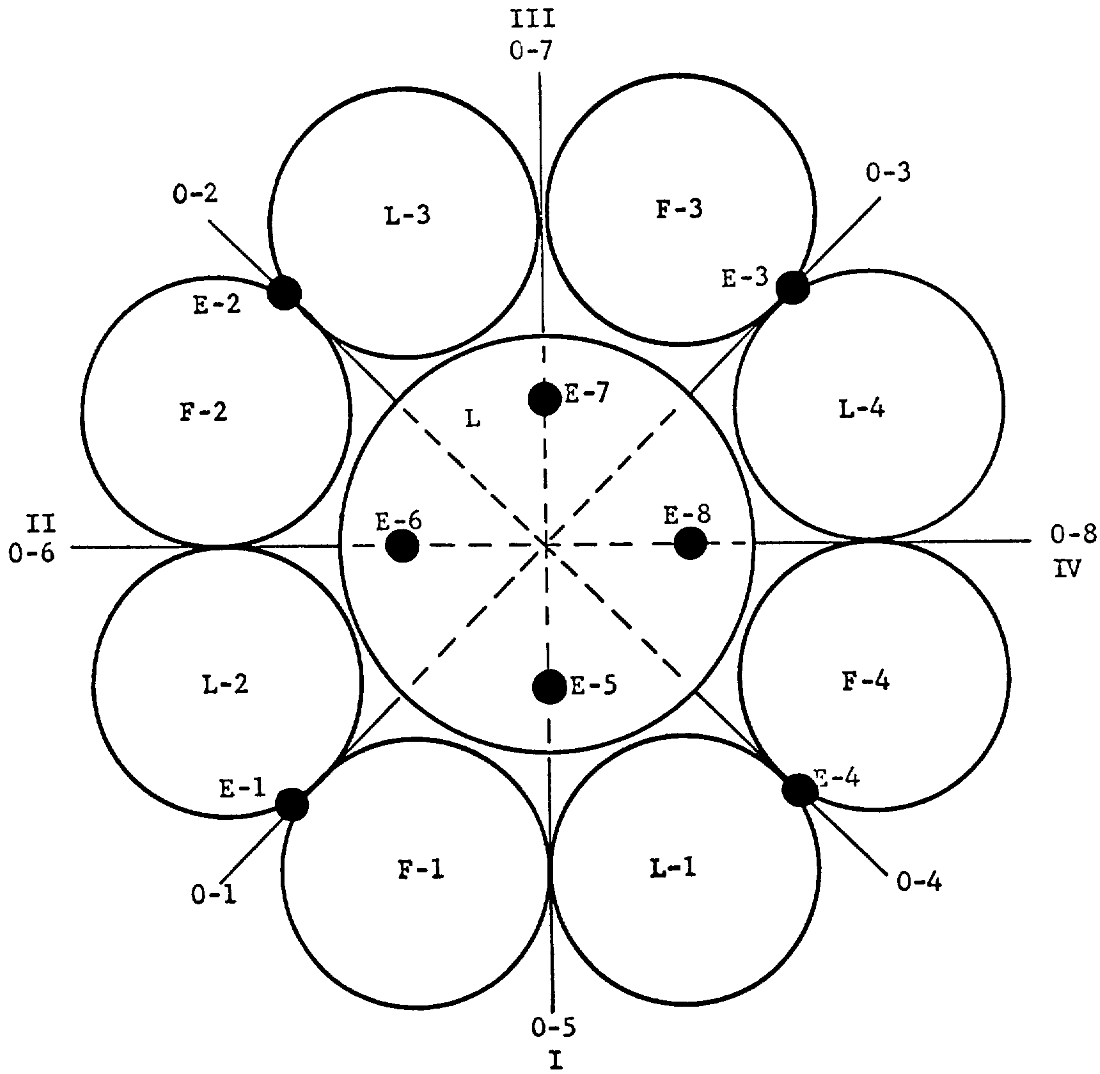


FIG. 4.2.A. TAIL FIN FOR SATURN C-1



	STA "A"	STA "B"	STA "C"	"X"	"Y"
315.00	178.48	127.97	107.08	8.80	11.20
295.00	185.92	134.05	108.43	8.99	11.94
275.00	193.37	140.13	109.78	9.19	12.69
255.00	200.81	146.21	111.13	9.38	13.43
235.00	208.26	152.29	112.48	9.57	14.18
215.00	215.70	158.37	113.83	9.77	14.92
195.00	223.15	164.45	115.18	9.96	15.67
175.00	230.59	170.53	116.53	10.15	16.41
172.00	231.71	171.44	116.74	10.18	16.52
155.00	238.04	176.61	117.88	10.35	17.16
135.00	245.48	182.69	119.24	10.54	17.90
127.00	248.46	185.125	119.775		

FIG. 4.2.B. TAIL FIN FOR SATURN C-1



Legend

L - LOX TANK
F - FUEL TANK
E - ENGINE THRUST PAD
O - OUTRIGGER HOLD-DOWN POINT

FIG. 4.3. PROPELLANT CONTAINER AND ENGINE SYMBOL DESIGNATION

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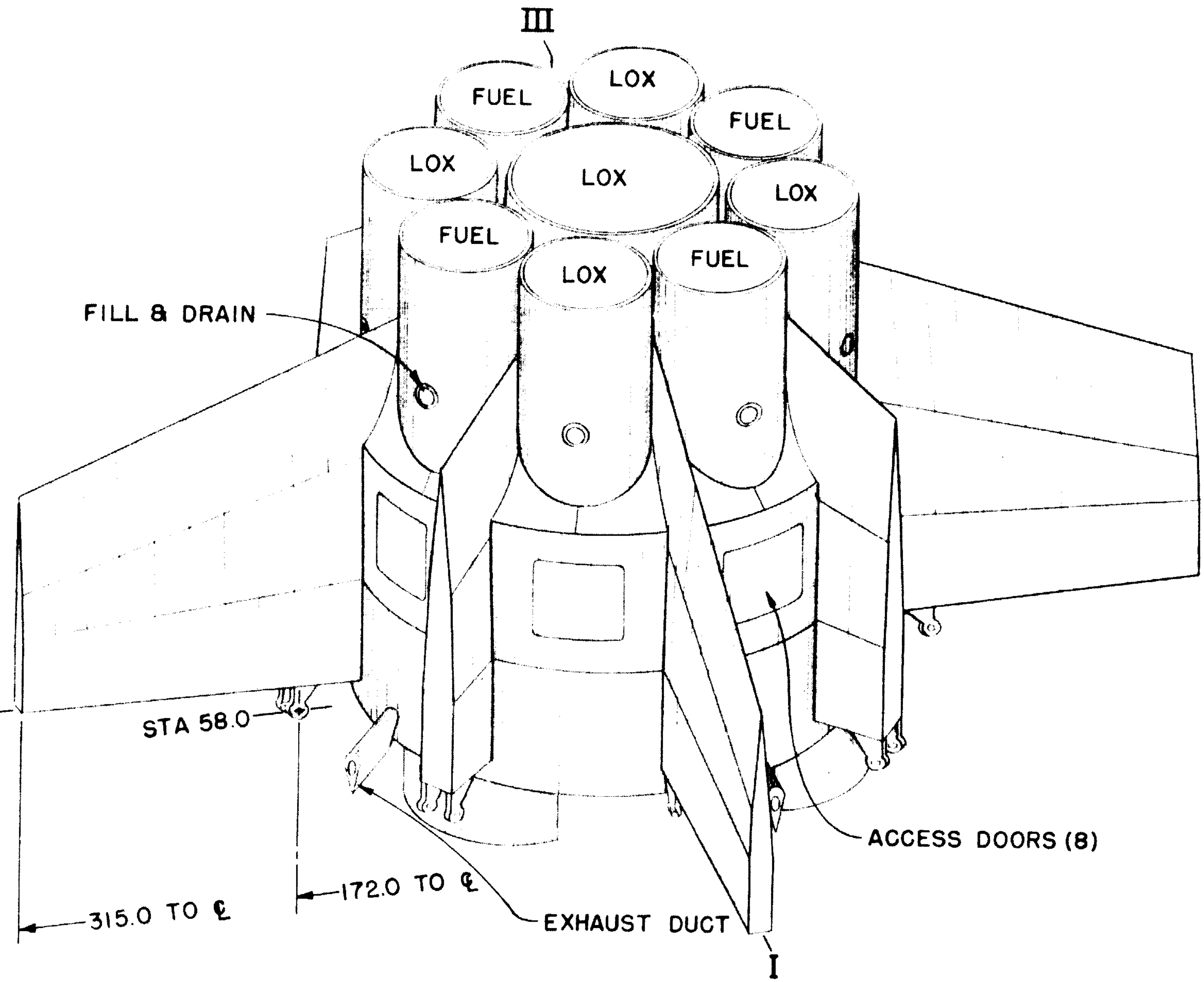

FIG. 4.4 STAGE S-I TAIL CONFIGURATION

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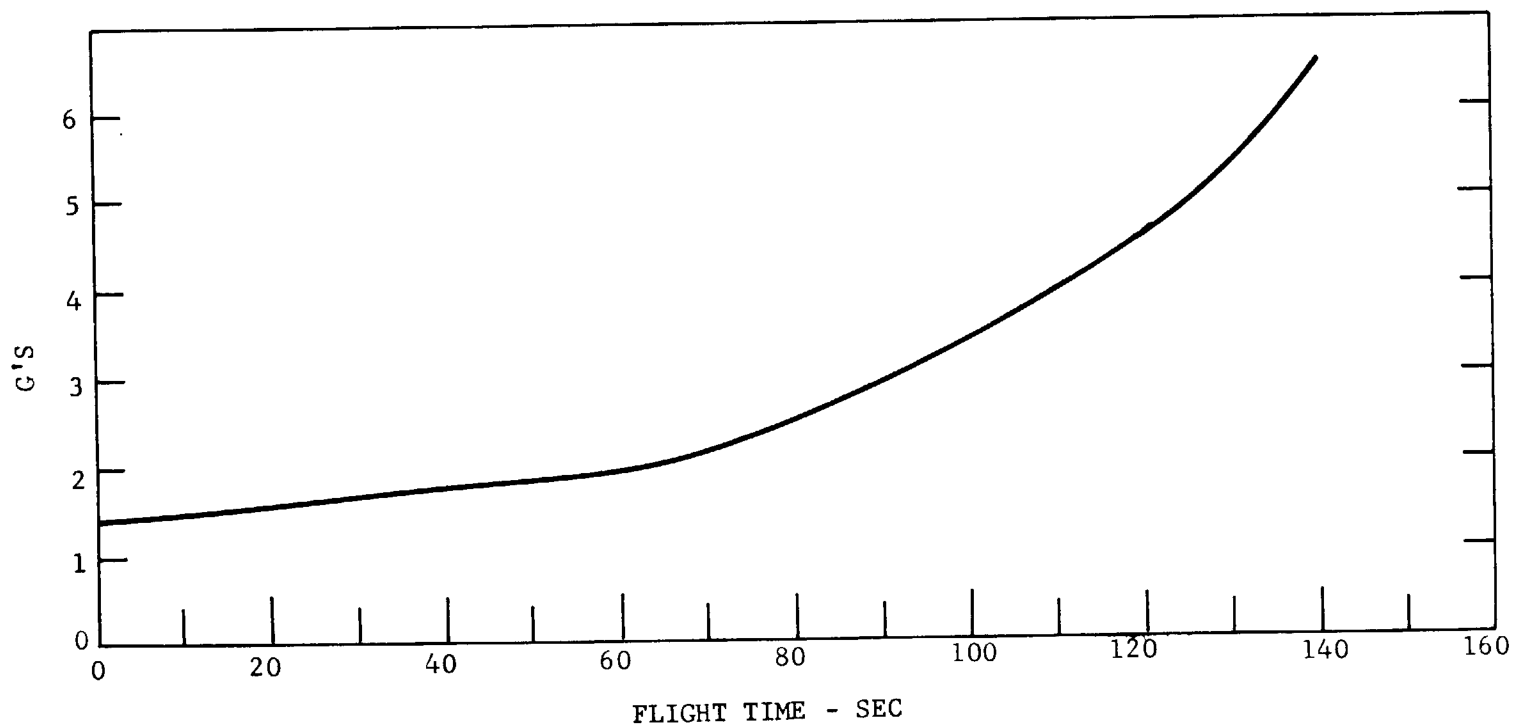

FIG. 4.5 SATURN C-1 STAGE I VEHICLE ACCELERATION VERSUS FLIGHT TIME

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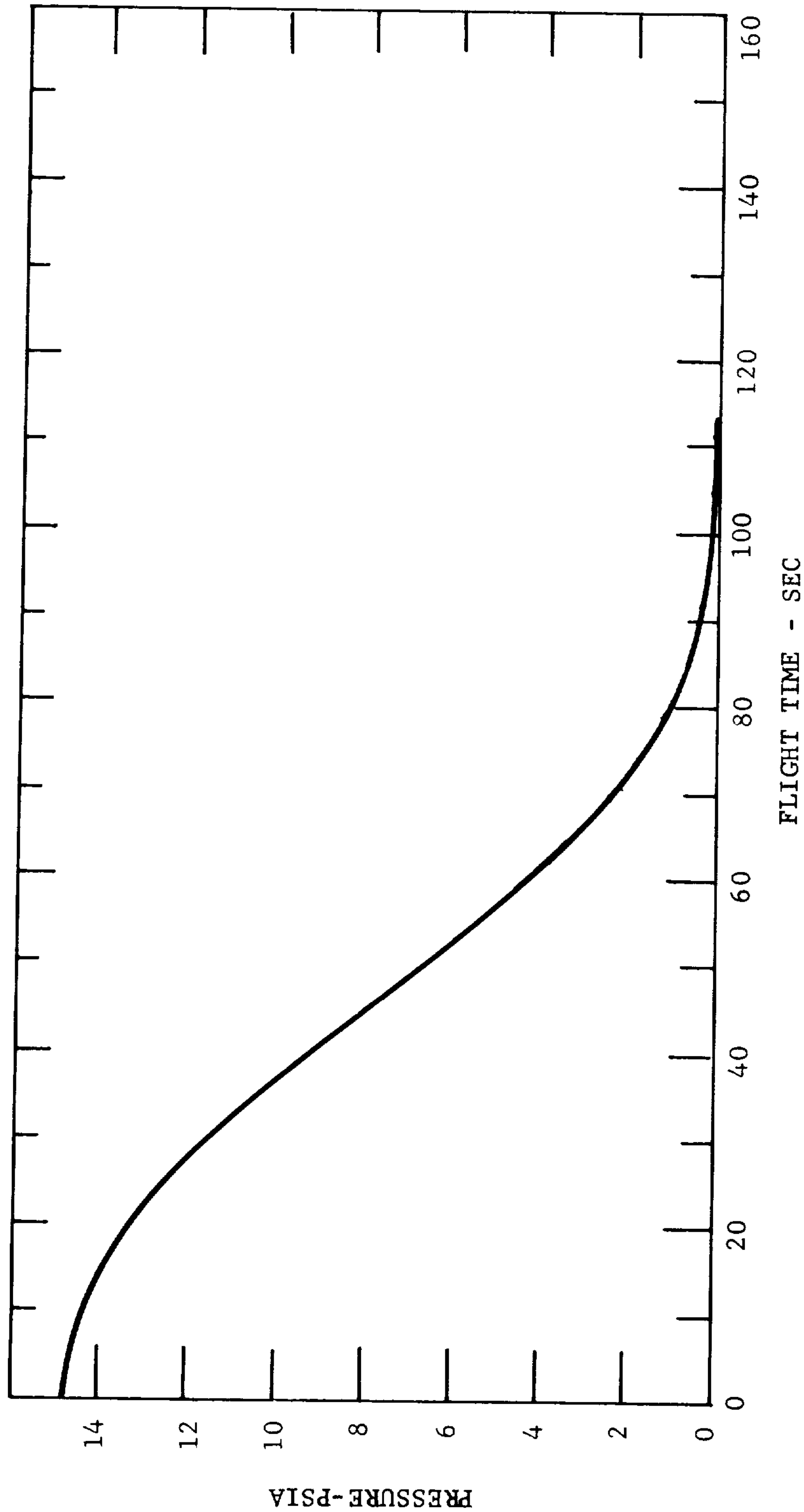

FIG. 4.6 SATURN C-1 STAGE I ATMOSPHERIC PRESSURE VERSUS FLIGHT TIME

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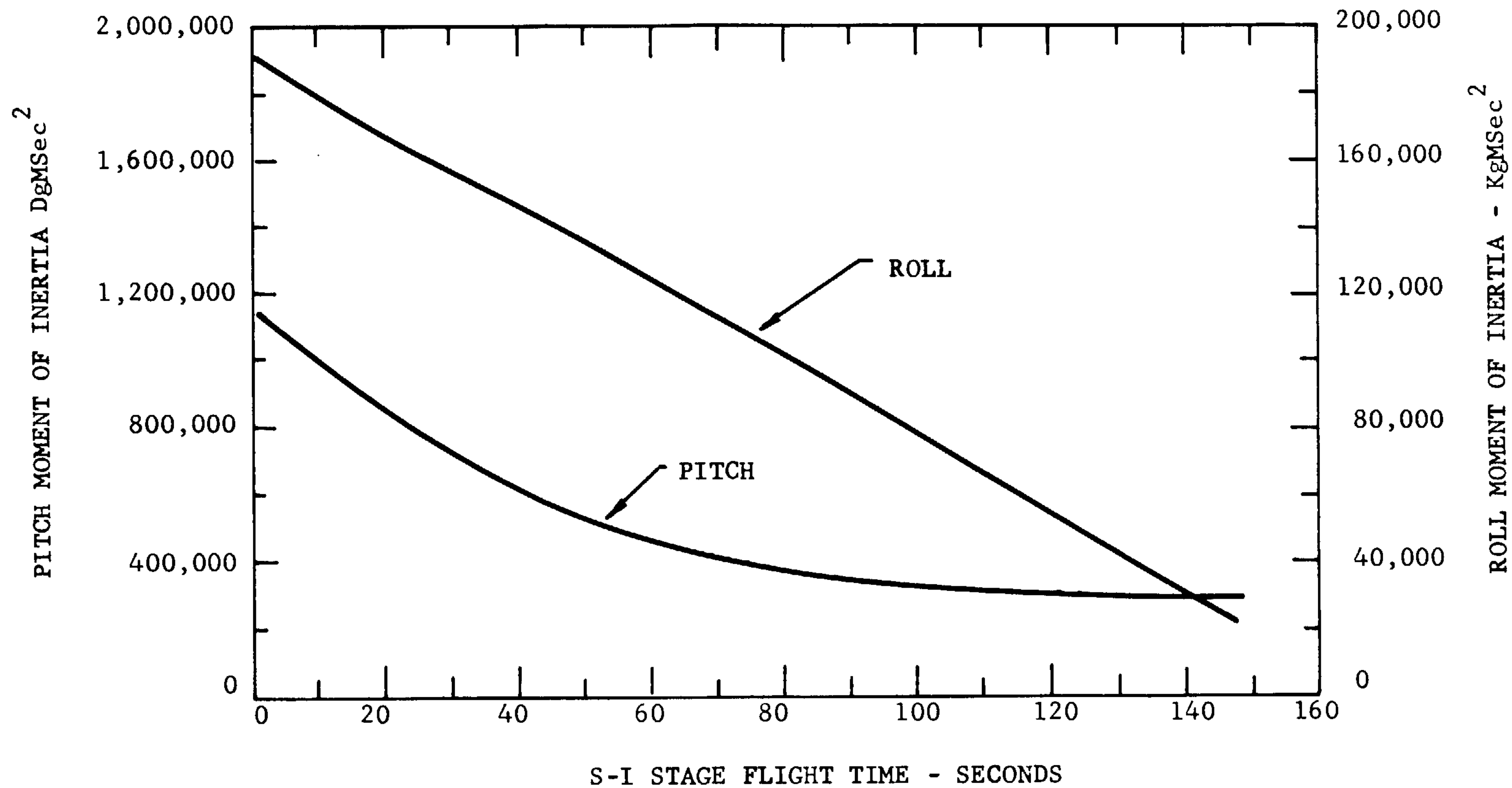

FIG. 4.7 MASS MOMENT OF INERTIA VERSUS BURNING TIME FOR SATURN C-1 STAGE S-I ONLY.

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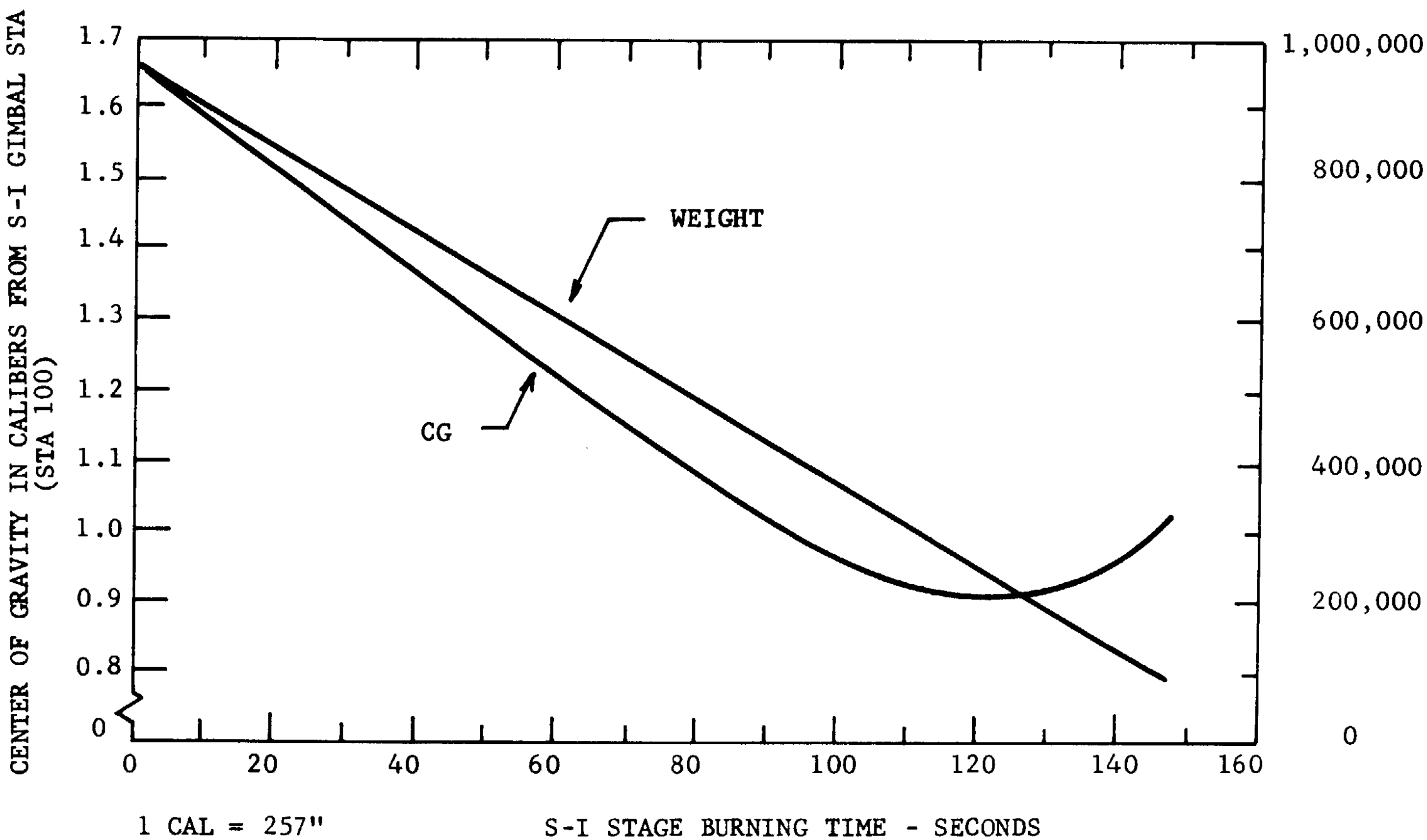

FIG. 4.8 CENTER OF GRAVITY SHIFT AND DECREASE IN WEIGHT VERSUS BURNING TIME FOR SATURN C-1 STAGE S-I ONLY.

(C) Table 4.1
SATURN C-1, S-I STAGE, DRY WEIGHT BREAKDOWN

		WEIGHT (LB)	CG STA
STRUCTURE		59,000	387
PROPULSION SYSTEM		22,000	187
INSTRUMENTATION & ELECTRICAL NETWORK		3,000	761
CONTINGENCY		6,000	361
	TOTAL	90,000	

DETAILED BREAKDOWN

STRUCTURE			
A. 105" Dia Lox Cont Section		6,850	601
1. Front Ring Frame	58.1		941
2. Longerons & Fittings (48)	228.5		915
3. Front Lox Manifold	101.3		922
4. Front Bulkhead	177.0		898
5. Bulkhead Reinforced Ring	16.4		904
6. Pressurant Distributor Ass'y	52.0		922
7. Backup Plates	95.9		888
8. Brackets - Front	19.2		937
9. Fasteners - Front	42.5		937
10. Front Skin Assembly	187.3		932
11. Skin Assembly No. 1	594.0		893
12. Skin Assembly No. 2	290.1		842
13. Skin Assembly No. 3	593.8		780
14. Skin Assembly No. 4	393.7		720
15. Skin Assembly No. 5	393.7		662
16. Skin Assembly No. 6	393.7		604
17. Skin Assembly No. 7	393.7		546
18. Skin Assembly No. 8	393.7		488
19. Skin Assembly No. 9	479.4		430
20. Skin Assembly No. 10	479.4		372
21. Skin Assembly No. 11	479.4		314
22. Rear Skin Assembly	485.5		266
23. Rear Bulkhead	255.0		252
24. Bulkhead Reinforcement	16.4		225
25. Rear Lox Manifold	129.3		218
26. Angle Section - Rear	70.8		247
27. Fasteners - Rear	8.0		247
28. Antivortex Device	1.4		232
29. Misc Ass'y Brackets	20.8		247
30. Paint & Corrosion Protection	30.0		571

(C) Table 4.1 (Continued)

			WEIGHT (LB)	CG STA
B.	70" Dia Lox Cont Section No.1		3,100	525
	1. Front Ring	34.2		939
	2. Z-Ring	8.4		899
	3. Angle Stiffeners - Front (8)	16.1		919
	4. Tee Stiffeners - Front (8)	42.8		919
	5. Support Fittings - Top	61.2		934
	6. Front Bulkhead	56.0		895
	7. Manhole Cover	15.1		912
	8. Brackets - Front	19.7		919
	9. Fasteners - Front	8.3		919
	10. Front Skin Ass'y	79.1		930
	11. Skin Segment No. 1	203.7		896
	12. Skin Assembly No. 2	146.3		842
	13. Skin Assembly No. 3	190.2		780
	14. Skin Assembly No. 4	146.3		712
	15. Skin Assembly No. 5	146.3		654
	16. Skin Assembly No. 6	146.3		596
	17. Skin Assembly No. 7	146.3		538
	18. Skin Assembly No. 8	161.8		480
	19. Skin Assembly No. 9	174.0		422
	20. Skin Assembly No. 10	181.6		364
	21. Skin Assembly No. 11	187.5		306
	22. Skin Segment No. 12	212.6		252
	23. Rear Skin Segment	155.8		208
	24. Rear Bulkhead	76.9		254
	25. Funnel Weldment	8.0		240
	26. Mounting Plate	62.0		236
	27. Support Fittings - Bottom (2)	61.2		94
	28. Longerons - Rear (8)	73.6		219
	29. Stiffeners - Rear (48)	67.2		219
	30. Lox Fill Flange & Wldmt	14.1		217
	31. Channel Frames (3)	26.5		230
	32. Bottom Ring	34.1		190
	33. Brackets - Rear	17.3		219
	34. Fasteners - Rear	14.0		219
	35. Prima Cord Conduit Ass'y	13.6		566
	36. Lox Elbow	25.9		220
	37. Antivortex Device	5.8		245
	38. Stand Pipe	3.0		245
	39. Misc Ass'y Brackets	37.2		225
	40. Paint & Corrosion Resistance	20.0		520
C.	70" Dia LOX Cont Section No. 2		3,100	525
D.	70" Dia Cont Section No. 3		3,100	525
E.	70" Dia Lox Cont Section No. 4		3,100	525

(C) Table 4.1 (Continued)

			WEIGHT (LB)	CG STA
F.	70" Dia Fuel Cont Section No. 1		1,950	504
	1. Front Ring Frame	39.4		935
	2. Z-Ring	8.4		928
	3. Front Bulkhead	42.0		888
	4. Manhole Cover	12.8		906
	5. Brackets - Front	10.3		932
	6. Fasteners - Front	11.3		932
	7. Front Skin Segment	45.9		924
	8. Skin Segment No. 1	99.7		888
	9. Skin Assembly No. 2	92.1		833
	10. Skin Assembly No. 3	124.6		780
	11. Skin Assembly No. 4	93.9		703
	12. Skin Assembly No. 5	95.8		645
	13. Skin Assembly No. 6	97.6		587
	14. Skin Assembly No. 7	99.5		529
	15. Skin Assembly No. 8	101.3		471
	16. Skin Assembly No. 9	103.1		413
	17. Skin Assembly No. 10	105.0		355
	18. Skin Assembly No. 11	82.3		307
	19. Skin Segment No. 12	111.2		263
	20. Rear Skin Segment	106.9		214
	21 Rear Bulkhead	42.0		265
	22. Funnel Weldment	3.4		248
	23. Mounting Plate, Fuel Tank	45.0		246
	24. Support Fittings - Bottom (2)	61.2		194
	25. Longerons - Rear (8)	67.1		221
	26. Stiffeners - Rear	28.8		221
	27. Z-Rings (4)	24.6		236
	28. Bottom Ring	27.8		189
	29. Brackets - Rear	9.7		221
	30. Fasteners - Rear	33.6		221
	31. Prima Cord Conduit Ass'y	16.9		566
	32. Fuel Elbow	11.9		237
	33. Antivortex Device	5.8		255
	34. Misc Ass'y Brackets	69.1		225
	35. Paint & Corrosion Resistance	20.0		500
G.	70" Dia Fuel Cont Section No. 2		1,950	504
H.	70" Dia Fuel Cont Section No. 3		1,950	504
I.	70" Dia Fuel Cont Section No. 4		1,950	504
J.	Tail Section Assembly		12,700	143
K.	Structural Requirements for 188K Engines		1,000	137
L.	Spider Beam Assembly		3,770	951
	1. 20" I-Beam	3,029		951
	2. Shroud	320		951
	3. Miscellaneous	421		929

(C) Table 4.1 (Continued)

			WEIGHT (LB)	CB STA
M.	Tail Fins		5,000	140
N.	Bending Moment Reinforcement		4,000	371
O.	Fire Wall		530	120
P.	Heat Shield & Support		650	69
	1. Heat Shield	98.7		56
	2. Support Structure	551.3		71
Q.	Flame Shield & Support		1,280	39
R.	Tail Shroud & Heat Protection		2,660	79
S.	Cable Ducts (4)		140	586
T.	Ice & Rain Shield		175	586
U.	Misc Paint & Corrosion Protection		200	422
V.	Misc Brackets & Hardware		845	100
	STRUCTURE TOTAL		59,000	
PROPULSION SYSTEM				
A.	Rocket Engine System		13,197	70
	1. Inboard Engines (4)	5,996.0		
	2. Inboard Exhaust Ducts and Heat Exchangers	445.0		
	3. Outboard Engines (4) Incl. Hydraulic Actuator System Heat Exchangers Exhaust Aspirators	6,756.0		
B.	High Pressure Supply System (16 Spheres)		1,800	951
C.	Suction Lines (Including Flow Meters)		4,329	140
	1. Lox Suction Lines IE (4)	1,092.4		
	2. Fuel Suction Lines IE (4)	1,032.2		
	3. Lox Suction Lines OE (4)	1,074.6		
	4. Fuel Suction Lines OE (4)	1,129.8		
D.	Tank Interconnecting Lines		700	225
	1. Fuel Transfer Ass'y	101.0		
	2. Inter Line Ass'y, Fuel	294.0		
	3. Inter Line Ass'y, Lox	305.0		
E.	Vent Lines & Tubing		700	524
	1. Lox Vent Line Ass'y 4" Dia	46.0		
	2. Lox Vent Inter Line Ass'y	107.2		
	3. Lox Fill & Drain Line Ass'y	57.4		
	4. Lox Replenishing Line Ass'y	26.5		
	5. Fuel Pressurizing Manifold	109.8		
	6. Fuel Vent Line Ass'y	33.6		
	7. Fuel Fill & Drain Ass'y (2)	38.4		
	8. Gox Line Ass'y	154.1		
	9. Misc Tubing	127.0		
F.	Water Deluge System		425	100

(C) Table 4.1 (Continued)

			WEIGHT (LB)	CG STA
G.	Tail Heating & Nitrogen Purge System		55	135
H.	Engine Purge System		75	125
I.	Control Supply & Gear Box Press (2 Spheres)		101	125
J.	Miscellaneous		618	490
	PROPULSION SYSTEM TOTAL		22,000	187
INSTRUMENTATION & ELECTRICAL NETWORK				
A.	Instrument Ass'y A		465	913
	1. Instrument Compartment Bulkhead	224.0		
	2. Instrument Group Ass'y A			
	a. Panel Ass'y	35.9		
	b. Measuring Racks (6)	60.0		
	c. Measuring Distributor	25.0		
	d. Donner Accelerometers (2)	1.7		
	e. Cooling System	32.4		
	f. Cable, Unit A, Running List	75.0		
	3. Miscellaneous	11.0		
B.	Instruemtn Ass'y B		640	913
	1. Instrument Compartment Bulkhead	225.8		
	2. Instrument Group Ass'y B			
	a. 216 Channel Commutator	15.0		
	b. Telemeter Ass'y (6)	148.8		
	c. Telemeter RF Multicoupler (3)	8.3		
	d. Telemeter Power Divider (2)	1.0		
	e. RF Power Amplifier (6)	55.5		
	f. Flow Rate Multiplexers (3)	6.0		
	g. TM Aux Equip Ass'y	8.0		
	h. Cooling System	18.4		
	i. Cable, Unit B, Running List	75.0		
	j. Instrument Panel Ass'y	35.9		
	k. Misc Hardware	27.4		
	3. Miscellaneous	14.9		
C.	AZUSA System		57	913
	1. Transponder	40.0		
	2. Power Supply	5.0		
	3. Antenna Mod 706	12.0		
D.	Command System		14	
	1. Receiver & Decoder	5.0		
	2. Power Supply	3.0		
	3. Antennas (2)	6.0		
E.	S-Band Radar System		14	913
	1. Transmitter	4.0		
	2. Receiver	3.0		
	3. Power Supply	5.0		
	4. Antenna	2.0		

(C) Table 4.1 (Continued)

		WEIGHT (LB)	CG STA
F. Power Supply System		145	913
1. Power Distributor	25.0		
2. Power Inverter	44.0		
3. Large Battery	76.0		
G. Pre-flight Cooling Ducts		35	913
H. Electrical Network System		1,030	470
1. Cable Trucks (4)	500.0		
2. Tail Section Cables	500.0		
3. Propulsion System Dist	23.0		
4. Heater Power Distributor	7.0		
I. Mounting Brackets & Misc		600	913
INSTRUMENTATION & ELECTRICAL NETWORK TOTAL		3,000	761

4.4. PROPULSION SYSTEM

4.4.1. Power Plant. The S-I stage power plant consists of eight engines (Fig. 4.9). Four of the engines, called inboard engines, are mounted in a closed square pattern of a 32-inch radius from the center and in a fixed three-degree canted position from the launch vehicle vertical. The other four engines, called outboard engines, are mounted on the points of a larger square rotated 45 degrees from the inner pattern. They are on a 95-inch radius from the center and canted to a six-degree null position from the vertical. Each outboard engine is designed to permit ± 10-degree square gimbal pattern in any direction from the null position by means of two hydraulic actuators. This gimbaling establishes proper forces for pitch, yaw, and roll control of the launch vehicle. Canting is used to direct the thrust vector of each engine through the vehicle center of gravity at cutoff of the S-I stage. This minimizes vehicle pitch or yaw at the time of separation in case of unpredictable impulse variation from any one of the engines or total loss of an engine.

The H-1 engine is a greatly simplified and repackaged S-3D engine (Fig. 4.10). The basic S-3 engine has been used in several missile programs. All the components have been thoroughly developed and have had extensive static test times accumulated. Some of the components have been extensively flight tested. All components have been successfully static test fired at thrust levels exceeding the design rating of 188,000 pounds. The simple pressure sequencing start system and the improved turbopump design have also had extenxive testing.

The Rocketdyne H-1 engine parameters are shown in Table 4.2.

4.4.2. Propellant System.

4.4.2.1. Propellant Container Water Volume. The total water volumes, including ullage, based on a 72-inch cylindrical extension on all S-I stage tanks (lengthened to accommodate the Saturn C-1 to Apollo "A"), at a fuel tank temperature of 60°F and a lox tank temperature of 4°C, are:

(1) Center Lox Container	24,271 gal
(2) Outboard Lox Container	10,917 gal
(3) Fuel Container	10,612 gal

4.4.2.2. Propellant Container Pressurization. The fuel containers are pressurized with gaseous nitrogen (GN_2). This pressurizing gas is controlled by four valves which are located between the GN_2 containers and the pressurizing gas manifold. The pressurizing valves are programmed by a timer which limits the flowrate during flight phases.

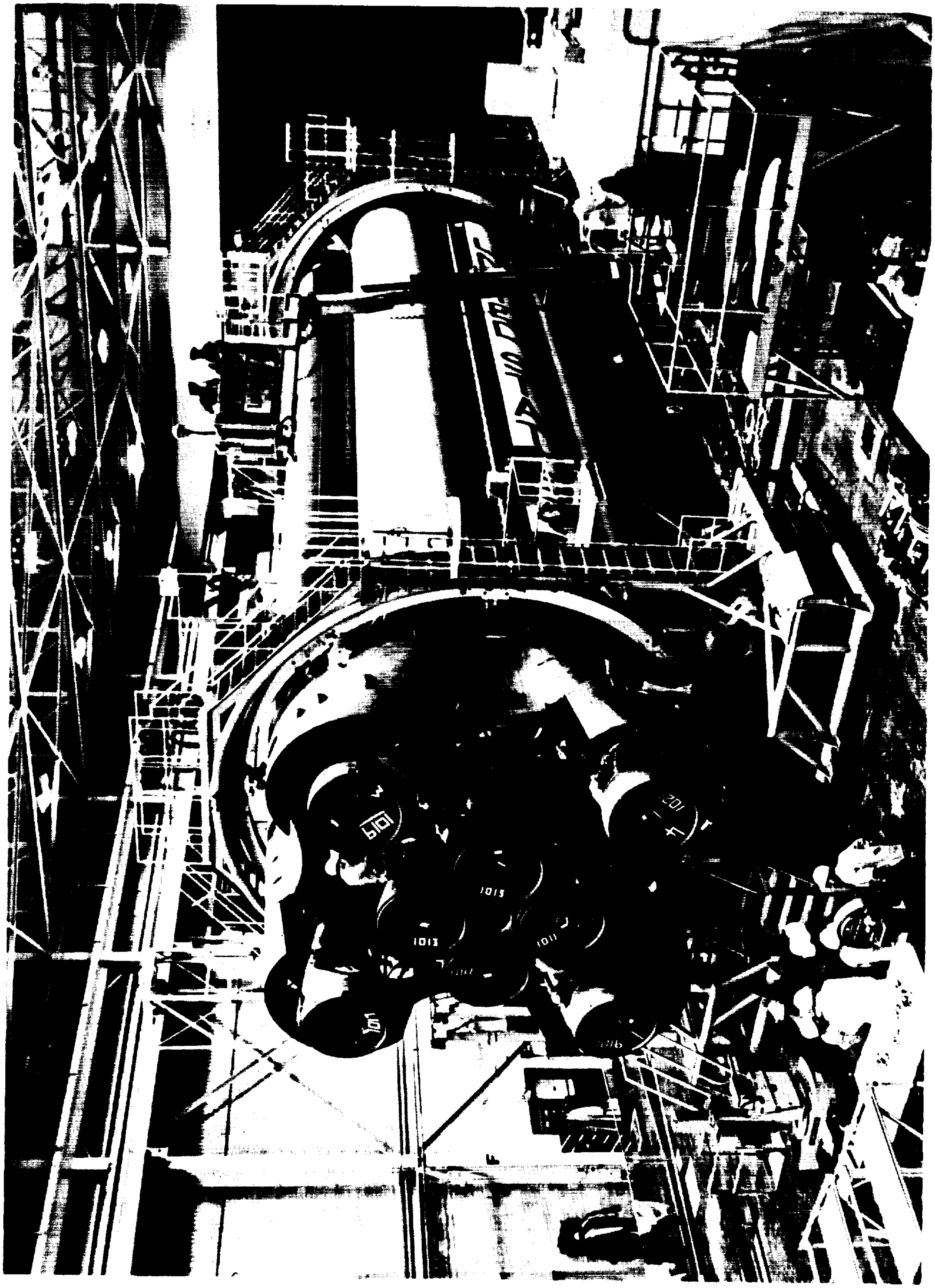

FIG. 4.9 BOTTOM VIEW OF SATURN STAGE I

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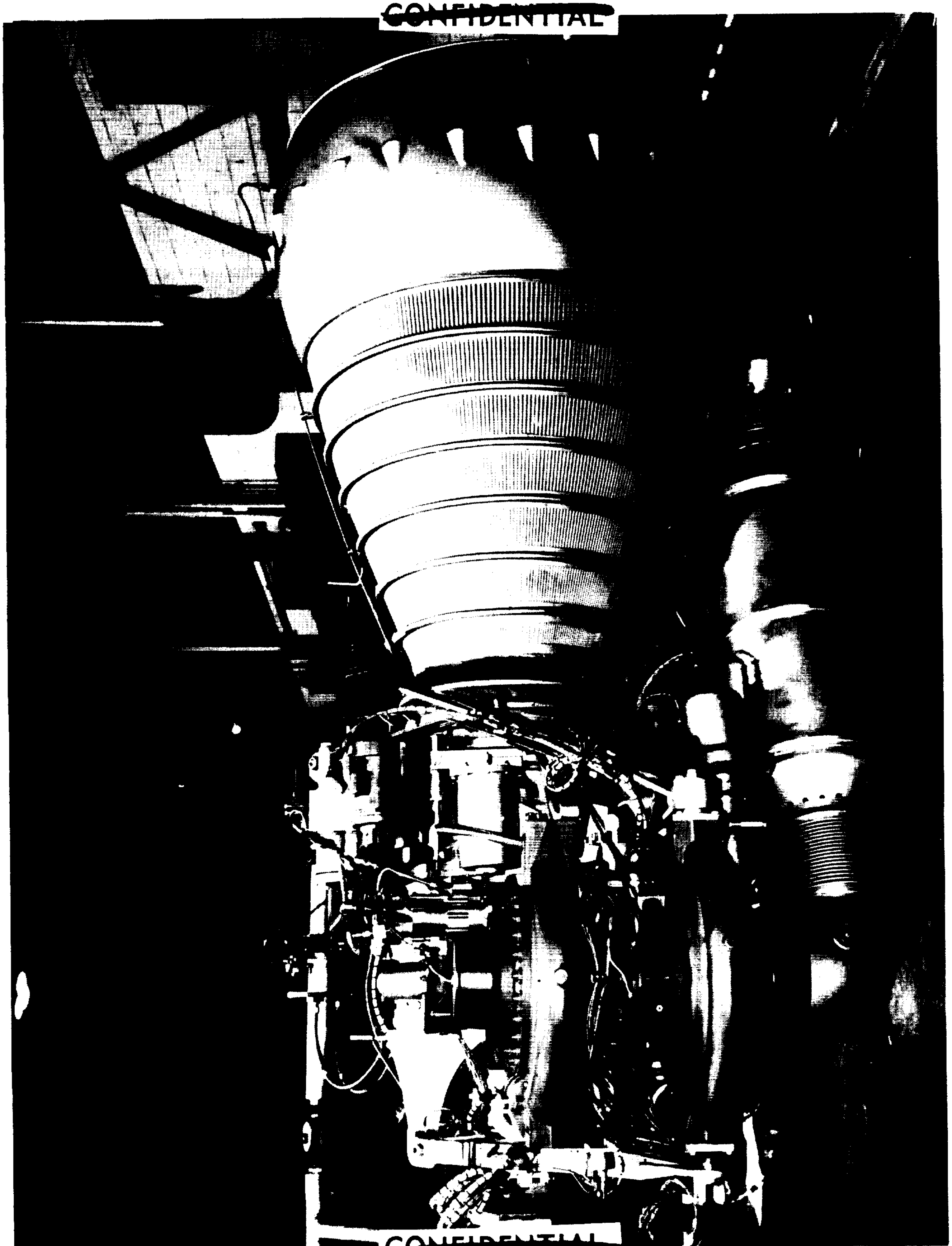

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FIG. 4.10.A.   ROCKETDYNE H-1 ENGINE FOR SATURN STAGE I.

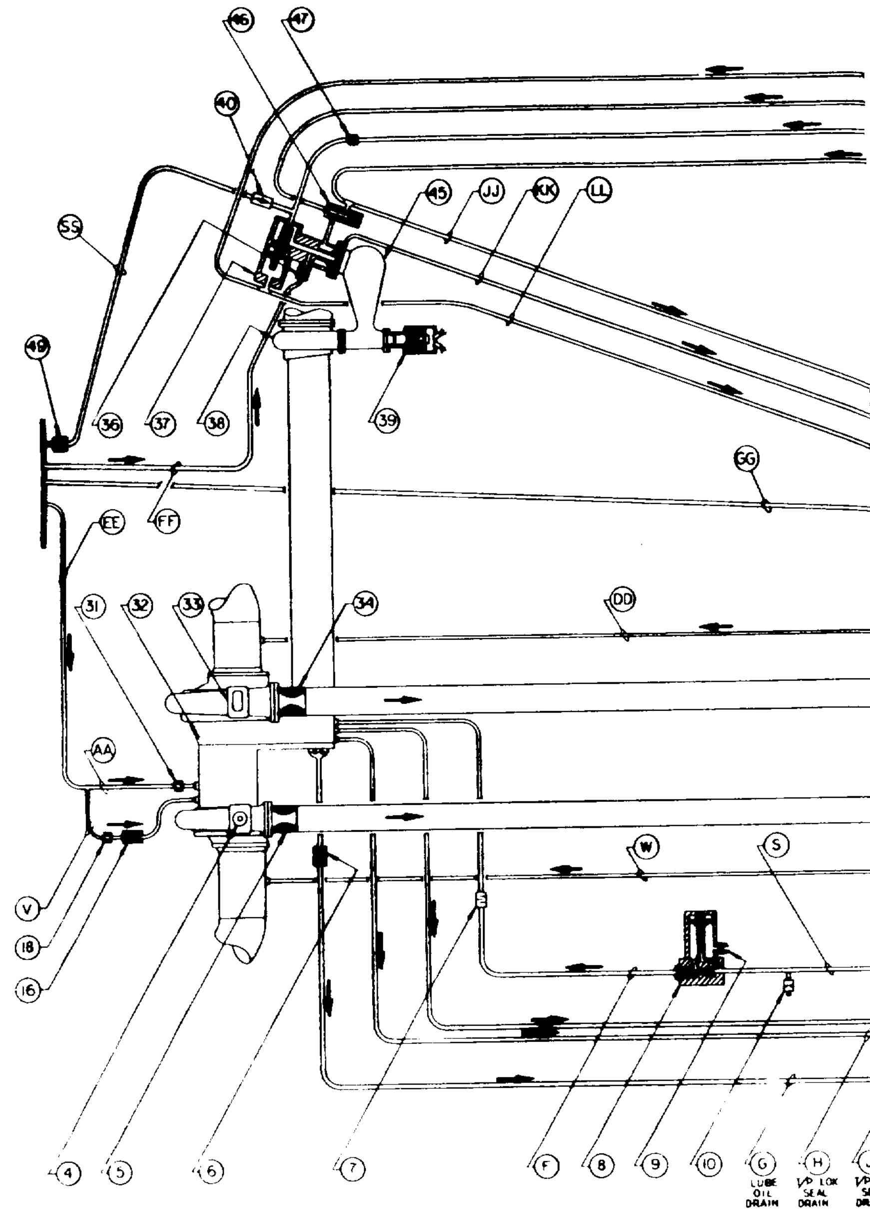
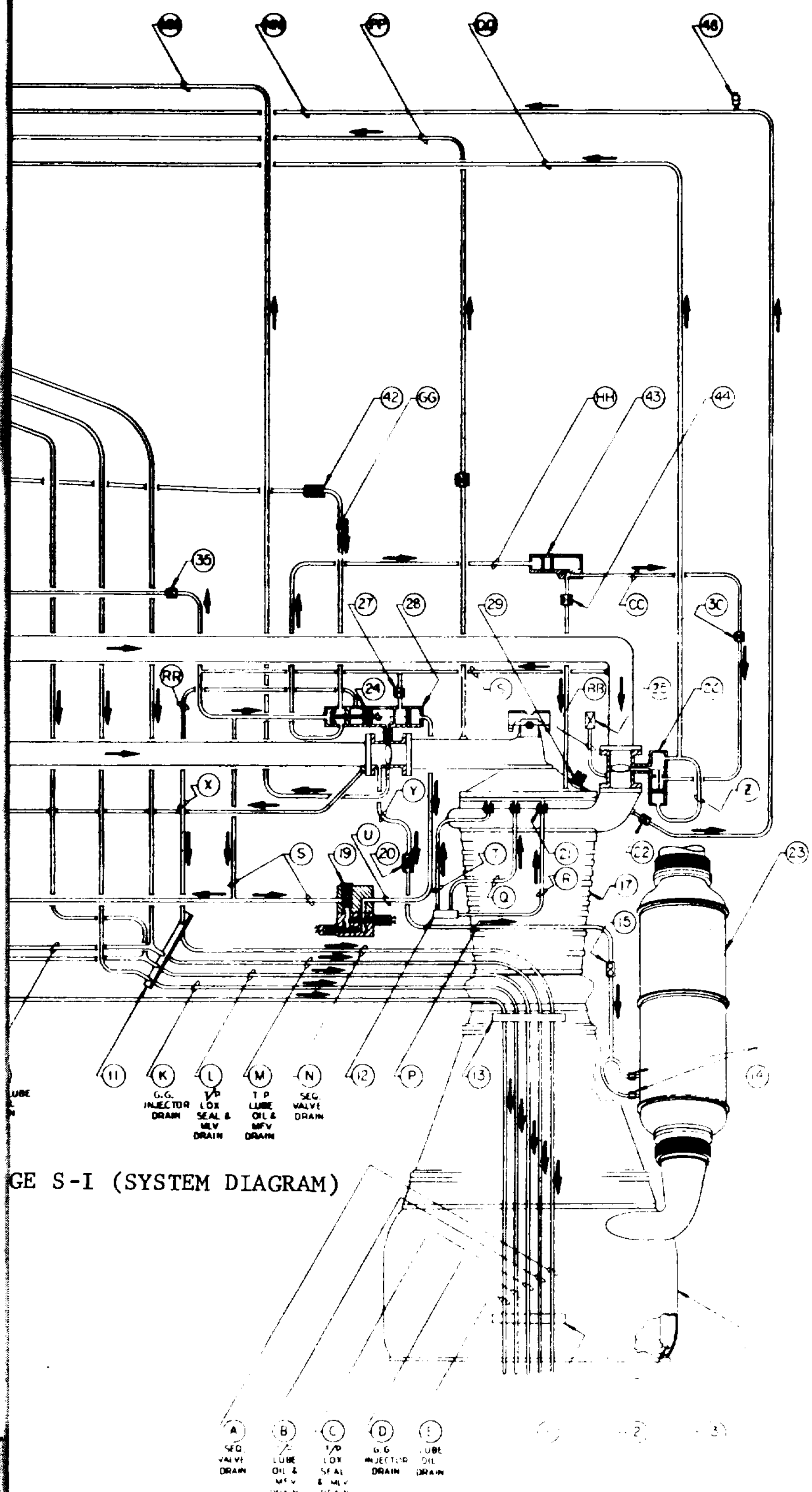


FIG. 4.10.B. ROCKETDYNE H-1 ENGINE FOR SATURN STA

FOLDOUT FRAME 1



GE S-I (SYSTEM DIAGRAM)

FOLDOUT FRAME 2

[illegible]	TUBE ASSEMBLY
A	VENT BKT TO ASPIRATOR BKT (SEQUENCE VALVE)
B	VENT BKT TO ASPIRATOR BKT (LUBE OIL AND FUEL)
C	VENT BKT TO ASPIRATOR BKT (LOX)
D	VENT BKT TO ASPIRATOR BKT (G.G. INJECTOR)
E	VENT BKT TO ASPIRATOR BKT (LUBE OIL)
F	BLENDER TO LUBE OIL FILTER (FLEX HOSE)
G	LUBE DRAIN TO T/C BRACKET
H	LOX SEAL DRAIN
J	LUBE SEAL DRAIN
K	GAS GENERATOR INJECTOR DRAIN
L	LOX SEAL DRAIN BKT TO T/C BKT
M	LUBE SEAL DRAIN BKT TO T/C BKT
N	SEQUENCE VALVE DRAIN BKT TO T/C BKT
P	CHECK VALVE TO HEAT EXCHANGER (FLEX HOSE)
Q	FUEL INJECTOR PURGE
R	FUEL INJECTOR PURGE
S	FUEL CONTROL LINE
T	FUEL INJECTOR PURGE
U	MAIN LOX VALVE TO CONAX VALVE
V	GEAR BOX PURGE
W	LOX BLEED
X	SEQUENCE VALVE DRAIN
Y	MAIN LOX VALVE TO CHECK VALVE (LOX LINE TO H.E.)
Z	MAIN FUEL VALVE VENT
AA	LOX SEAL PURGE
BB	MAIN FUEL VALVE TO PRESSURE SWITCH
CC	HYPERGOL TO MAIN FUEL VALVE
DD	FUEL BLEED
EE	T/P PURGE
FF	GAS GENERATOR INJECTOR PURGE (FLEX HOSE)
GG	HYPERGOL PURGE
HH	SEQUENCE VALVE TO HYPERGOL
JJ	GAS GENERATOR FUEL VALVE DRAIN
KK	GAS GENERATOR INJECTOR DRAIN
LL	GAS GENERATOR LOX VALVE DRAIN
MM	MAIN LOX VALVE DRAIN
NN	FUEL BOOTSTRAP (FLEX HOSE)
PP	LOX BOOTSTRAP (FLEX HOSE)
QQ	MAIN FUEL VALVE DRAIN (FLEX HOSE)
RR	SEQUENCE VALVE DRAIN
SS	AUXILIARY LOX DOME PURGE (FLEX HOSE)

FIND	COMPONENT
1	ASPIRATOR BRACKET
2	THRUST CHAMBER DRAIN
3	ASPIRATOR
4	T/P LOX DRAIN
5	LOX ORIFICE
6	CHECK VALVE (LUBE DRAIN LINE)
7	FILTER (LUBE OIL)
8	FUEL ADDITIVE ASSY
9	ORONITE FILL COUPLING
10	QUICK DISCONNECT COUPLING (FUEL DRAIN)
11	VENT BRACKET
12	MANIFOLD (FUEL INJECTOR PURGE)
13	T/C BRACKET
14	ORIFICE (HEAT EXCHANGER)
15	FLOW METER (LOX LINE TO HEAT EXCHANGER)
16	CHECK VALVE (GEAR BOX PURGE)
17	THRUST CHAMBER (T/C)
18	ORIFICE (GEAR BOX PURGE)
19	CONAX VALVE
20	CHECK VALVE (LOX LINE TO HEAT EXCHANGER)
21	CHECK VALVE (FUEL T/C INJECTOR PURGE)
22	ORIFICE (FUEL BOOTSTRAP)
23	HEAT EXCHANGER (H.E.)
24	SEQUENCE VALVE
25	PRESSURE SWITCH
26	MAIN FUEL VALVE (M.F.V.)
27	ORIFICE (CONTROL LINE)
28	MAIN LOX VALVE (M.L.V.)
29	CHECK VALVE (LOX DOME PURGE)
30	ORIFICE (M.F.V. CONTROL LINE)
31	ORIFICE (LOX SEAL PURGE)
32	TURBO PUMP GEAR BOX
33	FUEL DRAIN
34	FUEL ORIFICE
35	ORIFICE (FUEL BLEED)
36	CHECK VALVE (G.G. LOX INJECTION PURGE)
37	GAS GENERATOR LOX CONTROL VALVE
38	GAS TURBINE
39	TURBINE SPINNER
40	CHECK VALVE (AUX LOX DOME PURGE)
41	
42	CHECK VALVE (HYPERGOL PURGE)
43	HYPERGOL CONTAINER ASSY
44	ORIFICE (HYPERGOL TO LOX DOME)
45	GAS GENERATOR (G.G.)
46	GAS GENERATOR FUEL CONTROL VALVE
47	ORIFICE (LOX BOOTSTRAP)
48	QUICK DISCONNECT COUPLING
49	ORIFICE UNION

(C) Table 4.2

H-1 ENGINE PERFORMANCE PARAMETERS

Nominal Engine Thrust	188K lb (sea level)
Thrust Variation	± 3%
Thrust Overshoot	5%
Time from Ignition to 90 Percent Thrust (Maximum)	1.3 sec
Nominal Specific Impulse	255.5 $\frac{\text{lb-sec}}{\text{lb}}$ (sea level)
Engine Mixture Ratio, Lox to RP-1 by Weight	2.27:1
Cutoff Impulse (sea level)	42,000 lb-sec
Cutoff Impulse Variation (maximum) (sea level)	± 7,500 lb-sec
Lox Pump NPSH (required)	65 ft
Fuel Pump NPSH (required)	38 ft
Thrust Build-Up Rate (average)	71,300 lb/10 ms

The liquid oxygen containers are pressurized by gaseous oxygen supplied by eight heat exchangers (one on each engine turbine exhaust). The liquid oxygen is taken from a point downstream of the main lox valve and fed into the heat exchangers for vaporization. Heat exchangers for the outboard engines are mounted along the thrust chambers and swivel with their respective engines. Turbine exhaust gases from the inboard engines pass through the heat exchangers mounted horizontally above the firewall and discharge into the free air stream. Gaseous oxygen from the heat exchangers passes to a manifold, and is routed through an external line to a dome on top of the center tank. It is then distributed to the center tank and outer liquid oxygen tanks.

4.4.2.3. Propellant Feed System. The propellant system consists of the one 105-inch diameter center tank (for liquid oxygen), surrounded by eight 70-inch diameter tanks (four for liquid oxygen and four for fuel, alternately arranged). The aft or bottom header for the 105-inch tank has a sump supplying liquid oxygen through four interconnecting lines to the outer liquid oxygen tank sumps. The five lox tanks are interconnected at the bottom by lines feeding into a square shaped system of 8-inch transfer lines. The fuel tanks are also interconnected at the top by a pressurizing gas manifold.

Each outer tank supplies one inboard and one outboard engine through 8-inch suction lines. The 105-inch center tank is a reserve for the four outer liquid oxygen tanks and will supply approximately 36% of the lox consumed. If there is a premature cutoff of one engine, the propellant for that engine will flow through the interconnecting lines to supply the remaining engines.

Although the total maximum propellant capacity is about 866,000 pounds, the total available for mainstage (S-I) is approximately 850,000 pounds. The difference is accounted for by ullage, reserve, liquid oxygen for conversion to gaseous oxygen for pressurization, fuel-additive lubricant, etc.

4.5 S-I CONTROL SYSTEM

Each outboard engine is equipped with an independent closed-loop hydraulic system which supplies hydraulic pressure for engine gimbaling (Fig. 4.11). Two electrically controlled hydraulically operated actuators, a main hydraulic pump, an auxiliary motor and pump, and an accumulator-reservoir assembly are the major components in the S-I stage control system.

The hydraulic system fuctions when the main pump, driven by the turbopump, receives oil from the low pressure side (reservoir) of the accumulator-reservoir. The oil pressure is increased to approximately 2900

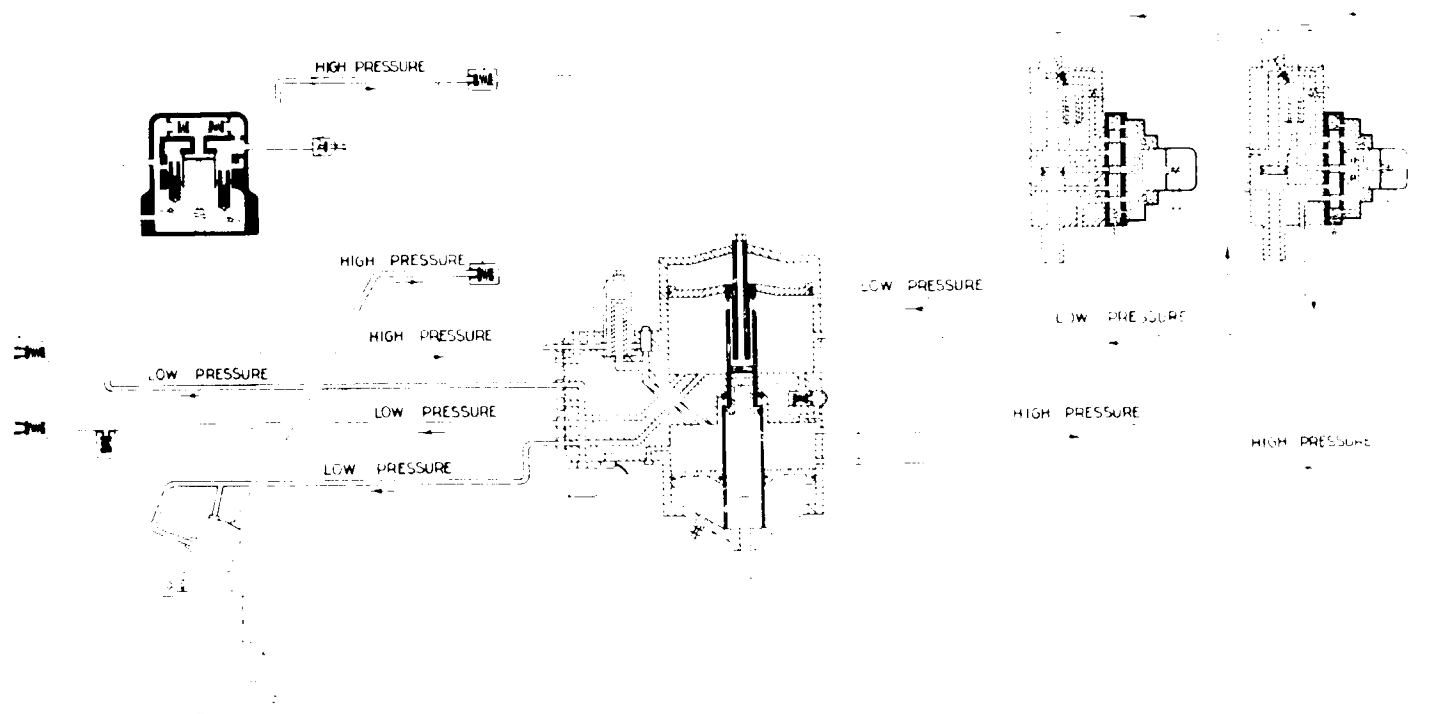


FIG. 4.11 HYDRAULIC SYSTEM FOR ENGINE GIMBAL ACTUATORS.

psig by the pump, and routed into the high pressure side of the accumulator-reservoir where the oil pressure is maintained between approximately 2600 and 3000 psig. From the accumulator-reservoir it is routed through filters to the actuator assemblies. Upon receipt of an electrical control signal, an electrohydraulic servo valve on each actuator allows the high pressure oil to flow into the actuator, extending or retracting the actuator arms. This movement provides gimbaling action to the engine and allows the engines to have a 10° square gimbal pattern. Displaced oil from the actuator assemblies is routed to the low pressure side of the accumulator-reservoir.

4.6 S-I TELEMETRY AND RF SYSTEMS.

The plan for telemetry on the Saturn C-1, S-I stage, is five rf links. The links are one FM/FM telemetry set, two Single Side Band FM sets and two PAM/FM/FM sets. This telemetry system then has 43 standard IRIG analog channels, 432 commutated channels and 15 wide band (3kc) analog channels for measuring purposes.

The only other rf link on the S-I stage is the range safety command destruct link.

The external type antennas for all of the links are placed on panels that fit across the area between two tanks in four places above each fin near the top of the tanks.

The frequencies are:

Telemetry	242.0 mc	253.8 mc
	246.3 mc	259.7 mc
	249.9 mc	
Command	(Classified SECRET)	

One PCM (pulse code modulated) - FM link in the 225 mc band duplicates transmission of data from T/M No. 1.

S-I telemetry for a typical Saturn C-1 is as follows:

T/M No. 1 and 2, PAM/FM-FM

216 channels sampled at 12/sec

14 continuous channels

T/M No. 3, PAM/FM-FM-FM

15 continuous FM channels

12 continuous FM/FM channels

T/M No. 4, SS/FM

15 continuous channels for vibration data, 50-3000 cps

4.6.1 S-I Measuring Program. Approximately 600 ground and inflight measurements in the Stage S-I may be required during a typical Block II mission. A condensed list of the measured parameters in Stage S-I, including the range of the measurements, is given in Table 4.3. This list of measurements may change from flight to flight depending on the evaluation of the previous flights. Usually the listed measurements are only fed into the stage telemetry system for the transmission to ground stations. Some critical signal voltages and propulsion measurements, however, also will be routed to the instrument unit of the Saturn C-1 Block II vehicle for eventual use in the abort system and in the spacecraft.

4.6.2. Tracking. An ultrahigh-frequency Doppler tracking system (UDOP) has been developed for Saturn. It operates over a frequency range of 430-470 mc/s. The transponder and amplifier together weigh only 18 pounds. Full power output is 15 watts. Two Jupiter UDOP antennas will be used for each flight.

Azusa and C-band radar will also be used for tracking throughout the Saturn C-1 program. S-band radar will be included for SA-1 and SA-2, but, thereafter, will be discontinued since it is not for compliance with range safety requirements.

4.7. S-I FABRICATION AND ASSEMBLY

The Saturn booster is basically a cluster of nine cylindrical containers structurally joined together forward and aft. These containers hold the fuel and oxidizer that feed the eight H-1 Rocketdyne engines. The four inboard engines are stationary and the four outboard engines gimbal. Four stabilizing fins, a tail shroud, firewall, heat shield, and flame shield are located aft.

4.7.1. Container Section Fabrication and Assembly. The 105-inch diameter and the 70-inch diameter containers are manufactured by about the same methods. Cylindrical sections are made of 5456 aluminum alloy sheet, formed and welded. The hemispherical bulkheads are formed from 5086 aluminum alloy plate by spinning and machining. Stiffening rings are spot welded to the skin sections, and bulkheads are circumferentially

(C) Table 4.3

TYPICAL STAGE S-I MEASURING PROGRAM

MEASURED PARAMETER	RANGE
Flight Sequencer Steps	On-Off
First Motion	On-Off
Cutoff Signal	On-Off
Engine Cutoff	On-Off
Lox Level Cutoff	On-Off
Fuel Level Cutoff	On-Off
Retrorocket Ignition Signal (EBW)	On-Off
Breakwire, R-Rocket Ignition	On-Off
Temperature, Base, Tail	0-1500°C
Temperature Actuator Support	-50 to +500°C
Temperature at Various Inboard and Outboard Locations	-50 to +900°C
Temperature Outrigger Thrust Assy. L. P.	-50 to +300°C
Temperature Lower Thrust Ring	-50 to +300°C
Temperature Outrigger Support Assy, L. P.	-50 to +300°C
Temperature Engine Shroud	0-500°C
Temperature Thrust Frame Area	-50 to +100°C
Temperature Radiation Shield	0-800°C
Temperature Tank Shroud	0-300°C
Temperature Shroud Skin	0-800°C
Temperatures of S-I/S-IV Interstage	0 to 825°C
Pressure, Base, Tail	0-20 psia
Pressure Retrorocket	0-2500 psia
Pressure Surface, Various Stations	0-25 psia
Strain, Mounting Stud	.0008 to .008 ins/in.
Vibration Radiation Shield, Longitudinal	±50g

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(C) Table 4.3 (Continued)

MEASURED PARAMETER	Range
Strain, Top Skirt	.0005 to .005 inches/in.
Vibration Retrorocket, Longitudinal	0 to 40g
Vibration, Spider Beam, Longitudinal	0 to 40g
Sound Intensity, Fine	10 db
Sound Intensity, Coarse	110-190 db
Experimental EBW, Voltage	0-5V
Experimental EBW, Trigger	On-Off
Experimental EBW, Breakwire	On-Off
EBW Voltage	0-5V
Measuring Voltage	0-5V
Frequency of Inverter	400±.25 cps
AGC Voltage-UDOP	0-5V
AGC Voltage-Command	0-5V
Beta Reference Voltage	0-5V
Command Voltage	55-65V
Power Trans. Bus Volt	0-28V
Thrust Commit Volt	0-28V
Bus Voltage	24-32V
Power Output, Azusa	0-2.4 VDC
AGC Voltage, Azusa	0-2.4 VDC
Frequency, Static Inverter	400±.25 cps
Temperature Fuel Pump Inlet	-20 to +40°C
ΔP Fuel Sloshing, Pitch	±0.3 psid
ΔP Fuel Sloshing, Yaw	±0.3 psid
ΔP Lox Sloshing, Pitch	±0.5 psid

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(C) Table 4.3 (Continued)

MEASURED PARAMETER	RANGE
ΔP Lox Sloshing, Yaw	±0.5 psid
ΔP P. U. Computer, Lox	0-30 psid (1)
ΔP P. U. Computer, Fuel	0-20 psid (1)
Pressure Fuel Pump Inlet	0-100 psi
Pressure Lox Pump Inlet	0-150 psia
Vib. Fuel Suction Line, Longitudinal	±50g
Vibration Fuel Tank Pitch	±0.5g
Vibration Fuel Tank, Yaw	±0.5g
Lox Emergency Press Switch	On-Off
Lox Relief Control Valve	On-Off
Temperature Fuel	0-40°C
Main Fuel Valve Pos	On-Off
Flow Rate Main Fuel, dc	0-40 gal/sec
Flow Rate Main Lox, dc	0-50 gal/sec
Main Lox Valve Position	On-Off
Flow Rate Main Fuel, ac	0-40 gal/sec
Flow Rate Main Lox, ac	0-50 gal/sec
P. U. Computer Output - Coarse	
P. U. Computer Output - Fine	
Lox Level, Discrete	On-Off
Fuel Level, Discrete	On-Off
Temperature Gas Top Fuel Tank	-50 to +50°C
Temperature Gas Top Lox Tank	-50 to +200°C
Temperature Skin Lox Tank	-185 to +150°C
Temperature Skin Fuel Tank	0-300°C

(C) Table 4.3 (Continued)

MEASURED PARAMETER	RANGE
Temperature Lox	-190 to -40°C
Temperature Lox Pump Inlet	-200 to -160°C
Flow Rate Lox to Heat Ex., ac	33 gpm
Temperature Heat Exchange Gox Outlet	0-300°C
Temperature High Pressure Spheres	-70 to +150°C
Temperature Gox Manifold	-200 to +700°C
Pressurization Gas in Fuel Tank	0-30 psi
Pressurization Gas in Lox Tank	0-100 psi
Pressurization Gas High Pressure Spheres	0-3500 psi
Pressurization Gox Manifold	0-400 psi
Temperature Gas Generator Chamber	0-1000°C
Pressurization Gas in Fuel Fank	0-30 psi
Pressurization Gas High Pressure Spheres	0-3500 psi
Pressurization Gas Generator Lox Injector	0-800 psia
Turbine RPM	0-7000 RPM
Temperature Lox Pump Bearing	-20 to +200°C
Temperature Intermediate Shelf Bearing	0-150°C
Temperature High Speed Bearings	0-150°C
Temperature Turbine Shaft Bearings	0-300°C
Temperature Turbine Exhaust	0-800°C
Temperature Gas Generator Chamber	0-1000°C
Temperature Hydraulic Oil	-20 to +135°C
Temperature Engine Compartment	0-1000°C
Temperature Fire Wall	0-1500°C
Pressure Combustion Chamber	0-600 psia

(C) Table 4.3 (Continued)

MEASURED PARAMETER	RANGE
Pressure Turbine Inlet	0-650 psia
Pressure Turbine Exhaust	0-40 psi
ΔP Turbine Outlet to Aspirator Inlet	0-20 psid
Pressure Gear Case Top	0-30 psi
Pressure Gear Case Lubrication High	0-800 psia
Pressure Gear Case Lubrication Low	0-200 psia
ΔP Across Aspirator	0-10 psid
Pressure Inside Tail	0-25 psia
Pressure Hydraulic Source	0-3500 psi
ΔP Yaw Actuator	$\pm$ 3000 psid
ΔP Pitch Actuator	$\pm$ 3000 psid
Pressure Hydraulic Oil Ret.	0-100 psi
ΔP Turbine Outlet to Exhaust Nozzle	0-20 psid
Pressure Combustion Chamber	0-6 psia
Pressure Ignition Monitor Valve Inlet	0-1000 psia
Vibration Thrust Chamber Dome, Lateral	$\pm$50g
Vibration Turbine Gear Box	$\pm$50g
Vibration Thrust Chamber Dome, Longitudinal	$\pm$50g
Vibration Pitch Actuator, Pitch	+40g
Vibration Yaw Actuator, Yaw	$\pm$40g
Vibration Gimbal Point Support, Longitudinal	+50g
Vibration Propulsion Unit Distribution 9A1, Longitudinal	$\pm$30g
Vibration Thrust Ch. Dome, Lateral	$\pm$50g
Position Pitch Actuator	$\pm$10°
Position Yaw Actuator	$\pm$10°

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(C) Table 4.3 (Continued)

MEASURED PARAMETER	RANGE
△ I Pitch Actuator	±10 ma
△ I Yaw Actuator	±10 ma
Level, Hydraulic Oil	0-3.75"
Temperature Lox Pump Bearing #1	-20 to +200°C
Temperature Hydraulic Oil	-20 to +135°C
Temperature Gear Case Lubricant	0-150°C
Temperature Turbine Spinner Case	-50 to -100°C
Temperature, Fire Detection	
Pressure Hydraulic Source	max 23 psi
Pressure Gas Generator Fuel Injector	0-800 psi
Vibration Combustion Stability Mon, Longitudinal	±100g

welded together to from the containers. These containers are hydrostatically pressure tested, cleaned, calibrated, and painted prior to transporting to the final assembly areas.

4.7.2. Spider Beam Fabrication and Assembly. The spider beam assembly supports the fuel and oxidizer containers at the forward end of the booster. It is composed of extruded, cast, and machined structural components which are mechanically fastened together. Precise optical control is maintained throughout manufacture.

4.7.3. Tail Section Assembly. The tail section assembly is composed of a cylindrical barrel which is built up of corrugated aluminum alloy skins and shaped plates for stiffening around the openings. The outrigger beam assemblies are fabricated from aluminum alloy structural members and mechanically fastened. The outrigger beam assemblies support the eight engines, the four stabilizing fins, and the aft end of the 70-inch diameter containers. The stabilizing fins are installed, aligned, match marked, and then removed from the tail assembly and transported to the launch site for final assembly. This is accomplished with the tail section assembly in a vertical position.

4.7.4. Center Unit Assembly. The tail section assembly is mounted in the aft tooling ring and fuel pressurization sphere assemblies and other components are installed. The 105-inch diameter container is then fastened in place, completing the lox container center unit assembly.

Propellant containers are assembled to the lox container center unit assembly in the main assembly fixture. After the first 70-inch diameter container is installed, the assembly is rotated 180° for installation of the next container. The remaining containers are installed in alternate positions to maintain balance of the cluster. The cluster is optically aligned during container installation. Propellant fee, vent and pressurization systems are installed. Control wiring, electrical components, and other miscellaneous hardware are then installed to complete the propellant container assembly. Structural and assembly manufacturing sequence is shown in (Fig. 4.12).

4.7.5. Power Unit Assembly. The power unit assembly consists of the propellant container unit assembly, eight H-1 rocket engines, lower suction lines, inboard engine turbine exhaust system, lox line assemblies, and measuring and control instruments. Measuring and control components are installed in the compartments at the forward end of numbers one and two fuel tanks through access doors. Alignment within the instrument compartments is accomplished by employing precise optical equipment. Electrical, hydraulic, and measuring components are installed on the H-1 rocket engines. The engines are then transported to the assembly area for installation on the power unit. This is accomplished by means of the press ray inflatable beam and overhead crane.

4.7.6. Stage Assembly. The heat shield, flame shield, tail shroud, water quench systems, and miscellaneous components are installed to complete the Saturn stage assembly.

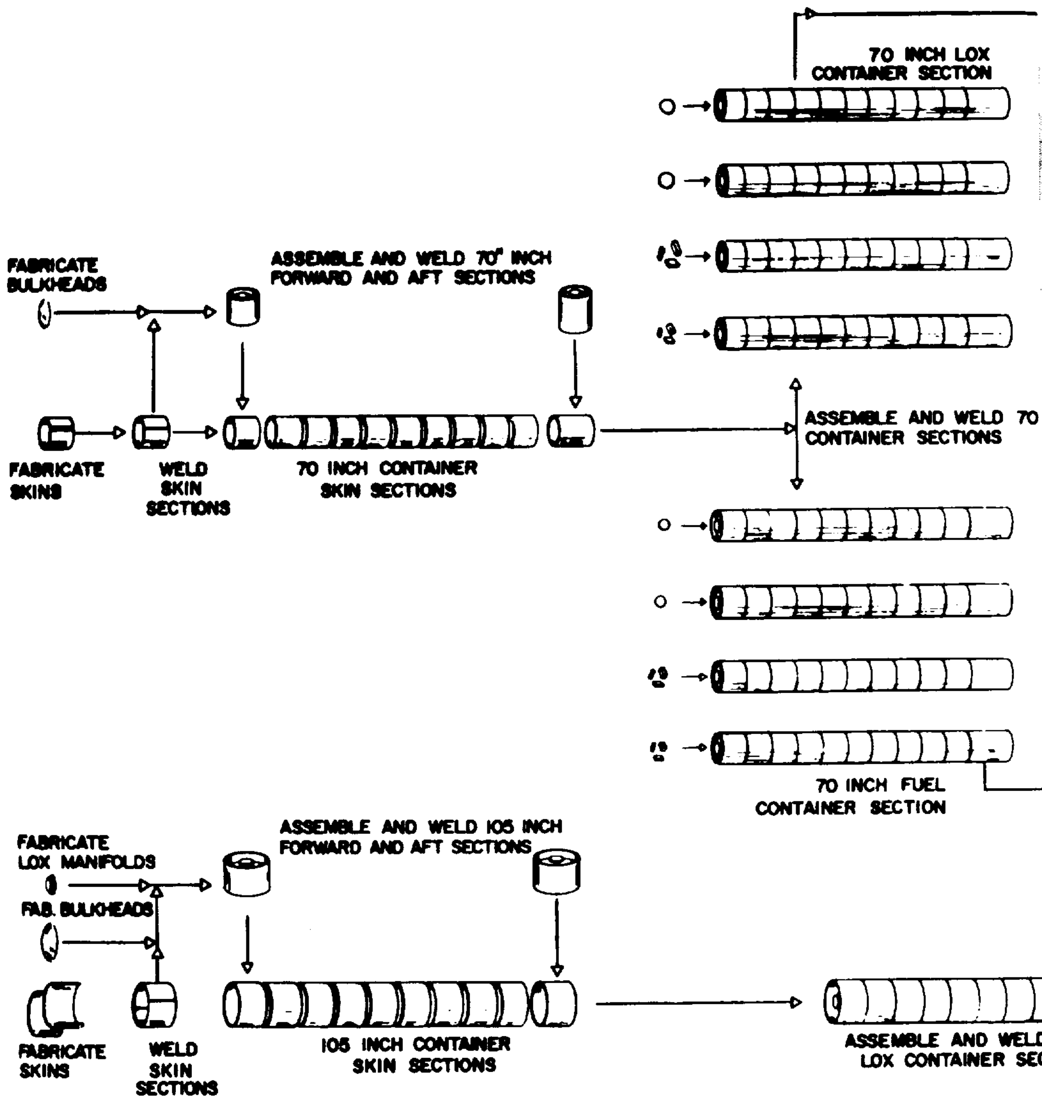

FIG. 4.12 SATURN C-1 ST

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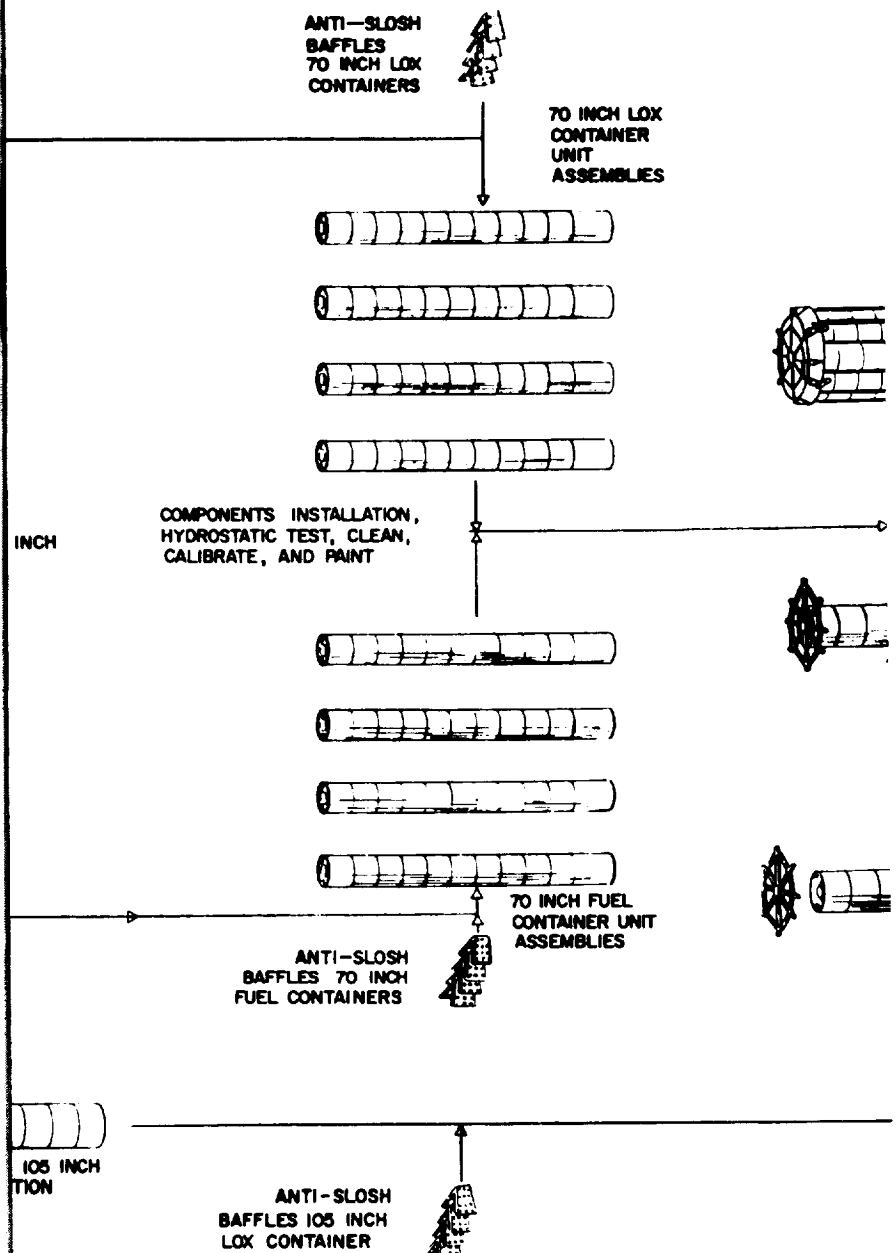
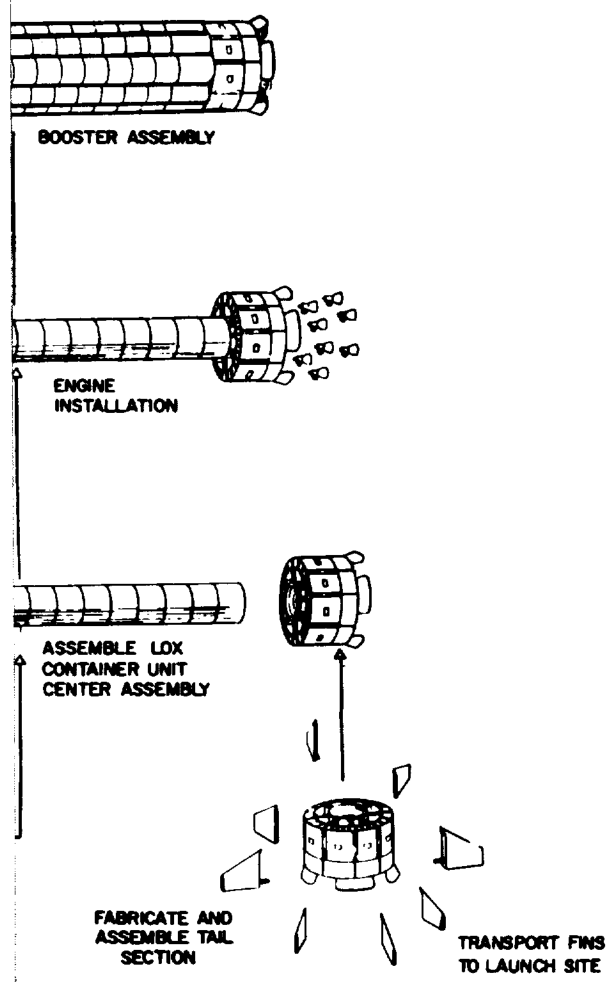


AGE S-I STRUCTURAL AND ASSEMBLY MANUFACTURING SEQUENCE.

FOLDOUT FRAME 2



FOLDOUT FRAME 3

4.8 S-I TESTING

The primary objective of the test program is to provide and demonstrate vehicle systems which will operate safely and reliably so as to accomplish the complete flight mission. Since the Saturn will eventually carry men, and thus must be man-rated, reliability and safety become doubly important and require careful scrutiny throughout the entire effort. This testing may be subdivided into several categories.

4.8.1 Research Tests. These are usually laboratory tests conducted under closely controlled conditions. An example is the investigation of base heating at launch. Cold flow tests and 1/20 scale hot jet model tests were conducted wherein prelaunch and liftoff phases were simulated to determine exhaust gas backflow and base heating. The results of these tests were substantiated by providing special instrumentation in the boattail area during subsequent full scale hot firings of the Saturn S-I stage on the MSFC static test tower.

Similar testing is being performed in conjuction with Arnold Engineering Development Center to determine the probable base heating in flight.

4.8.2. Structural Test Program. Flight qualification testing of components and structures permits a determination of load carrying capabilities and safety margins under maximum expected service operational environments. These tests simulate flight cutoff, maximum dynamic pressures, launch hold down and rebound conditions, and maximum aerodynamic lift and drag loads acting on the booster fins. Test results serve to disclose any areas of submarginal design and permit corrective actions to be taken prior to Apollo/Saturn launch dates. Experimental evaluation of designs also provides guide lines for material specifications and for quality control during fabrication.

A special static structural test tower capable of sustaining loads of 3,000,000 pounds has been constructed to accommodate the Saturn tail section and a short stage assembly. Engine thrust loads, tank inertia loads, flight shear loads, fin lift, and drag loads are simulated by hydraulic cylinders acting simultaneously and under accurately controlled conditions for load synchronization. Figure 4.13 shows preliminary test setups for a finned stage configuration.

The requirements for elevated temperature tests are based on the need for simulating aerodynamic heating and loading conditions simultaneously. Under transient heating, thermal gradients and thermal stresses can become large enough to cause failure or severe distortion even in a structure which experiences low aerodynamic loads. Consequently, both thermal stresses and load stresses as well as changes in material properties must be considered in the design criteria.

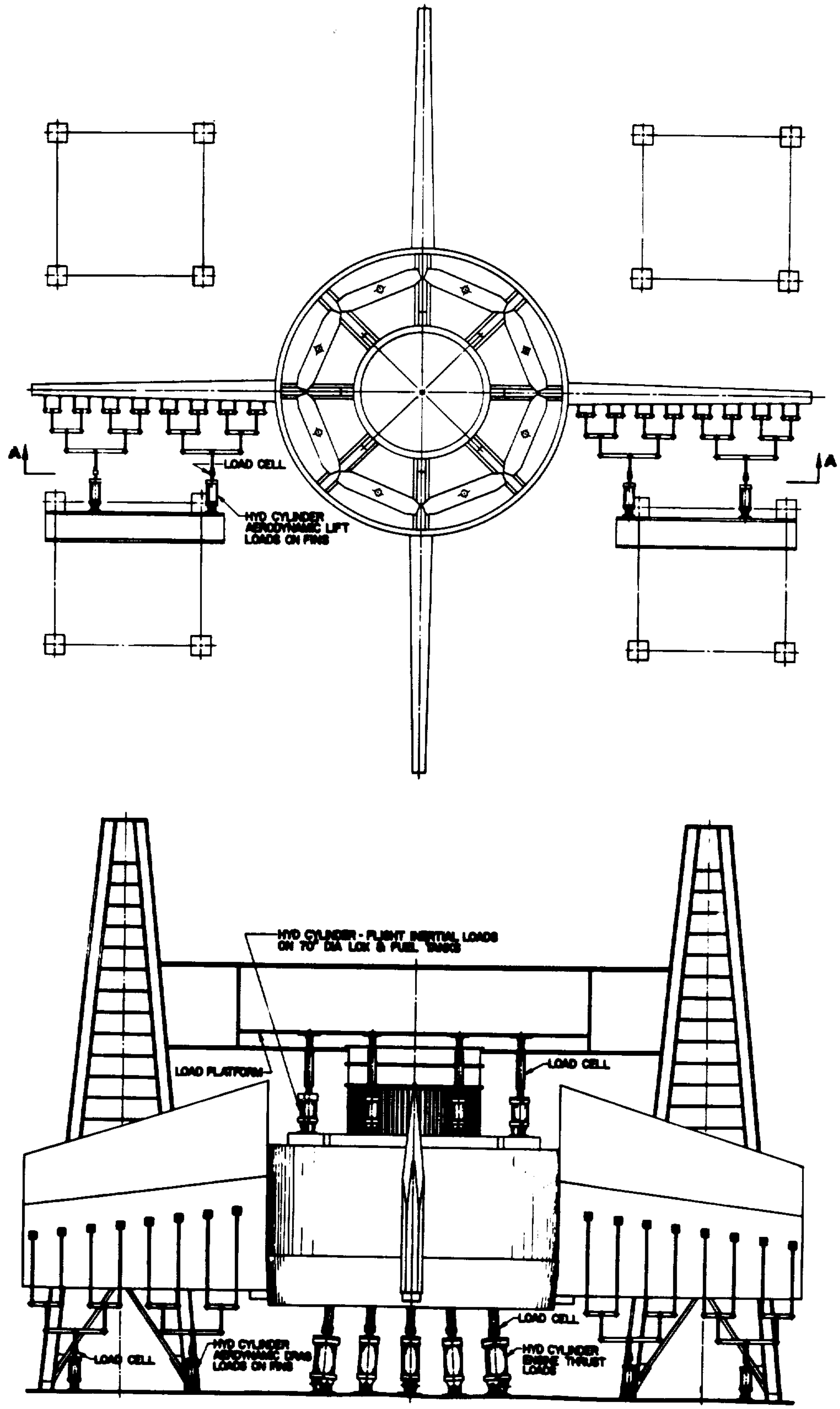


FIG. 4.13 STRUCTURAL TEST - TAIL SECTION ASSEMBLY, SATURN C-1 LAUNCH VEHICLE.

The aerodynamic heating simulator consists of six complete control and power channels capable of handling three thousand kilowatts of power as normal load. Infrared heat lamps are used as the heat source and the heat output is controlled by a computer and ignition controller to regulate the skin temperature in the structure or the heat rate applied according to any foreseeable combination of flight environments.

Size and configuration of reflectors are made to conform to the test configuration. Figure 4.14 shows a test set up to apply proper aerodynamic heating conditions and inertial loads simultaneously to the booster fins.

4.8.3. Vehicle Acceptance Tests. All vehicles will be subjected to a complete checkout prior to captive test firings or shipment to the launch site.

The vehicle tanks, pneumatic systems, and power plants will be leak checked. Electromechanical and functional tests will demonstrate the proper operation of engine ignition and sequence control systems, pressure switches, and valve interlock circuits.

Upon completion of subsystem checkout, a complete electrical system checkout will be conducted on internal power simulating the inflight sequence. All instrumentation will be operated with the data recorded on magnetic tape. The reduced data will be evaluated and utilized for vehicle acceptance.

Captive test firings of the S-I stage are performed at MSFC. These stages are tested without fins attached so that existing stand equipment can be used (Fig. 4.15). Two tests are made for each of the stages tested. The first is a short duration test to establish proper functioning of the system. The second test is of full duration to establish proper inflight sequencing of the system and to qualify the stage for launch.

A successful full duration firing is required to qualify the stage under test. Ability of the systems to work together properly is demonstrated. The test also demonstrates compliance with performance rating prescribed in the engine model specifications. Performance of the engine cluster is compared to individual engine performance by utilizing calibration data supplied by the engine manufacturer. The vehicle is equipped with a flight instrumentation package during the static firing. Prediction of flight performance of each cluster is accomplished by combining vehicle acceptance test performance data, the engine manufacturers' gain characteristics (for example, thrust change as a function of oxygen pump inlet pressure), and predicted values of engine input flight histories such as pump inlet pressure and temperatures.

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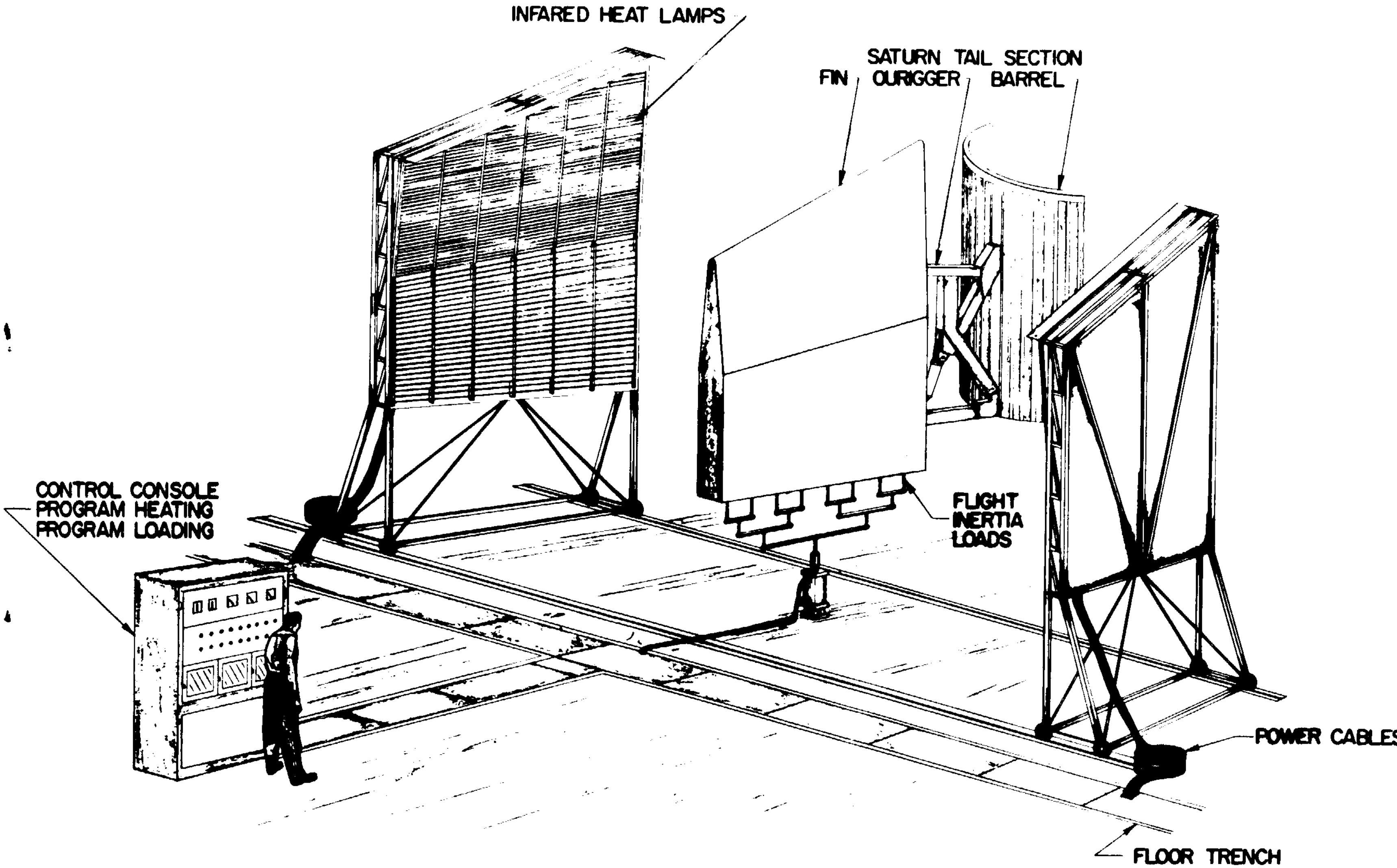

FIG. 4.14 AERODYNAMIC HEATING TEST - TAIL SECTION FINS, SATURN C-1 LAUNCH VEHICLE.

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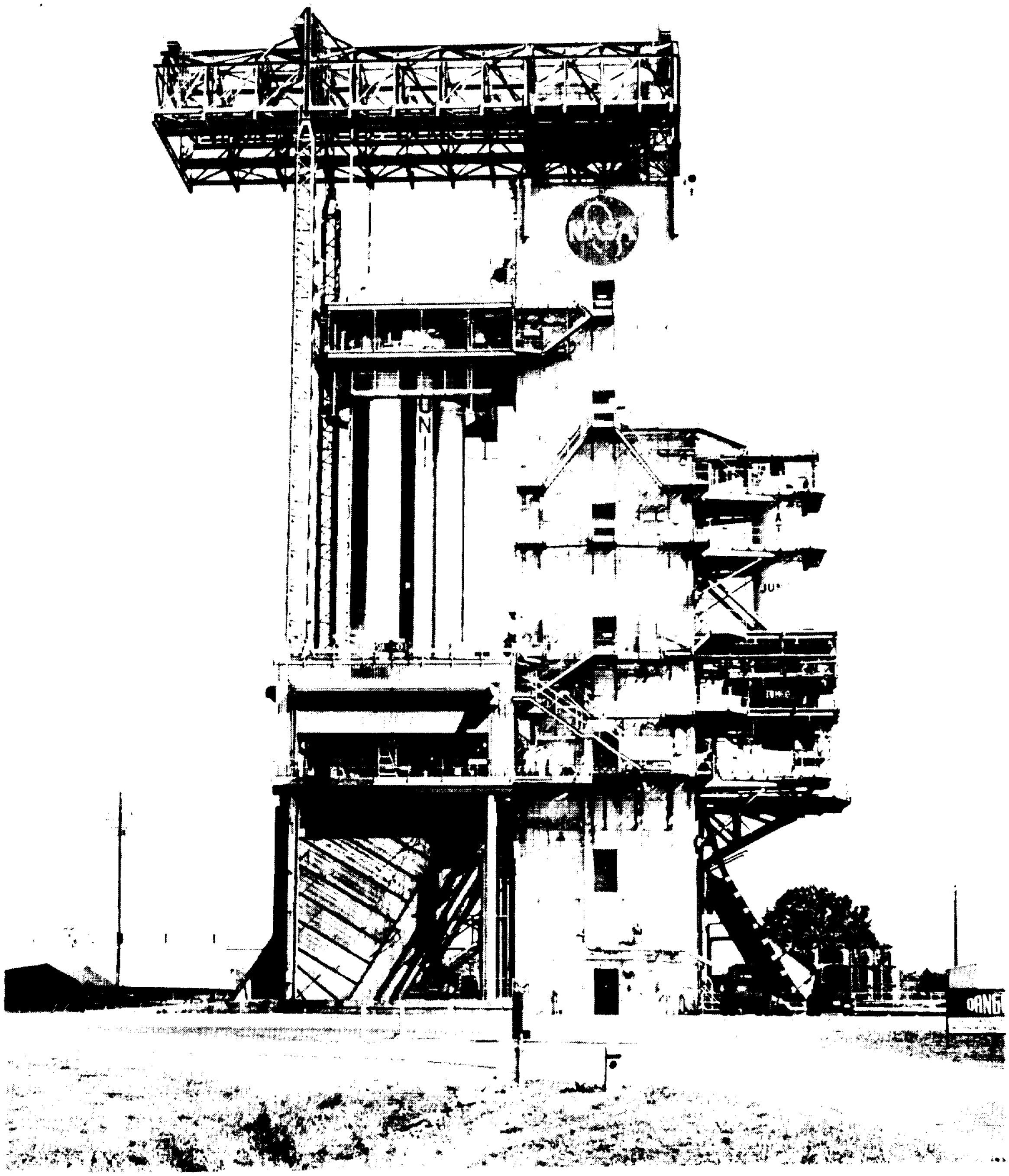

FIG. 4.15 SATURN CAPTIVE TEST TOWER.

All S-I stage captive firing tests are performed at the Saturn Test Facility Figure 4.16 shows a typical S-I stage during acceptance testing.

4.8.4. Launch Site Testing. All components will be subjected to a rigorous inspection to determine the nature and extent of any damage received in transit. Subsystems and components will receive routine operational and functional checks prior to mating tests and subsequent assembly of the vehicle on the launch site. Complete subsystem and systems checks will be performed prior to launch.

4.9. S-I ENVIRONMENT

The Saturn booster is probably the most powerful sustained source of sound existing today. A conservative estimate of the acoustic power developed by the motors while the vehicle is in flight exceeds 1,000 kilowatts. It is generally recognized that the sound pressures created by this large energy radiation may be severe enough to create problems around the launch complex, such as damage to structural components, malfunctions of instrumentation, damage to hearing, and annoyance due to noise. These conditions need a base for the development of solutions to the problems that could occur.

The estimates concern (1) the acoustic power generated by the engine, (2) the near-field sound pressures as a function of frequency, and (3) the overall far-field sound pressures as a function of time, distance, and frequency. The peak overpressure generated by the ignition of the engines is an important part of the conditions, too, but not enough information is available to permit an estimate of the magnitude of the overpressure.

The predictions are based on a calculation of the sound pressure level at several radii from the pad centerline, considering the position of the vehicle, the directivity of the radiation, and atmospheric absorption losses. As the vehicle proceeds on its launch, these relations change, so the sound pressures are related to time in the estimates.

The following procedure was used to make the estimates: First, an estimate of the amount of mechanical power developed by the booster, converted into acoustic power by the turbulence, is made. This estimate also considers the distribution of energy in the frequency spectrum. Then, for a particular location in the space around the sound source, the sound pressure can be calculated as a function of acoustic power developed at the source, the distance from the source, and the direction from the source.

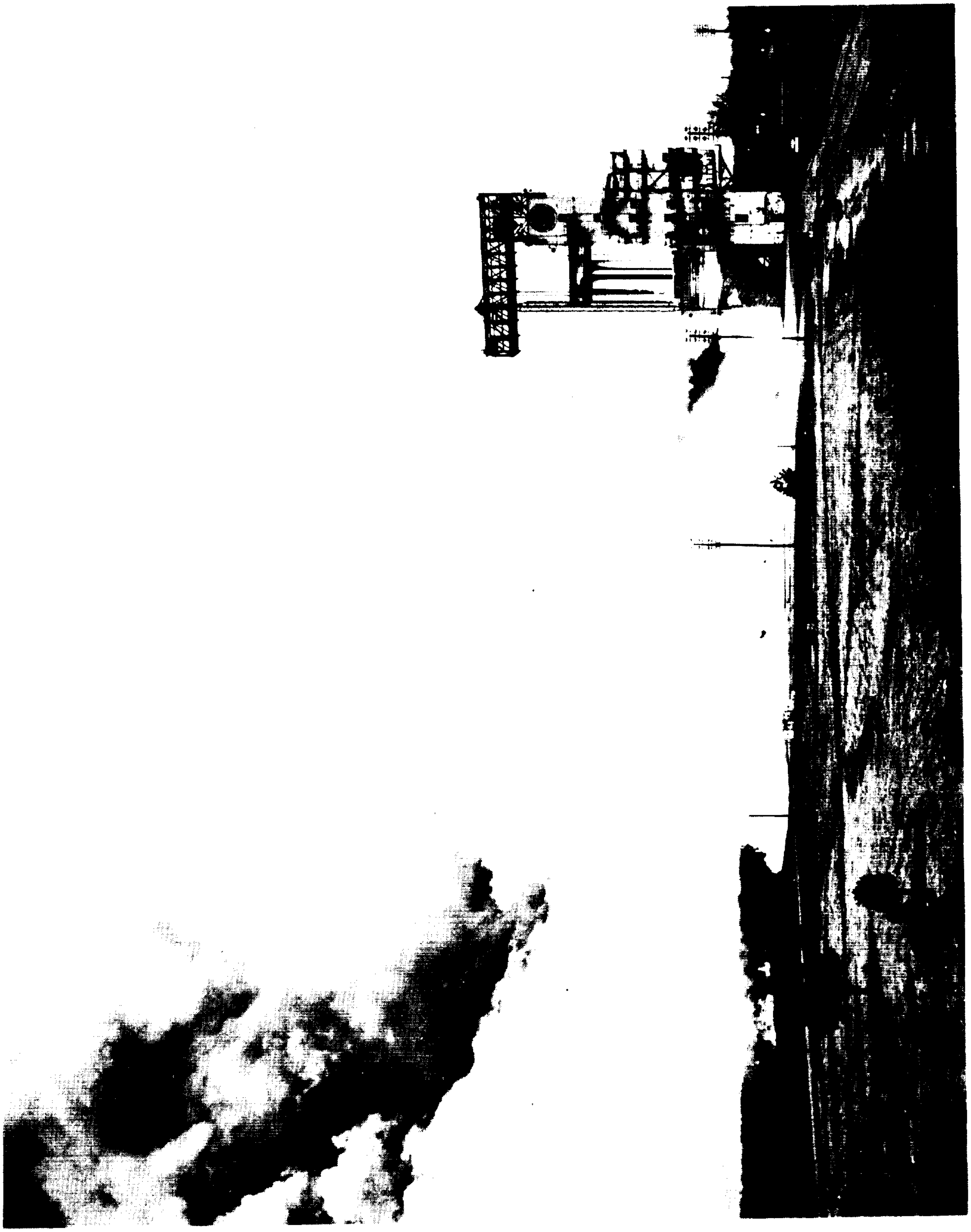

FIG. 4.16 CAPTIVE TEST - SATURN C-1 STAGE S-I.

The calculated sound pressure is also a function of many conditions that affect the radiation of the energy. Two of these effects that have been evaluated include the way the source radiates sound and the atmospheric absorption of the energy.

The sound source radiates into a half sphere of space while the source is close to the ground. Sometime after liftoff, the source begins to radiate into a full sphere of space, and after this transition occurs only half the previous energy will be radiated through a particular plane in space. The correction based on the decrease in the sound pressures due to this transition is also included in the estimate. The best illustration of this is the curve of sound pressure versus time in Figure 4.17. Other items that could affect the radiation have been disregarded and will be mentioned later.

After liftoff, the distance and angle relationship between the sound source and the location will change as the vehicle proceeds along its trajectory. Therefore, the estimate is made for different times during the launch when evaluations of the instantaneous relationships are calculated, and the resulting sound pressure levels are presented with reference to time.

The same general procedure is also used to estimate the spectral distributions of the sound pressure. The power, in frequency bandwiths at the source is estimated and the related calculations of sound pressure within given bandwidths is made, considering all corrections previously mentioned.

Some of the effects on the sound radiation that had to be disregarded in the estimate include: the effects of the flame deflector on sound generation and radiation while the vehicle is resting on the pedestal; effects of air velocity and air temperature gradients; effects of local turbulences in the medium; and effects due to cloud presence in the atmosphere. Another important influence is the finite amplitude distortion of the sound pressures in the near field; this distortion causes the sound energy to shift from lower frequencies to higher frequencies. However, the degree of such a shift is not contained in the prediction with the consequence that the high frequency sound pressure estimations might be low.

The estimates of the distribution of the power generated by the source as a function of frequency is given in Figure 4.18. The estimate is given for both single engine operation and eight engine operation and, as a basis of comparison, the measured power generated by the Atlas missile during static tests is also plotted. Special consideration of the near-field estimate, in Figure 4.19, is needed. The sonic near-field of the source moves with the motion of the vehicle, and while the structure of the vehicle is always in the near-field, the complex is only

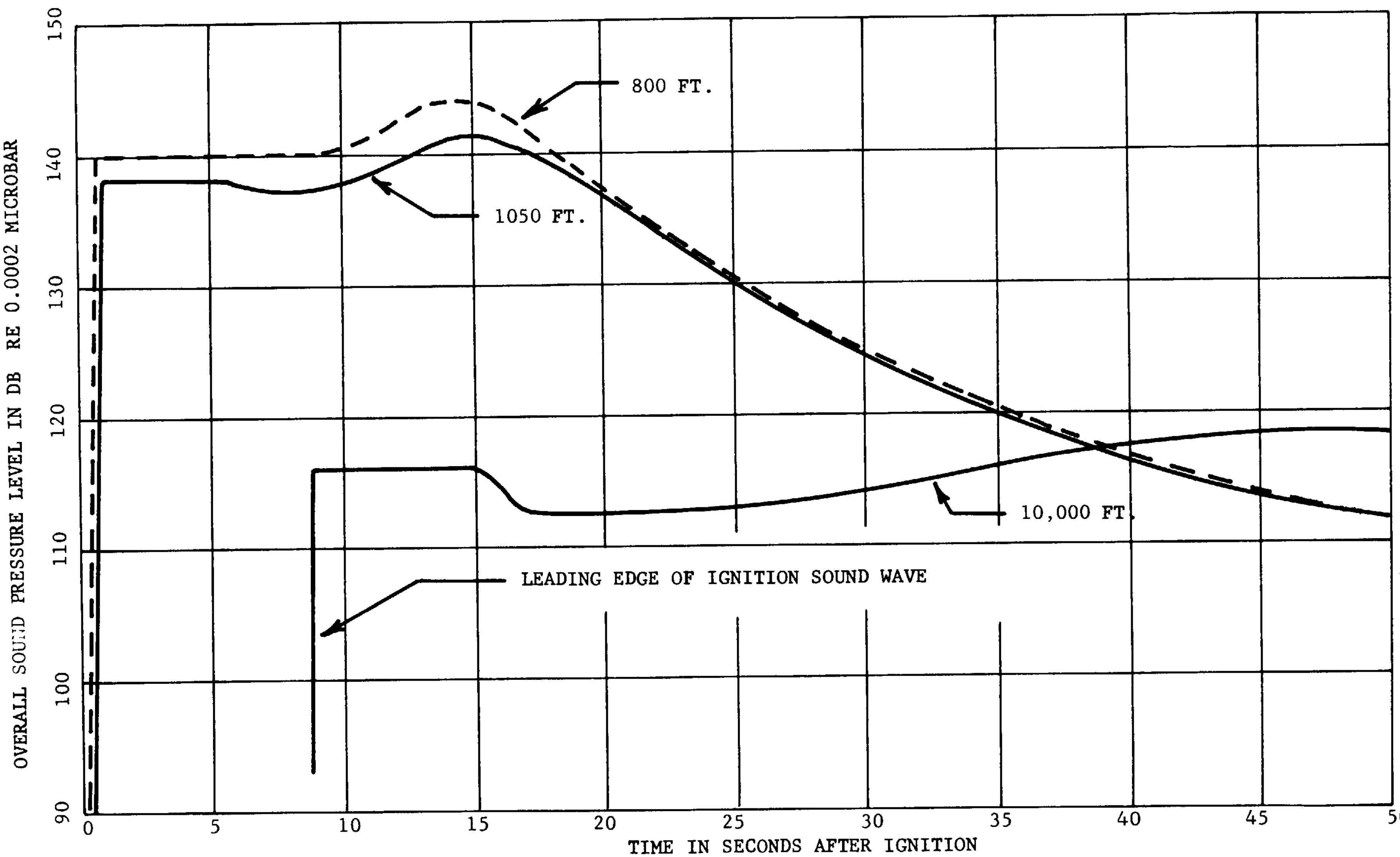

FIG. 4.17 OVERALL SOUND PRESSURE LEVEL VERSUS TIME FOR THREE RADII FROM THE SATURN LAUNCH PEDESTAL DURING THE FIRST FIFTY SECONDS OF THE FLIGHT.

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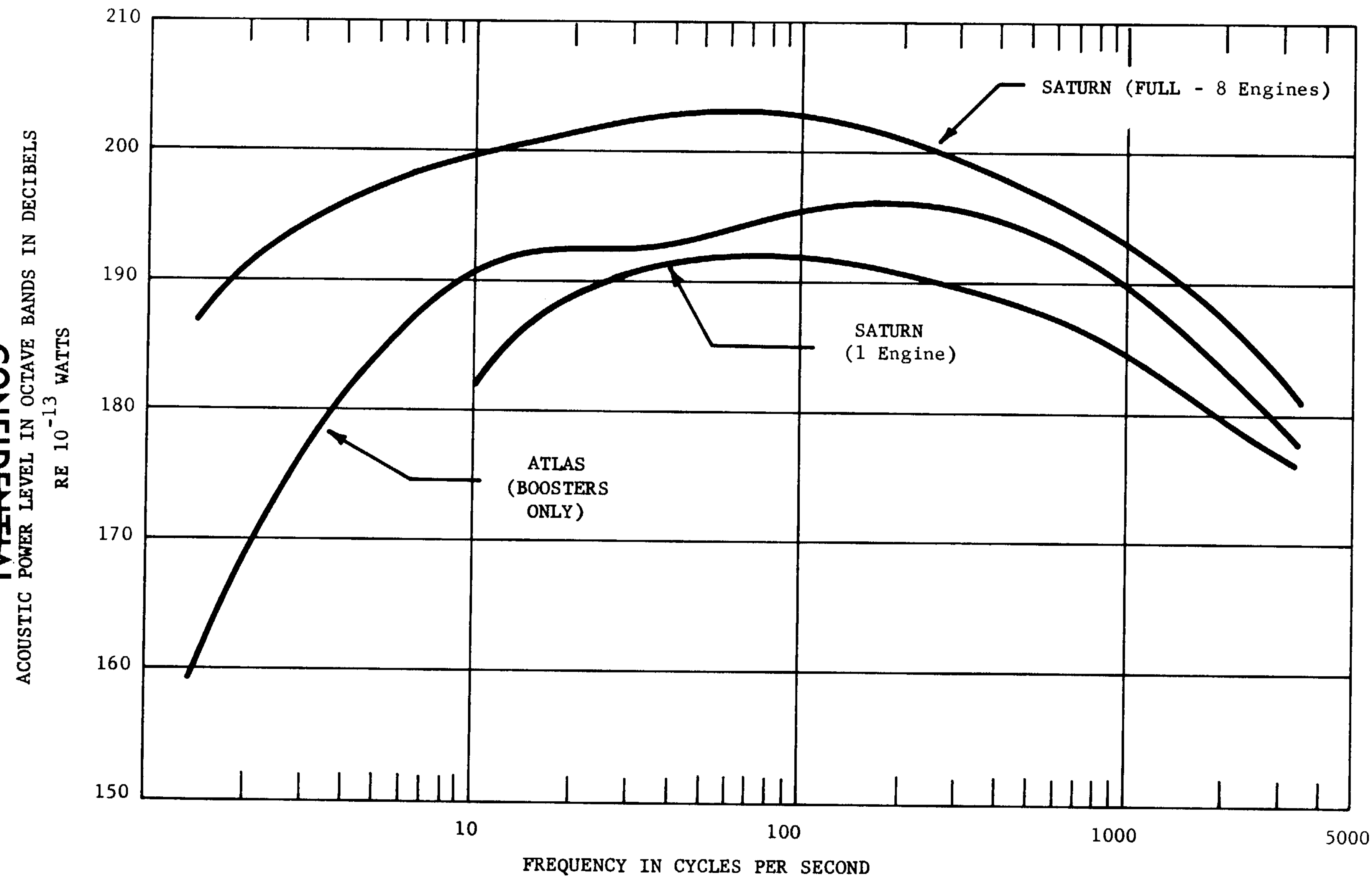

FIG. 4.18 ESTIMATED OCTAVE BAND POWER LEVELS OF ACOUSTIC RADIATION FROM THE SATURN VEHICLE WITH MEASURED POWER LEVELS FROM THE ATLAS MISSILE.

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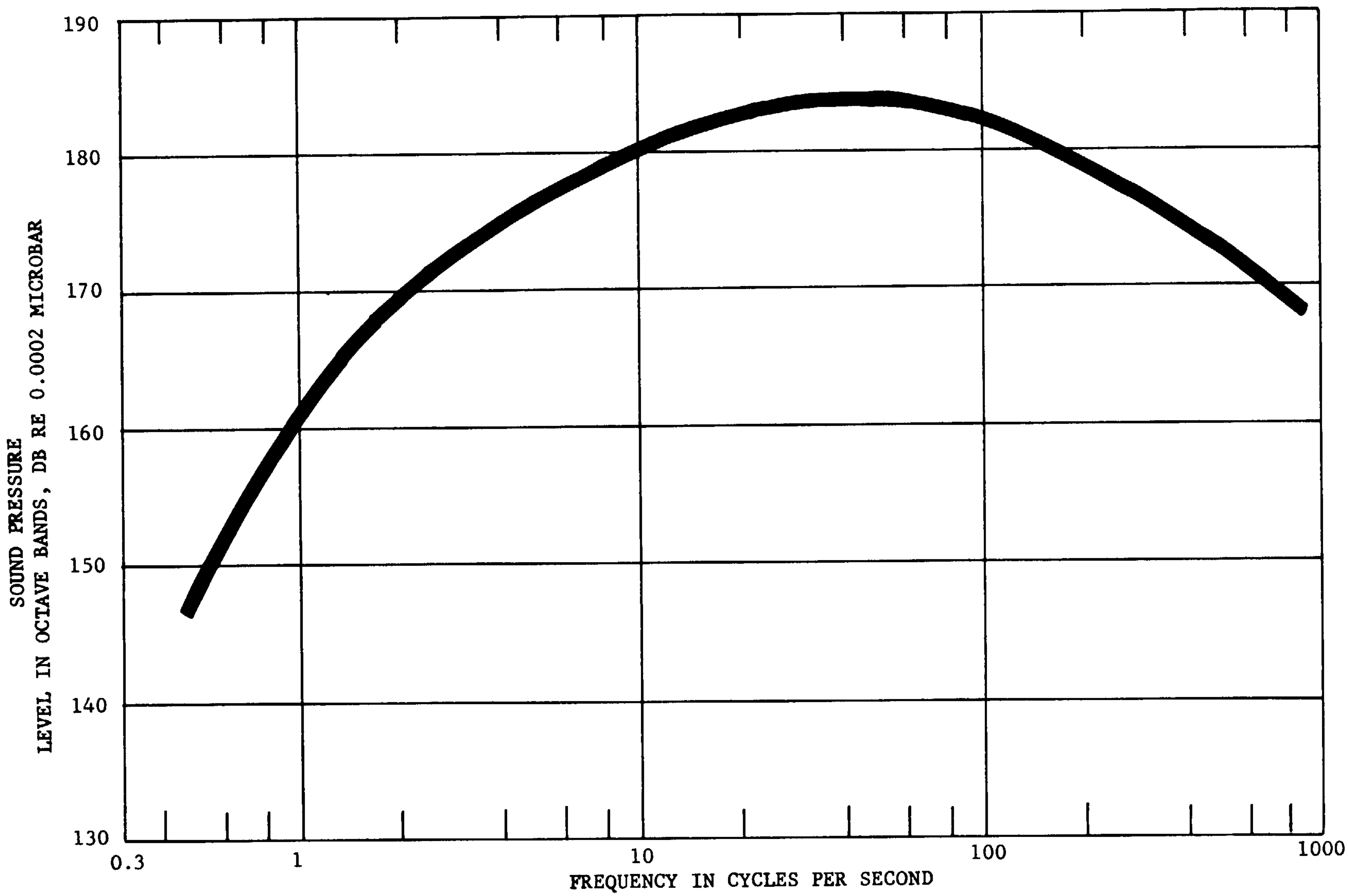

FIG. 4.19 ESTIMATED NEAR FIELD SOUND PRESSURE LEVEL FOR A SATURN STAGE I AS A FUNCTION OF FREQUENCY.

exposed to the acoustic near-field for the first 5 to 10 seconds after liftoff. The near-field sound pressures shown in this figure actually apply only to the near-field below the tail of the vehicle and very near the sound sources. The spectrums in Figures 4.20 and 4.21 show the effects of near-field amplitude distortion and atmospheric attenuation. The evidence of these effects is shown by the shift in the peak of the curve of the sound pressures higher in frequency and by the large drops in the sound pressures in the frequencies above 1,000 cps.

A profile of the overall sound pressure level along one bearing for three different times during the launch is given in Figure 4.22. This profile could be for almost any bearing because the sound radiation is considered to be symmetrical about the centerline of the booster while it is in flight.

Noise level and frequency outputs from eight engines of the Saturn booster are:

140 db close to the test tower and

105 db approximately 6-8 miles away.

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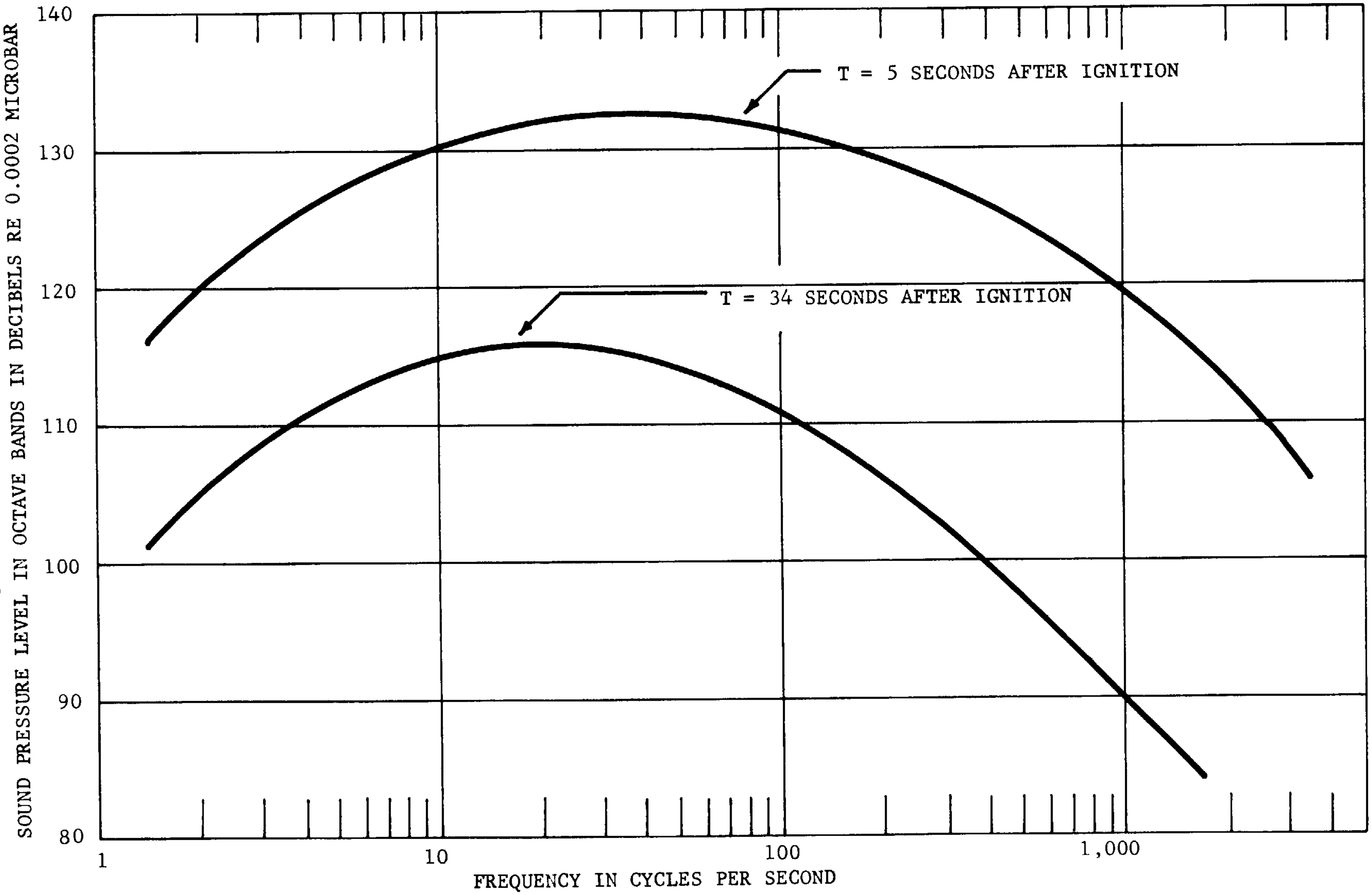

FIG. 4.20 ESTIMATED OCTAVE BAND SOUND PRESSURE LEVELS FOR A RADIUS OF 1050 FEET FROM THE SATURN LAUNCH PEDESTAL FOR TWO DIFFERENT TIMES DURING THE LAUNCH.

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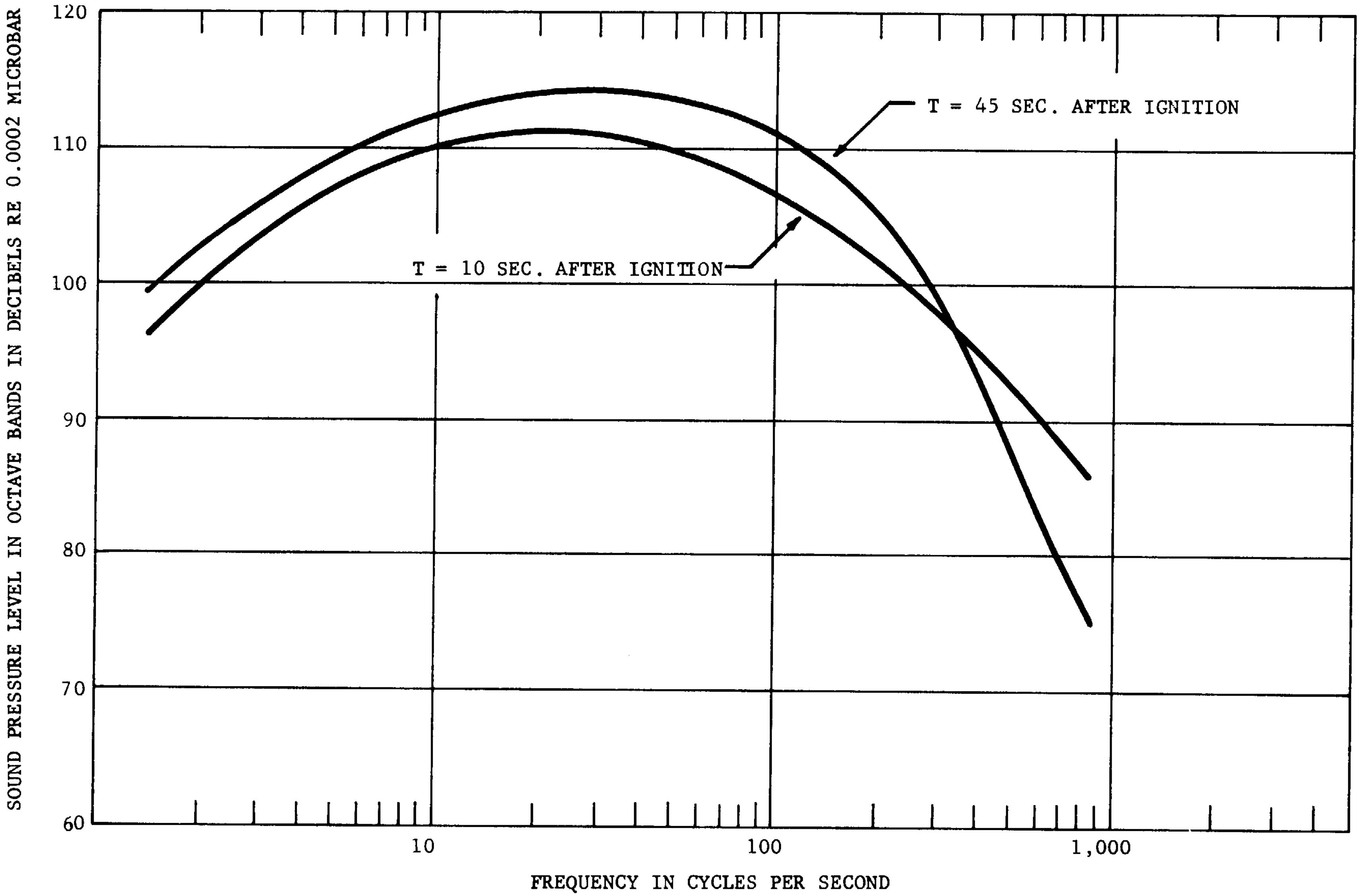

FIG. 4.21 ESTIMATED OCTAVE BAND SOUND PRESSURE LEVELS FOR A RADIUS OF 10,000 FEET FROM THE SATURN LAUNCH PEDESTAL FOR TWO DIFFERENT TIMES DURING THE LAUNCH.

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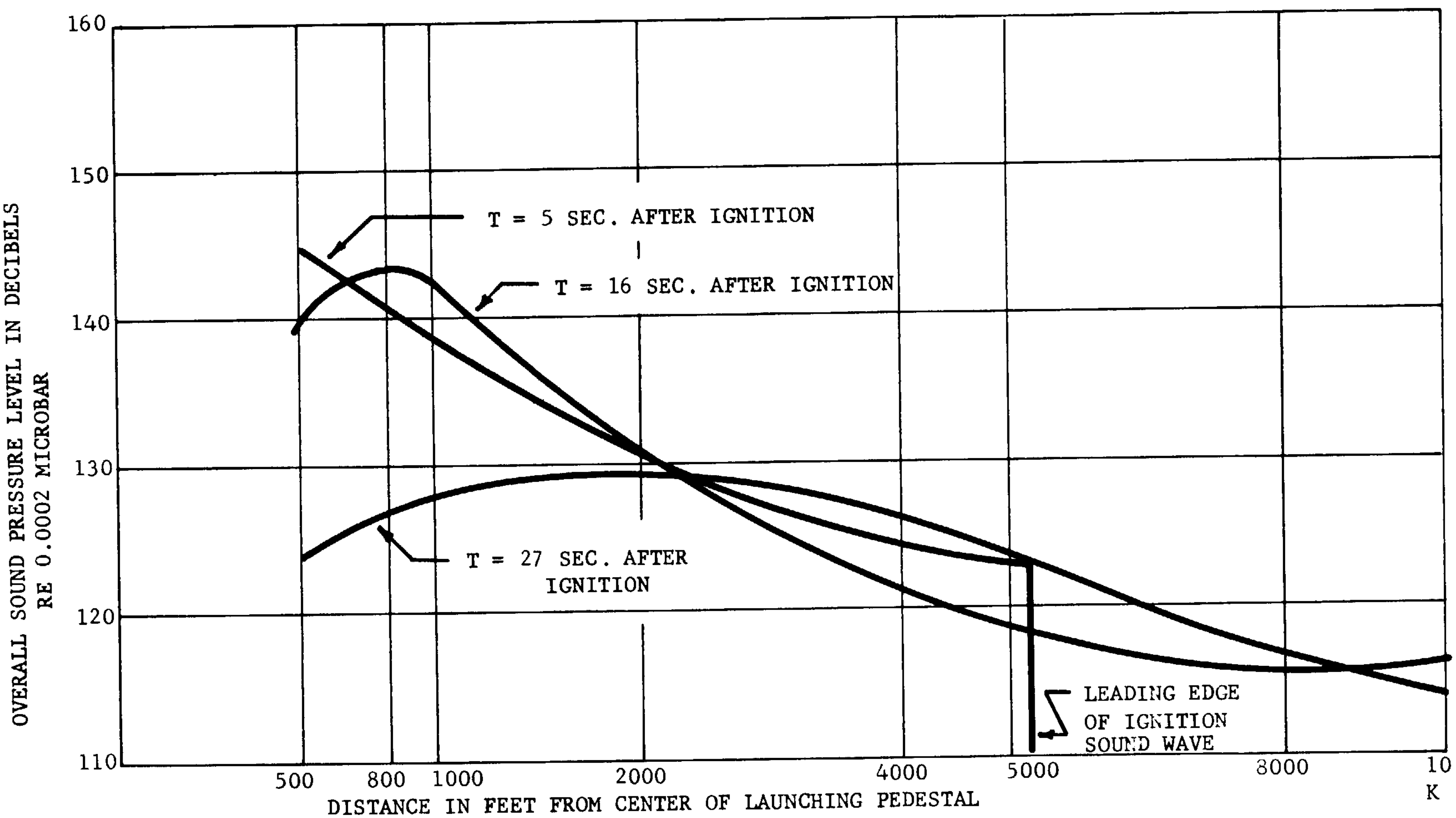

FIG. 4.22 SOUND PRESSURE LEVELS VERSUS DISTANCE ALONG A BEARING N.42°E.

SECTION 5. (C) STAGE S-IV

5.1. S-IV DESCRIPTION

The S-IV stage (Fig. 5.1) is a cylindrical configuration, 220 inches in diameter. It is powered by six Pratt and Whitney RL10-A-3 liquid propellant engines of 15,000 pounds thrust per engine. Propellants for these engines consist of 100,000 pounds of useable liquid oxygen and liquid hydrogen for a maximum burning time of 470 seconds.

The engines are mounted in a circular pattern at a cant angle of 6° relative to the vehicle longitudinal center line. Each engine is gimbaled for flight control. The gimbaling system is capable of deflecting the engine thrust in a 4° square gimbal pattern. Each engine is gimbaled into its controlled position by two linear type hydraulic actuators. A total of twelve actuators are arranged in a star pattern.

The S-IV stage, exclusive of engines, consists of four major units: the propellant tanks, the forward interstage structure, the aft interstage structure, and the heat shield.

The propellant tank is a 220-inch diameter welded aluminum alloy dome-ended tank. The tank is divided into two chambers by a welded-in and insulated double walled common bulkhead. The forward chamber is for the liquid hydrogen and the smaller aft chamber is for the liquid oxygen. The internal domed bulkhead prevents freezing of the lox during a hold period of up to 12 hours.

The forward and aft interstage structures consist of aluminum honeycomb panels attached to an aluminum alloy reinforcing structure.

The heat shield is located between the aft bulkhead of the lox tank and the exhaust nozzles of the engines. It serves to protect the tanks, engines, and control equipment from radiated and convective heat transfer.

5.2. S-IV STRUCTURES AND LOADS

Structures and loads are broken down as follows:

5.2.1. Handling and Transportation Loads. The S-IV stage is designed to withstand maximum transportation, handling, and erection loads of 3g in the longitudinal plane and 2g in the lateral and vertical planes.

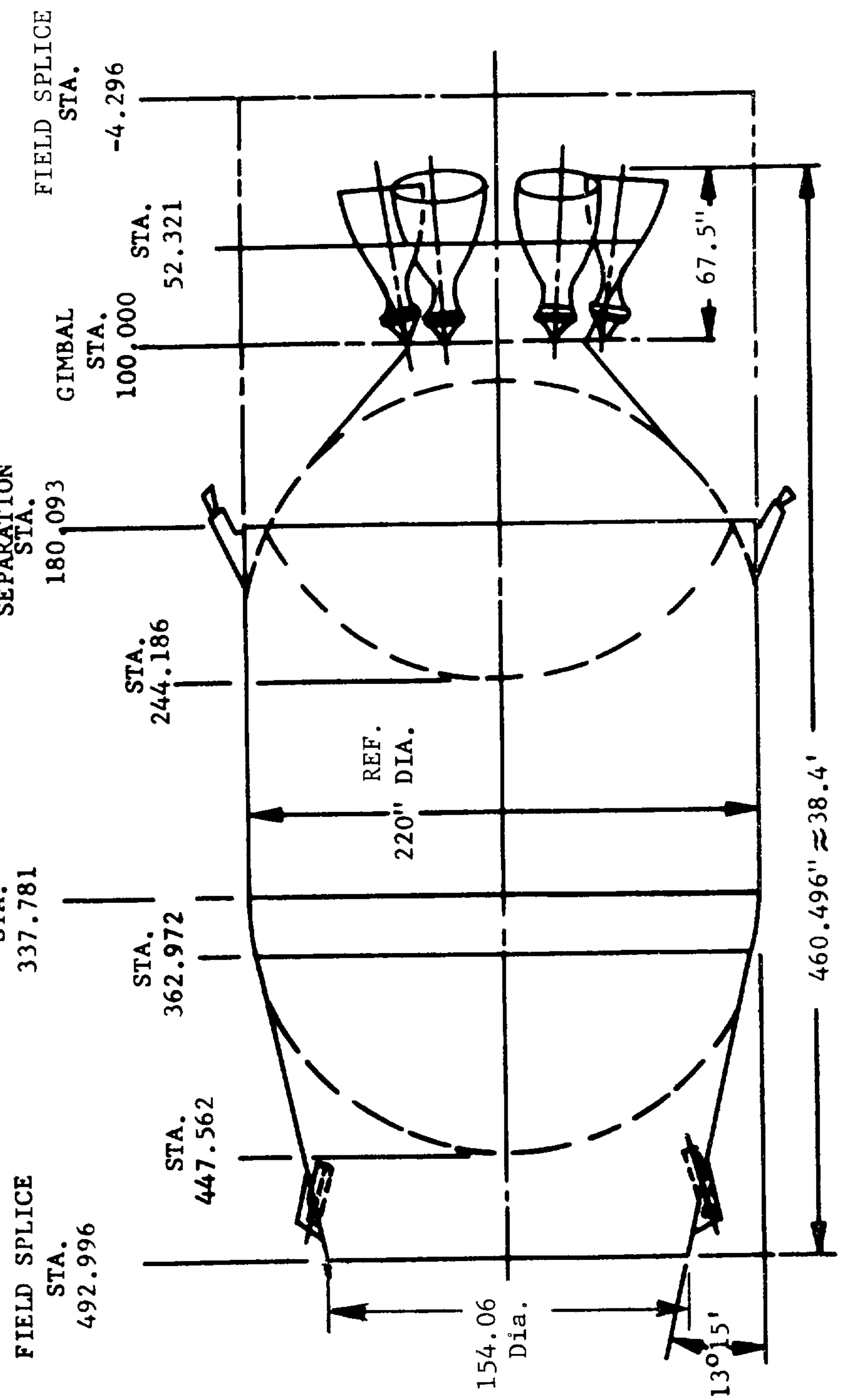

FIG. 5.1 STAGE S-IV DIMENSIONS

5.2.2. Ground Wind Loads. The S-IV stage is designed to withstand ground winds not exceeding 65 knots. This is predicated on the roll rings, installed at stations 889.3 and 1321.6, and secured to the service tower by cables.

5.2.3. Flight Loads. The allowable flight maneuver loads are:

Longitudinal Acceleration	7g
Lateral Acceleration	1g
Gimbal Acceleration	38 rad/sec^2

5.2.4. Internal and External Pressures. The aft interstage structure is designed for a maximum of 13 psi internal pressure.

The design of the Saturn S-IV stage, like the S-I stage, is such that the pressurization system is not required for structural integrity during ground handling, including pad operations. However, since satisfactory operation of the pressurization system is a necessary condition for successful flight, beneficial effects to be derived from this system may be useful for structural design. Thus, the flight stresses and allowables have been calculated based on satisfactory pressure system operation. Redundant vent control systems are used to prevent over-pressurization of the tanks.

5.3. S-IV WEIGHT STATEMENT

The Saturn C-1 stage S-IV dry weight breakdown is given in Table 5.1. Mass moment of inertia versus burning time, for the S-IV stage only, is given in Figure 5.2. Center of gravity shift and decrease in weight versus burning time for the S-IV stage only is given in Figure 5.3.

(C) Table 5.1
SATURN 6-ENGINE S-IV STAGE DRY WEIGHT BREAKDOWN

MAJOR UNITS	WEIGHT (LB)	CG STA
A. FUSELAGE & EQUIPMENT	8,008	260
B. PROPULSION SYSTEM & ACCESSORIES	3,494	126
C. CONTINGENCY	198	136
S-IV STAGE DRY WEIGHT TOTAL	11,700	218

DETAILED BREAKDOWN

		WEIGHT (LB)	CG STA
A. FUSELAGE & EQUIPMENT			260
1. Forward Transition Structure		661	433
a. Skin, Doublers, Core	408		-
b. Longerons	39		-
c. Frames	29		-
d. Access Doors Cutouts	63		-
e. Joining Parts & Bonding	122		-
2. Forward Skirt		194	368
a. Skin, Doublers, Core	108		-
b. Frames	61		-
c. Joining Parts & Bonding	25		-
3. Propellant Container		4,033	261
a. Forward LH_2 Bulkhead	765		387
b. Container Wall	1,074		283
c. Intermediate Bulkhead	850		207
d. Aft Bulkhead	1,046		188
e. Fuel Antislosh Baffles	136		372
f. LO_2 Antislosh Baffles	94		199
g. Fuel Antivortex Screen	16		193
h. LO_2 Antivortex Screen & Sumps	50		115
4. Container Insulation		702	300
a. Internal Fwd Bulkhead	207		-
b. Internal Side Wall	314		-
c. External Fwd Bulkhead	72		-
d. External Aft Bulkhead	109		-
5. Aft Skirt & Transition		506	207
a. Skin, Doublers, Skirt	214		-
b. Skin, Doublers, Trans Sect	193		-
c. Frames	41		-
d. Joining Parts & Bonding	58		-
6. Basic Thrust Structure		348	-
a. Skin and Doublers	128		-
b. Longerons	36		-
c. Frames	68		-
d. Engine Mount	116		-
7. Tail Section		273	81
a. Heat Shield	265		-
b. Air Conditioning	8		-
8. Tunnels		45	299
9. Paint & Marking		20	246
10. Electrical System		129	146
a. Power Supply	38		-
b. Power Distribution	10		-
c. Umbilical Wiring & Supports	81		-

DETAILED BREAKDOWN (Continued)

			WEIGHT (LB)	CG STA
11. Conduit & Cabling		61		335
12. Control Systems		414		175
a. Actuator Control	59			-
b. Engine & Sequence Control	209			-
c. Prop Utilization Control	120			-
d. Propulsion Control	26			-
13. Telemetry & Instruments		535		246
a. Antenna System	71			-
b. Telemetry	101			-
c. Instruments	303			-
d. Instr & Telemetry Electrical	60			-
14. Destruct System		87		382
B. PROPULSION SYSTEM & ACCESSORIES			3,494	126
1. Engines GFE		1,710		75
2. Pneumatic System		125		135
a. Pneumatic Supply	95			-
b. Engine Pneumatics	15			-
c. Propellant System Pneumatics	15			-
3. Fuel System		561		163
a. Suction Lines & Valves	244			-
b. Fill & Drain Lines & Valves	19			-
c. Vent System	70			-
d. Pressurization System	26			-
e. Replenishing Lines & Valves	-			-
f. Chill-Down Lines & Valves	202			-
4. Oxidizer System		538		163
a. Suction Lines & Valves	183			-
b. Fill & Drain Lines & Valves	28			-
c. Vent System	69			-
d. Pressurization Supply	233			-
e. Pressurization Lines & Valves	9			-
f. Replenishing Lines & Valves	-			-
g. Chill-Down Lines & Valves	16			-

DETAILED BREAKDOWN (Continued)

		WEIGHT (LB)	CG STA
5. Hydraulic Actuating System		366	81
a. Hydraulic Pump	9		-
b. Hydraulic Control	55		-
c. Lines & Fittings	12		-
d. Ground Check	55		-
e. Fluid & Gas	45		-
f. Actuator & Servo Valves	190		-
6. Separation System - Aft		9	180
a. Retention & Release	6		180
b. Ullage Rockets	3		181
7. Forward Retrorockets		185	463
C. CONTINGENCY		198	136
S-IV STAGE TOTAL		11,700	218

5.4. S-IV PROPULSION SYSTEM

The S-IV propulsion system consists of the following:

5.4.1. Engines. RL10-A-3 Engines - Six 15,000 pounds vacuum thrust lox/LH_2 engines are used on the S-IV stage. Each engine is a regeneratively-cooled turbopump-fed rocket engine (Fig. 5.4). Pumped fuel, after cooling the thrust chamber, is expanded through the turbine which drives the propellant pumps. The fuel is then injected into the combustion chamber. The pumped oxidizer is supplied directly to the propellant injector.

Thrust control is accomplished by sensing engine chamber pressure. This is used to control the amount of gaseous hydrogen that is allowed to bypass the turbopump turbine; thereby controlling the amount of propellant arriving at the engine. Propellant utilization is accomplished by a throttling valve in the oxidizer line. The throttling valve is controlled by signals from the propellant utilization computer.

Ignition is accomplished by means of an electric spark ignitor projecting into the combustion chamber. Starting and stopping are controlled by valving, actuated by supplied helium pressure, which is in turn controlled by electrical signals from the vehicle. A hydraulic pump, which provides power for the gimbal actuator, is mounted on the engine accessory drive pad. The accessory drive is rated at 13 inch pounds torque at a nominal rpm of 12,500.

The major performance parameters of the RL10-A-3 engine are summarized in Table 5.2.

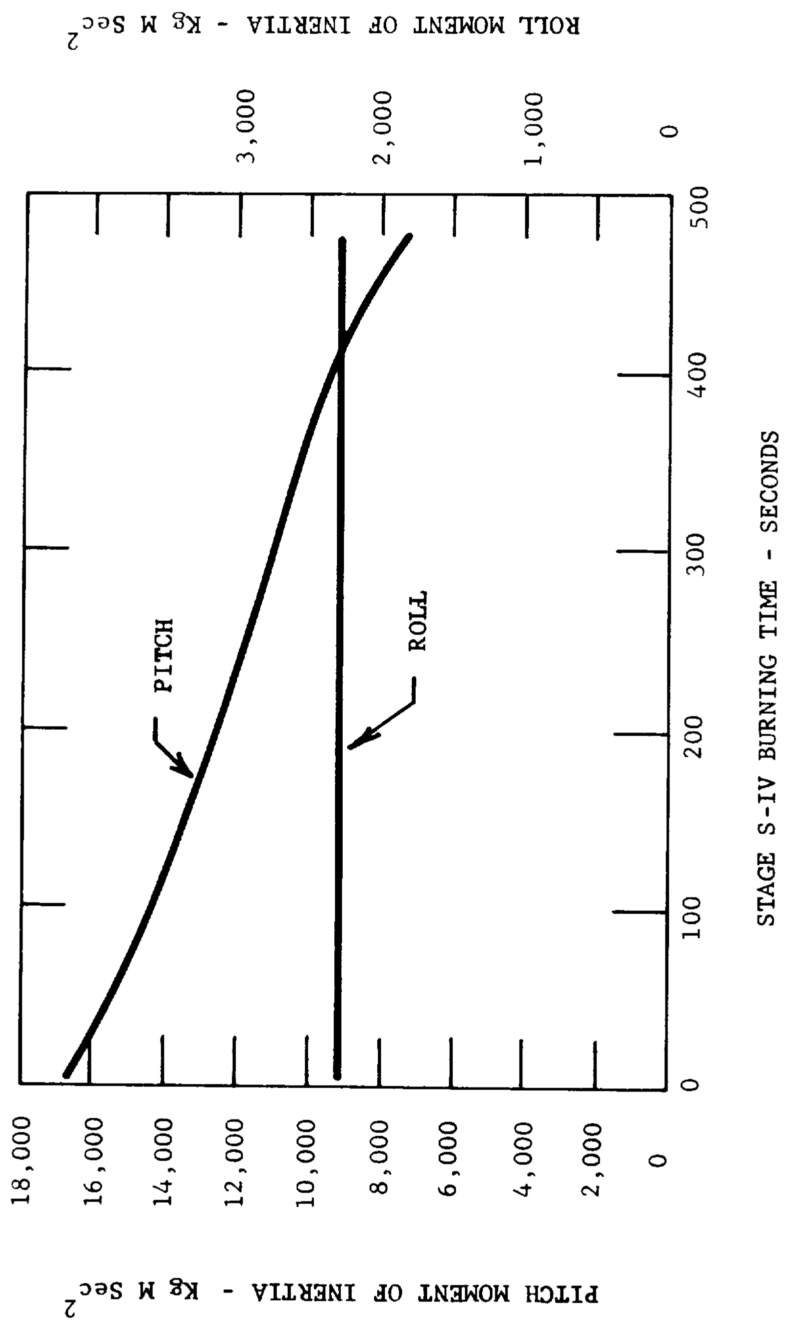

FIG. 5.2 MASS MOMENT OF INERTIA VS. STAGE S-IV BURNING TIME FOR STAGE S-IV ONLY

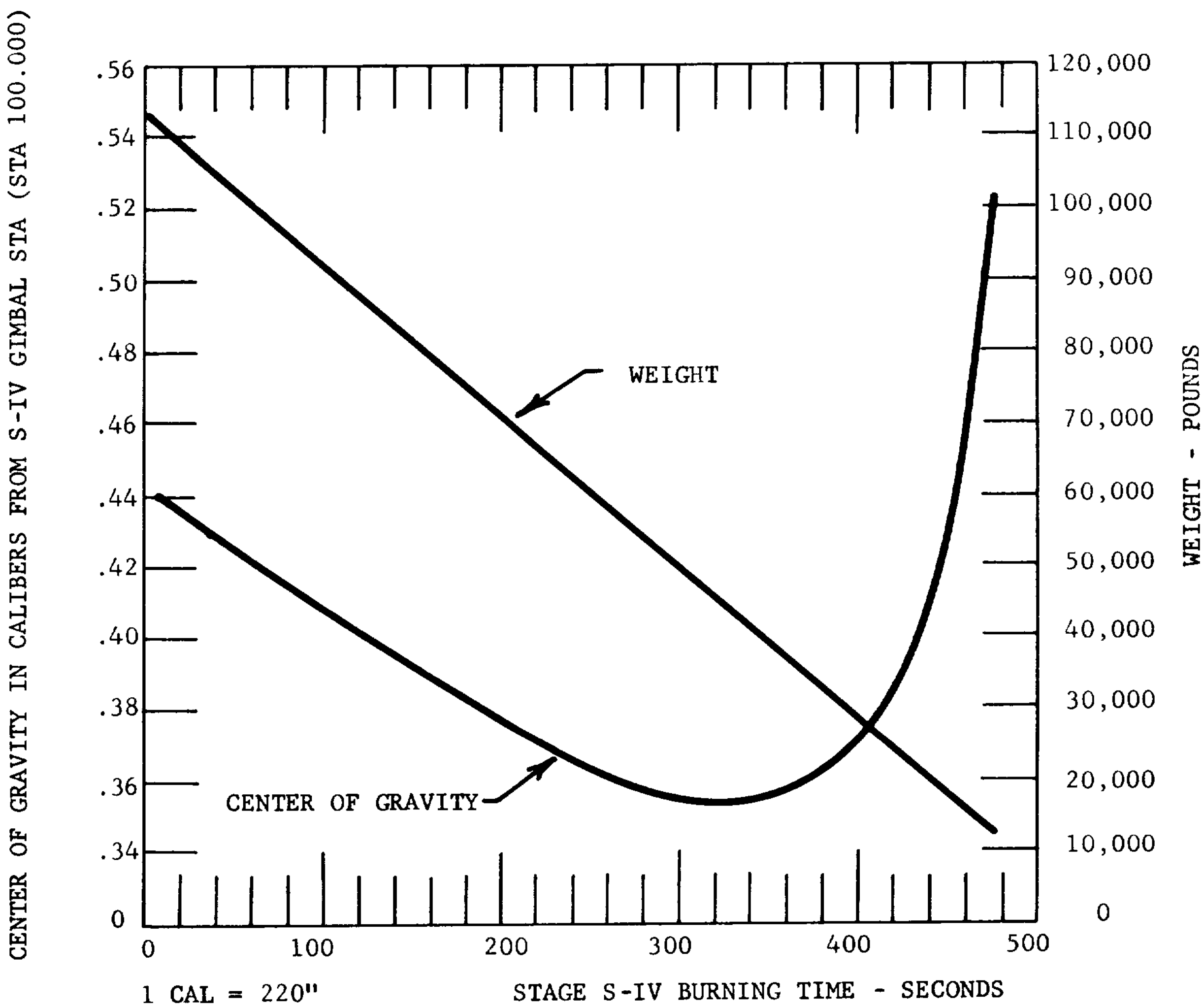

FIG. 5.3 CENTER OF GRAVITY SHIFT & DECREASE IN WEIGHT VS. STAGE S-IV BURNING TIME FOR STAGE S-IV ONLY

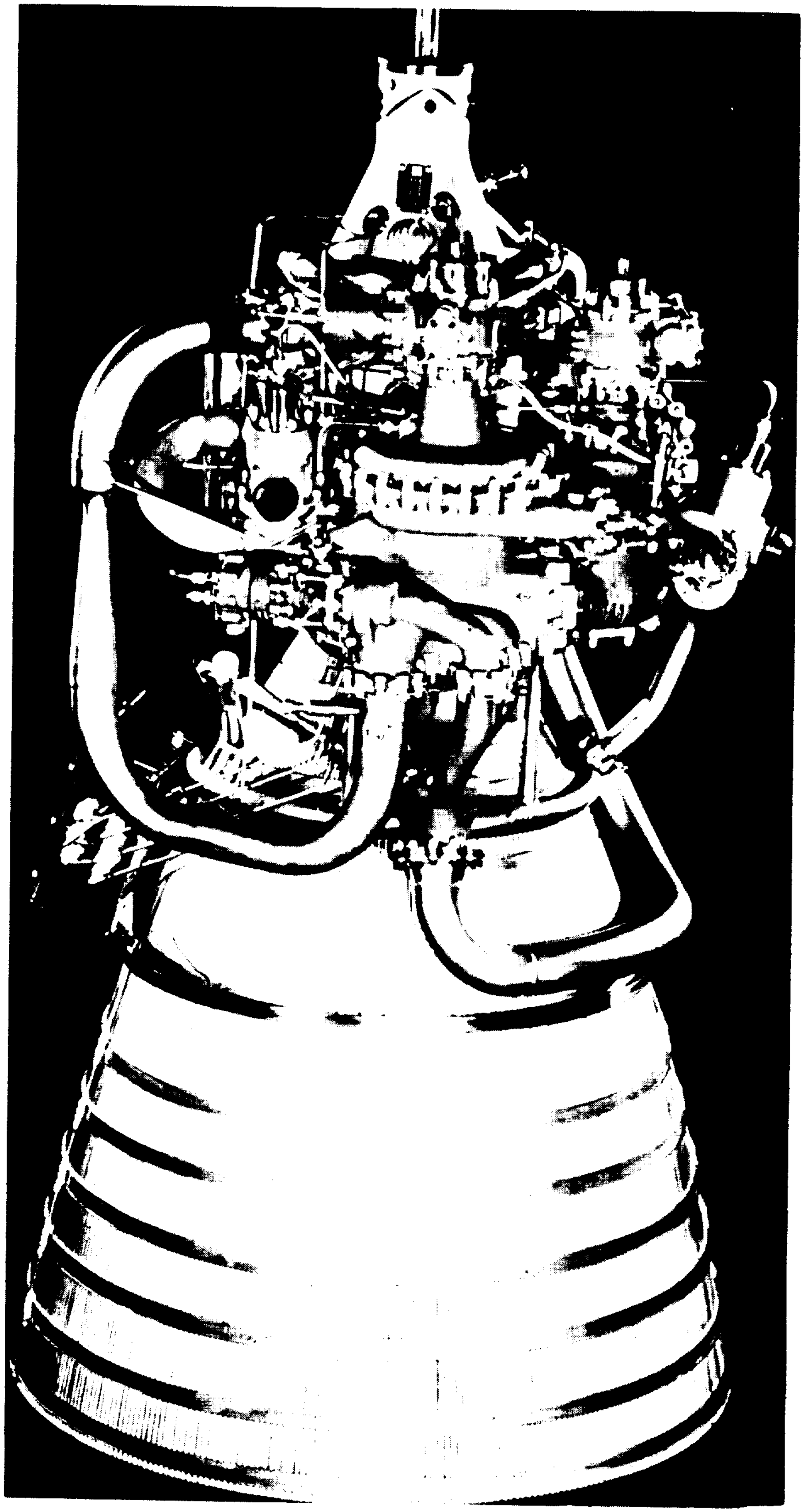

FIG. 5.4 THE RL10-A-3 ENGINE

(C) Table 5.2

RL10-A-3 ENGINE PERFORMANCE PARAMETERS

Nominal Thrust (Vacuum)	15,000 lb
Thrust Variation (Vacuum)	$\pm$2% ($\pm$300 lb)
Maximum Time from Ignition to 90% Thrust	2 sec
Minimum Specific Impulse (I_{sp}) (Vacuum)	412 $\frac{\text{lb-sec}}{\text{lb}}$
Engine Mixture Ratio	5.0:1 $\pm$1.67%
Cutoff Impulse (Vacuum)	2500 lb sec
Cutoff Impulse Variation (Vacuum)	$\pm$ 140 lb sec
Lox Pump NPSH (required)	46.5 psia at 163°R
Fuel Pump NPSH (required)	31.5 psia at 37°R
Rate of Thrust Increase	250 $\frac{\text{lb}}{\text{10 ms}}$
Nozzle Area Ratio	40:1
Chamber Pressure	300 psia

5.4.1.1. Prestart. A vehicle-supplied electrical prestart signal opens the engine prestart solenoid valve. This allows helium pressure to open the fuel pump inlet shutoff valve and the oxidizer pump inlet shutoff valve. Liquid oxygen flows through the oxidizer pump, controlled by a bypass orifice in the mixture ratio and propellant utilization control valve, through the injector plate and is expelled to atmosphere through the thrust chamber. Liquid hydrogen enters the fuel pump, passes through the first stage, and enters the interstage connecting tube where a portion of the flow, controlled by the port area of the interstage cooldown valve, is diverted overboard. The remainder flows through the second fuel pump stage, and is diverted overboard through the port area of the pump discharge cooldown valve.

5.4.1.2. Start. Upon receipt of the start signal, the start solenoid is energized and routes helium pressure to both cooldown valves and the main fuel shutoff valve. The two cooldown valves close to the preset overboard bleed area required to prevent fuel pump stall during acceleration. Simultaneous with the start signal, the ignition system is energized, thereby providing a source of ignition in the combustion chamber. As the turbopump accelerates, the fuel pump discharge pressure closes the bleed ports in the cooldown valves.

5.4.1.3. Steady State Operation. During steady-state operation, the thrust control senses combustion chamber pressure with a bellows, which in turn positions a force-type pneumatic servo. The servo unit controls the area of the bypass port of the thrust control to maintain the preset chamber pressure.

5.4.1.4. Shutdown. The shutdown signal consists of simultaneous removal of the electrical supply to the start and prestart solenoid valves. In the de-energized position, the helium supply pressure is removed from all propellant shutoff valves by venting all helium-controlled valve actuators overboard. This permits the valves to return to their normal position. All valves and systems assume a position of readiness for the next prestart phase.

5.4.1.5. Restart Considerations

5.4.1.5.1. Restart Capability of RL10-A-3 Engines. The S-IV engine, Pratt and Whitney's RL10-A-3, is covered by the model specification on restart capability. The restart specification was predicated on the Centaur requirements; the Centaur and Saturn S-IV use the same type of engine. The specification states that the engine life is 2,820 seconds during which time three inflight starts of 470 seconds each are possible. This specification covers all flight engines on Saturn C-1 S-IV stages.

5.4.1.5.2. Coast Attitude Control System for Stage S-IV. In order to provide a restartable S-IV, a coast attitude control system must be added to maintain constant ullage as well as adequate attitude control.

Such a system has been sized for the S-IV vehicle utilizing gaseous H_2O_2 as the propellant. Control is maintained by swiveling the nozzles according to signals received from the guidance system. The addition of the coast attitude control system to the S-IV stage is the major portion of the change required to provide restart capability for the S-IV.

5.4.1.5.3. Other Modifications of S-IV for Restart. The other modifications of the S-IV stage to accommodate a restartable engine are minor. The tank insulation appears to be adequate to maintain LH_2 boiloff rates at reasonable levels. The flight sequencer circuitry would require changes as would the inflight pressurization schedules.

5.4.2. Ullage Rockets. Four TX-280 Thiokol rocket motors (Fig. 5.5) are provided for proper settling of propellants during the S-I/S-IV separation phase.

Table 5.3
TX-280 ROCKET PERFORMANCE CHARACTERISTICS

Ignition Delay	0.04 sec (max)
Min. Thrust (Sea Level Static) Over Temperature Range of -10°F to 110°F	3000 pounds (Per rocket)
Average Thrust Level Tolerance Over Constant Burning Time	±11.2%
Total Impulse Tolerance	±6.6%
Min. Burning Time, Over a Temperature Range of -10°F to 110°F	3.1 sec

5.4.3. Propellant Containers. The propellant containers are designed with sufficient volumes to allow a minimum of 100,000 pounds of propellants to be consumed during S-IV stage flight, plus the amount required during engine cool down period, engine start sequence, thrust decay, etc.

The propellant replenishing systems are automatically controlled. The vehicle-to-ground connections are manually connected at the umbilical points and are automatically disconnected just prior to vehicle liftoff. Emergency drain provisions are provided only while the umbilical lines are connected to the vehicle.

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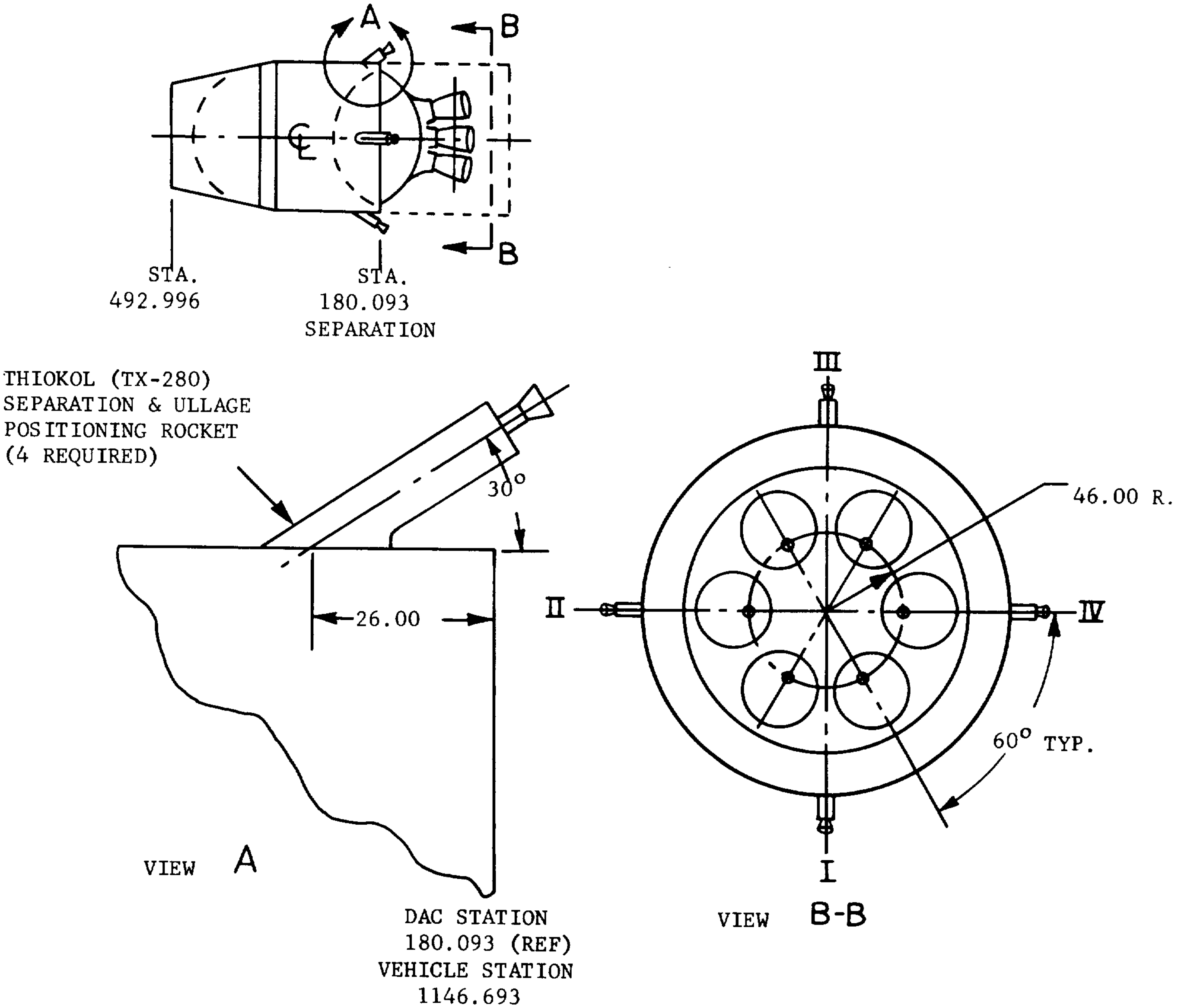

FIG. 5.5 LOCATION OF S-IV STAGE ULLAGE ROCKETS

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The propellant containers are calibrated during construction to facilitate accurate loading of propellants at the launch site. Propellants are loaded with a maximum error of ± 1/4% (design goal).

The propellant utilization system provides for simultaneous depletion of useable propellants, with an allowable exception of ±1/2% of the initial total propellant mass. Propellant utilization is accomplished by varying the lox consumption rate by a throttling valve in the oxidizer line that is controlled by signals from the propellant utilization computer.

5.4.4. Propellant Container Pressurization. The lox container is pressurized by ambient helium during ground operation and in flight as required prior to actuation of the helium heater, which produces a heated gaseous helium flow to maintain pressurization during lox consumption.

Container Pressurization values are:

Regulation	Minimum (psia)	Nominal (psia)	Maximum (psia)
Ground and Flight	45	46.5	48

The LH_2 container is pressurized by ambient helium during ground operation and in flight as required prior to gaseous hydrogen flow from the engines to the pressurization system.

Container pressurization values are:

Regulation	Minimum (psia)	Nominal (psia)	Maximum (psia)
Ground and Flight	30	31.5	33

5.4.5. Ventilation. During launch preparations, conditioned air or gaseous nitrogen is supplied to the aft interstage of the S-IV. The distribution system design provides a means for gaseous nitrogen purging or for the air conditioning for personnel and installed equipment to temperatures between 35°F and 100°F.

5.4.6. Helium Systems. Two helium systems are provided for the S-IV stage as follows:

(1) The cold helium system consists of three spheres mounted in the LH_2 container, one heat exchanger which is supplied with LH_2 and lox directly from the propellant containers and the required controls, valves, piping, etc.

(2) The ambient helium system consists of one sphere mounted in the engine thrust structure and the required controls, valves, piping, etc.

The cold helium system is used exclusively for pressurization of the lox container. The ambient helium system is used for propellant pressurization of both containers and to operate the engine controls. The characteristics of the systems are as follows:

Type	No. of Spheres	Storage Pressure	Volume	Regulated Pressure Flow
Cold Helium	3	3000 psi	5850 cu in.	- - - -
Ambient Helium	1	3000 psi	6000 cu in.	- - - -

5.5. S-IV CONTROL SYSTEM

The S-IV control system consists of the following:

5.5.1. <u>Hydraulic System</u>. The S-IV stage contains six independent closed-loop flight-control hydraulic systems. Each system consists of:

(1) Two actuator assemblies with integral accumulators and reservoirs

(2) Engine driven fixed displacement pump

(3) Electrically driven auxiliary pump

(4) Filter, relief valves, piping, etc.

Two linear type hydraulic actuator assemblies are connected to each engine to provide gimbal motion. Proper gimbal positioning of the engines for pitch, yaw, and roll control of the S-IV vehicle during flight is provided through flight correction signals from the guidance and control system to the actuator assemblies. All engines provide gimbal action to correct pitch and yaw of the S-IV vehicle. In addition, four engines, numbers 1, 2, 3, and 4 provide gimbal action to counteract roll of the vehicle. Engines 5 and 6 do not provide roll control (Fig. 5.6).

5.5.2. <u>Electrical Power Supplies</u>. The power requirements of the electrical systems are satisfied by the power supplies listed in Table 5.4.

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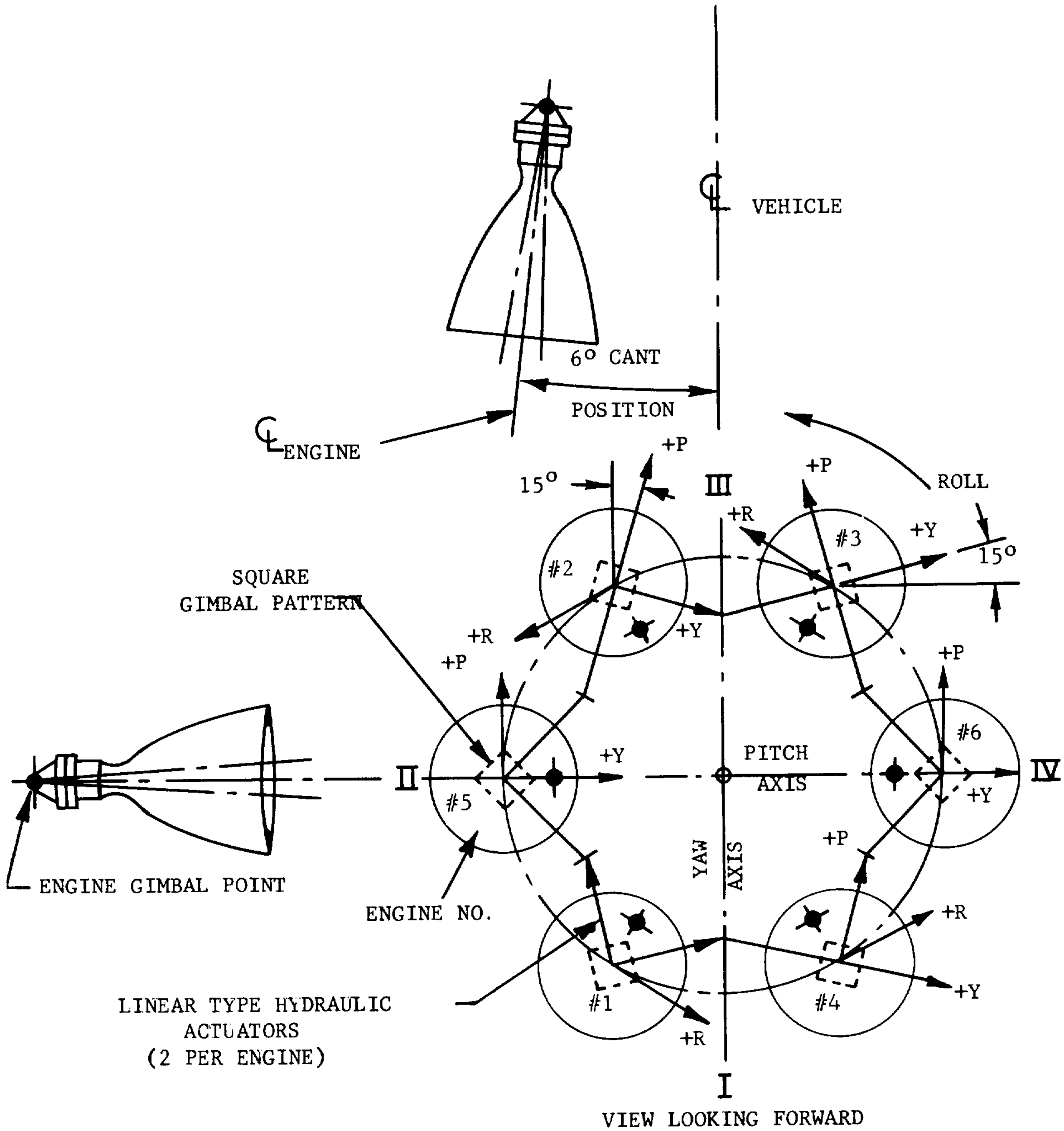

FIG. 5.6 STAGE S-IV SIX-ENGINE GIMBAL PATTERN AND CANT

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(C) Table 5.4

S-IV ELECTRICAL POWER SUPPLIES

Quantity	Power Supply		Voltage	Requirement
1	28V 5 Amp-Hr Battery		28 VDC	Constant loads
1	28V 10 Amp-Hr Battery		28 VDC	Transient loads
1	Inverter-Converter Combination	Inverter Section	115 V 400 cycle	Propellant utilization
		Converter Section	6 VDC 12 VDC	Computer Logic and sequencing systems

5.6. S-IV TELEMETRY AND RF SYSTEMS

The telemetry of the S-IV stage uses three rf links. The links are three pulse duration modulation/frequency modulation/frequency modulation sets. This telemetry system has 24 standard IRIG analog channels and 405 commutated channels.

The only other rf link in the stage S-IV is the range safety command destruct link.

The antennas, located on the conical interstage, are

DOG	Slot Transverse
STA 456.495	TM
STA 417.561	Longitudinal Slot

The frequencies are:

Telemetry	251.5 mc, 255.1 mc, 258.5 mc
Command	(Classified SECRET)

Inflight voltage calibration is provided on all FM channels. Figure 5.7 is a block diagram of the system.

PDM/FM/FM sets No.1 and No. 2 are each capable of handling 86 data channels at a sampling rate of 10 samples per second plus 43 data channels at a sampling rate of 2.5 samples per second. Nine channels of FM/FM information will also be handled by nine subcarrier oscillators ranging from Interrange Instrumentation Group (IRIG) standard frequency 2.3 kc through 40 kc. The equipment required for PDM/FM/FM set No. 1 is listed as follows:

Quantity	Item
1	90 x 10 Commutator-Keyer
1	45 x 2.5 Commutator-Keyer
1	FM Transmitter
1	Wide Band Amplifier

Sets No. 2 and 3 are the same except set No. 3 does not have the 90 x 10 commutator-keyer.

The commutator-keyer components of the system are mounted in the engine compartment. The remaining components of the system are mounted in the forward interstage area. The equipment mounted in the forward interstage area consists of subcarrier oscillators, wide band amplifiers, FM transmitters, triplexer, rf power amplifier, and antennas.

5.6.1. S-IV Measuring Program. Approximately 300 ground and inflight measurements in the S-IV stage may be required during a typical Block II mission. A condensed list of the measured parameters in the S-IV stage, including the range of the measurements, is given in Table 5.5. This list of measurements may change from flight to flight as required by the previous flight evaluation. Usually the listed measurements are only fed into the stage telemetry system for the transmission to ground stations. Some critical signals, voltages and propulsion measurements, however, also will be routed to the instrument unit of Saturn C-1 Block II vehicle for eventual use in the abort system and in the spacecraft.

(C) Table 5.5

TYPICAL STAGE S-IV MEASURING PROGRAM

Measured Parameter	Range
Prestart Signal to Sequencer	0, 28 VDC
S-I Separation Signal	0, 28 VDC
Engine Start Pressure Switch Pickup - Engine 1	0, 28 VDC
Engine Prestart Pressure Switch Pickup - Engine 1	0, 28 VDC
Engine Cutoff Signal	0, 28 VDC
S-IV Engines Start Signal	0, 28 VDC
Ullage Rocket Jettison Current	
Ullage Rocket Pressure Switch	0, 28 VDC
Retrorocket Pressure Switch	0, 28 VDC
Payload Abort Signal to Sequencer	0, 28 VDC
Engine Buss Enable Signal	0, 28 VDC
Control Switching Network Start Command	0, 28 VDC
Ignite Retrorockets Signal	0, 28 VDC
Blow S-I to S-IV Explosive Bolts Command	0, 28 VDC
Payload Separation Signal	0, 28 VDC
Base Calorimeter Temperature - Black	
Base Calorimeter Temperature - Gold	
Base Pressure	
Base Temperature	
S-I to S-IV Extensiometer	0 - 24 ft.
Axial Acceleration	-1 to +2 g's
Detonation Pressure Switch	0, 28 VDC
Vibration Pickup - Engine Compartment Pitch Axis	0 - 1200 cps
Vibration Pickup - Engine Compartment	0 - 660 cps
Vibration Pickup - Fwd interstage - Thrust Axis	0 - 1200 cps

(C) Table 5.5 (Cont'd)

Measured Parameter	Range
Vibration Pickup (Fwd Interstage - Pitch Axis)	0 - 600 cps
Vibration (Electronics Package in Engine Compt)	0 - 20 g's RMS
Vibration (Oxidizer Feed Line)	0 - 20 g's RMS
Vibration (Fuel Feed Line)	0 - 20 g's RMS
Vibration (Turbopump)	0 - 20 g's RMS
Vibration (Aft Skirt - Radial Direction)	0 - 20 g's RMS
Vibration (Aft Skirt - Thrust Axis)	0 - 20 g's RMS
Vibration (Telemetry Package)	0 - 20 g's RMS
Vibration (Outer Tank Surface)	0 - 20 g's RMS
Static Inverter Inner Temperature	0 - 160° F
Battery Temperature	0 - 160° F
Reference Voltage - High	+5 V DC
Reference Voltage - Low	0 V DC
Static Inverter 115V Voltage	110 - 120 VAC
Battery Voltage	27 - 29 VDC
DC Converter Negative 6-volt Output	-5.5 to -6.5 VDC
DC Converter Negative 12-volt Output	-10 to -14 VDC
DC Converter Positive 28-volt Output	26 - 30 VDC
Range Safety AGC Voltage	
Telemetry Antenna SWR	
EBW Converter Output Voltage	
DC Power Signal to Interface	0, 28 VDC
LH_2 Temperature - Suction Line Inlet	
LO_2 Temperature - Suction Manifold	
Oxidizer Tank Ullage Temperature - 85% Level	-100° to -300°F

(C) Table 5.5 (Cont'd)

Measured Parameter	Range
Fuel Tank Ullage Temperature - 85% Level	-200° to -423°F
Temperature Probe - LH_2 Tank - 40% Level	-323° to -423°F
Temperature Probe - LH_2 Tank - 5% Level	-413° to -423°F
Temperature Probe - LH_2 Tank - 1% Level	-413° to -423°F
Temperature Probe - LO_2 Tank - 40% Level	-187° to -297°F
Temperature Probe - LO_2 Tank - 5% Level	-287° to -297°F
Temperature Probe - LO_2 Tank - 1% Level	-287° to -297°F
Fuel Tank Gas Supply Orifice Differential Pressure	0 - 500 psid
Fuel Tank Ullage Pressure	0 - 40 psia
Oxidizer Tank Ullage Pressure	0 - 50 psia
Fuel Tank Inner Skin Temperature	
Base Thrust Structure Temperature	
Aft Dome Stress	
Side Tunnel Temperature (Internal)	
Engine Mount Stress	
Thrust Beam Stress	
LH_2 Tank Mass - Fine	0 - 5 VDC
LH_2 Tank Mass - Coarse	0 - 5 VDC
LO_2 Tank Mass - Fine	0 - 5 VDC
LO_2 Tank Mass - Coarse	0 - 5 VDC
LH_2 Tank Vent Valve Closed	0, 28 VDC
LO_2 Tank Vent Valve Closed	0, 28 VDC
Fuel Tank Step Pressure Signal	0, 28 VDC
LH_2 Point Level Sensor	0, 28 VDC
LO_2 Point Level Sensor	0, 28 VDC

(C) Table 5.5 (Cont'd)

Measured Parameter	Range
Cold Helium Bottle Gas Temperature	-225° to -423°F
Cold Helium Outlet Temperature	-200° to -400°F
Helium Heater Chamber Pressure	0 - 20 psia
Cold Helium Bottle Gas Pressure	0 - 3500 psia
Cold Helium Orifice Differential Pressure	0 - 600 psid
Ambient Helium Bottle Pressure	0 - 3500 psia
Ambient Helium Regulator Outlet Pressure	0 - 750 psia
Helium Heater Combustion Temperature	0° to +1500°F
Outer Surface Temperature Helium Heater	
Ambient Helium Bottle Gas Temperature	-100° to +100°F
Helium Heater Oxidizer Valve Closed	0, 28 VDC
Helium Heater Activate Signal	0, 28 VDC
Helium Heater Fuel Valve Open	0, 28 VDC
Helium Heater Fuel Valve Closed	0, 28 VDC
Helium Heater Oxidizer Valve Open	0, 28 VDC
Helium Heater Ignition Exciter Command	0, 28 VDC
Thrust Chamber Pressure	0 - 400 psia
Oxidizer Pump Speed	(0 - 60 cps)
Turbine Inlet Temperature	-100° to 250°F
Fuel Pump Inducer Inlet Temperature	-415° to -423°F
Oxidizer Pump Inlet Temperature	-275° to -300°F
Fuel Pump Housing Temperature	+110° to -423°F
Oxidizer Pump Housing Temperature	+110° to -423°F
Thrust Chamber Pressure	325 - 375 psia
Thrust Chamber Pressure - (Low Range)	0 - 50 psia

(C) Table 5.5 (Cont'd)

Measured Parameter	Range
Fuel Injector Differential Pressure	0 - 50 psid
Oxidizer Injector Differential Pressure	0 - 80 psid
Venturi Inlet Pressure	- 1200 psia
Turbine Differential Pressure	0 - 600 psid
Fuel Pump Outlet Pressure	0 - 1500 psia
Oxidizer Pump Outlet Pressure	0 - 600 psia
Oxidizer Pump Inlet Pressure	0 - 50 psia
Fuel Pump Inducer Inlet Pressure	0 - 50 psia
Accumulator GN_2 Pressure - Engine Yaw Actuator	0 - 3400 psia
Accumulator GH_2 Pressure - Engine Pitch Actuator	0 - 3400 psia
Reservoir Oil Pressure	0 - 300 psia
High Pressure Oil - Vehicle Pump Engine	0 - 3400 psia
Hydraulic Pump Inlet Oil Temperature-Engine	-40° + 350°F
Yaw Actuator Reservoir Surface Temperature - Engine	0° - 500°F
Pitch Actuator Reservoir Surface Temperature - Engine	0° - 500°F
Engine Pitch Position	
Engine Yaw Postion	
Engine Position Monitor Reference Voltage	+5 VDC
Engine Position Monitor Reference Voltage	-5 VDC
P.U. Ratio Valve Position	0 - 5 VDC
P.U. Error Signal	
P.U. Start Signal to Sequencer	0, 28 VDC
P.U. Depletion Signal to Sequencer	0, 28 VDC
Hydraulic Accumulators Solenoid Valve Open Command	0, 28 VDC
Ignition Exciter Box Output	0, 6 VDC

5.7. S-IV FABRICATION AND ASSEMBLY

5.7.1. <u>Fabrication and Subassembly Operation</u>. Fabrication and assembly of the S-IV stage is accomplished by the Douglas Aircraft Company and consists of the following:

5.7.1.1. Tank Shell. The cylindrical tank shell is composed of three individual skin segments, welded internally to form the complete cylinder.

5.7.1.2. Forward and Aft Bulkheads. The forward and aft bulkheads, each consisting of six segments, are formed and welded in a radial pattern to produce a hemispherical dome.

5.7.1.3. Common Bulkhead. The common bulkhead is an assembly consisting of an inner and outer dome, fiberglass honeycomb core, and two attach rings.

The inner and outer domes are manufactured in the same manner as the forward and aft domes.

5.7.1.4. Engine Mount. The engine mount assembly is a sheet metal structure, reinforced at the points of engine thrust with machined fittings.

5.7.1.5. S-IV to Instrument Unit (Fwd) Interstage. The forward interstage structure consists of eight bonded aluminum honeycomb panels and a four piece attaching ring at the forward end.

5.7.1.6. Aft Skirt. The aft skirt structure consists of four bonded aluminum honeycomb panels and a single piece attaching ring at the forward end. The aft end and edge members of each panel are sheet metal channels.

5.7.1.7. S-I to S-IV (Aft). The aft interstage structure is a riveted assembly of eight bonded aluminum honeycomb panels.

5.7.1.8. Lox Tank and Engine Mount Subassembly. This assembly consists of three components: the aft bulkhead, the engine mount assembly, and the common bulkhead.

5.7.1.9. Propellant Tank Assembly. All operations of assembling the propellant tank are accomplished in the tank assembly tower with the vehicle axis vertical.

5.7.1.10. Firewall Assembly. The firewall assembly consists of a heat shield baffle, tubular support struts, and seal rings for each engine.

5.7.1.11. Engines. The six Pratt and Whitney engines of the Saturn S-IV stage are government furnished equipment, (GFE).

5.7.2. Final Assembly. Final assembly operations on the Saturn S-IV vehicle are accomplished with the vehicle in a horizontal attitude. Following the calibration operation, the aft interstage structure is removed and the propellant tank assembly placed on the insulation installation fixture. In this position, all possible installations are made with the exception of the engines and firewall assembly.

The aft interstage structure is then re-installed. The forward and aft roll rings and the engine section support spider of the transport kit are installed, and the vehicle is placed on the transporter.

In this position, the engines and firewall assembly are installed and aligned. Final systems checkout is conducted, and the remainder of the transport kit items installed in preparation for shipping.

5.8 S-IV TESTING

An extensive ground test program will be carried out in order to establish and demonstrate safe and reliable operation of the S-IV stage on the ground before the beginning of the flight tests.

5.8.1. Research and Development Testing. The R&D test program encompasses a wide variety of tests ranging from metallurgical tests in the liquid hydrogen environment, and component and subsystem tests, to full-scale captive test firings. During the initial captive test firings, at the contractor's Sacramento facilities, a 20-second all-engines firing test will be performed to test the exhaust diffuser system, the flame deflector water systems, and the general performance of the vehicle and the test stand. Data for flight performance predictions, engine chamber pressures, individual engine thrust, specific impulse and mixture ratio of the cluster, and similar data will be acquired in these firings. Demonstration of normal engine prestart (chill-down) and startup characteristics in the six-engine stage environment will be an important objective in the R&D test firing program. The test program will also include (1) a determination of the effect the shutting down of one engine has on the accuracy of the closed-loop propellant utilization system, (2) determination of whether geysering from inactive engine lines is a problem, and (3) establishing the capability of the other vehicle subsystems (such as the tank pressurization and pneumatic supply) and the engine cluster itself, to operate for the longer duration of engine operation which will be encountered with one engine out.

5.8.2. Stage S-IV Acceptance Tests. Before beginning of the flight tests, the Stage S-IV will be subject to a complete acceptance test. This test will start with a subsystem and stage checkout and will be completed by a full duration captive test firing. The subsystem checkout generally will consist of

(1) End-to -end checkout of the instrumentation transducers under ambient conditions of pressure, temperature, g-levels, rates, etc.

(2) Closed loop checkout of telemetry transmitters and range safety receivers to determine power output levels, receiver sensitivities, frequencies, etc.

(3) Checkout of engine electrical components by operation from electrical and pneumatic consoles

(4) Checkout of inflight engine start and restart sequence, to determine complete system operation, valve timing, response to cutoff malfunction, and one engine out capability.

(5) Operation of engine gimbaling in all axes, with actuators controlled by an interface simulator

(6) Checkout of propellant utilization subsystems by simulation of sensor signals to the computer circuits and ringout of capacitance gage installations

(7) Functional checkout of the power circuits such as battery power, static inverter power, battery heaters, etc.

(8) Complete antenna subsystem test. This will consist of insertion loss measurements between the antenna subsystem terminals and the terminals of an antenna test coupler which has been previously calibrated for its contribution to total insertion loss.

(9) The vehicle tanks, pneumatic systems, and power plants will be leak checked.

(10) Electromechanical and functional tests will demonstrate the proper operation of engine ignition and sequence control systems, pressure switches, and valve interlock circuits.

Upon completion of subsystem checkout, a complete electrical system checkout will be conducted on internal power simulating the inflight sequence. All instrumentation will be operated with the data recorded on magnetic tape. The reduced data will be evaluated and utilized for vehicle acceptance.

A successful full duration captive test firing will complete the acceptance test. Ability of the systems to work together properly will be demonstrated. The test will also demonstrate compliance with performance ratings prescribed in the engine model specifications. Performance of the engine cluster will be compared to individual engine performance by utilizing calibration data supplied by the engine manufacturer. Engine calibration will be confirmed by verifying that all engine parameters fall within limits specified by the engine model specifications. The vehicle will be equipped with a flight instrumentation package during the acceptance firing. Data for prediction of flight performance of each cluster will be accomplished by combining vehicle acceptance test performance data, the engine manufacturer's gain characteristics (for example, thrust change as a function of oxygen pump inlet pressure), and predicted values of engine input flight histories (such as pump inlet pressure and temperatures). This procedure has been quite successful in other programs in which an IBM-704 trajectory was flown to determine propellant loading quantities and to predict flight performance. The accuracy of prediction would be expected to increase considerably following the first S-IV stage flight; primarily, because the effects of boost flight aerodynamic heating upon propellant tank ullage pressures and propellant temperature stratification cannot be completely determined by static testing.

5.9. ENVIRONMENT OF S-IV FORWARD INTERSTAGE

The forward interstage of Stage S-IV extends from Station 1304.381 to Station 1459.596. The structure of the interstage consists mainly of eight bonded aluminum honeycomb panels and a four piece attachment ring at the forward end. The environmental conditions presently considered in the design by the Douglas Aircraft Company for the area of the forward interstage of the S-IV are of interest to the designers of the instrument unit and of the Apollo spacecraft itself. In the following, a brief summary is given of the predicted and designed environment of the S-IV forward interstage.

5.9.1. Ground Environment. The ground environment is specified in DAC Spec Drawing No. 7857406 and MIL-STD-210A. It represents conditions expected to be encountered in the fabrication, testing, storage, handling, and transportation of the S-IV stage. The ground temperature environment of the forward S-IV interstage, which is of particular interest, is summarized in Table 5.6.

5.9.2. Flight Environment. The flight environment of the S-IV forward interstage is summarized in Table 5.7. The vibration environment is given in the following paragraph.

5.9.3. Vibration. The vibration encountered in the flight environment is of a random nature with sinusoidal components. The sinusoidal test levels are of sufficient magnitude to insure that no random and/or sinusoidal vibration components will exceed these levels. If combined environments of vibration and temperature are appropriate, the vibration levels indicated, in conjunction with high and/or low temperatures will be used, otherwise testing will be done at room temperature.

(C) Table 5.6

GROUND ENVIRONMENT (TEMPERATURE) OF S-IV FORWARD INTERSTAGE

ITEM	PREDICTED VALUE	DESIGN CRITERIA
Temperature Maximum		
Not fueled, Outer Skin	+ 160°F	+ 300°F
Not fueled, Inner Skin	+ 125°F	+ 300°F
Not fueled, Internal	+ 125°F	+ 150°F
Temperature Minimum		
Fueled, Outer	- 10°F	- 10°F
Fueled, Inner Skin	- 10°F	- 10°F
Fueled, Internal	Air conditioned at + 50°F with 2500 cfm	- 10°F

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(C) Table 5.7

FLIGHT ENVIRONMENT OF S-IV FORWARD INTERSTAGE

ITEM	PREDICTED VALUE	DESIGN CRITERIA
Temperature Maximum		
Outer Skin	+ 272°F (Fig. 5.8)	+ 500°F
Inner Skin	+ 236°F	+ 400°F
Internal	+ 150°F	+ 150°F
Temperature Minimum		
Outer Skin	- 10°F	- 10°F
Inner Skin	- 10°F	- 10°F
Internal	- 10°F	- 10°F
Thermal Shock	- 10°F to 272°F at 4°F/sec	- 10°F to 500°F at 4°F/sec
Acceleration, Axial, Stage S-I Flight		7.0 g's
Acceleration, Axial, Stage S-IV Flight		1.5 g's

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(C) Table 5.7 (Continued)

ITEM	PREDICTED VALUE	DESIGN CRITERIA
Acceleration, Lateral		2.9 g's
Vibration	(See Text) par. 5.9.3	(See Text) par. 5.9.3
Shock		20 g's for 10 milliseconds. Shocks shall be of sawtooth form and shall be applied 6 times in each of three mutually perpendicular axes.
Acoustic Noise, External, Overall Sound Pressure Level	See Fig. 5.10	
Acoustic Noise, Internal, Overall Sound Pressure Level	See Fig. 5.11	+145 db (RE: 0.00002 DYNES/CM^2)
Acoustic Noise, Subsonic Flight, Sound Pressure Level Versus Frequency	See Fig. 5.12	

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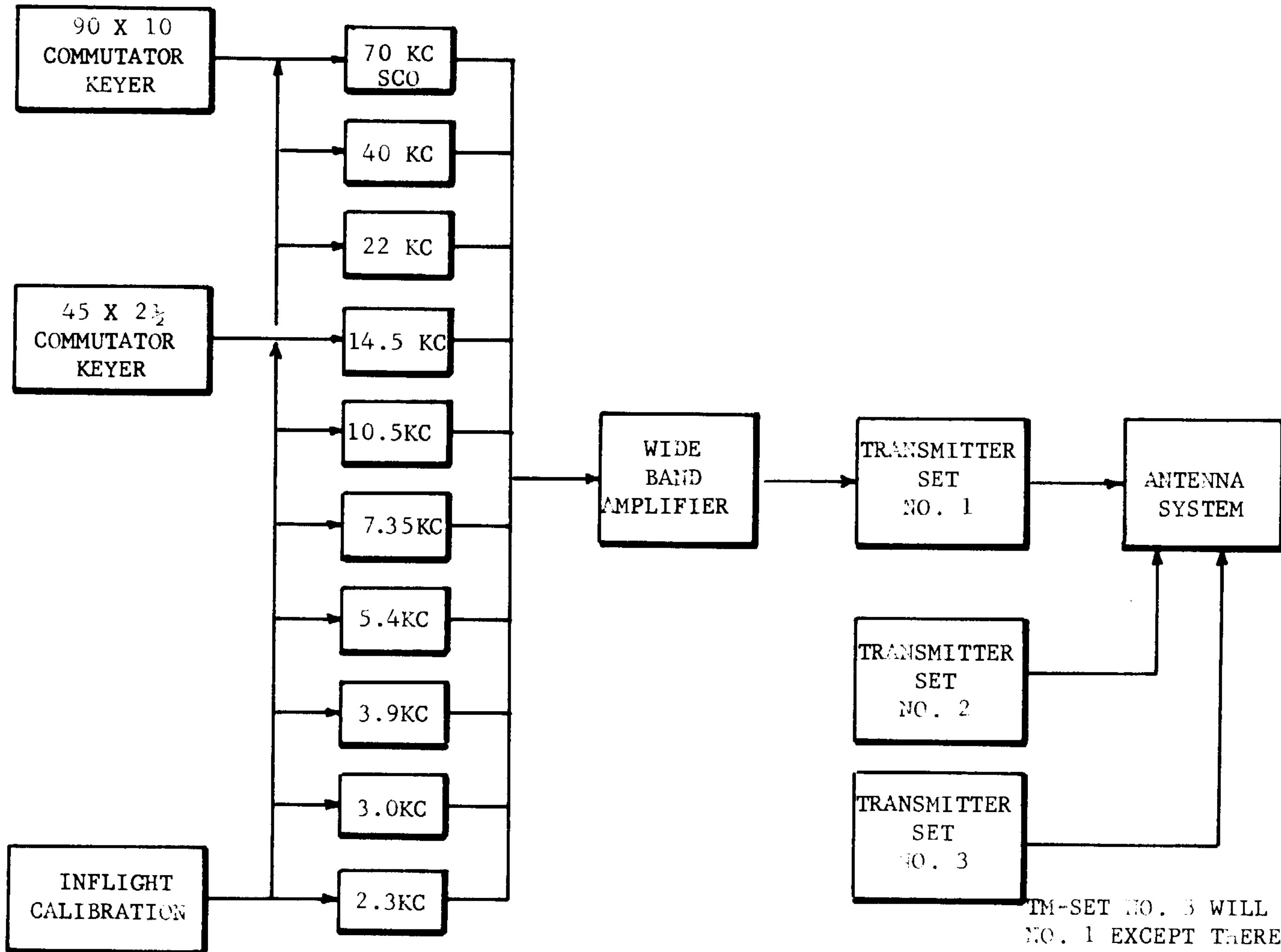

TM-SET NO. 3 WILL BE THE SAME AS NO. 1 EXCEPT THERE WILL BE NO 90 X 10 COMMUTATOR KEYER

*TM SET NO. 2 WILL BE SAME AS NO. 1

FIG. 5.7 BLOCK DIAGRAM OF THE S-IV TELEMETRY SYSTEM

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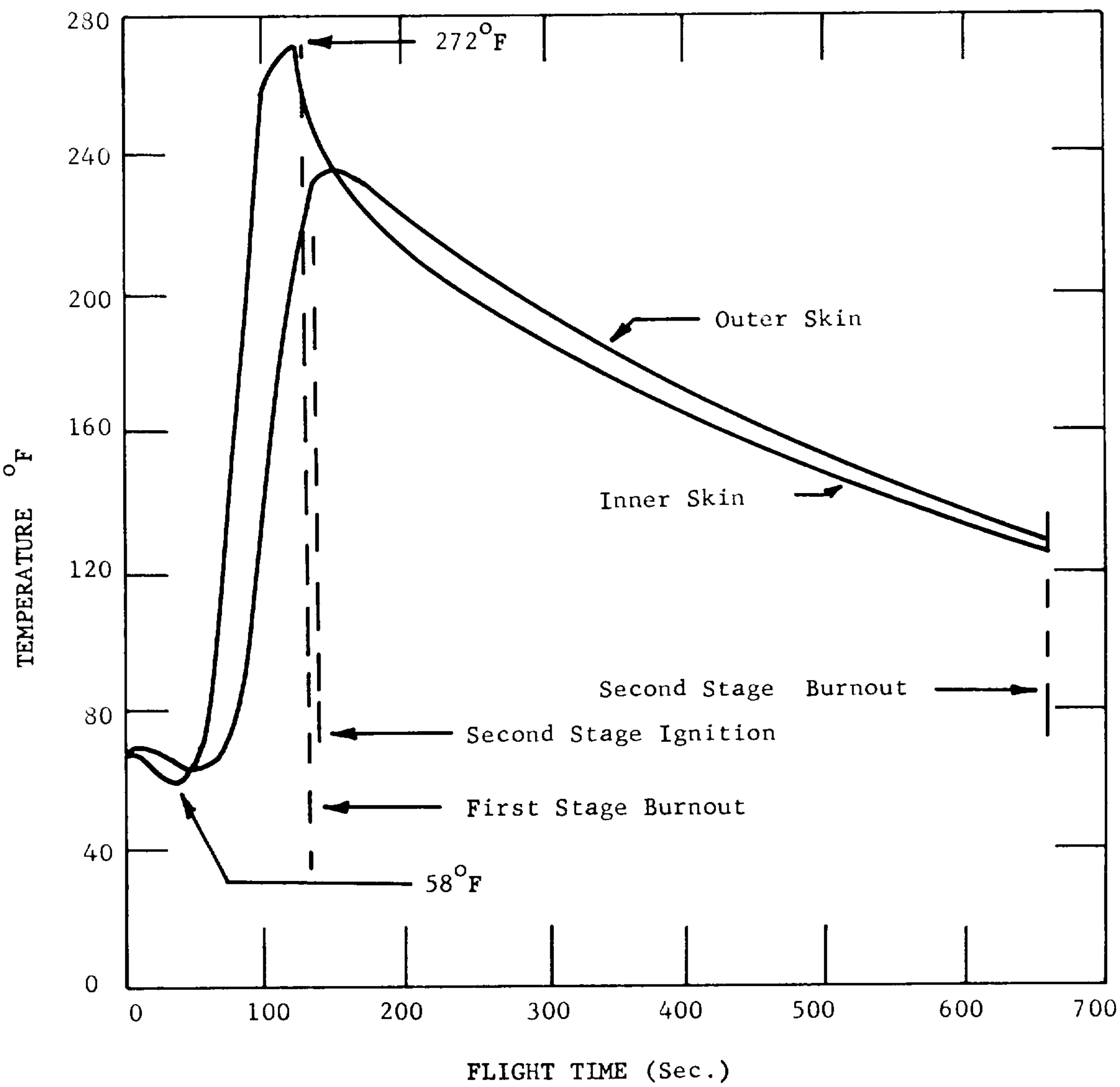

FIG. 5.8 PREDICTED TEMPERATURE OF THE S-IV FORWARD INTERSTAGE VS. FLIGHT TIME.

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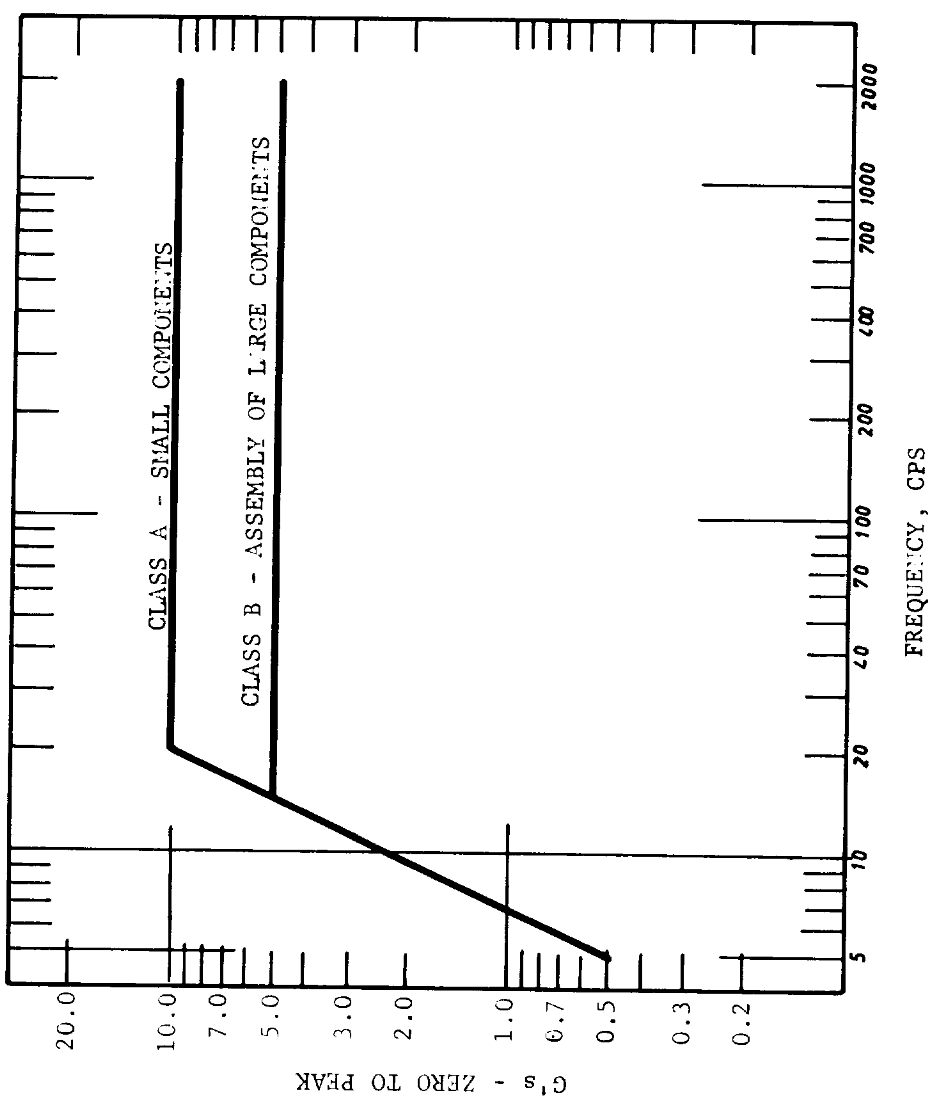

FIG. 5.9 SINE WAVE VIBRATION TEST LEVELS VS. FREQUENCY FOR EQUIPMENT IN S-IV FORWARD INTERSTAGE.

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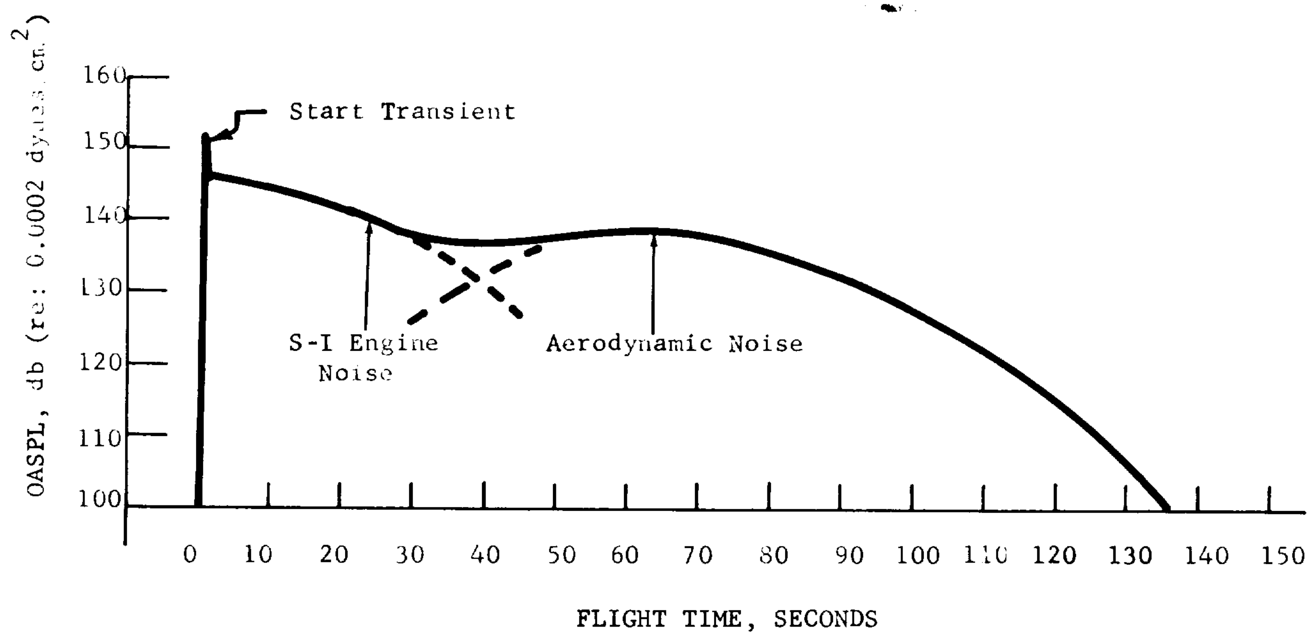

FIG. 5.10 IN-FLIGHT OVERALL SOUND PRESSURE LEVEL (OASPL) VS. FLIGHT TIME FOR EXTERNAL PARTS OF STAGE S-IV AT STA. 1400.

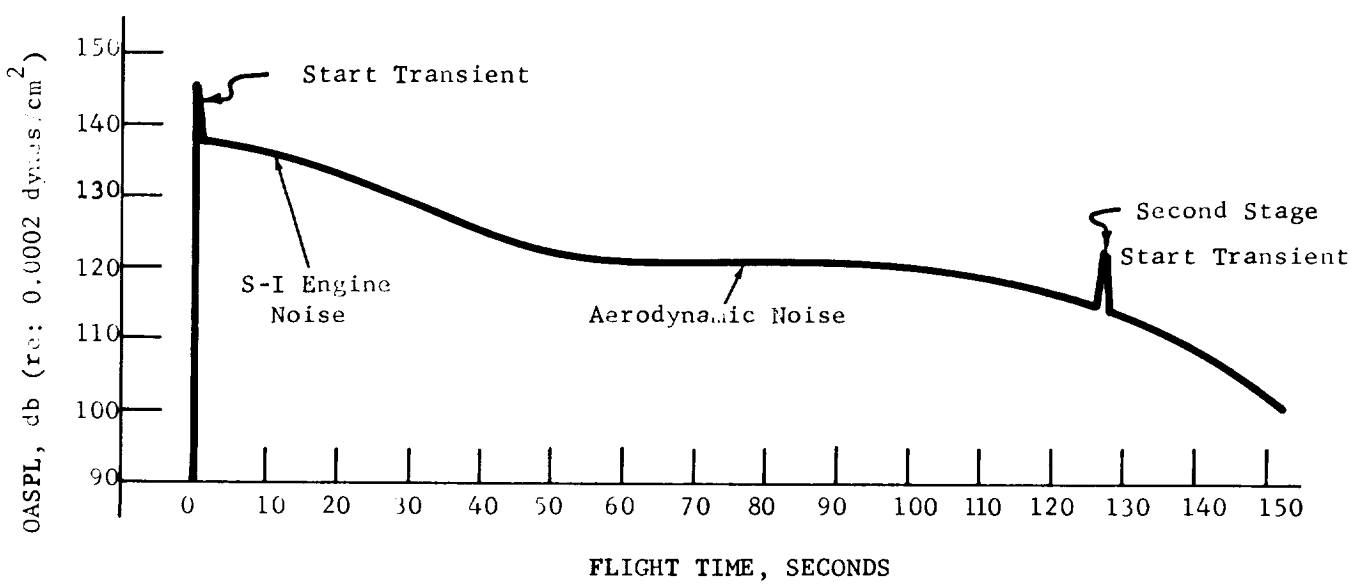

FIG. 5.11 IN-FLIGHT OVERALL SOUND PRESSURE LEVEL (OASPL) VS. FLIGHT TIME FOR INTERNAL PARTS OF STAGE S-IV AT STA. 1400.

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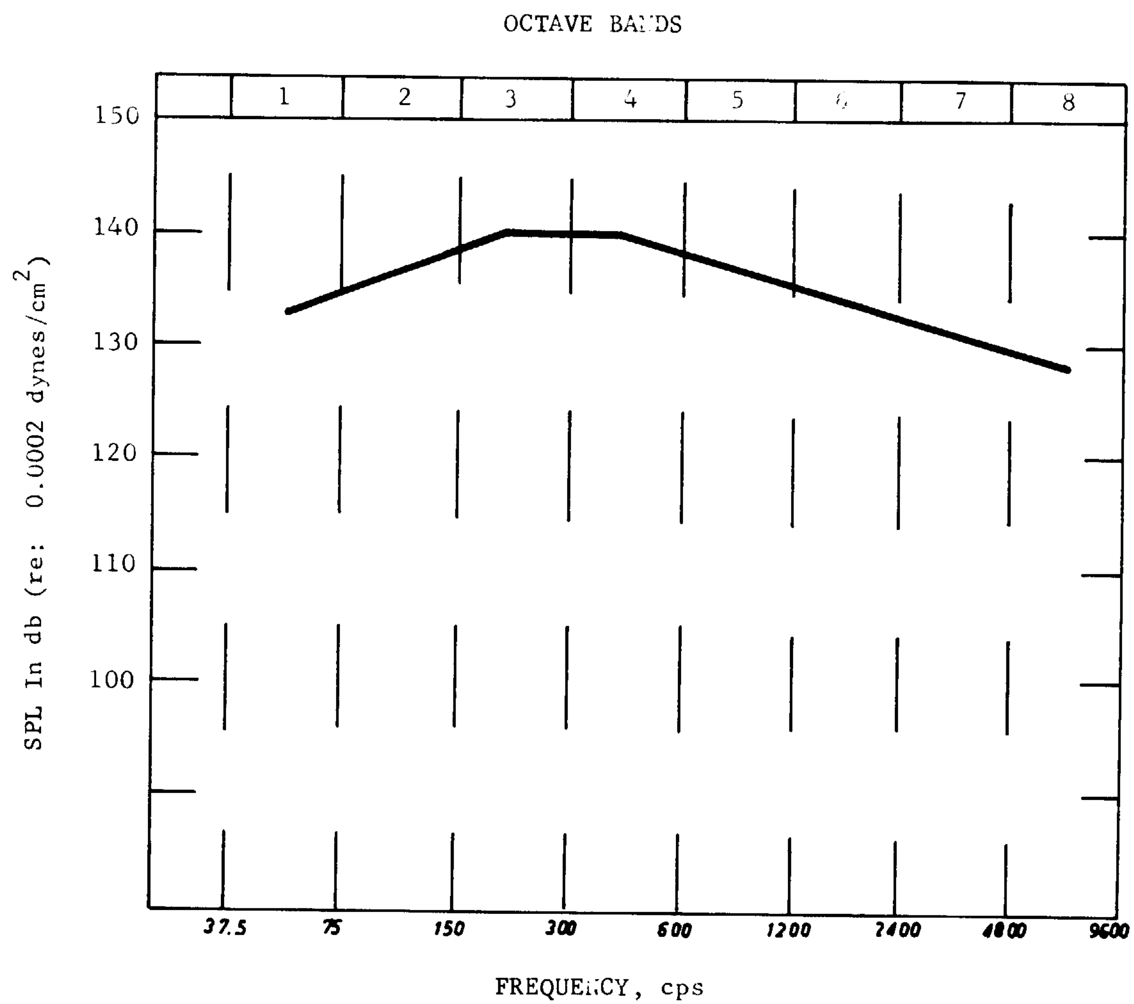

FIG. 5.12 SUB-SONIC FLIGHT SOUND PRESSURE LEVEL (SPL) VS. FREQUENCY IN OCTAVE BANDS FOR STAGE S-IV AT STA. 1400.

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The possibility of vibration damage is considered in designing equipment to insure that the equipment will pass the environmental vibration tests, which in this case are also the design criteria. Figure 5.9 shows the vibration test levels. A constant logarithmic sweep rate of about one minute per octave between the frequency limits shown in Figure 5.9 is used. The total vibration time is 1.5 hours with vibration applied in each of the three mutually perpendicular axes for one half hour at the normal operating temperature. If temperature extremes are expected, vibration will be applied at the high, low, and normal temperatures for ten minutes in each axis at each temperature.

The flight vibration criteria shown in Figure 5.9 are preliminary and may change, depending upon the results of captive test firings and initial flight tests. There are not sufficient data to mathematically compute the vibration environment at each point along the vehicle body. Theoretically, it is expected that at frequencies well above the fundamental compression and bending modes (i.e., greater than 200 cps), the vibration is very intense in the engine nozzle section and rapidly attenuates going forward. Therefore, the vibration levels shown in Figure 5.9 should only be used for all locations sufficiently forward of the engine compartment.

It is desirable that equipments be designated as Class A and Class B for vibration consideration. These classes are defined as follows:

Class A: Class A equipment consists of small components whose structural attach points are not more than three inches apart. Examples of equipment in this category include relays, solenoids, small junction boxes, valves, clips for hydraulic lines and wire bundles, etc.

Class B: Class B equipment consists of assemblies of Class A equipment in large pieces of equipment whose structural attach points lie more than three inches apart. Examples of equipment in this category would include hydraulic power units, actuator assemblies, assemblies of electronic and guidance equipment.

SECTION 6. (C) INSTRUMENT UNIT

6.1. INSTRUMENT UNIT DESCRIPTION

The Instrument Unit (previously called Instrument Compartment) is mounted to the forward end of the S-IV stage. The forward end of the unit's fuselage provides the interface mount for the payload body. In order to satisfy the requirement of noninterference with the payload body, no portion of the unit's structure extends above the payload interface. However, the hub container protrudes into the S-IV upper interstage to within 16 inches of the dome of the LH_2 container bulkhead. The form of the unit is much like four spokes attached to the hub of a wheel, forming a hollow cavity of 5 elements (Figs. 6.1 and 6.2). One spoke, located $22\frac{1}{2}°$ off Fin I toward Fin II, will carry the active guidance; the second, $22\frac{1}{2}°$ off Fin II toward Fin III, will house the passenger guidance, if used, and the remaining two spokes will contain other instrumentation. Access for installation and servicing of the equipment is provided by removable outer doors and the center hub in which one or two men can work. Entry into the hub is made through the S-IV forward interstage access door. The outside diameter and length of the fuselage rim are 154 inches and 58 inches, respectively; the inside diameter of each spoke tube container is 40 inches. The inside diameter and length of the hub container are 70 inches and 103.434 inches, respectively.

The active guidance and passenger guidance stabilized platforms are mounted on a basic support frame, which, in conjunction with an adapter plate, will allow substitution of a passenger guidance for the present active platform. In addition, interchanging of at least two different passenger guidance platform configurations may be required on relatively short notice.

Cooling of the instrument unit is facilitated since the cooled volume is one unit, separated only by the geometry of the unit. The apportionment of the cooling medium is dictated by the instrument placement and temperature range requirement.

A fixture to facilitate fabrication and handling of the instrument unit is required since the bottom of the unit projects below the 154 inch diameter field splice which forms the lower interface at the vehicle skin. This fixture must mate to the lower interface and provide support for the unit with its flight direction vertical.

Transportation of the unit with its support fixture is allowed only in the vertical position; therefore, handling requires the use of locating points as necessary for a cylinder of equivalent size and weight.

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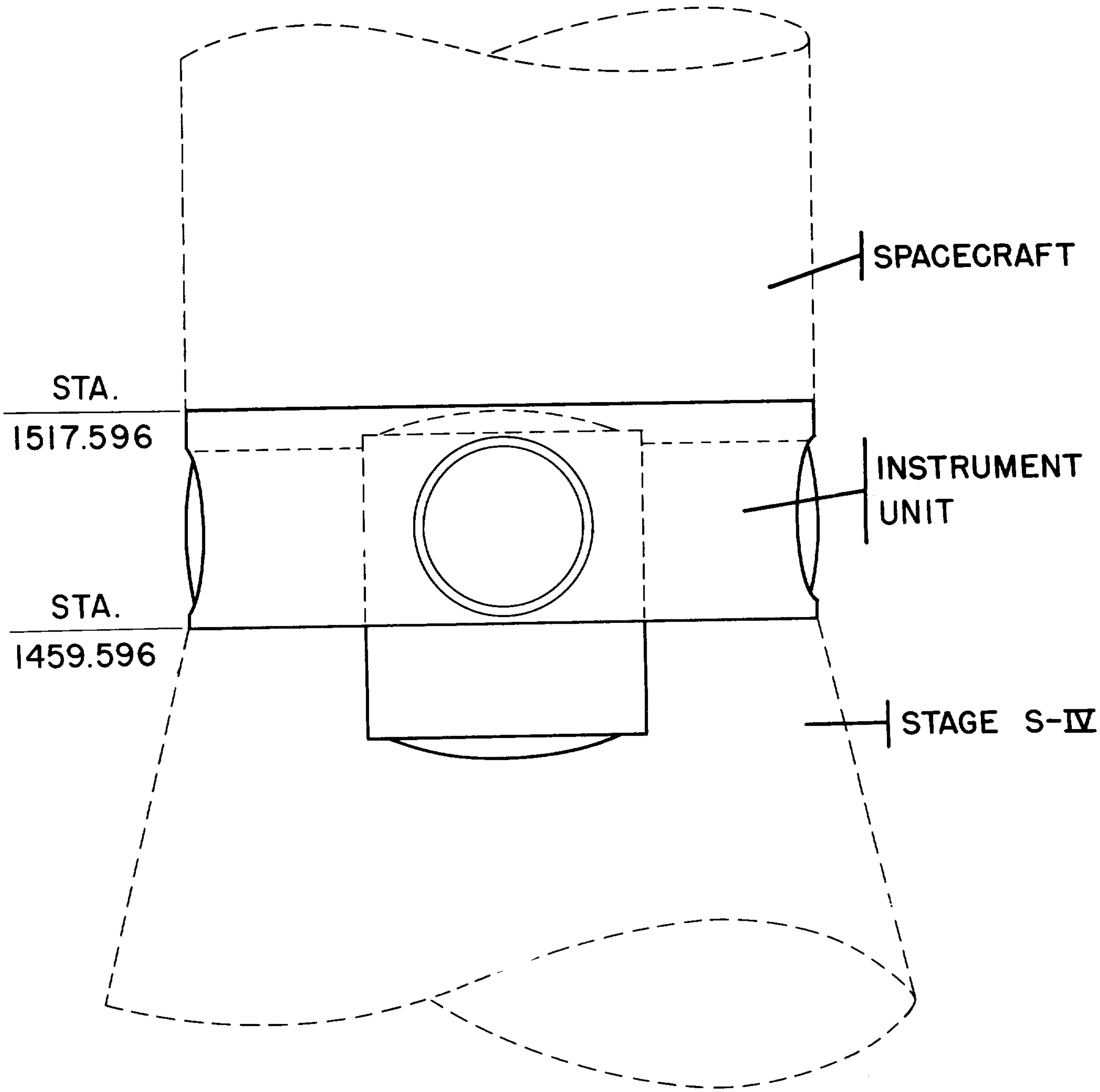

FIG. 6.1 LOCATION OF INSTRUMENT UNIT IN SATURN C-1 SPACE VEHICLE

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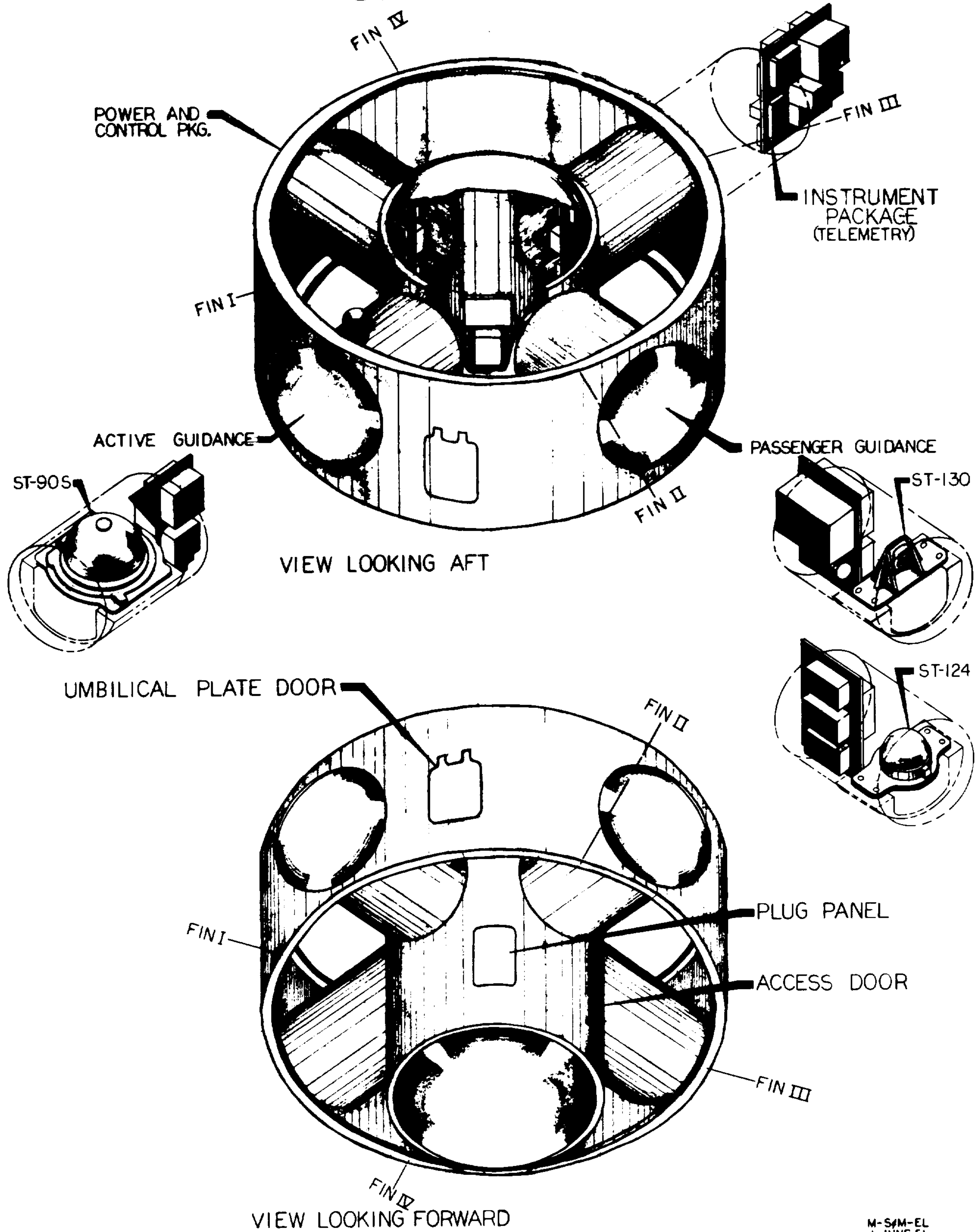

FIG. 6.2 INSTRUMENT UNIT, FORWARD AND AFT VIEWS

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Mounting for the instrumentation within the instrument unit is basically panel boards.

Requirements for internal pressurization for the instrument unit have been fixed prior to the adoption of this type unit; therefore, no change is necessary for this unit.

Orientation of the instrument unit assures a clear line-of-sight for azimuth laying of both active and passenger guidances. Some alterations to the passenger guidances may be indicated, however, to position the prism of each in the proper relation. Such modifications are reasonable to make.

6.2. INSTRUMENT UNIT STRUCTURE

6.2.1. Structural Capability. The unit provides a stable support for the guidance and instrumentation equipment and structure for the vehicle assembly under the environmental loads during static handling and transportation and flight conditions.

The spoke tube containers and hub container of the unit are pressure-tight vessels with the capability of maintaining an internal pressure of at least 10 psia without the employment of an auxiliary pressurizing system.

6.2.2. Requirements for Unit Handling, Transportation and Checkout Fixture, and Adapter Hoist Points. The fixture provides a platform which supports and maintains the unit oriented in an upward vertical flight attitude during its handling, transportation, and checkout phases. The fixture mates with and supports the unit completely at the unit's vehicle assembly splice connections only, without the use of secondary supports from the spoke tube containers or the hub container.

Hoist points are provided in the fuselage rim structure of the unit. These hoist points are used in transfering the unit between the fixture and vehicle.

6.2.3. Antennas and Other Exterior Equipment. The design of the instrument unit dictates that skin cutouts be made only where mandatory, therefore, all antennas and other exterior equipment are surface mounted.

6.3. INSTRUMENT UNIT EQUIPMENT

Table 6.1 contains a list of equipment installed in the instrument compartment and canisters located in the S-I stage. Descriptions of the guidance and control equipment and the electrical equipment are found in Sections 7 and 8, respectively. The telemetry and rf systems are described below.

(C) Table 6.1

EQUIPMENT INSTALLED IN INSTRUMENT UNIT

1. VEHICLE GUIDANCE SYSTEM

   a. ST-90-S GUIDANCE SYSTEM

      ST-90-S Stabilized Platform

      ST-90-S Servo Amplifier Box

      Guidance Computer

      Guidance Signal Processor

      Azimuth Ring Control Box

      ST-90-S Missile Mounting Frame

   b. PASSENGER GUIDANCE SYSTEM

      Gyro Stabilized Platform

      Platform Electronics

      Digital Coupler

      Guidance Computer

      Guidance Signal Processor

      Vehicle Mounting Frame

2. CONTROL SYSTEM

   Transducer, Angle-of-Attack

   Control Computer

   Pitch Control Accelerometer

   Yaw Control Accelerometer

   Rate Control Gyros

   Control Signal Processor

   Pitch and Yaw Actuator Assembly

(C) Table 6.1 (Cont'd):

3. TIME BASE SELECTOR

   Program Device "J" Mod "O"

4. NETWORKS

   Platform Distributor

   Propulsion Distributor

   Heater Power Distributor

   Measuring Distributors

   Junction Boxes

   Power Distributors

   Main Distributor

   EBW Firing Unit

   Control Distributor

   Switch

   Abort Distributor

   Power Supplies

   750 VA Invertor and Regulator

   Battery Assemblies

   Measuring Voltage Supply 5V

   Control Voltage Supply 60V

   Battery Shunt Box Assembly

   Flight Sequencer

(C) Table 6.1 (Cont'd):

5. INSTRUMENTATION AND MEASURING

UDOP Transponder Set

UDOP rf Amplifier

Radar Altimeter

Azusa

C-Band Radar

Telemeter XO-4

Telemeter 216 Channel

Telemeter SSFM

Telemeter PCM

Command Receiver and Decoder

Command Receiver Power Supply

6. AIR BEARING SYSTEM

Air Bottle

Air Bearing Regulator No. 1 and Heater

Heater Thermostat Assembly

7. SPECIAL EQUIPMENT

Continuous Light Assembly

Horizon Sensor

Horizon Sensor Power Supply

6.3.1. Telemetry and Rf Systems. The telemetry in the instrument unit consists of three FM/FM sets and one PCM/FM/FM set. This telemetry system then has 48 standard IRIG analog channels and a PCM bit rate of 36,000 per second.

All of the rf links for tracking are carried in the instrument compartment. This includes the UDOP, Azusa, Radar Altimeter, C-Band radar, and a developmental tracking system now in the design stage. The UDOP tracking system is a UHF doppler system that consists of a ground CW transmitter, a vehicle born transponder that receives the transmitted wave, doubles its frequency and transmits it, and several ground based receivers located near the launching site at points geometrically spaced to give best results. These receivers receive both the basic wave and the doubled wave and compare the relative frequencies. Information from three receivers is sufficient to locate a vehicle in space. The Azusa system is similar to the UDOP system except that the Azusa transmitter transmits a modulated wave and the Azusa transponder does not double the frequency but shifts the carrier several megacycles away from that originally transmitted. The modulation is used for fine measurements in the doppler system. The C-Band radar is a standard pulse type radar operating at 5555 mc/sec and using a high gain parabolic antenna on the ground and a repeater type transponder in the vehicle.

The antennas are all of the external configuration and are located on the skin of the instrument compartment.

The frequencies are:

Telemetry	248.6 mc 252.4 mc	256.2 mc 1 unassigned
UDOP	900 mc & 450 mc	
Azusa	500 mc	
Radar Altimeter	Unassigned	
C-Band Radar	5555 mc	
Development Tracking	Unassigned	

6.3.2. Instrument Unit Measuring Program. Approximately 100 ground and inflight measurements in the instrument unit will be required during a typical Block II mission. A condensed list of the measured parameters in the instrument unit, including the range of the measurements, is given in Table 6.2. This list of measurements may change from flight to flight as required by the evaluation of the previous flight. Usually the listed measurements are only fed into the telemetry system of the instrument unit for the transmission to ground stations. Some critical signals, voltages, and guidance measurements may also be routed to the abort system and to the spacecraft.

6.3.3. Research and Development. Analytical and experimental studies and developmental efforts have been carried out at MSFC on digital techniques for transmission and acquisition of data. The result of this effort is an extremely flexible digital data link capable of efficiently handling data from a variety of sources. These include analog-to-digital converters, digital guidance computers, horizon sensing system, platform position measurements, sequence and event type measurements, radio altimeter, as well as other measurements.

The PCM telemetry system design for Saturn includes the latest state of the art concepts in digital data transmission. Among the new concepts are an extremely stable clock source, phase lock techniques for bit rate synchronization, and the use of pseudo-random codes for frame and subframe identification. These techniques will eliminate most of the problem areas in earlier PCM system designs.

The system was designed to complement the latest automatic checkout and monitoring systems developed in the Guidance and Control Division.

The Analog to Digital converter designed for the PCM system is completely compatible with the system concept of automatic checkout and will be used in all stages of the Saturn vehicle.

6.3.4. Coordination. The coordination of efforts in this field is done by the Saturn System Instrumentation Working Group which consists of permanent members from MSFC personnel concerned with instrumentation, representatives from all stage contractors, and payload contractors, where applicable. Some of the responsibilities of this group are as follows:

(1) To solve immediate instrumentation system problems concerning the complete Saturn system

(2) Authorize the instrumentation program for each stage

(3) Define and offer solutions for all interface instrumentation problems.

(C) Table 6.2

INSTRUMENT UNIT MEASURING PROGRAM

MEASURED PARAMETER	RANGE
Temperature, Internal, Q Ball	0-150°C
Temperature, Total	0-800°C
Temperature Wall, Density Gauge	0-100°C
Temperature Air, Density Gauge	0-100°C
Temperature Gas Nitrogen Manifold	-50 to +50°C
Temperature Azusa, Internal	10-65°C
Pressure Instrument Compartment	0-20 psia
Pressure Air Bearing Supply ST-90	0-60 psid
Pressure Control Equipment, Supply	0-3500 psi
Pressure Control Equipment, Regulator	0-800 psia
Dynamic Pressure (Q)	5 to 750 PSF
Differential Pressure, Pitch, Q Ball	± 5 psi
Differential Pressure, Yaw, Q Ball	± 5 psi
Differential, Pressure Q Compensation	15 psi
Pressure Instrument Compartment	0-20 psia
Vibration - ST-90, Y Axis	± 3g
Vibration Instrument Panel, Lateral	± 5g
Vibration Lower Support, Longitudinal	± 20g
Vibration Instrument Panel, Pitch	± 3g
Angular Velocity, Pitch	± 10°/sec
Angular Velocity, Yaw	± 10°/sec
Acceleration, Longitudinal	0-5g
Acceleration, Longitudinal Coarse	0-5g
Acceleration, Longitudinal	± 1g
Acceleration, Pitch, Control	± 5m/sec. sq

(C) Table 6.2 (Cont'd):

MEASURED PARAMETER	RANGE
Acceleration, Yaw, Control	± 5m/sec. sq
Angular Velocity, Pitch, Control	± 10°/sec
Angular Velocity, Yaw, Control	± 10°/sec
Angular Velocity, Roll, Control	± 10°/sec
Local Angle-of-Attack	± 10°
Angular Velocity, Pitch	± 100°/sec
Angular Velocity, Yaw	± 100°/sec
Angular Velocity, Roll	± 100°/sec
Angle-of Attack, Pitch, Q Ball	± 15°
Angle-of-Attack, Yaw, Q Ball	± 15°
Density, Ram Air	10^2 to 10^{-3} mb, in 5 range steps
Pitch Position, ST-90 Minus Prog.	± 15°
Yaw Position, ST-90	± 15°
Roll Position, ST-90	± 15°
Servo Signal, Cross Range Accel	± 100 MV
Servo Signal, Slant Range Acceleration	± 100 MV
Servo Signal, Slant Range	± 100 MV
Time Base Selector Output	0-5V
Time Base Selector Zero Indication	on-off
Temperature Instrument Compartment	0-60°C
Temperature Inlet Air for Air Bearing	20-30°C
Pressure Air Bearing Supply	0-60 psid
Pressure Air Bearing Supply, ST-90	0-3500 psig
Pressure Control Equipment, Supply	0-3500 psig
Pressure Control Equipment Regulator	0-800 psi

SECTION 7. (C) GUIDANCE AND CONTROL SYSTEM

7.1. DESCRIPTION

The Saturn guidance and control system which has been developed is an all-inertial type using a digital computer for the guidance computations and an analog computer for the control functions. Acceleration measurements for the guidance system and accurate attitude angle measurements for the control system are obtained from the stabilized platform subsystem. This integrated system has been based on attaining an overall reliable and proven vehicle system. The characteristics of the propulsion systems, structure, ground support equipment, instrumentation, and the associated aerodynamic and launch considerations are all integrated into the established Saturn guidance and control system. These basic factors when combined with the programmed launching plans for Saturn have defined the C-1 version. The techniques and guidance modes used on C-1 are applicable to and designed toward advanced guidance and control systems for later versions of the space vehicle.

The overall system has been designed for possible integration with the spacecraft in mind and offers, with its early flight testing in the Saturn vehicle, a high reliability and performance assurance. The system, operating in the Path Adaptive Guidance Mode, will be capable of performing all the remaining portions of Saturn missions in addition to launching the space vehicle. Thus, the guidance system should be considered either as the primary or as a redundant system for the spacecraft. In this way, full utilization of the Research and Development which has gone into the Saturn system will be made, and as advances in the state-of-art are reflected in the system these will be available for use in the spacecraft. Some of the specific factors which were taken into account for the guidance and control system are discussed below.

The prime consideration in the establishment of the Saturn guidance and control system has been to attain an overall high reliability for the complete vehicle system. The G&C system concept, the mechanization, and program planning have evolved from this basis. In the discussion which follows, it is readily seen that the requirements and demands on the Saturn G&C system are much more stringent in many respects than in previously developed and presently conceived missiles or space vehicles.

The three major factors that were considered in establishing the G&C systems concept are:

(1) The elastic booster and the associated bending modes affecting control stability and the allowable magnitude of any command inputs

(2) The requirement of an engine-out capability to increase the reliability of the propulsion systems

(3) The requirement that the Saturn launch vehicles will be used for a large variety of missions.

Although the G&C system concept was primarily based on the above factors, other normally encountered practical considerations, not including those of the G&C equipment itself, also were taken into account.

These include the following:

(1) Control torques available as limited by the swiveled engines' deflection

(2) Wind profile which the vehicle may be expected to fly through

(3) Sloshing effects of the propellants

(4) Probability of wide dispersions of the launch time due to countdown delays

(5) Margin of static instability during part of boost phase.

In the final analysis the G&C systems' concept has been modified and realigned to the practical limitations of the G&C equipment mainly considering the status of development of this equipment and the confidence limits which can be placed on its reliability. The practical limitations also include, in varying degrees, the following:

(1) The accuracies attainable with various guidance schemes and hardware

(2) The complexity of the system required to mechanize the concept

(3) Size, weight, and power consumption of the equipment

(4) Any appropriate trade-offs between G&C equipment parameters and other subsystems' reliability, weight savings, etc.

(5) Excessively costly installation, and/or complex launch operations.

The R&D program planning has further defined the G&C equipment and its systems' concept, considering the phasing-in of the equipment with the established schedules. The general approaches have been to attain maximum reliability of the equipment in the period of the specified schedules and has considered the following:

(1) Initially to use control hardware which has a history of high reliability performance

(2) Phase-in guidance hardware actively after adequately proving the propulsion and control systems in flight

(3) Use guidance hardware which will have a large usage rate and a high confidence factor at the time of its introduction in the Saturn launch vehicle

(4) Determine as early as practical the weak points of the system and hardware and establish a sound R&D program to phase-in major improvements

(5) Maintain a continuous program of reliability testing to discover any previously undetected deficiencies or those introduced through some unrefined manufacturing process change.

For the Saturn guidance and control system, there are two modes of operation and these are definitely interrelated, functionally and in the manner in which they are mechanized. These modes are the control mode and the guidance mode. The control mode takes account of the basic stability requirements of the vehicle and provides proper control torques to overcome any disturbance of the vehicle's attitude from a desired reference. The guidance mode determines the trajectory which the space vehicle will follow in attaining the correct velocity vector, space coordinates, and time of arrival of the space vehicle with respect to its destination. Many practical and desirable considerations also were taken into account for the guidance mode, including optimizing certain chosen parameters, such as minimizing the amount of propellant required for a given mission, thereby increasing the allowable size of the spacecraft.

All inputs to the guidance mode and control mode are primarily obtained from the stabilized platform of the all-inertial system. The outputs of the platform are the space fixed inertial data from three mutually perpendicular accelerometers required for the guidance mode, and highly accurate reference attitude information in pitch, yaw, and roll for stabilization of the vehicle in the control mode. Control accelerometer outputs are combined with the accurate attitude information to limit any excessive buildup of the vehicle's angle-of-attack during critical periods of flight. Rate gyros (S-I stage only) and differentiating networks provide attitude rate signals. Complex shaping networks, based on the total stability requirements, are included in the control computer.

Computations using the guidance signals and control signals are accomplished in a general purpose digital computer and an analog computer, respectively. Initially a 3000 word digital guidance computer will be used for flight computations and portions of the prelaunch checkout. For the later vehicles a digital computer using advanced technology and reliability approaches, and a larger expandable memory will be provided.

The guidance system also includes a launch computer which will communicate with the space vehicle's computer and platform. This will allow a continuous change of the flight program constants and platform aiming azimuth. This hardware capability combined with the allowable launch window variations afforded by the Adaptive Guidance Mode will meet any foreseeable injection requirements.

The analog control computer utilizes a combination of magnetic amplifiers and transistor amplifiers for summation together with an electromechanical device for programming gain changes. Studies are under way to determine the feasibility and necessity of using the digital computer for both the guidance and control functions in later Saturn versions.

The guidance and control system also computes or provides functions (e.g., staging, cutoff, etc.) for the propulsion system and the overall vehicle.

The presently developed four-gimbal, full-freedom stabilized platform, and the guidance system which has been developed for the initial vehicles, will provide an accuracy of injection with 0.95 meter/sec (3 σ value for the total velocity vector). Advanced platforms are being developed which will improve this figure or provide the equivalent value for much longer periods of flight. Stellar supervision of the prime inertial reference is one of the means for this improvement. This development maintains the prime requirement of always increasing the reliability along with reduction in weight, size, and power consumption.

Table 7.1 gives a brief description of the major components of the system and their functions, and Figure 7.1 shows a block diagram of the system.

The initial Saturn vehicles will carry the guidance and control system in the instrument unit. For the application to the Saturn vehicle and Apollo spacecraft, a thorough investigation should be made to determine what approaches can be taken to provide the best technically integrated guidance and control system for the entire space vehicle.

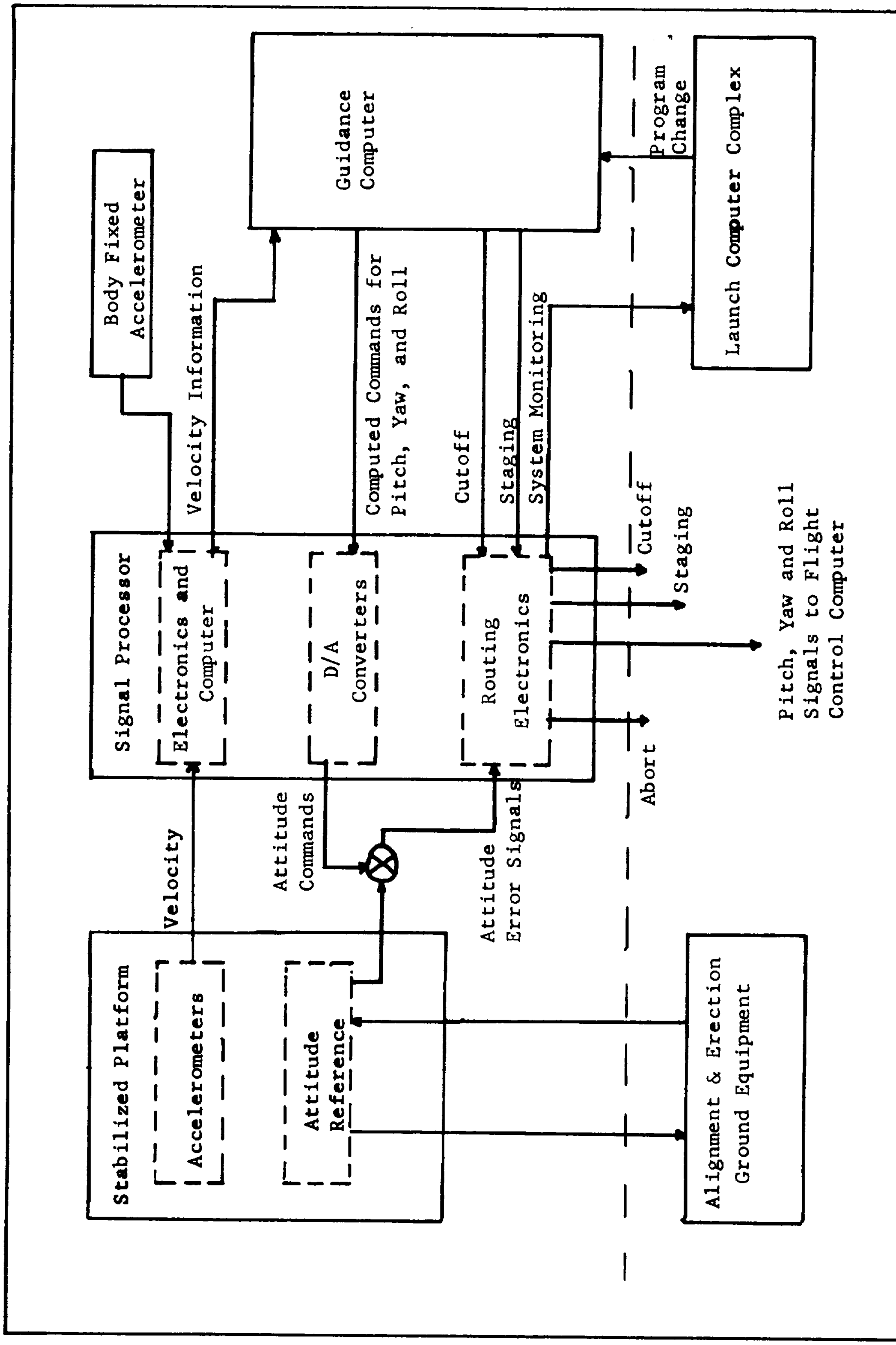

FIG. 7.1 BLOCK DIAGRAM OF THE GUIDANCE AND CONTROL SYSTEM

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Table 7.1

MAJOR COMPONENTS OF THE C-1 LAUNCH VEHICLE G&C SYSTEM

ITEM	DESCRIPTION	FUNCTION
Stabilized Platform	Four gimbal, full freedom	Provides inertial reference for accurate measurement of missile attitude and accelerations
Guidance Signal Processor	Analog/digital circuitry, electromechanical module, and power amplifiers	1. Processes input data for use by digital computer. 2. Processes output data for use with platform and control computer. 3. Amplifies discrete signals to network. 4. Buffers digital data to and from GSE.
Guidance Computer	Digital, general purpose, drum memory, approximately 3000 words permanent and 448 words temporary storage word length 24 bit plus sign, silicon circuitry.	1. Computes, from input data, vehicle velocity and position and derives guidance commands. 2. Computes discrete signals for engine cutoff, staging, etc.
Control Computer	Analog circuits, silicon semi-conductors, magnetic amplifiers, gain change program mechanism.	1. Combines properly all control measurements. 2. Provides proper bending mode and stabilizing filtering. 3. Provides signals to actuation system.

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Table 7.1 (Continued)

ITEM	DESCRIPTION	FUNCTION
Control Signal Processor	Demodulator-Amplifier for Sensors	1. Converts outputs of control sensors (rate gyros & control accelerometers) for use by the control computer.
Control Accelerometers	Analog output, spring mass type	Used in control system to minimize angle-of-attack.
Rate gyro	Standard rate gyro package	Used during first stage burning to provide attitude rate signals in pitch and yaw.
Actuation System	Hydraulic, closed, 3000 psi system. Uses separate electrical auxiliary pump motor for ground checkout, and a turbine driven pump for flight.	1. Swivel 4 outer engines on S-I 2. Swivel 6 engines on S-IV

Special consideration should be given to the fact that the very powerful and flexible guidance mode and hardware utilized in Saturn can be used to great advantage throughout the total space mission. This system will be proven by the time the Saturn becomes operational.

A brief and simplified description of the Saturn Guidance Mode is given and it should be considered by the Spacecraft Contractor since this technique has been established and applied by MSFC to the entire mission. Use of a single type of guidance mode will simplify the launch and checkout operations and provide more efficient use of the space vehicle guidance equipment.

7.2. SATURN GUIDANCE SYSTEM CONCEPT

The theoretical guidance concept under development at MSFC is termed the "adaptive guidance mode". The mathematical structure of this concept is sufficiently comprehensive to include all forseeable specific guidance schemes. The specific guidance scheme is derived from the general mathematical structure when the conditions and requirements of the particular flight are specified.

The fundamental principle of the "adaptive guidance mode" is that any specific guidance scheme derived from the general mathematical structure will cause the vehicle to fly an optimum trajectory. The feature to be optimized is to be dictated by the assigned mission. Examples of features to be optimized are:

(1) Propellant consumption

(2) Maneuvering angles

(3) Angular rates

(4) Aerodynamic heating

In developing a specific guidance scheme the feature to be optimized will often be that of minimizing the propellant consumption. Since the propellant flow rate per engine is essentially constant, this reduces to a condition of minimum propelled flight time.

The logic of the descriptive term "adaptive" and the power of the mathematical structure are illustrated by the following examples of its flexibility and capability:

(1) Mission Flexibility

a. Earth fixed target,

b. Orbital injection,

c. Lunar landing,

d. Circumlunar flight and return,

c. Planetary flight.

(2) Vehicle and Propulsion Flexibility

a. Single stage vehicle,

b. Multistage vehicle,

c. High or low thrust engines,

d. Continuous thrust periods,

e. Thrust periods interrupted by coast periods.

(3) Absorption of Disturbances

a. Large thrust deviations,

b. Engine failure with clustered engines,

c. Winds.

(4) Independence from Intelligence Mechanism

a. Inertial mechanisms,

b. Optical mechanisms,

c. Radio.

(5) Independence from Guidance Computer Location

a. Vehicle borne,

b. Ground based and linked to vehicle by radio command channels.

(6) Application

a. Control of certain phases of a flight mission,

b. Control of the total flight history of the mission.

The method of the Adaptive Guidance Mode is to evaluate the current vehicle coordinates and flight status and select an optimum flight for the remainder of the trajectory. The thrust level is assumed to remain constant at the current value for the remainder at the stage and to take on standard value for the subsequent stages. The trajectory selection is repeated continuously throughout the flight.

In addition the "time-to-go" is continuously computed during last stage in order to provide a cutoff signal, wherever required.

The mathematical techniques employed in formulating the Adaptive Guidance Mode are those of the Calculus of Variations. A simple case for a flat earth with a constant gravitational field (g) and with no atmosphere will illustrate the results obtained with the Calculus of Variations.

(1) Given: (Fig. 7.2)

a. Engine characteristics - F, $\dot{m}$

b. Vehicle characteristics - m_o

c. Terminal conditions - x_t, y_t, T, $\dot{x}_t$, $\dot{y}_t$

d. Equation of vehicle motion

$$\ddot{x} = \frac{F}{m} \sin(\chi)$$
$$\ddot{y} = \frac{F}{m} \cos(\chi) - g \qquad (1)$$

where χ is the angle between the vehicle's longitudinal axis and the launch vertical.

e. Condition for Optimum - Minimum powered flight time.

(2) Find an expression for χ and cutoff time, t_c, in terms of measurable quantities, and which will provide the optimum trajectory.

The cutoff coordinate of the vehicle may be written in terms of the target coordinates from an inspection of Figure 7.2.

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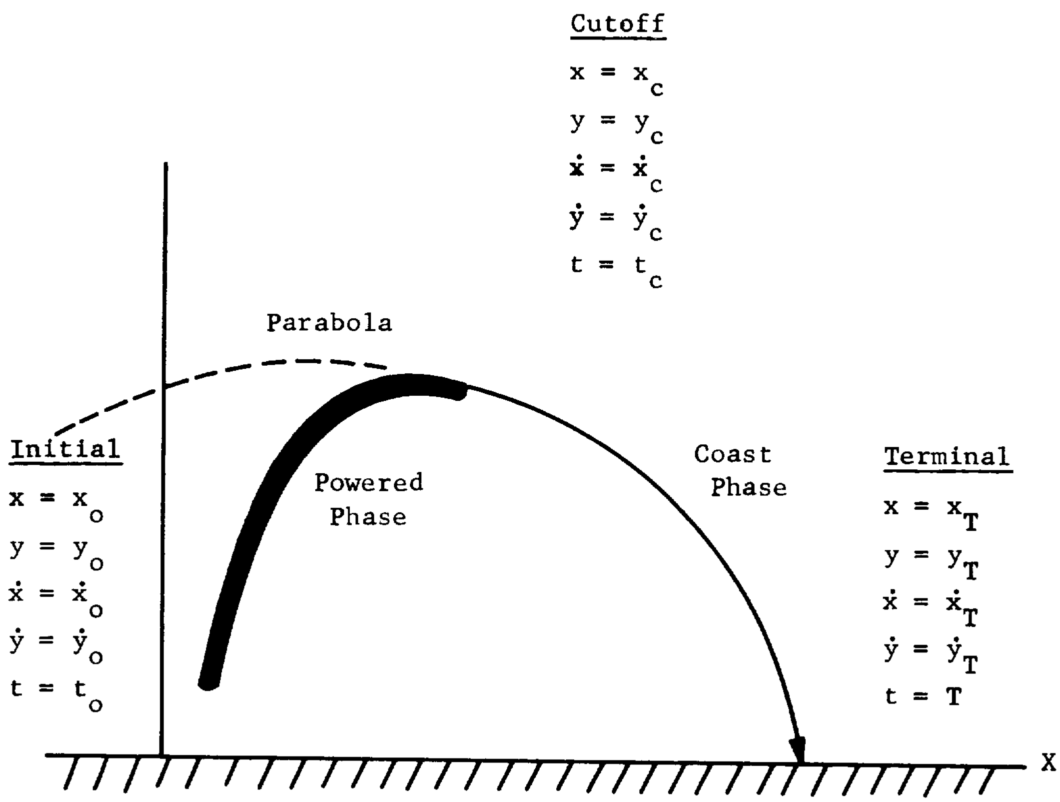

FIG. 7.2. TRAJECTORY SHOWING TARGET COORDINATES

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Thus,

$$\begin{aligned}
\dot{x}_c &= \dot{x}_t \\
x_c &= x_t - \dot{x}_t \,(T - t_c) \\
\dot{y}_c &= \dot{y}_t - g\,(T - t_c) \\
y_c &= y_t - \dot{y}_t \,(T - t_c) - \frac{1}{2}\, g\,(T - t_c)
\end{aligned} \tag{2}$$

The Calculus of Variation then provides a solution to the tilt program, χ, of the form

$$\tan \chi = \left(\frac{c_1 t + 1}{c_2 t + 1}\right) \cdot \chi_o \tag{3}$$

where c_1 and c_2 are parameters which must be determined in addition to χ_o.

If Equation 3 is substituted into Equations 1 and the first and second integrals taken, the resulting equations may be solved simultaneously with Equations 2 to give the desired expression.

Thus,

$$\bar{\chi}_o = \bar{\chi}_o \left[x_o,\ y_o,\ x_o,\ y_o,\ \left(\frac{F}{m}\right)_o,\ t_o\right]$$

$$t_c = t_c \left[x_o,\ y_o,\ x_o,\ y_o,\ \left(\frac{F}{m}\right)_o,\ t_o\right]$$

The subscript o refers to a variable "initial" point taken as any point during flight, the coordinates of which are supplied by the intelligence mechanism. The measured coordinate χ_o represents the attitude of the thrust vector at that time. These equations give the ideal value $\bar{\chi}_o$ that should be assumed there. The series form of the corresponding equations for a spherical earth and more realistic missions are shown in Figure 7.3. For convenience, the $\bar{\chi}_o$, x_o, y_o, $\dot{x}_o$, $\dot{y}_o$, $\left(\frac{F}{m}\right)_o$, and t_o have been replaced by χ, x, y, $\dot{x}$, $\dot{y}$, $\left(\frac{F}{m}\right)$, and t, respectively.

Thrust Direction:

$$\chi = A_0 + A_1 x + A_2 y + A_3 \dot{x} + A_4 \dot{y} + A_5 t + A_6 \frac{F}{m} + A_7 x^2 + A_8 y^2 + A_9 \dot{x}^2 + A_{10} \dot{y}^2 + A_{11} t^2 + A_{12} \left(\frac{F}{m}\right)^2 + A_{13} xy$$

$$+ A_{14} x\dot{x} + A_{15} x\dot{y} + A_{16} xt + A_{17} x \frac{F}{m} + A_{18} y\dot{x} + A_{19} y\dot{y} + A_{20} yt + A_{21} y \frac{F}{m} + A_{22} \dot{x}\dot{y} + A_{23} \dot{x}t + A_{24} \dot{x} \frac{F}{m}$$

$$+ A_{25} \dot{y}t + A_{26} \dot{y} \frac{F}{m} + A_{27} t \frac{F}{m}$$

where A_0, A_1 . . . . are stored constants different for each stage

Cutoff:

$$t_c = B_0 + B_1 x + B_2 y + B_3 \dot{x} + B_4 \dot{y} + B_5 t + B_6 \frac{F}{m} + B_7 x^2 + B_8 y^2 + B_9 \dot{x}^2 + B_{10} \dot{y}^2 + B_{11} t^2 + B_{12} \left(\frac{F}{m}\right)^2$$

$$+ B_{13} xy + B_{14} x\dot{x} + B_{15} x\dot{y} + B_{16} xt + B_{17} x \frac{F}{m} + B_{18} y\dot{x} + B_{19} yy + B_{20} yt + B_{21} y \frac{F}{m} + B_{22} \dot{x}\dot{y} + B_{23} \dot{x}t$$

$$+ B_{24} \dot{x} \frac{F}{m} + B_{25} \dot{y}t + B_{26} \dot{y} \frac{F}{m} + B_{27} t \frac{F}{m}$$

where B_0, B_1, ... are stored constants

FIGURE 7.3. SATURN GUIDANCE EQUATIONS.

SECTION 8. (C) ELECTRICAL SYSTEMS

8.1. GENERAL

Electrically, the Saturn C-1 Launch Vehicle System is divided into three areas: Stage S-I, State S-IV, and the instrument unit (or instrument compartment) located above Stage S-IV. The instrument unit will house most equipment to guide the vehicle during ascent. It contains a control computer which generates the signals for the actuators of the stages S-I and S-IV. Tracking and telemetering equipment will also be housed in the instrument unit. In addition to these, the instrument unit will have abort distributor which will receive all abort signals for distribution and execution. This arrangement provides enough flexibility to sense abort conditions in all stages, including the Apollo Spacecraft, and to give abort parameters to the spacecraft. A block diagram of the Saturn C-1 instrument unit is shown in Figure 8.1.

Since the Saturn C-1 Launch Vehicle System has already been established, the Apollo Spacecraft should be designed to fit into this system. The S-I stage, as well as the S-IV stage, have been defined between MSFC and the S-IV stage contractor, and a system has been worked out to assure compatibility of the overall vehicle. The following paragraphs define the existing Electrical System of the C-1 Launch Vehicle. A description of the major components of the electrical system will also be given.

8.2. ELECTRICAL SYSTEM DESIGN SPECIFICATIONS

For the electrical system, the following list of specifications, standards, drawings, and publications are the basis for the design and process requirements in manufacturing of the S-I and S-IV stages. This list reflects the latest official design specifications for the Saturn C-1 system.

MIL-B-5087	Bonding, Electrical (for aircraft).
MIL-I-6051	Interference Limits and Methods of Measurements, Electrical and Electronic Installation in Airborne Weapons Systems and Associated Equipment.
MIL-I-6181	Interference Limits, Tests and Design Requirements Aircraft Electrical and Electronic Equipment.

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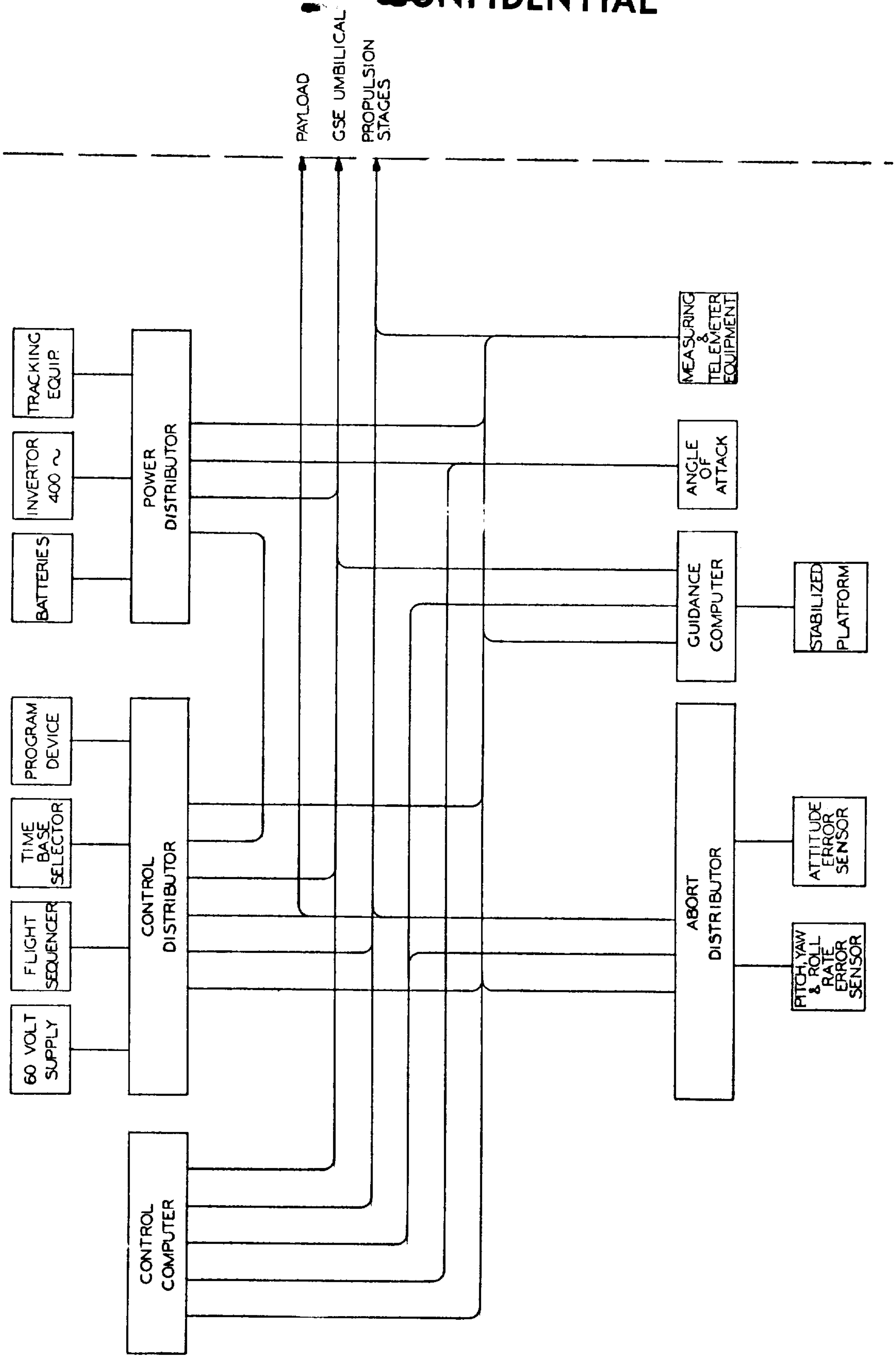

FIG. 8.1 BLOCK DIAGRAM OF THE SATURN C-1 INSTRUMENT UNIT

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MIL-W-8160B	Wiring, Guided Missile, Installation of.
MIL-I-26600	Interference Control Requirements, Aeronautical Equipment.
ABMA-PD-E-53	Electrical Wiring Procedures.
ABMA-PD-R-187	Relays, Hermetically Sealed, Missile Equipment.
ABMA-PD-C-711	Cable and Harness Assemblies, Electrical, Missile System, General Specifications for.
ABMA Drawing 8931071	Printed Circuit Soldering and Inspection Procedure.
MSFC Drawing 10M01071	Environmental Protection when using Electrical Equipment with the Areas of Saturn Complexes where Hazardous areas exist, Procedures for.
MSFC Drawing 10509300	Soldering Electrical Connections for Space Vehicles, Procedure for.
MSFC Drawing 10509322	Electrical Wiring Procedures.
MIL-STD-16	Electrical and Electronic Reference Designations.
ABMA-STD-54	Accepted Standards of Electrical Engineering Design.
ABMA-STD-428	Printed Circuit Design and Construction Standard.
XDOD-STD-0001A	Electrical and Electronic Reference Designations, Saturn Ground Support Equipment.
XDOD-STD-0002B	Saturn Ground Support Equipment Hardware List.

In addition to these electrical specifications, it is recommended that the contractor of the Apollo "A" should also use, wherever possible, components already developed by the S-I and S-IV Saturn stage contractors. This action is necessary to achieve standardization within the Saturn system.

8.3. ELECTRICAL SYSTEMS GENERAL

Each stage is electrically self-contained and has its own power system except for the control and feedback power supplies which are centrally located for all stages in the instrument unit. The electrical interface between stages is well defined, thus allowing, when electrical substitutes for adjacent stages and the GSE are provided, the checkout of each stage to take place prior to overall systems mating. To insure uniformity throughout the vehicle and GSE, MSFC works jointly with the stage contractors to define the design of the stage electrical circuits which include all electrical subsystems such as:

(1) Propellant utilization

(2) Propulsion and pneumatic control

(3) Measuring and telemetry

(4) Tracking and range safety

(5) Power supplies

(6) Command destruct and ordnance systems.

The design shall include provisions for automated checkout, testing and launch operations, utilizing digital computers. The contractor shall obtain concurrence from MSFC for all electrical circuit designs outlined under the paragraph herein entitled Required Electrical Documentation. The stage contractors provide an electrical systems scheme for the stage system defining power supplies, network, polarity diagram for the control system, and inflight sequence of events as well as start and cutoff sequence.

8.4. ELECTRICAL DESIGN CRITERIA

Consideration and design analysis is given in the following:

(1) Flight sequence design interface

(2) Control system interface

(3) Engine, pneumatic and electronic package

(4) Instrumentation system

(5) Command destruct and ordnance systems.

A brief description of each of these areas is given in the following paragraphs.

8.4.1. Flight Sequence Design Interface. The forward instrument unit will provide a stimulation source for all inflight time based events. Signals that are required by the S-IV stage prior to separation from S-I stage will come through the S-I/S-IV interface. This philosophy is used to assure that all signals required by the adjacent stages are routed through the lower stage to prevent a signal race. The S-IV contractor will specify what signals are required from the S-I stage. The S-I stage will presently furnish the following signals to the S-IV stage:

(1) Prestart signal. S-I/S-IV stage separation minus 20 to 28 seconds. (Times are not firm to date).

(2) Blow-off vent ports. Same time as prestart signal, except that S-I stage will furnish all circuits to the two vent port EBW units.

(3) Separation of S-I/S-IV stages.

The methods to be utilized in transmitting signals between stages is as follows:

> If Stage "A" requires signal from Stage "B", it will furnish Stage "B" plus 28 volts and "B" in turn will switch the plus 28 volts and return the switched signal to Stage "A". The plus 28 volts for signal transmission need only be furnished one time for a set of signals between stages. This method of transmitting signals allows the power in each stage to be completely independent and eliminates the problem of current transfer in the negative side of power systems.

8.4.2. Control System Interface. A trunk line or cable is carried through all stages of the C-1 vehicle. Each stage contains a switching device which ties its control system to the trunk line. Circuitry will be provided in the forward instrument unit making it possible for two stages to be electrically connected into the trunk at the same time. The switching device used will be from the source indicated by, and controlled by, the Douglas Aircraft Corporation Drawing No. 7866039. The excitation voltage for the engine position potentiometer, both for control and measuring instrumentation, is supplied from the forward instrument unit.

8.4.3 Engine, Pneumatic and Electronic Package. It is the combined responsibility of the stage contractor and engine contractor to design the most suitable circuits to incorporate the engine sequence into the stage sequence. This has been done in the S-I stage with the H-1 engine, as well as in the S-IV stage with the Pratt and Whitney RL10-A-3 engine.

8.4.4. Instrumentation System. The description of this area appears elsewhere in this report.

8.4.5. Command Destruct and Ordnance Systems. The Exploding Bridge Wire (EBW) command destruct system, as shcwn in Figure 8.2, fulfills the requirements of Regulation ARMTC 80-7 and has been utilized for the Saturn vehicle. Each stage will contain a command destruct system consisting of:

(1) An omnidirectional antenna system of several antennas coupled together, feeding both command receivers

(2) Two command receivers, GFE, operating on different batteries

(3) Two EBW firing units; one for each receiver and operating from the same battery as the associated receiver

(4) Two destruct system controllers, GFE, standardizing the control circuits for destruct

(5) Two EBW initiators; one for each firing unit

(6) An ordnance system capable of being initiated from either EBW initiator.

In addition, the system contains a GSE-controlled safe arming unit, identical in S-I and S-IV stages. An ordnance interface between adjacent stages is provided to assure destruction of the entire vehicle in all conditions. This interface shall be the same as used in all stages within the Saturn system. The contractor provides monitoring equipment necessary to assure the safety of the EBW initiation system.

8.5. GSE MONITORING EQUIPMENT

The precisely defined GSE monitoring equipment is furnished to the stage contractor. EBW pulse checkers and test sets are provided by the contractor from a MSFC-designated source by electrical GSE drawing number. The contractor provides complete GSE, in accordance with the referenced standards, required to check out his stage in all of the contractor facilities in the Staging Building at the launch site. Any check out of an individual stage, as a stage and not assembled with other stages of the C-1 vehicle, is accomplished with contractor-furnished GSE.

8.6. CHECKOUT AND LAUNCH OPERATIONS

For the Saturn C-1 vehicle these operations will be performed with contractor-furnished GSE integrated into a specified countdown sequence. This sequence is controlled by MSFC GSE to maintain overall countdown control and vehicle safety up to liftoff. The Saturn vehicle will be checked out with the general purpose digital computer complex.

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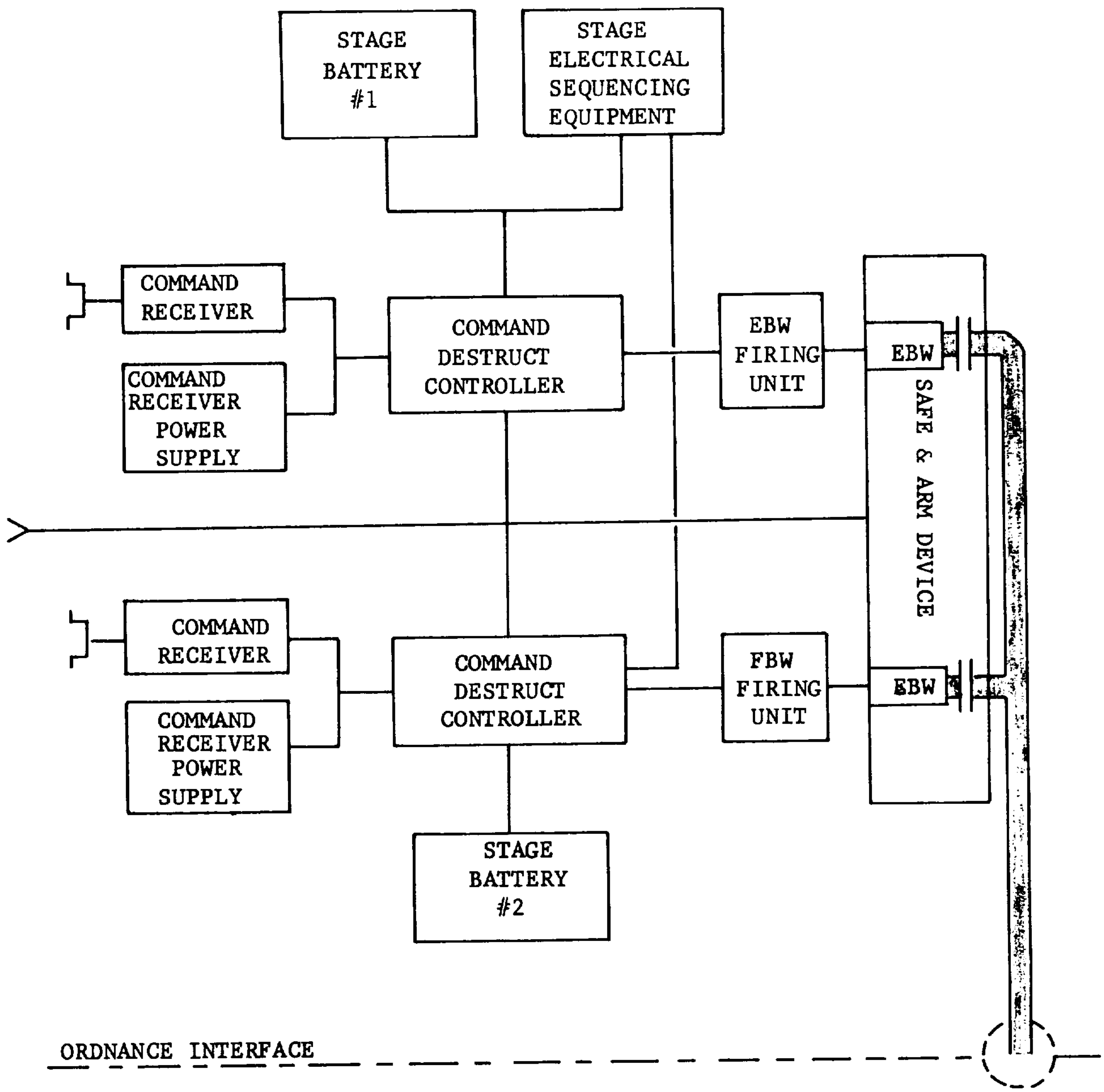

FIG. 8.2 BLOCK DIAGRAM OF EXPLODING BRIDGE WIRE (EBW) COMMAND DESTRUCT AND ORDNANCE SYSTEM

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This complex will be provided by MSFC and operated by MSFC personnel. The design concept of checkout and launch operations shall be specified by MSFC. Contractor personnel shall assist MSFC as required during the design phase to assure a complete system checkout of all their stage functions by the MSFC-furnished hardware. The digital computer complex will be programmed to perform the complete electrical systems test such as Simulated Flight Tests and Launch Sequencing for the entire Saturn C-1 space vehicle system.

The Saturn electronic ground support equipment is part of integrated checkout and control system for the many facets of launch site activity. It is capable of real-time monitoring and testing of hundreds of analog and digital signals, monitoring the status of discrete fuctions, communicating with the vehicle guidance and control system, controlling and monitoring fueling operations, certifying that all channels of telemetry are working, etc. In addition, the ground support equipment is self-checking and designed from state-of-the-art equipment.

The GSE as described above is designed around a general purpose digital computer. The computer system gives the speed and precision required yet retains reliability. It has flexibility of program and input/output equipment, and is a commercially available, medium size, solid state device. The computer which has been selected as the prototype is the RCA-110 computer. Input/output equipment has been designed to meet the needs of the Saturn system.

The first computer and its associated peripheral equipment will be delivered to MSFC in July 1961. At that time the system will be checked out and the simulation of the launch site activities will be performed at MSFC.

The capacity of the RCA-110 computer can be easily expanded, if required. It can be used for many monitoring and checkout functions in the entire Saturn space vehicle. The utilization of digital electronic checkout techniques will allow a compression of the time required for the launch checkout and, therefore, an increase in the allowable launch rate for a given launch facility.

8.7. STAGE SUBSTITUTES

The contractors will design and fabricate the stage substitutes for their own stages. The concept and philosophy will be defined by MSFC. The substitutes are required during checkout of other stages to assure adjacent stage electrical compatibility with the stage to be checked out. Requirements exist for the following:

(1) Substitutes for stage contractors' plant operations

(2) Substitutes for stage contractors' test operations

(3) Substitutes for stage contractors' Cape Canaveral Staging Building operations

(4) Substitutes for MSFC operations in Huntsville

(5) Substitutes for MSFC Cape Canaveral operations

The number of these substitutes necessary for the various checkouts of a stage is worked out jointly between MSFC and the contractors to maintain compatibility of all checkouts in the various areas and to guarantee proper function of the stages prior to mating.

8.8. POWER SUPPLY AND DISTRIBUTOR SYSTEM

The stage contractor defines the GSE power requirements and design, utilizing existing power supplies at Cape Canaveral where feasible. Electrical power available at the Cape facilities is described below:

8.8.1. 28 Volt dc Power

Source: Aircraft Energizer, 500 amp dc (manufactured by General Electric Company)

Input Power: 220/440 volts, 60 cps, 3-phase

RPM: 3600 (NO-load speed)

Generator Output: 500 amp continuous
1000 amp intermittent

Voltage Regulating Range: 14-35, 22-45 (accomplished in two steps)

Voltage Regulation: ± 1 from NO-load to FULL-load

Ripple: Wave analysis of ripple voltage content with MAGAMP regulator. Voltage 28.381 volts.

NO LOAD

Freq. kc	Voltage Millivolts (rms)
16.10	45
15.65	20
15.20	20
14.75	20
14.30	50*
13.35	40
11.60	50*
10.20	40
9.80	30
8.90	20
8.00	60
6.25	40
5.40	30
4.50	20
4.30	60*
3.60	50
2.20	400*
1.90	40
1.80	70*
1.20	160*
106 Cycles	30
52 Cycles	2-22 (Varying)

250 AMP LOAD

Freq. kc	Voltage Millivolts (rms)
10.90	25
8.10	20
5.40	25
3.65	35
3.25	25
2.72	160*
1.80	35
1.10	110*
800 Cycles	90*
360 Cycles	14
240 Cycles	16
120 Cycles	45
65 Cycles	25

* Indicates values greater than 50 mv.
Values less than 10 mv were neglected.

Drift Rate: 50 Millivolts for 8 hours at 80°F ambient. Transient response time when 200% inductive load is applied equals 750 millisecond to return within ± 2% of steady-state voltage. Nominal voltage will drop to 19 volts minimum during transient. Normal dc load center bus voltage is 28 volts ± 1 volt dc

Radio-noise Suppression: See MIL-I-6181B

8.8.2. <u>115 Volt, 400 Cycle, 3-Phase Power</u>

Source: 1800 VA Inverter; Ford Instrument Company

Input Power: 25 to 29 volt dc

Output: 1800 VA, 400 cycle, 115 volts, 3 phase

Voltage Regulation: 115 volts ± .5 volts per phase from NO-load to FULL-load. This voltage shall be the average of the three rms line voltages.

Phase Balance: After a fifteen minute warm-up, the rms voltage of each of the three output phases shall be within 1.5 volts rms of each other while the unit is operating with NO-load, and any constant input voltage, in the range of 29 to 25 volts dc. For unbalanced loads the voltage variation between phases will increase, depending on the unbalance condition.

8.8.3. <u>208 Volt, 400 Cycle, 3-Phase</u>

Source: 45 kw, Motor Generator Set: J.R. Hollingsworth

Input Power: 400 volts, 60 cps, 3 phase

RPM: 1200 (synchronous)

Generator Output: 45 kw, 400 cycles, 3 phase, WYE connected, 208 volts line-to-line. Current equals 156 amps at FULL-load.

Voltage Adjust Range: 208 volts ± 2% from NO-load to FULL-load.

Frequency Regulation: Synchronous machine.

8.9. EXPLOSION PROOFING

MSFC Specification, Drawing No. 10M01071, concerning electrical equipment in hazardous areas is used as a reference. The stage contractor provides a list of potentially hazardous operations and the measures proposed to alleviate or eliminate any possibility of explosions from such operations. Particular reference is made to such items of GSE and other components which must operate in a potentially explosive atmosphere.

8.10. REQUIRED ELECTRICAL DOCUMENTATION

The contractor will provide sufficient electrical systems documentation to permit adequate review and concurrence by MSFC. The documentation includes, but is not limited to, the following:

<u>(1) GSE</u>

a. Elementary functional schematics of the system

b. Cable interconnect diagram of system

c. In addition to all panel layouts, the contractor supplies layouts of general packaging arrangement and interconnect system.

d. Breakdown of equipment into racks, panel assemblies, and units

e. Location of this equipment in the various operational complexes, i.e., VLF 37, MSFC/Contractor facilities

f. Functional description of each of the above assemblies or units as to purpose, including list of functions monitored or operated on

g. Definitions of any proposed items of hardware not listed in NASA Standards (for GSE, XDOD-STD-0002). MSFC will be notified immediately by the contractor of any desire to deviate prior to procurement action

h. Interface definitions.

(2) Space Vehicles

a. Functional schematics

b. Cable interconnect diagrams

c. Logic sequence drawings

d. Mechanical layout drawings of electrical assemblies

e. Interface definitions

Information or documentation listed above must be reviewed for approval at each of the three phases of development listed below:

(1) Design concept drawings or sketches prior to start of detailed design

(2) Detail drawings parallel with the release for manufacture

(3) Detail drawings to accurately define vehicles and GSE upon completion of assembly will be transmitted to MSFC parallel with start of all systems checkout.

Convenient access to documentation and equipment must be provided to MSFC representatives at contractor installations.

8.11. DOCUMENTATION, FORMAT AND ELECTRICAL SYMBOLS

The documentation required shall be in accordance with MSFC Drawing No. 1044336.

8.12. APOLLO "A"/SATURN C-1 ELECTRICAL SYSTEMS INTERGRATION

Electrically, the Saturn C-1 system is defined as an intergrated system due to cooperative efforts between contractors and MSFC to achieve and maintain standardization of the system. Apollo "A" will be intergrated into the established overall vehicle complex as well as the launch facilities at AFMTC. The Electrical Systems Design Integration Working Group established under the Saturn Vehicle System Management Plan should be responsible for the integration of Apollo electrical systems into the C-1 system. The Working Group's membership now includes representatives of Saturn system contractors as well as MSFC design and launch operations personnel and would, accordingly, include representatives of the Apollo contractors. Specific responsibilities of the group should include:

(1) The establishment of electrical standards and system parameters to be applied in the design of the Apollo stage

(2) The coordination of MSFC and contractor design activities which are interrelated

(3) The detection of problem areas in electrical systems integration and the determination of corrective courses of action to be taken

(4) To act as consultants to the Saturn systems board in all matters affecting C-1 system electrical integration.

SECTION 9. (C) ATMOSPHERIC ENVIRONMENT

9.1. PHYSICAL PROPERTIES

For determination of the structural design, trajectory, control requirements and aerodynamic heating, the atmospheric characteristics (pressure, ambient temperature, density, etc.) are based on NASA TN D-595 and MTP-AERO-61-11.

References

1. NASA TN D-595, dated March 1961, "A Reference Atmosphere for Patrick AFB, Florida."

2. MTP-AERO-61-11, dated February 23, 1961, "Range of Density Variability from Surface to 120 Km Altitude."

9.2. WIND ENVIRONMENT - ALTITUDE VARIATION OF WIND SPEED AND WIND SHEAR.

The wind data used for the design of the Saturn C-1/Apollo "A" is given in the following paragraphs. These data are specifically applicable for flights to be conducted from Cape Canaveral, Florida.

9.2.1. Idealized Wind Profile Envelope. This profile, Figure 9.1, provides an envelope about the wind speeds which are not expected to be exceeded on an average of 95% of the time during the windiest monthly period. The windiest monthly period is defined as the month having the highest average wind speeds in the 10-14 km altitude region. Since the various monthly periods that the 75 m/sec wind speed is exceeded has also been included for reference purposes.

9.2.2. Wind Shear Spectrum as Function of Altitude. Figure 9.2 provides information on the vertical wind shears (partial derivative of the wind velocity profile with respect to altitude) associated with differentials in altitude from 100 m to 5000 m over the altitude range from 1 km to 80 km. These data should be combined with the idealized wind profile envelope data to establish synthetic wind build-up profiles leading to the 95% probability level wind magnitudes at the various altitudes.

9.2.3. Wind Speed Change Spectrum as Function of Altitude. Figure 9.3 shows the conversion of the wind shear spectrum to the magnitude of the wind speed change associated with the scale-of-distances.

9.2.4. Selected Wind Shear Spectrums. Figure 9.4 provides a selection of the major wind shear versus scale-of-distance values for comparative purposes.

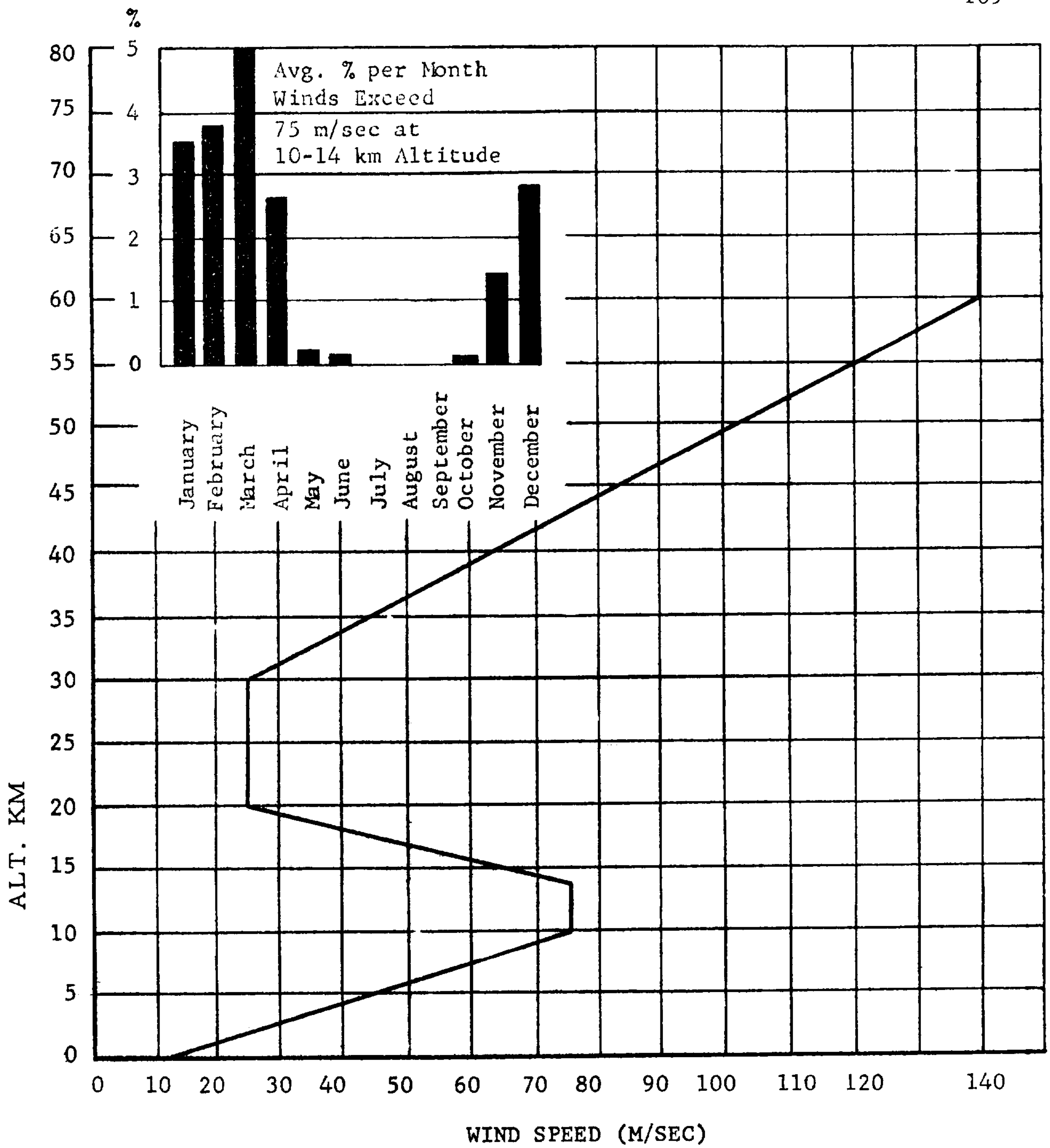

FIG. 9.1 IDEALIZED WIND PROFILE ENVELOPE (95% PROBABILITY LEVEL) FOR CAPE CANAVERAL, FLORIDA

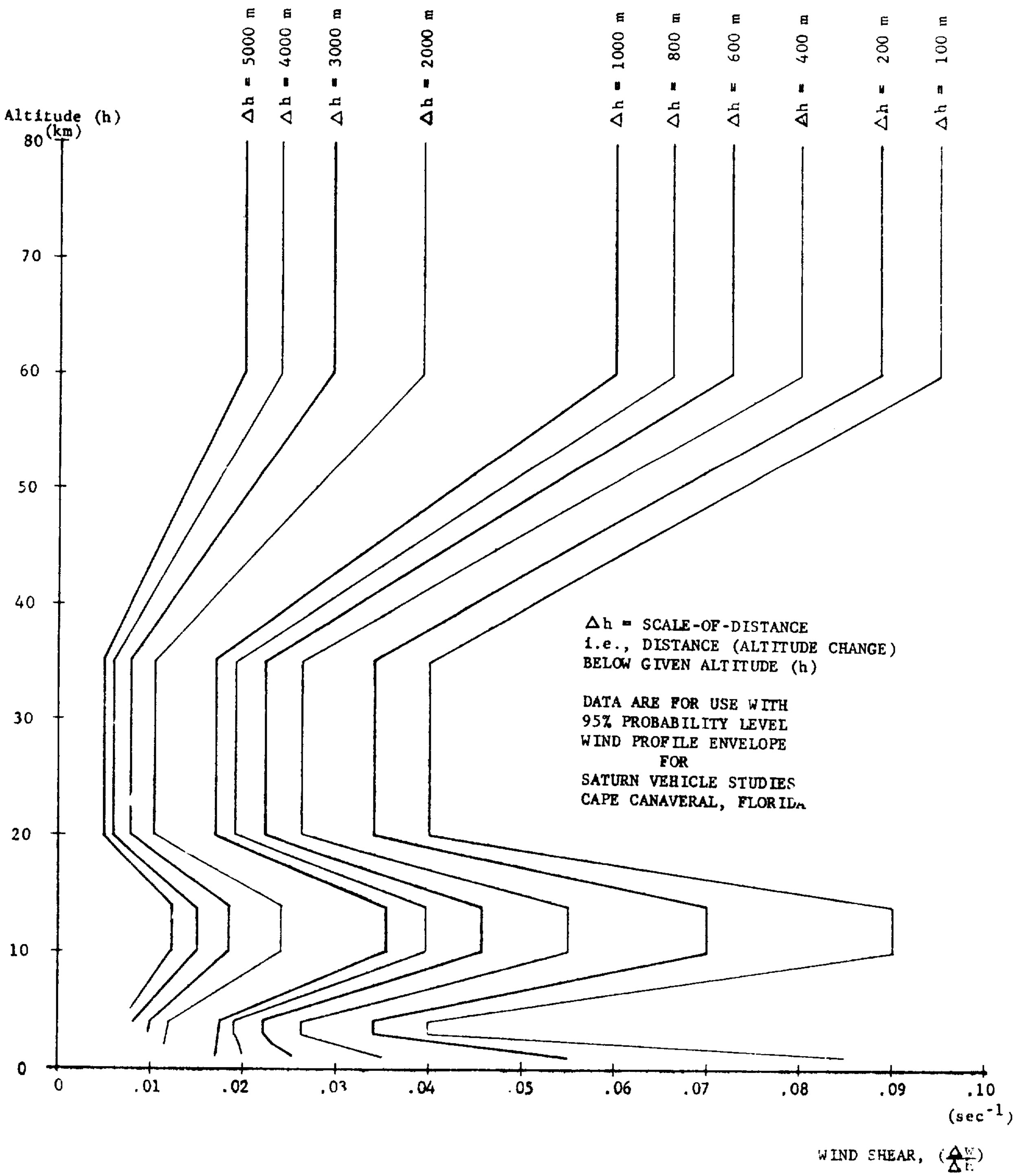

FIG. 9.2 VERTICAL WIND SHEAR SPECTRUM AS A FUNCTION OF ALTITUDE (1-80km) FOR 100m TO 5000m SCALE-OF-DISTANCE

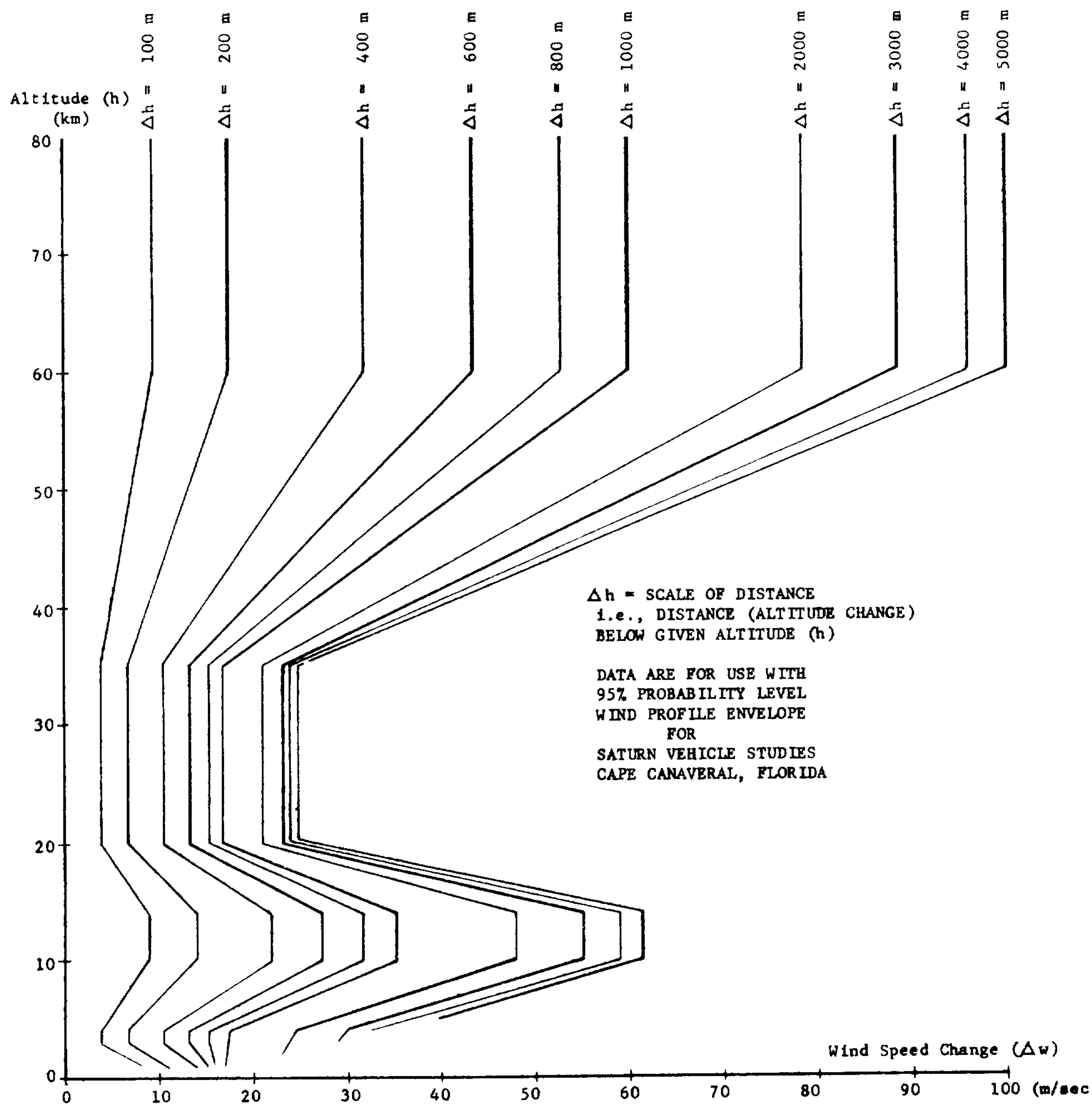

FIG. 9.3 VERTICAL WIND SPEED CHANGE SPECTRUM AS A FUNCTION OF ALTITUDE (1-80km) FOR 100m TO 5000m SCALE-OF-DISTANCE

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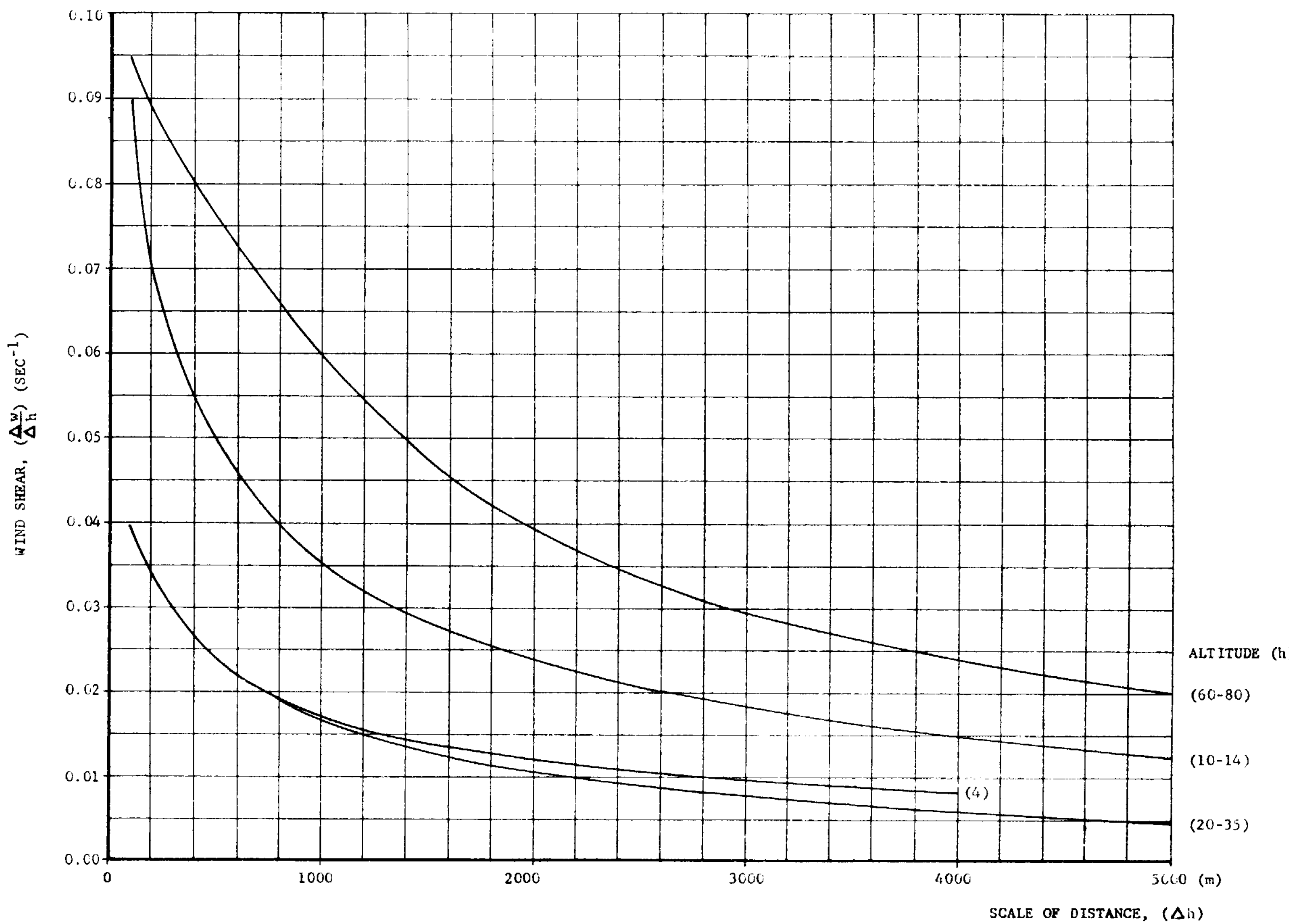

FIG. 9.4 SELECTED VERTICAL WIND SHEAR SPECTRUMS FOR USE WITH 95% PROBABILITY LEVEL WIND PROFILE ENVELOPE, CAPE CANAVERAL, FLORIDA

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9.2.5. Selected Wind Speed Change Spectrum. As shown in Figure 9.5, a conversion is made of the selected wind shear spectrums to enable easy determination of the wind speed change associated with the various scale-of-distances.

9.2.6. Summary of Wind Data. See Summary Table 9.1.

9.3. WIND ENVIRONMENT - LAUNCH SITE WIND VELOCITY CRITERIA

The expected steady state wind speeds at altitudes from 2 to 120 meters above ground level is given in Figure 9.6 for Cape Canaveral, Florida, for the condition of 10 meters/sec at 10 feet. To obtain the total expected peak wind speed (including gust), the steady state wind speed profile must be multiplied by a gust factor of 1.4.

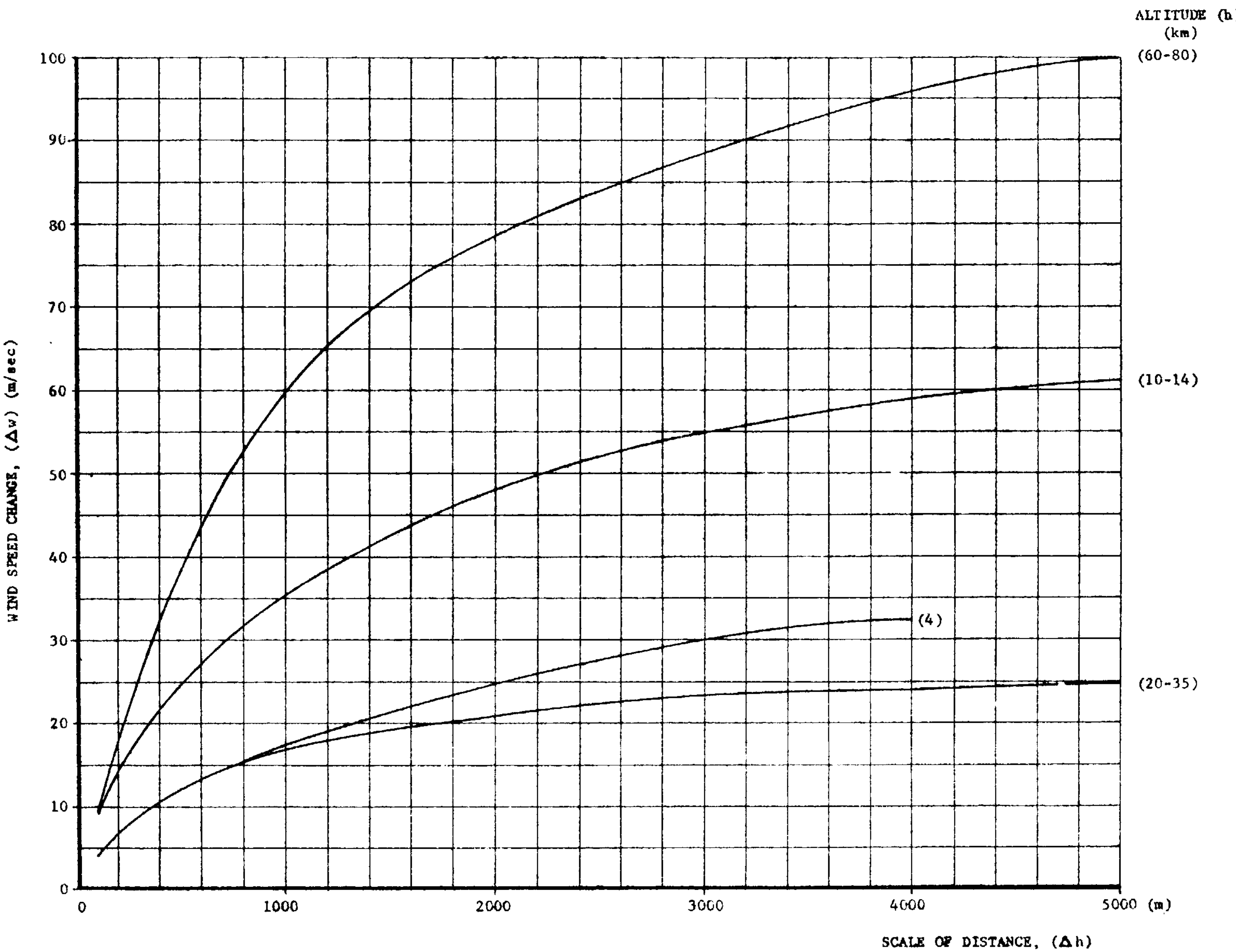

FIG. 9.5 SELECTED VERTICAL WIND SPEED CHANGE SPECTRUMS FOR USE WITH 95% PROBABILITY LEVEL WIND PROFILE ENVELOPE, CAPE CANAVERAL, FLORIDA

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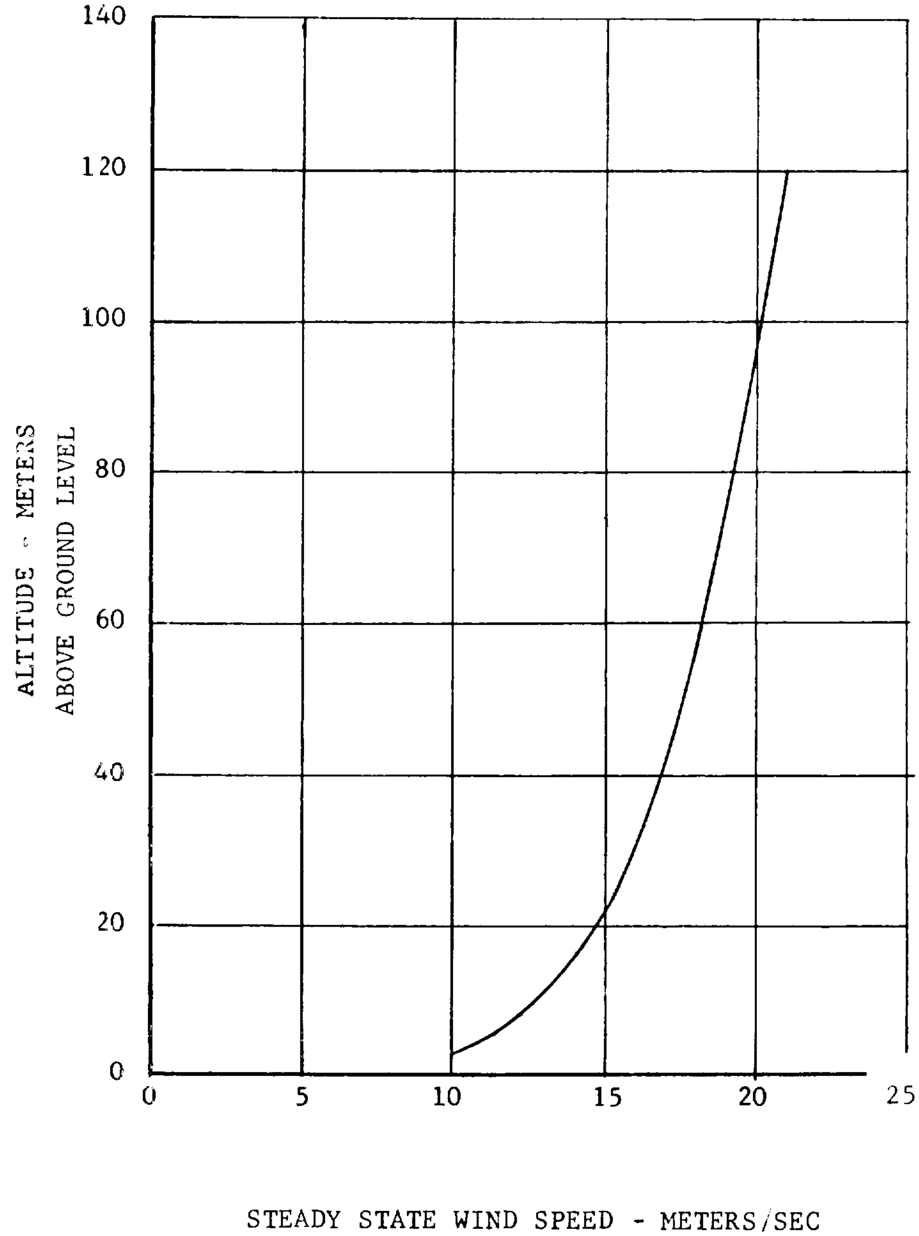

GUST FACTOR: - To obtain the total expected peak wind speed (including gust), the above steady state wind speed profile must be multiplied by a gust factor of 1.4 over the altitude range of 3 to 120 meters.

FIG. 9.6 LOW LEVEL WIND SPEED PROFILE FOR CAPE CANAVERAL, FLORIDA

Table 9.1

SUMMARY OF WIND DATA

A summary table of wind speed, wind shear, and wind speed change data as presented in Figures 9.1, 9.2, and 9.3.

Idealized 95% Probability Level Wind Profile Envelope (Fig. 9.1)

Alt (km)	Wind Speed (m sec)
0	10
10	75
14	75
20	25
30	25
60	140
80	140

Wind Shear (sec-1) Spectrum (Fig. 9.2)

Alt	Scale-of-Distance (m)									
(km)	5000	4000	3000	2000	1000	800	600	400	200	100
1	--	--	--	--	.0170	.0200	.0253	.0350	.0550	.0850
2	--	--	--	.0116	--	--	--	--	--	--
3	--	--	.0097	--	--	.0190	.0222	.0263	.0340	.0400
4	--	.0081	.0100	.0122	.0175	.0190	.0222	.0263	.0340	.0400
5	.0079	--	--	--	--	--	--	--	--	--
10	.0123	.0148	.0184	.0240	.0353	.0396	.0457	.0550	.0700	.0900
14	.0123	.0148	.0184	.0240	.0353	.0396	.0457	.0550	.0700	.0900
20	.0050	.0060	.0078	.0105	.0170	.0191	.0222	.0263	.0340	.0400
35	.0050	.0060	.0078	.0105	.0170	.0191	.0222	.0263	.0340	.0400
60	.0200	.0240	.0295	.0393	.0600	.0662	.0727	.0800	.0885	.0950
80	.0200	.0240	.0295	.0393	.0600	.0662	.0727	.0800	.0885	.0950

Wind Speed Change (m/sec) Spectrum (Fig. 9.3)

Alt	Scale-of-Distance (m)									
(km)	5000	4000	3000	2000	1000	800	600	400	200	100
1	--	--	--	--	17.0	16.0	15.2	14.0	11.0	8.5
2	--	--	--	23.2	--	--	--	--	--	--
3	--	--	29.1	--	--	15.2	13.3	10.5	6.8	4.0
4	--	32.4	30.0	24.4	17.5	15.2	13.3	10.5	6.8	4.0
5	39.5	--	--	--	--	--	--	--	--	--
10	61.5	59.2	55.2	48.0	35.3	31.7	27.4	22.0	14.0	9.0
14	61.5	59.2	55.2	48.0	35.3	31.7	27.4	22.0	14.0	9.0
20	25.0	24.0	23.4	21.0	17.0	15.3	13.3	10.5	6.8	4.0
35	25.0	24.0	23.4	21.0	17.0	15.3	13.3	10.5	6.8	4.0
60	100.0	96.0	88.5	78.6	60.0	53.0	43.6	32.0	17.7	9.5
80	100.0	96.0	88.5	78.6	60.0	53.0	43.6	32.0	17.7	9.5

SECTION 10. (C) TRAJECTORIES AND ORBITS

10.1. TRAJECTORY OPTIMIZATION

The primary mission of the C-1 class of vehicles is to carry a payload into circular orbit. The height of the orbit is determined by the performance capability of the vehicle and by the desired orbital lifetime of the payload.

Two modes of ascent are considered;

(1) Directly ascending into the desired orbital altitude and

(2) Parking in an intermediate orbit of low altitude and using a Hohmann transfer to orbits above the parking orbit.

The advantage of the Hohmann transfer method is that more weight can be carried into high orbital altitudes. This transfer method is superior by 5,700 pounds in a 300 nautical mile orbit. The full gain in weight could not be utilized as payload since the transfer method requires more guidance and control equipment weight.

10.1.1. Engine-Out Capability. The capability of both stages to continue flight with one engine failing enhances considerably the mission reliability of the vehicle. The reduced performance, described as "reduced equivalent payload", would in most instances result in an alternate mission, e.g., lower orbit, or less space maneuvering.

As the engine-out trajectories are less severe than normal flight with respect to dynamic pressure and loads, only the "equivalent payload" losses will be presented in addition to the detailed data of the normal trajectories.

10.1.2. Tilt Program. Programming the tilt angle is the principal means for shaping the first-stage flight path. First-stage tilting is achieved by introducing a small angle-of-attack for a short period of time early in the flight. Zero lift is then continued to ignition of the second stage, which occurs in near vacuum conditions. Second-stage tilt programs have been determined by calculus of variation methods to optimize payload into different orbit altitudes.

10.1.3. Aerodynamic Heating. Aerodynamic heating does not impose a serious trajectory limitation, although some heat protection is required because of the low injection point. Further discussion of heating appears in Section 13.

10.1.4. <u>The Separation Procedure</u>. The standard separation procedure for Saturn C-1, i.e., cutting off the four inner engines 6 seconds before cutting off the four outer engines, was used in the trajectory calculations. Ignition of the second stage is assumed to occur with cutoff of the first stage.

Table 10.1

SUMMARY OF ORBITAL CAPABILITY

Orbital Altitude (N. M.)	Mode of Ascent	Orbital Weight (lbs) 8 Engine First Stage	7 Engine First Stage
100	Direct	40,700	37,270
100	Hohmann	40,700	37,270
200	Direct	37,300	33,800
200	Hohmann	40,000	36,270
300	Direct	32,870	29,670
300	Hohmann	39,000	35,570

Performance curves for the Saturn C-1 booster configuration and design payloads are presented in Figures 10.1 and 10.2 for various injection altitudes for both a direct ascent and the Hohmann transfer.

Tables 10.2 through 10.9 give the flight mechanical data for the powered phase of the 100 and 300 nautical mile orbits. Both the seven- and eight-engine S-I boost phase are given.

10.2. ABORT FROM SATURN C-1 TRAJECTORIES

In order to make the spacecraft contractor cognizant of the influence of pilot abort considerations on Saturn trajectory shaping, a brief resume of studies conducted by the Aeroballistics Division is presented,[1,2,3,4]. Off the pad, abort problems and abort considerations,

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1. R. F. Hoelker, R. Teague, and B.J. Lisle, <u>Studies of Pilot Abort From Saturn C-2 Trajectories</u>, MSFC Report MTP-AERO-61-22, March 21, 1961.
2. R. F. Hoelker, <u>Satellite Re-entry, Flight Mechanical Discussion</u>, ABMA Report DA-TN-82-58, February 6, 1958.
3. L. L. McNair, <u>Satellite Re-entry, Dive Studies for Flights with Constant Body Geometry and Varied Re-entry Conditions</u>, ABMA Report DA-TN-36-58, July 10, 1958.
4. L. L. McNair, <u>Satellite Re-entry Dive with Constant Lift Program</u>, ABMA Report DA-TN-41-58, July 22, 1958.

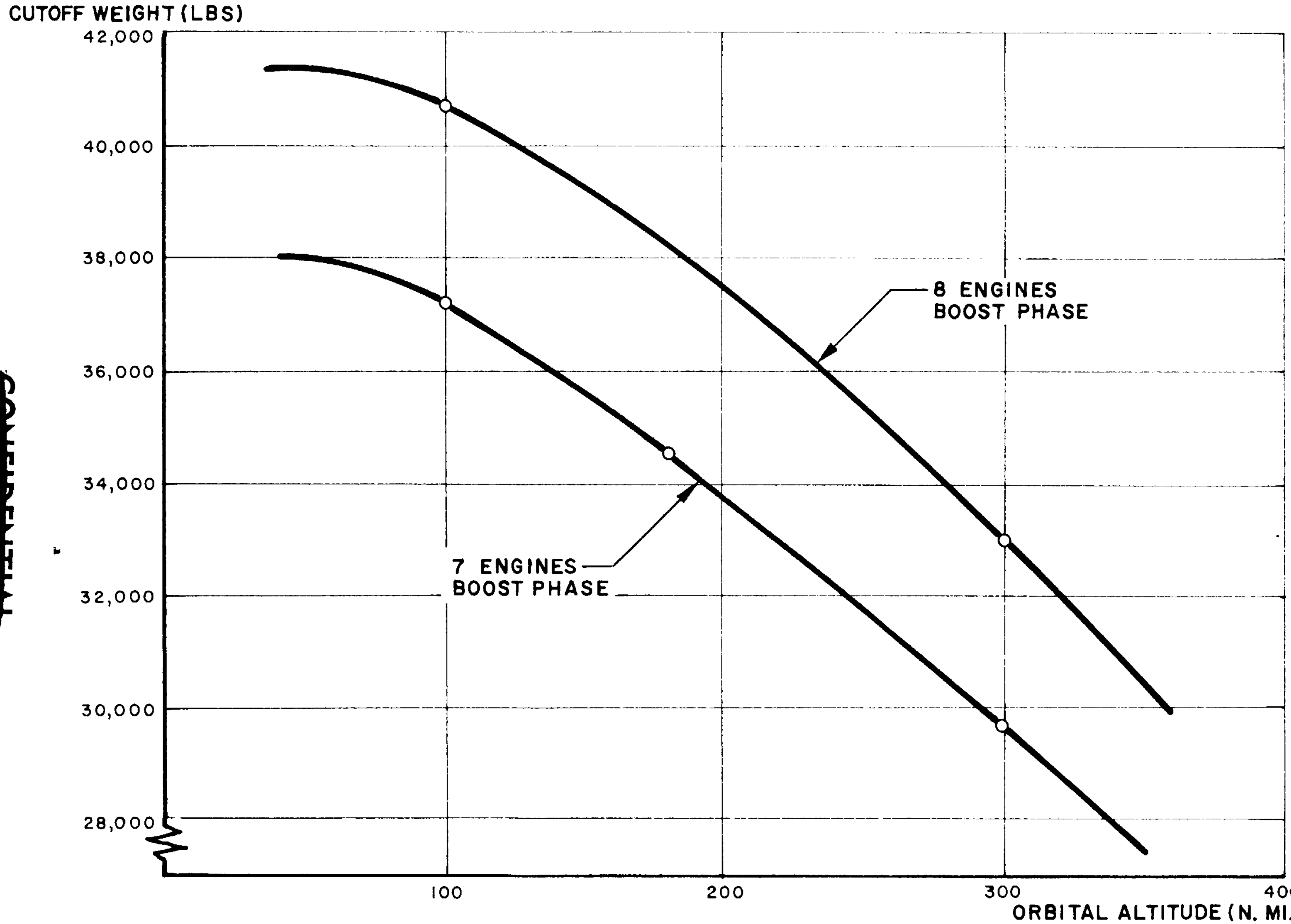

FIG. 10.1 ORBITAL CAPABILITIES OF SATURN C-1 FOR DIRECT ASCENT

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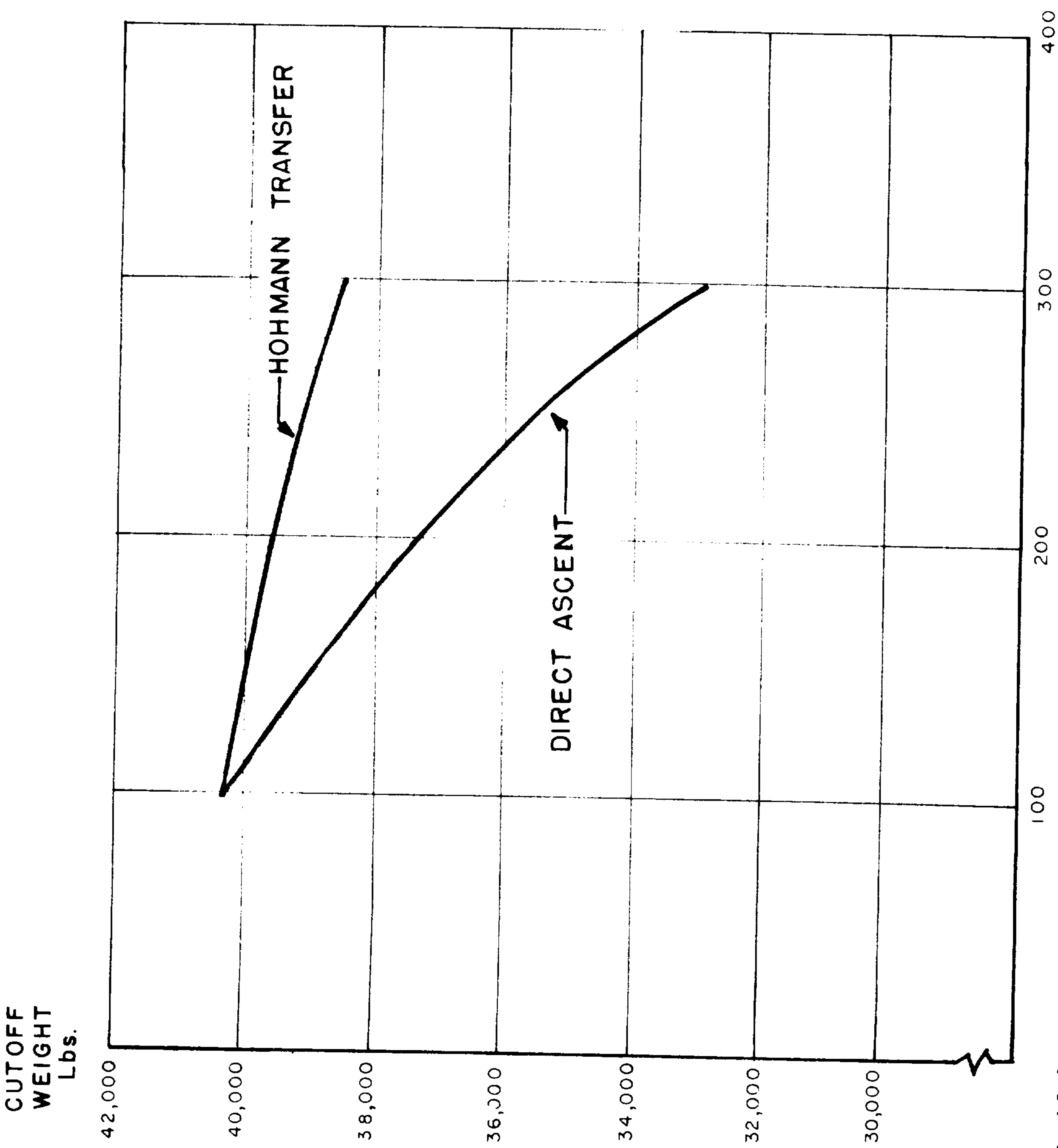

FIG. 10.2 ORBITAL CAPABILITIES OF SATURN C-1 FOR DIFFERENT MODES OF ASCENT.

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during the S-I propelled phase with its characteristic - high dynamic pressure, are not studie , since they are similar to problems encountered during the Mercury program. The investigation is based on Saturn C-1 trajectory profiles given in Figure 10.3. These trajectories are optimized for the largest payload for each combination of boundary conditions chosen. These are the altitude at injection, horizontal path angle at injection, and circular velocity, locally injected. Trajectories are injected only into circular orbits at 100 and 300 nautical miles.

Though the trajectories of Figure 10.3 were derived for a rotating earth (launching due east), the abort studies presented here are, for simplicity, based on a stationary earth. The atmosphere used is the Wright Air Development Center, 1956 version.

The first ground rule which was adopted is that the pilot after abort may not be able to command a lifting program, and also that the execution of it by automatic programming might not be considered feasible. However, it is recognized that a re-entry vehicle may be designed that will trim naturally at some fixed lift-to-drag ratio. Therefore, in the following, we are concerned with ballistic flights of zero angle-of-attack and re-entry flights employing constant lifting modes. An L/D of 0.5 is used in the latter mode of re-entry. The characteristic features of the reentry flights, e.g., the deceleration peak magnitude, depend on the capsule parameters. For our studies, the load parameter ($W/c_D A$) of the magnitude of 50 lb/sq ft was used. Applying the derived results to bodies with load factors larger by a factor 3 to 6 is permissible, in view of the preliminary character of this study.

Figure 10.4 shows the resulting deceleration on the re-entry vehicle for abort from the two trajectories of Figure 10.3, utilizing both a ballistic re-entry and a re-entry mode with constant L/D = 0.5. It is assumed that a small impulse is given to separate from the Saturn S-IV, and then no further propulsion is utilized. It is seen that a constant lifting mode, utilizing an L/D = 0.5, decreases the peak deceleration by 30 percent for the 100 nautical mile trajectory, and by about 20 percent for the 300 nautical mile case. However, the deceleration reaches 29 g's in the 300 nautical mile trajectory; even employing a constant L/D =0.5. For re-entry velocities under circular velocity (7800 M/S), the peak deceleration is least for a horizontal re-entry path angle. Also, the sensitivity of peak deceleration with deviations from the optimum re-entry angle is very low for subcircular re-entry speed. The time of the impulse application can be chosen to effect, most economically, the suppression of the re-entry deceleration. By a systematic variation of velocity changes in both direction and magnitude at different altitude levels of typical abort trajectories, departing at various points from the powered phase, it was found that the application of the impulse should be made at altitudes as low as possible above the sensible atmosphere. Staying above the atmosphere for thrusting is dictated by the desire of avoiding any nominal righting effects from aerodynamic moments which would require larger attitude control torques. The altitude of impulse application may be thought of as the region between 80 and 120 km.

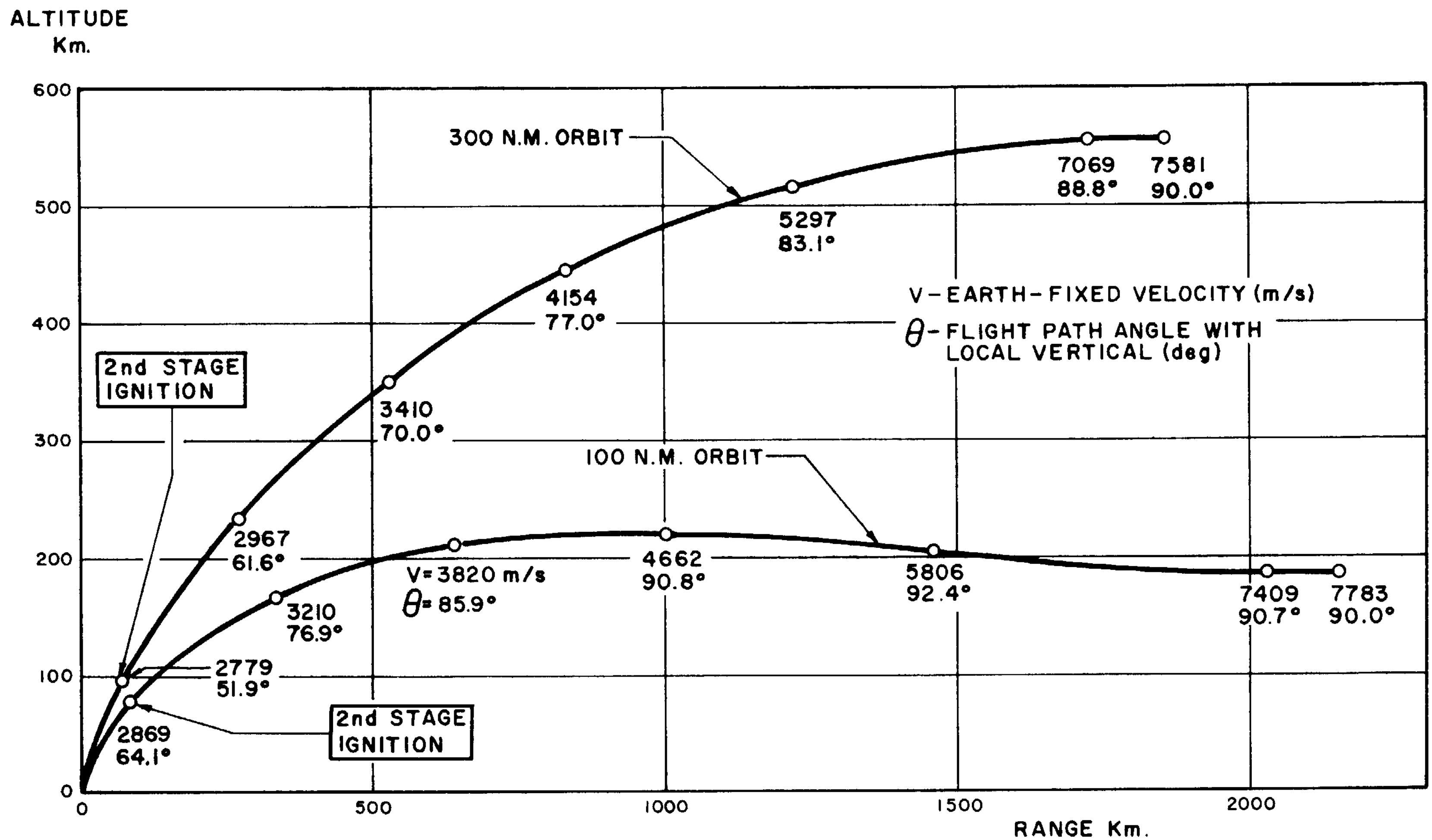

FIG. 10.3 FLIGHT HISTORIES OF C-1, 2-STAGE, SEVEN-ENGINE TRAJECTORIES FOR INJECTION INTO CIRCULAR ORBITS AT ALTITUDES OF 100 N.M. AND 300 N.M.

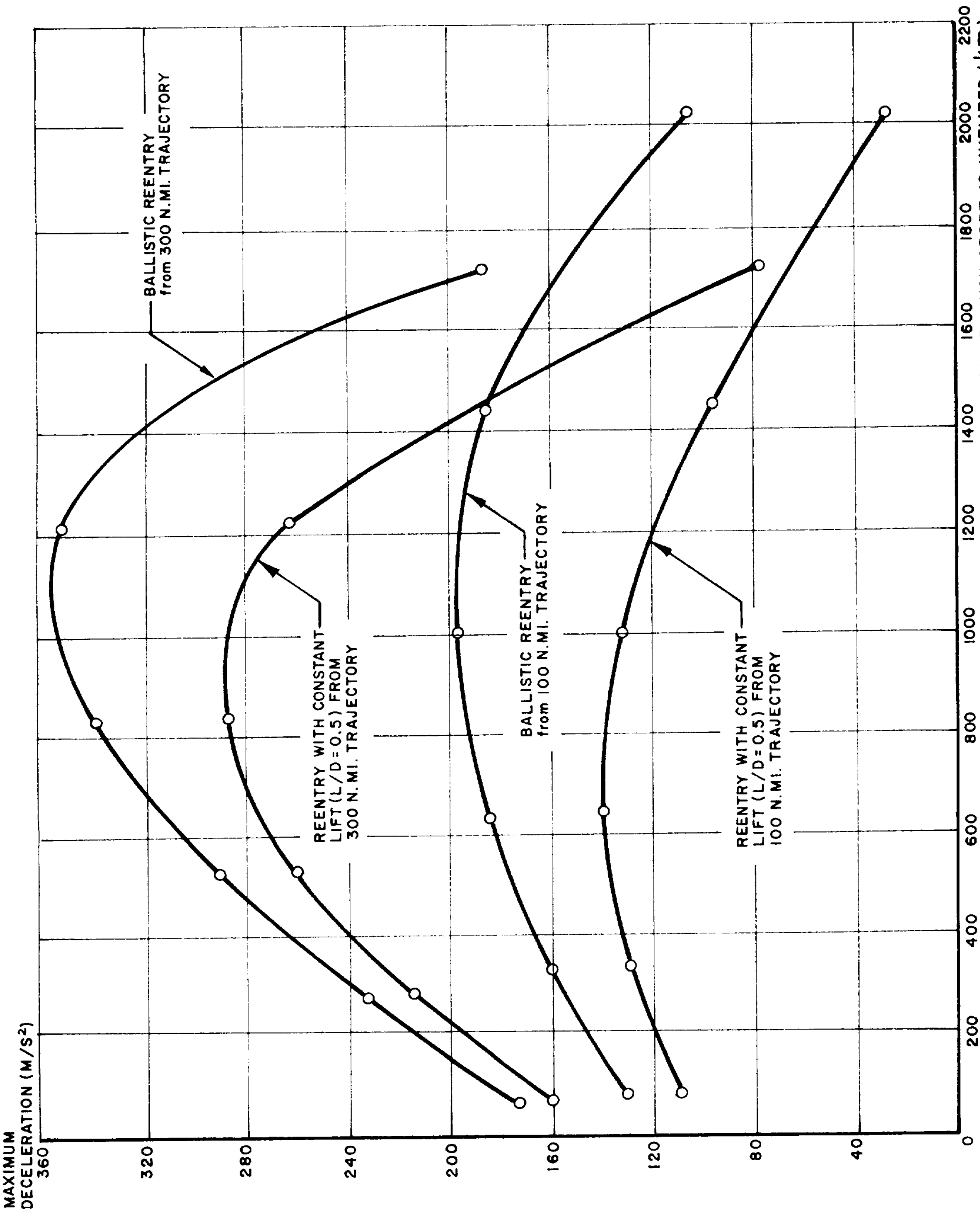

FIG. 10.4 MAXIMUM DECELERATION ENCOUNTERED DURING TWO REENTRY MODES AFTER ABORT FROM CIRCULAR INJECTION TRAJECTORIES OF 100 and 300 NAUTICAL MILES

Figures 10.5 and 10.6 show the maximum deceleration encountered during re-entry after abort from the circular 100 and 300 nautical miles trajectories, respectively, utilizing also an impulse of varied magnitude to suppress the deceleration. An altitude of 100 km was used for impulse application. Hence, for the results shown in Figures 10.5 and 10.6, an impulse of a given magnitude was applied at 100 km to turn the path angle to 90 degrees (local horizontal direction), or as near to the local horizontal as possible with that amount of impulse.

The abort point in the trajectory resulting in the highest deceleration may be called the critical abort point. The direction of impulse application for the critical phase is well defined and must be enforced. The capsule altitude (chi) was determined for the 100 and 300 nautical mile phases to be 11 and 15 degrees, respectively, measured in plane of flight with respect to a space-fixed reference that is vertically upward at launch. An essential feature of the scheme is that this angle is maintained space-fixed for all abort cases. Other schemes have also been studied.[1]

Figure 10.7 is a conversion of impulse into propellants predicated on the assumption that engines with specific impulses of 250 seconds or 420 seconds are used, and that the weight of the capsule plus the abort propulsion system varies from 11,000 pounds to 18,000 pounds before impulse application.

It is stressed that the reference to propellant weight neglecting tank or casing weight and propulsion system will not do full justice to the trade-offs caused by the flight mechanical properties.

10.3. ORBITAL LIFETIME

Preliminary orbital lifetime data for the Apollo capsule have been calculated based upon the following numerical data:

(1) A standard 1959 ARDC atmosphere was used. No diurnal, seasonal, or latitudinal variations were considered

(2) The attitude of the vehicle was assumed stabilized with zero angle-of-attack, and the drag area of the body was taken as

$A = \frac{\pi d^2}{4} = 12.017\ m^2$, where d, the diameter of the vehicle, was taken as 154 inches.

1. R. F. Hoelker, R. Teague, and B. J. Lisle, Studies of Pilot Abort From Saturn C-2 Trajectories, MSFC Report MTP-AERO-61-22, March 21, 1961.

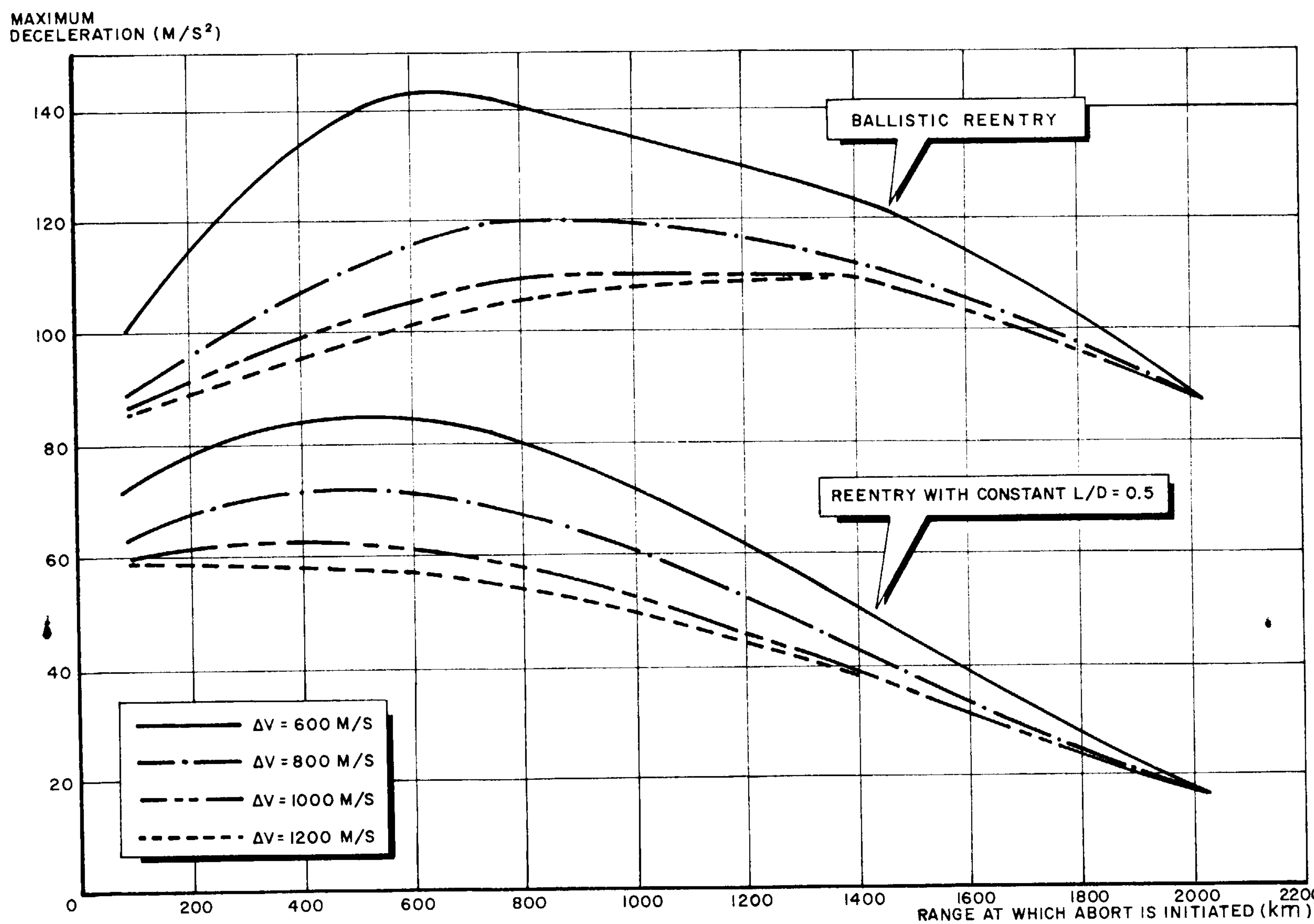

FIG. 10.5 MAXIMUM DECELERATION ENCOUNTERED DURING REENTRY AFTER ABORT FROM 100 N.M. CIRCULAR TRAJECTORY FOR TWO REENTRY MODES WITH VARIED IMPULSE APPLICATIONS

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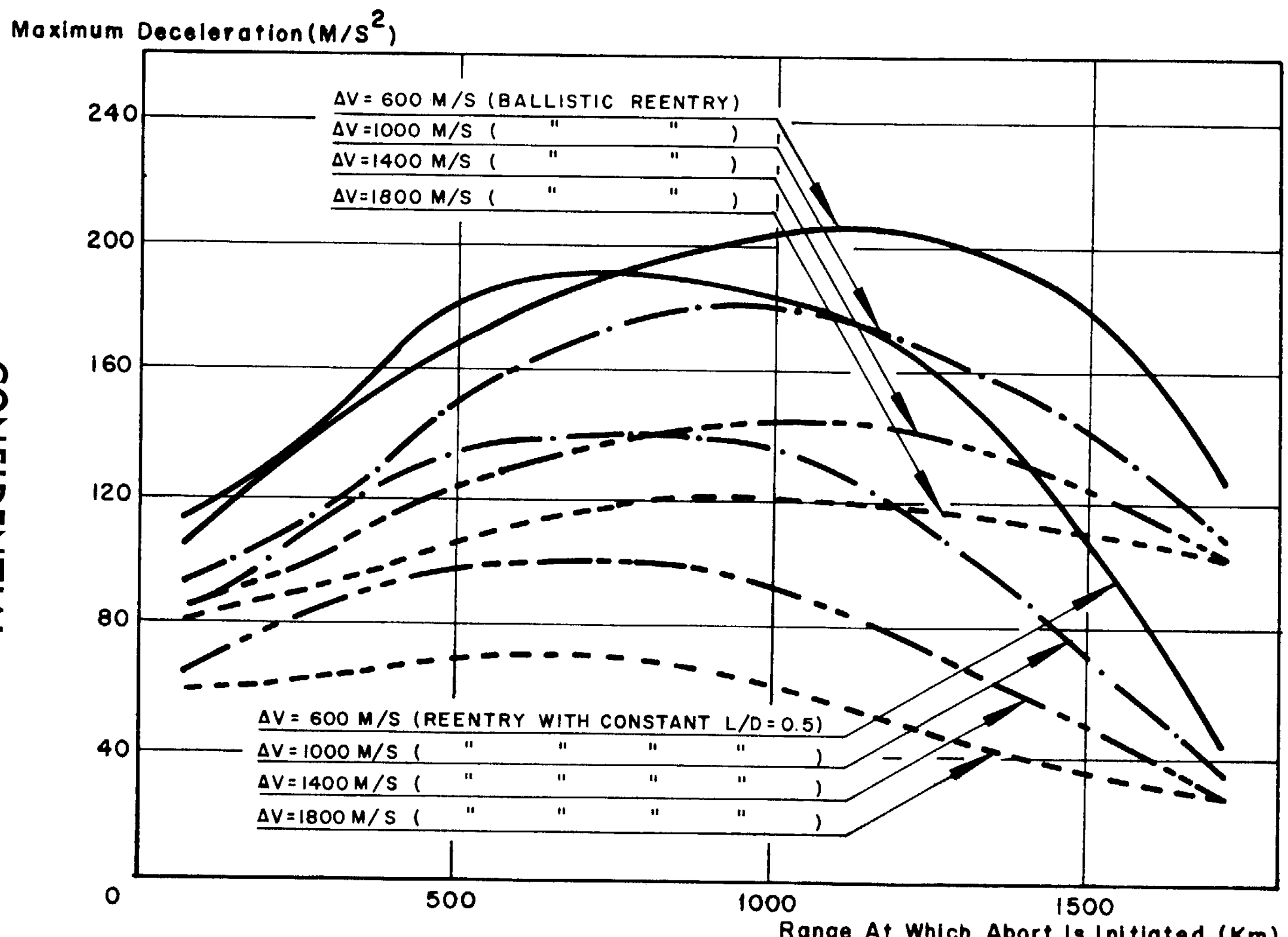

FIG. 10.6 MAXIMUM DECELERATION ENCOUNTERED DURING REENTRY AFTER ABORT FROM 300 N.M. CIRCULAR TRAJECTORY FOR TWO REENTRY MODES WITH VARIED IMPULSE APPLICATIONS

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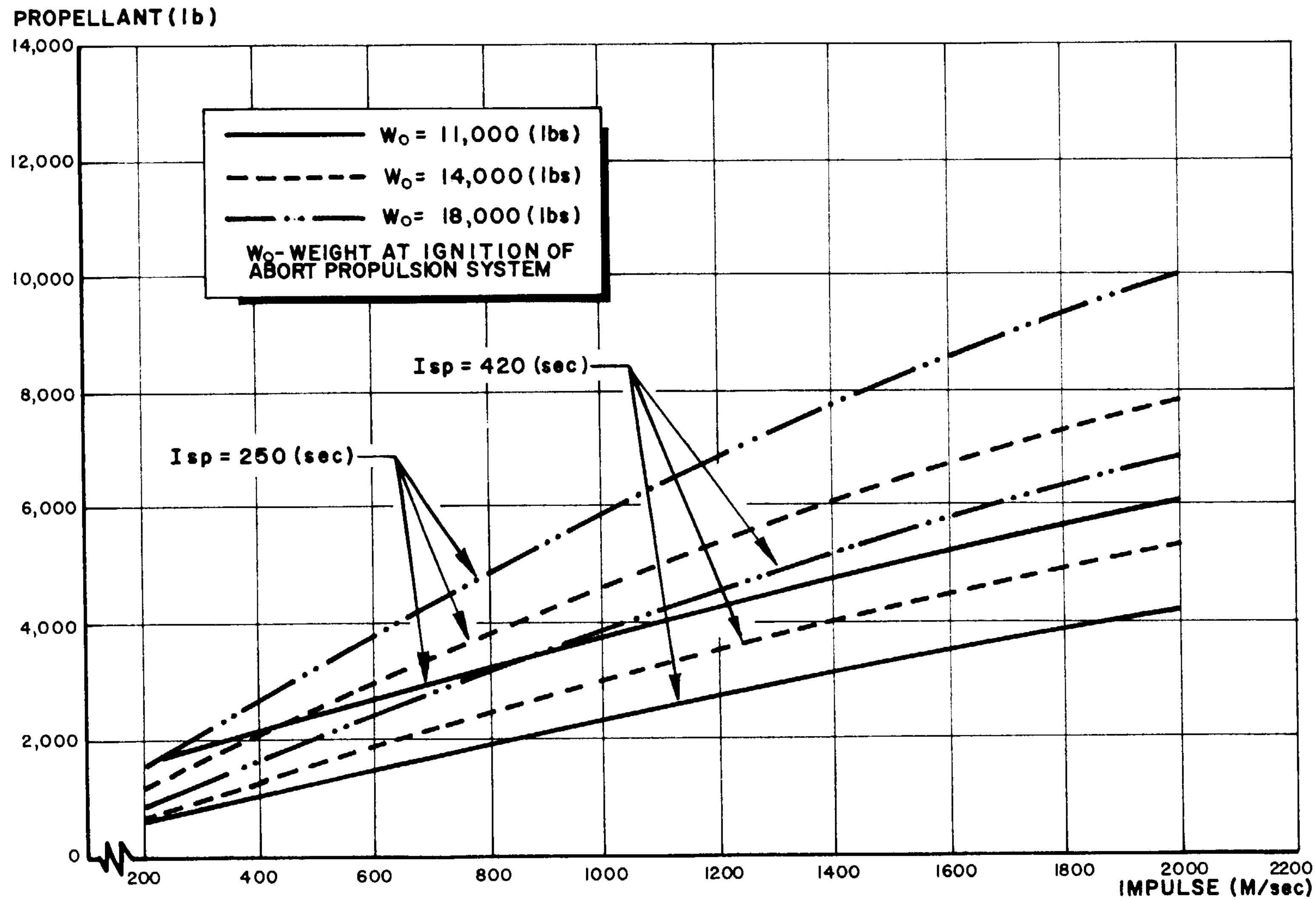

FIG. 10.7 ABORT PROPELLANT CONSUMED VERSUS IMPULSE FOR VARIED INITIAL ABORT WEIGHT and SPECIFIC IMPULSE

(3) The weight of the vehicle was assumed to be 20,000 pounds

(4) The drag coefficient of a sphere in a state of free molecular flow was assumed ($C_d = 2.0$).

(5) The vehicle orbit was assumed circular.

The method of lifetime calculation employed was an adaptation of a method due to Sterne and is described in ABMA Technical Note DA-TN-9-59. The computative process has been employed in the flight evaluation of the majority of the Juno-launched satellites, and its accuracy has also been verified on a number of additional satellites.

The results of the lifetime analysis are shown in Figures 10.8 and 10.9. Figure 10.8 shows the total orbital lifetime and the initial rate of altitude change as a function of the initial altitude. Figure 10.9 shows the altitude and rate of altitude change after two weeks in orbit as a function of initial orbital altitude.

The Apollo orbit requires a two-week stay time with no significant altitude decay. The minimum permissible altitude satisfying this condition may be required. Under the assumptions given above, the altitude satisfying these requirements may be seen from Figures 10.8 and 10.9 to lie between 130 and 150 nautical miles. A closer estimate would depend on the permitted safety factor and a detailed error analysis.

Concerning the realism of the assumptions made in deriving the data shown in the figures, the assumption of zero angle of attack throughout the orbit leads to a drag area that is too optimistic by a factor of about two. The assumption of a drag coefficient of 2.0 is pessimistic, on the other hand, by a factor of something less than 2. Diurnal and seasonal variations in atmospheric density at the altitude of interest amount to about $\pm$ 20 percent. The magnitude of latitudinal variations depends upon the inclination of the orbit, but for relatively low inclination orbits is of the same order as the diurnal variations.

The over-all error in the lifetime estimate at a given altitude due to uncertainties in the present assumptions is something less than a factor of 2. The lifetime study does not take into account expected injection errors but these errors are small due to the use of the adaptive guidance scheme. Since the shape of the lifetime-initial altitude curve is approximately constant on a semi-logarithmic scale over the altitude range of interest, an uncertainty of about 15 nautical miles exists in the initial orbital altitude necessary to meet the given requirements.

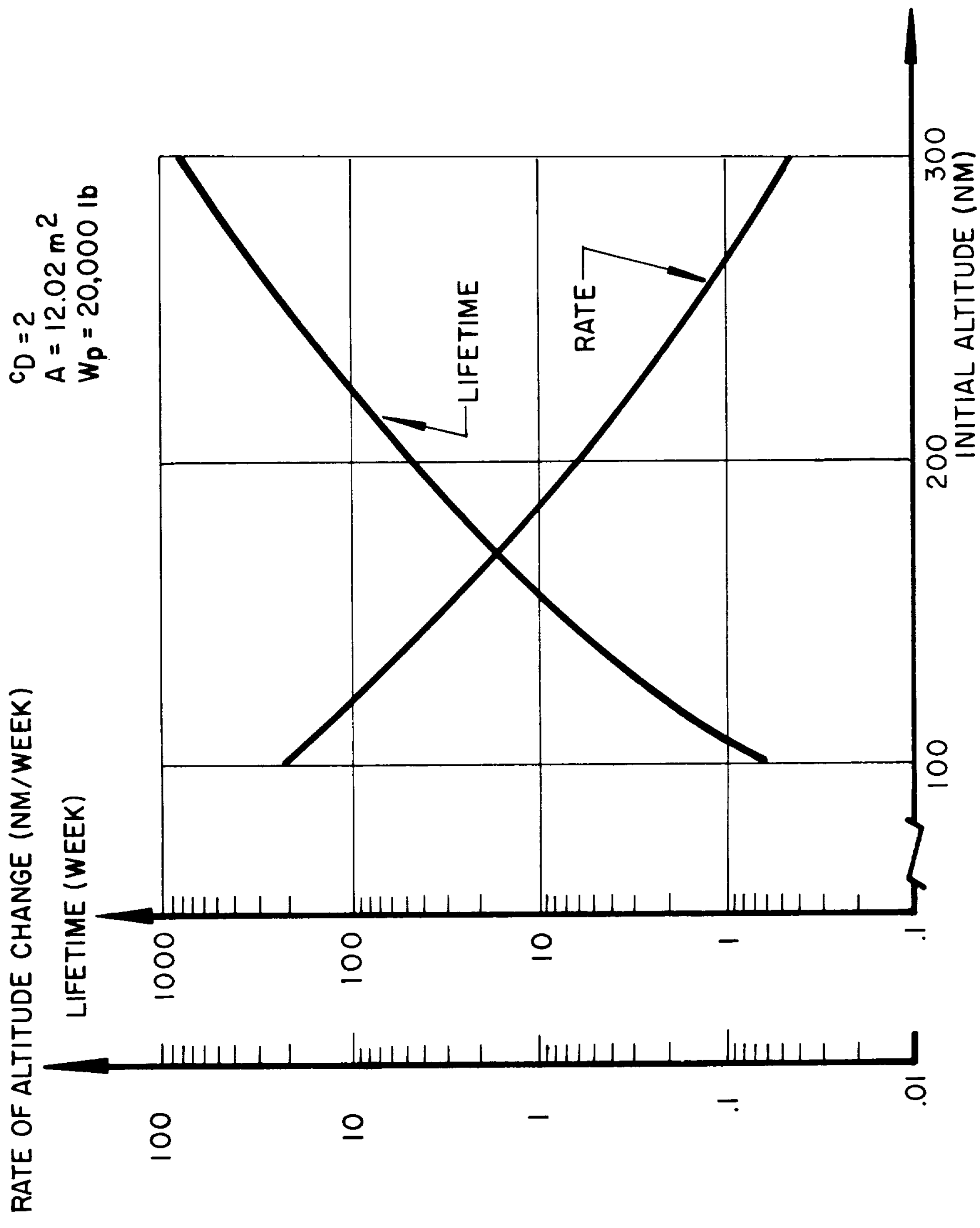

FIG. 10.8 LIFETIME OF APOLLO FOR VARIOUS ORBITAL ALTITUDES

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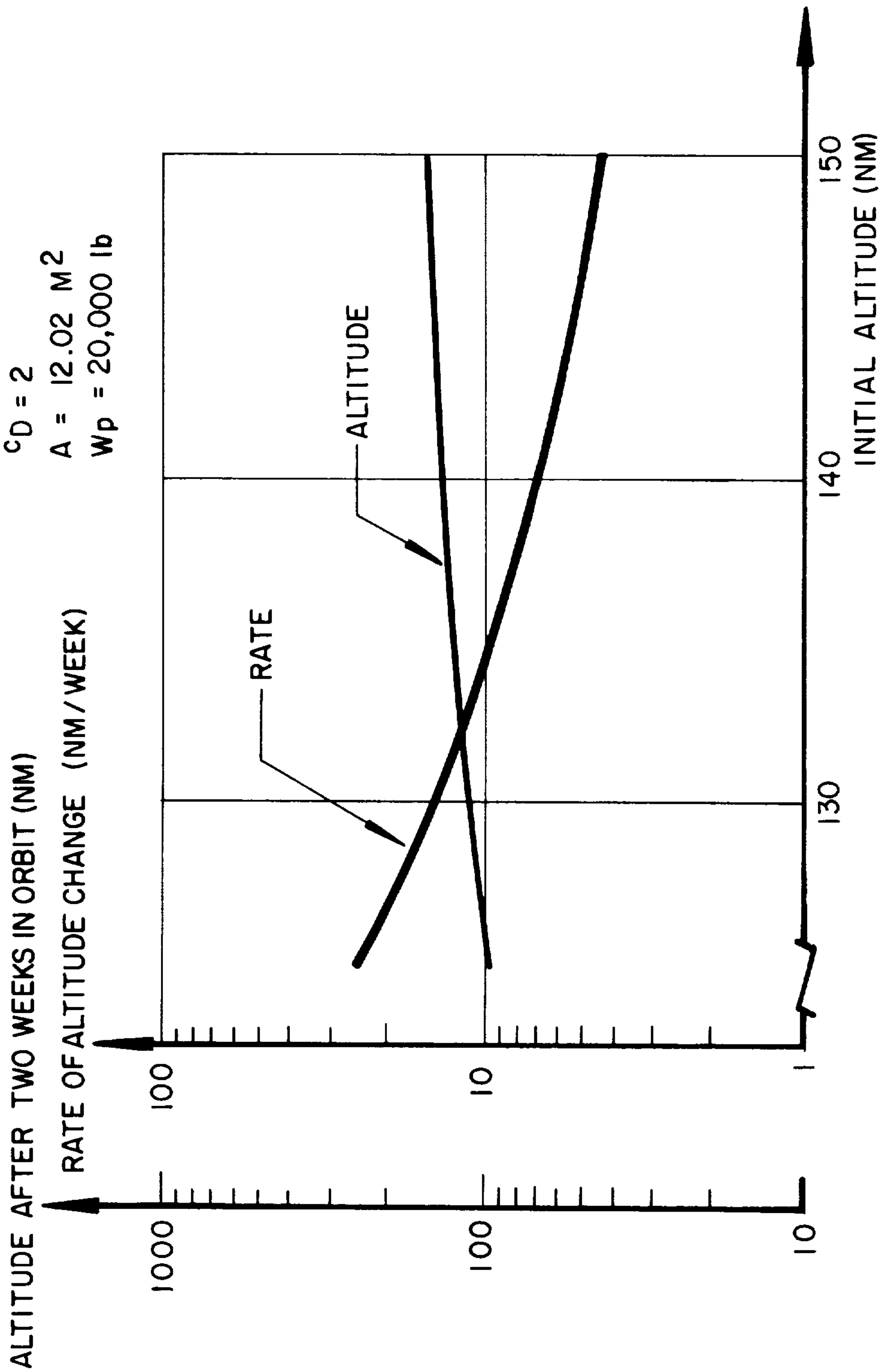

FIG. 10.9 ALTITUDE OF APOLLO AFTER TWO WEEKS IN ORBIT

(C) TABLE 10.2

FIRST STAGE FLIGHT C-1 DESIGN TRAJECTORY,
(100 NM ORBIT) (7-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
0	0	0	0	0	11.7	0	0	1,102,625	595,060	0	0
10	0	.1	0	22	12.3	30	0	1,049,830	596,020	1,051	0
20	.4	.4	0	49	12.9	143	.5	921,750	599,000	4,164	.1
30	2.1	1.1	0	83	13.6	389	2.2	948,100	604,450	6,334	.2
40	5.6	2.1	.1	126	14.6	807	5.6	896,600	612,400	8,760	.4
50	10.8	3.6	.3	180	15.6	1,404	10.8	845,090	622,560	13,478	.6
60	17.1	5.7	.8	246	16.6	2,104	17.1	793,580	634,060	25,794	.8
70	24.0	8.3	1.8	322	16.9	2,666	24.0	742,080	645,500	66,180	1.0
*78	29.7	10.9	3.1	392	18.0	2,894	29.7	700,870	653,570	69,552	1.3
80	31.1	11.6	3.5	411	18.4	2,876	31.1	690,570	655,330	67,299	1.4
90	37.9	15.4	6.2	527	20.8	2,586	37.9	639,060	662,350	46,448	1.8
100	44.0	19.9	10.1	675	23.5	2,094	44.1	587,560	666,660	28,624	2.3
110	49.3	25.1	15.6	857	26.4	1,490	49.4	536,050	669,000	15,440	2.9
120	53.8	31.1	23.1	1,076	29.6	886	54.0	484,540	670,100	7,475	3.5
130	57.5	37.9	33.0	1,335	33.3	493	57.8	433,040	670,560	3,398	4.2
140	60.6	45.5	45.7	1,641	37.9	261	61.0	381,530	670,740	1,527	4.9
150	63.2	54.0	61.6	2,003	43.9	139	63.7	330,020	670,820	645	6.0
160.6	65.4	64.2	82.7	2,468	52.7	66	65.9	275,380	670,850	225	8.0
166.6	66.4	70.4	96.4	2,584	23.7	32	65.9	262,130	287,610	107	8.9

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 7 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)
Path Angle is the angle measured between the local vertical and the velocity vector.
* Max (q)

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec (VAC)
Tilt Angle is the angle measured between the launch vertical and the vehicle axis.

(C) TABLE 10.3

SECOND STAGE FLIGHT C-1 DESIGN TRAJECTORY,
(100 NM ORBIT) (7-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
166.6	69.6	70.4	96.4	2,967	6.4	N.A.*	57.5	137,050		N.A.*	N.A.*
196.6	73.1	99.3	181.0	3,068	6.8		59.9	130,620			
226.6	76.3	124.0	269.9	3,193	7.1		62.4	124,190			
256.6	79.2	144.8	363.7	3,342	7.5		64.9	117,760			
286.6	81.7	161.8	462.6	3,515	7.9		67.4	111,340			
316.6	83.9	175.2	567.4	3,712	8.4		70.1	104,910			
346.6	85.8	185.4	678.4	3,933	9.0		72.8	98,480			
376.6	87.4	192.6	796.3	4,179	9.6		75.5	92,050			
406.6	88.7	197.0	921.8	4,452	10.3		78.3	85,620			
436.6	89.6	199.0	1,055.6	4,753	11.1		81.2	79,190			
465.1	90.3	199.0	1,191.2	5,068	12.1		84.1	73,090			
495.1	90.8	197.3	1,343.9	5,434	13.2		87.1	66,660			
525.1	91.1	194.5	1,507.8	5,842	14.6		90.3	60,230			
555.1	91.1	191.1	1,684.4	6,298	16.4		93.5	53,800			
585.1	90.8	187.8	1,875.3	6,815	18.6		96.7	47,370			
615.1	90.4	185.5	2,082.3	7,409	21.5		100.1	40,940			
632.3	90.0	185.0	2,209.0	7,793	23.7		102.9	37,270			

*N.A. - Not Applicable

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec. (VAC)

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 7 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

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(C) TABLE 10.4

FIRST STAGE FLIGHT,
C-1 DESIGN TRAJECTORY, (100 NM ORBIT), (8-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
0	0	0	0	0	13.3	0	0	1,105,425	679,867	0	0
10	0	.2	0	40	14.1	96	.1	1,044,288	681,869	3,102	.1
20	3.0	.8	0	85	14.9	420	3.6	986,895	687,756	6,972	.3
30	8.8	1.9	.1	142	16.0	1,046	9.4	928,030	697,834	10,357	.4
40	16.2	3.6	.5	213	17.2	1,961	17.0	869,165	711,232	19,671	.7
50	23.9	6.0	1.4	297	17.8	2,965	23.9	810,300	726,213	57,908	.9
60	31.4	9.0	3.0	391	18.8	3,630	31.5	751,435	740,337	88,076	1.3
*66	35.8	11.2	4.4	459	20.2	3,827	35.9	716,116	747,558	79,697	1.6
70	38.6	12.7	5.6	511	21.3	3,720	38.7	692,570	751,633	68,892	1.7
80	45.0	17.1	9.5	667	24.4	3,199	45.1	633,705	758,984	44,343	2.3
90	50.4	22.2	15.1	862	27.8	2,403	50.5	574,840	763,117	24,738	2.9
100	54.9	28.1	22.8	1,099	31.6	1,503	55.1	515,975	765,142	12,129	3.7
110	58.5	34.8	33.1	1,383	36.0	827	58.8	457,110	765,982	5,449	4.4
120	61.5	42.5	46.5	1,721	41.5	426	62.0	398,245	766,307	2,319	5.2
130	64.0	51.3	63.4	2,126	48.8	214	64.5	339,380	766,428	931	6.3
140	65.9	61.3	84.7	2,621	59.1	107	66.6	280,515	766,474	358	8.3
146.5	66.9	68.4	100.8	2,813	31.9	50	66.6	259,725	383,243	173	9.5

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 8 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec. (VAC)

* Max (q)

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(C) TABLE 10.5
SECOND STAGE FLIGHT,
C-1 DESIGN TRAJECTORY, (100 NM ORBIT), (8-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
146.5	69.9	68.2	100.6	3,191	6.3	N.A.*	62.4	139,850		N.A.*	N.A.*
176.5	73.3	98.8	191.6	3,292	6.6		64.7	133,421			
206.5	76.5	124.9	287.0	3,417	6.9		66.9	126,993			
236.5	79.3	146.7	387.3	3,567	7.3		69.2	120,564			
266.5	81.9	164.5	492.8	3,740	7.7		71.6	114,136			
296.5	84.1	178.5	604.0	3,936	8.2		73.9	107,707			
326.5	86.0	189.0	721.4	4,156	8.7		76.2	101,279			
356.5	87.6	196.1	845.7	4,399	9.3		78.6	94,850			
386.5	88.8	200.3	977.5	4,668	10.0		81.0	88,421			
416.5	89.8	201.9	1,117.4	4,963	10.8		83.5	81,993			
447.1	90.6	201.2	1,269.2	5,293	11.7		86.0	75,448			
477.1	91.0	198.9	1,428.2	5,651	12.8		88.6	69,019			
507.1	91.2	195.3	1,598.3	6,046	14.1		91.1	62,591			
537.1	91.2	191.3	1,780.6	6,485	15.7		93.8	56,162			
567.1	90.9	187.6	1,976.5	6,979	17.7		96.4	49,733			
597.1	90.3	185.3	2,187.9	7,541	20.4		99.1	43,305			
609.2	90.0	185.1	2,278.5	7,793	21.7		100.2	40,706			

*N.A. - Not Applicable

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec. (VAC)

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 8 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

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(C) Table 10.6

FIRST STAGE FLIGHT C-1 DESIGN
TRAJECTORY, (300 NM ORBIT), (7-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
0	0	0	0	0	11.7	0	0	1,095,775	595,059	0	0
10	0	.1	0	23	12.3	32	0	1,042,980	596,060	1,125	.1
20	.3	.5	0	50	13.0	152	.3	992,760	599,140	4,341	.2
30	1.4	1.1	0	86	13.7	412	1.4	941,250	604,740	6,771	.3
40	3.7	2.2	0.1	130	14.7	847	3.8	889,750	612,900	8,953	.4
50	6.9	3.8	.2	184	15.7	1,452	6.9	838,240	623,280	13,952	.7
60	10.9	5.9	.5	250	16.7	2,125	10.9	786,730	635,020	26,940	.8
70	15.4	8.7	1.2	324	17.2	2,592	15.4	735,230	646,710	62,777	1.1
77*	18.6	11.0	1.9	382	18.2	2,699	18.6	699,170	653,960	65,425	1.3
80	20.0	12.2	2.3	410	18.8	2,607	20.0	683,720	656,680	61,135	1.4
90	24.6	16.4	4.1	520	21.4	2,148	24.6	632,210	663,610	39,074	1.8
100	28.8	21.7	6.7	660	24.1	1,521	28.9	580,710	667,670	21,318	2.2
110	32.6	28.1	10.5	832	27.0	862	32.7	529,200	669,670	9,452	2.8
120	36.0	35.7	15.7	1,037	30.2	406	36.1	477,690	670,460	3,674	3.3
130	38.8	44.9	22.7	1,280	34.0	171	39.1	426,190	670,740	1,300	3.8
140	41.3	55.8	31.8	1,569	38.7	71	41.6	374,680	670,820	432	4.7
150	43.3	68.6	43.4	1,914	44.9	23	43.7	323,170	670,850	95	6.5
160.6	45.1	84.8	59.0	2,363	54.0	2	45.5	268,530	670,860	8	9.2
166.6	46.0	94.9	69.2	2,465	24.4	0	45.5	255,280	287,610	1	9.6

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust 7 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec. (VAC)

* Max (q)

(C) TABLE 10.7
SECOND STAGE FLIGHT,
C-1 DESIGN TRAJECTORY, (300 NM ORBIT), (7-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
166.6	51.9	94.9	69.2	2779	6.8	N.A.*	42.7	130,200		N.A.*	N.A.*
196.6	55.2	144.7	135.4	2815	7.1		45.7	123,770			
226.6	58.5	191.3	205.0	2878	7.5		48.9	117,340			
256.6	61.6	235.1	278.4	2967	7.9		52.5	110,910			
286.6	64.5	276.1	356.2	3085	8.4		56.4	104,490			
316.6	67.3	314.7	438.8	3232	9.0		60.6	98,060			
346.6	70.0	350.8	526.9	3410	9.6		65.2	91,630			
376.6	72.5	384.7	621.2	3621	10.3		70.2	85,200			
406.6	74.8	416.4	722.5	3868	11.2		75.6	78,770			
436.6	77.0	445.6	831.8	4154	12.2		81.3	72,340			
466.6	79.1	472.4	949.9	4484	13.4		87.4	65,910			
496.6	81.1	496.4	1078.1	4863	14.8		93.7	59,490			
526.6	83.1	517.2	1217.8	5297	16.6		100.1	53,060			
556.6	85.1	534.2	1370.6	5798	18.9		106.5	46,630			
586.6	87.0	546.8	1538.3	6380	21.9		112.8	40,200			
616.6	88.8	554.2	1723.6	7069	26.1		118.8	33,770			
635.7	90.0	555.6	1852.4	7581	29.7		122.5	29,670			

*N. A. - Not Applicable

S-I Main Stage Propellants 850,000 lbs.
S-I Thrust 7 x 188 K lbs.
S-I F/dm/dt = 255.2 Sec (S.L.)

S-IV Main Stage Propellants 100,000 lbs.
S-IV Thrust = 6 x 15 K lbs.
S-IV Isp = 420 Sec (VAC)

(C) Table 10.8

FIRST STAGE FLIGHT, C-1 DESIGN
TRAJECTORY, (300 NM ORBIT), (8-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (KG/M^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
0	0	0	0	0	13.4	0	0	1,098,275	679,867	0	0
10	0	.2	0	41	14.1	100	.1	1,037,940	681,910	3209	.1
20	2.2	.8	0	87	15.0	436	2.6	980,540	687,910	7,111	.3
30	6.2	2.0	.1	145	16.1	1078	6.6	921,680	698,180	10,579	.4
40	11.3	3.7	.4	215	17 3	1990	11.8	862,810	711,870	20,130	.7
50	16.6	6.2	1.0	298	18.0	2920	16.7	803,950	727,240	57,983	.9
60	22.1	9.4	2.1	389	19.1	3426	22.1	745,080	741,800	83,124	1.3
*64	24.2	10.9	2.8	432	20.0	3490	24.2	721,540	746,900	78,720	1.5
70	27.3	13.5	4.0	504	21.8	3219	27.4	686,220	753,290	60,415	1.7
80	32.1	18.5	6.9	654	25.0	2470	32.2	627,360	760,460	35,027	2.2
90	36.3	24.6	11.0	840	28.4	1561	36.4	568,490	764,190	16,536	2.8
100	39.9	32.0	16.9	1,067	32.2	751	40.1	509,630	765,740	6,435	3.5
110	42.9	41.0	24.8	1,338	36.7	317	43.1	450,760	766,270	2,220	4.1
120	45.4	51.7	35.1	1,662	42.3	124	45.7	391,900	766,430	725	4.9
130	47.5	64.5	48.4	2,053	49.8	44	47.9	333,030	766,480	180	6.7
140.5	49.2	80.5	66.4	2,566	61.2	6	49.7	271,030	766,490	23	9.9
146.5	50.0	90.8	78.3	2,719	32.7	1	49.7	253,370	383,250	4	10.5

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 8 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 Sec. (VAC)

* Max (q)

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(C) Table 10.9

SECOND STAGE FLIGHT, C-1 DESIGN
TRAJECTORY, (300 NM ORBIT), (8-ENGINE S-I)

TIME (SEC)	PATH ANGLE (DEG)	ALTITUDE (KM)	RANGE (KM)	VELOCITY (M/SEC)	LONGITUDINAL INERTIAL ACC. (M/SEC^2)	DYNAMIC PRESSURE (LB/FT^2)	TILT ANGLE (DEG)	WEIGHT (LBS)	THRUST (KG)	DRAG (KG)	MACH NO.
146.5	55.0	90.8	78.3	3,049	6.6	N.A.	44.4	132,700		N.A.	N.A.
176.5	58.0	141.6	153.7	3,092	7.0		47.6	126,270			
206.5	60.9	189.2	232.3	3,160	7.4		51.0	119,840			
236.5	63.7	233.9	314.7	3,254	7.8		54.6	113,410			
266.5	66.3	275.8	401.3	3,374	8.2		58.7	106,990			
296.5	68.8	315.2	492.7	3,522	8.8		63.1	100,560			
326.5	71.2	352.1	589.3	3,700	9.4		67.8	94,130			
356.5	73.5	386.6	692.1	3,909	10.1		73.0	87,700			
386.5	75.6	418.7	801.6	4,151	10.8		78.5	81,270			
416.5	77.7	448.3	918.8	4,430	11.8		84.4	74,840			
446.5	79.7	475.2	1044.6	4,750	12.9		90.5	68,410			
476.5	81.7	499.1	1180.1	5,114	14.2		96.9	61,990			
506.5	83.6	519.6	1326.6	5,529	15.9		103.2	55,560			
536.5	85.4	536.1	1485.4	6,005	17.9		109.6	49,130			
566.5	87.3	548.1	1658.5	6,553	20.6		115.7	42,700			
596.5	89.1	554.6	1847.9	7,194	24.3		121.5	36,270			
612.4	90.0	555.6	1955.7	7,581	26.8		124.5	32,870			

S-IV Main Stage Propellants 100,000 Lbs.
S-IV Thrust = 6 x 15 K Lbs.
S-IV Isp = 420 sec. (VAC)

S-I Main Stage Propellants 850,000 Lbs.
S-I Thrust = 8 x 188 K Lbs.
S-I F/dm/dt = 255.5 Sec. (S.L.)

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SECTION 11. (C) FLIGHT SEQUENCE AND SEPARATION

11.1. FLIGHT SEQUENCE

The first and second stage flight configurations of the Apollo "A"/Saturn C-1 space vehicle are shown in Figure 11.1. The major flight sequence events for a typical C-1 design trajectory of 100 nautical mile orbit, (7-engine S-I), is given in Table 11.1.

11.2. SEPARATION POLICY

It is an adopted NASA/MSFC policy that the contractor responsible for a stage (or spacecraft) of the Saturn space vehicle system is also responsible for the design, fabrication, test, and operation of both sides of the lower inflight separation plane. This is made possible by providing as part of the stage an interstage with a field splice located below the inflight separation plane. During space vehicle assembly, the stage (or spacecraft) is mounted on the stage below by the (nonseparable) field splice. Inflight separation during the staging sequence is then performed at the (inflight) separation plane.

11.3. STAGE SEPARATION CONSIDERATIONS

The separation system is designed so the separation may be accomplished without unscheduled contact between stages or jettisoned parts at any time after the tie bolts have been broken; also, so the upper stage motion after separation will be within the limits defined by the ability of the control system to maintain or regain control of the vehicle.

The standard Saturn S-I stage cutoff method will be used. The four inboard engines of the S-I stage will be cut off on a fuel level signal, then approximately six seconds later the four outer or control engines will be cut off.

11.3.1. <u>Present Separation Sequence</u>. The present design seperation sequence of the Apollo "A"/Saturn C-1 consists essentially of:

(1) A 4-4 engine shutdown of the S-I stage

(2) Ignition of explosive tie bolts

(3) A retardation of the S-I by four retrorockets, Type 2KS-36250

(4) An acceleration of the S-IV by four ullage rockets of Type TX-280, and

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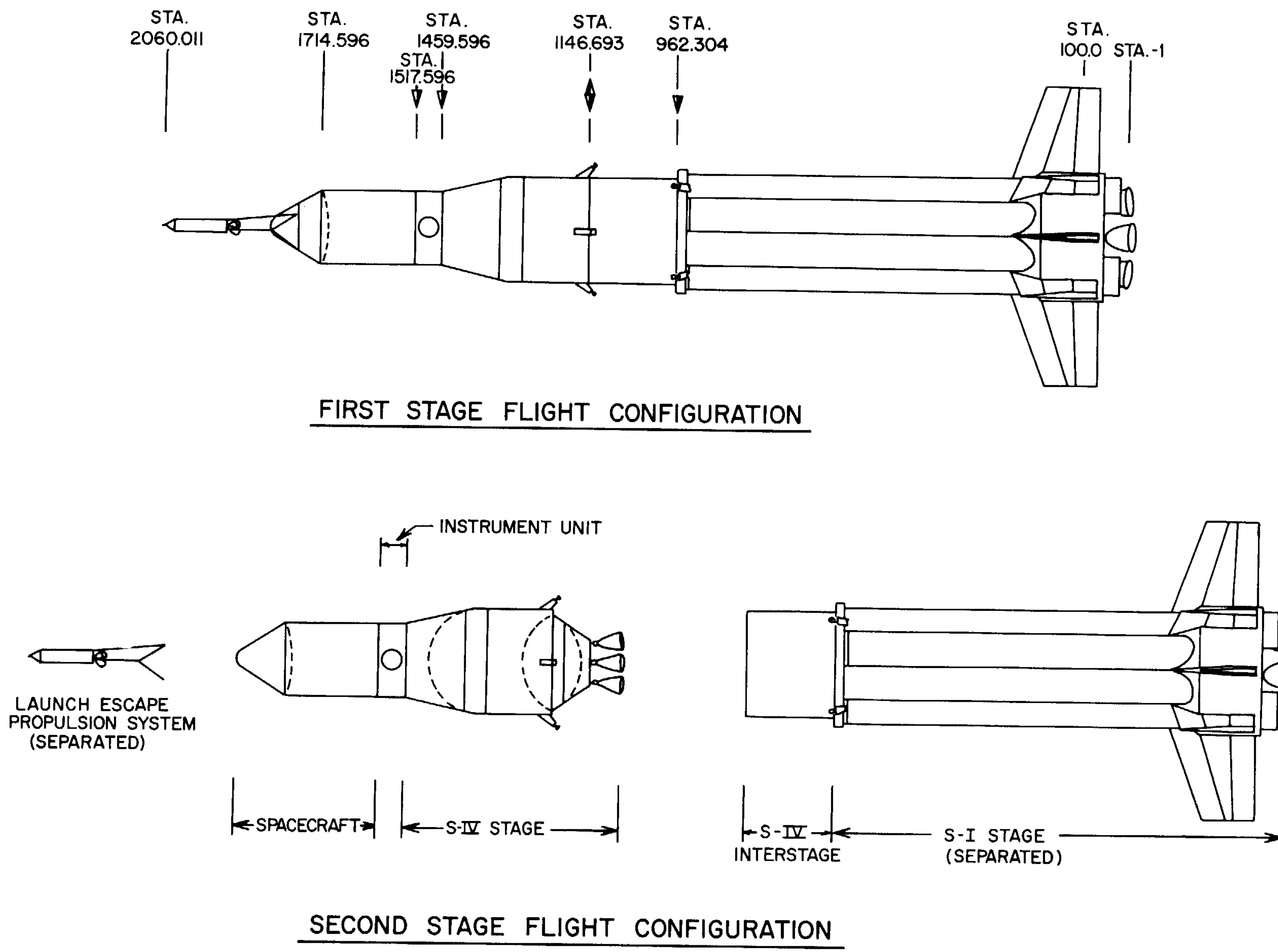

FIG. 11.1 FIRST AND SECOND STAGE FLIGHT CONFIGURATIONS OF THE APOLLO-A/ SATURN C-1 SPACE VEHICLE

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(5) A coast time of the S-IV, from separation form the S-I to ignition of the S-IV engines, of 1.7 seconds.

This separation method results in a coast distance of approximately 30 feet between the S-I and the S-IV at ignition of the S-IV engines.

11.3.2. <u>Controllability During Separation.</u> The problem of controllability during separation involves an examination of the control dynamics to insure that the vehicle will remain under control during the coast separation. To evaluate the separation system, it is necessary to determine the effects on controllability of initial conditions of dynamic pressure, angle-of-attack, and pitch rate, and to consider the effects of engine malfunctions.

Controllability was evaluated using the following guide lines:

(1) Dynamic pressure (q) at S-I control engine burnout is less than or equal to 50 kg/m^2, and after separation, during coast, it decays exponentially according to the formula:

$$q = q_o e^{-0.238t}$$

(2) Attitude deviations during separation must not exceed $\pm$ 10°.

(3) The settling time for transients after S-IV separation will not exceed 20 seconds.

The auto pilot gains were adjusted so that the natural frequency of the auto pilot was 0.3 cps with a damping ration of 0.7. Figures 11.2 and 11.3 show the maximum attitude deviation as a function of coast time.

The initial conditions are dynamic pressure of 50 kg/m^2, pitch rate of 1° /second and angle-of-attack of 8 degrees. The initial attitude error is over the range from 0.4° to 1.4°.

Coast times up to 3 seconds would be allowable for the 6-engine S-IV. The loss of one S-IV engine at start-up would reduce the allowable coast time to 2.8 seconds.

Figure 11.4 shows a typical thrust-time history for second stage separation.

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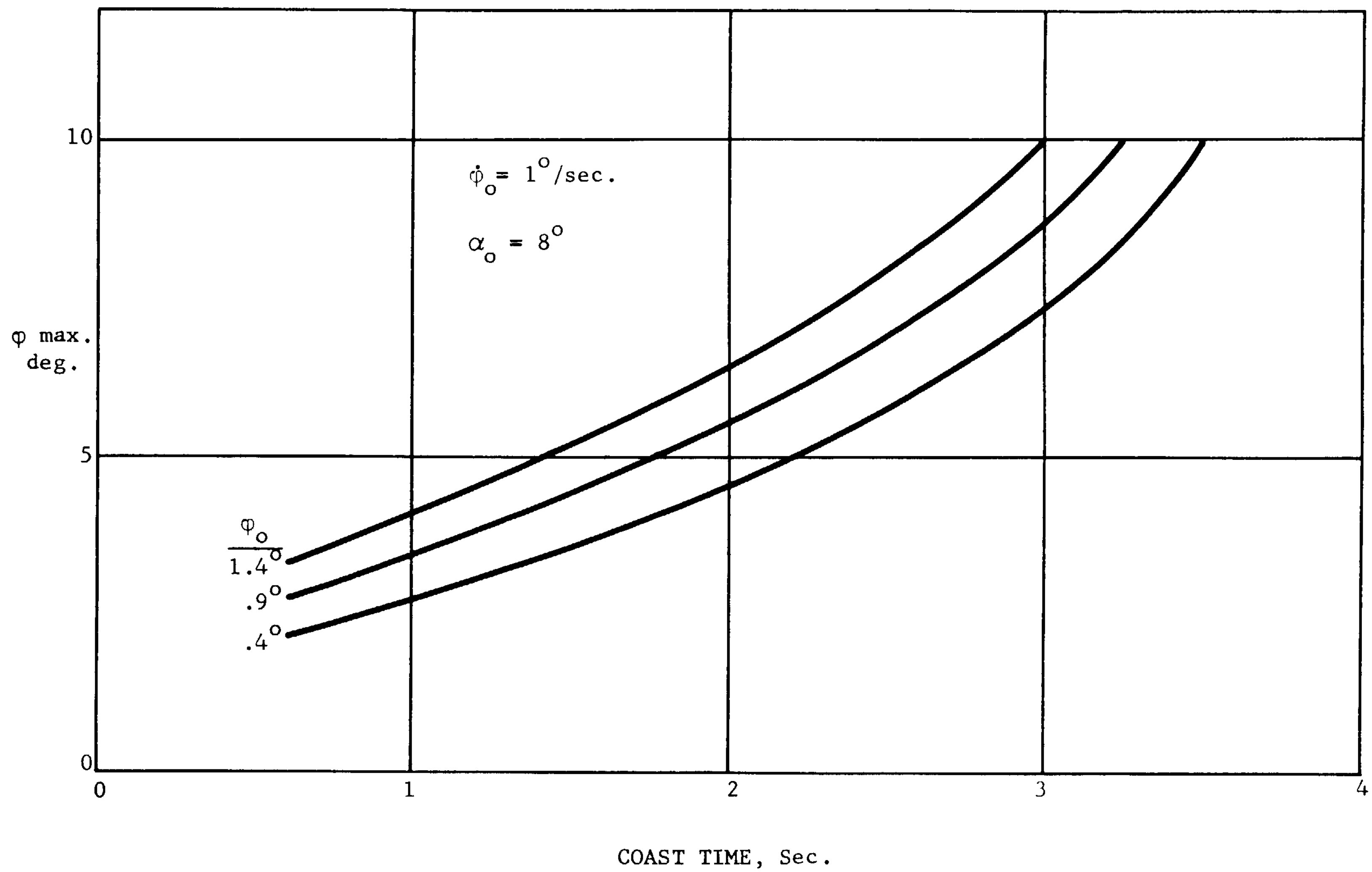

FIG. 11.2 MAXIMUM ATTITUDE DEVIATION OF S-IV AS A FUNCTION OF COAST TIME FOR SEPARATION OF S-IV FROM S-I, (6-ENGINE S-IV, 2 σ WIND)

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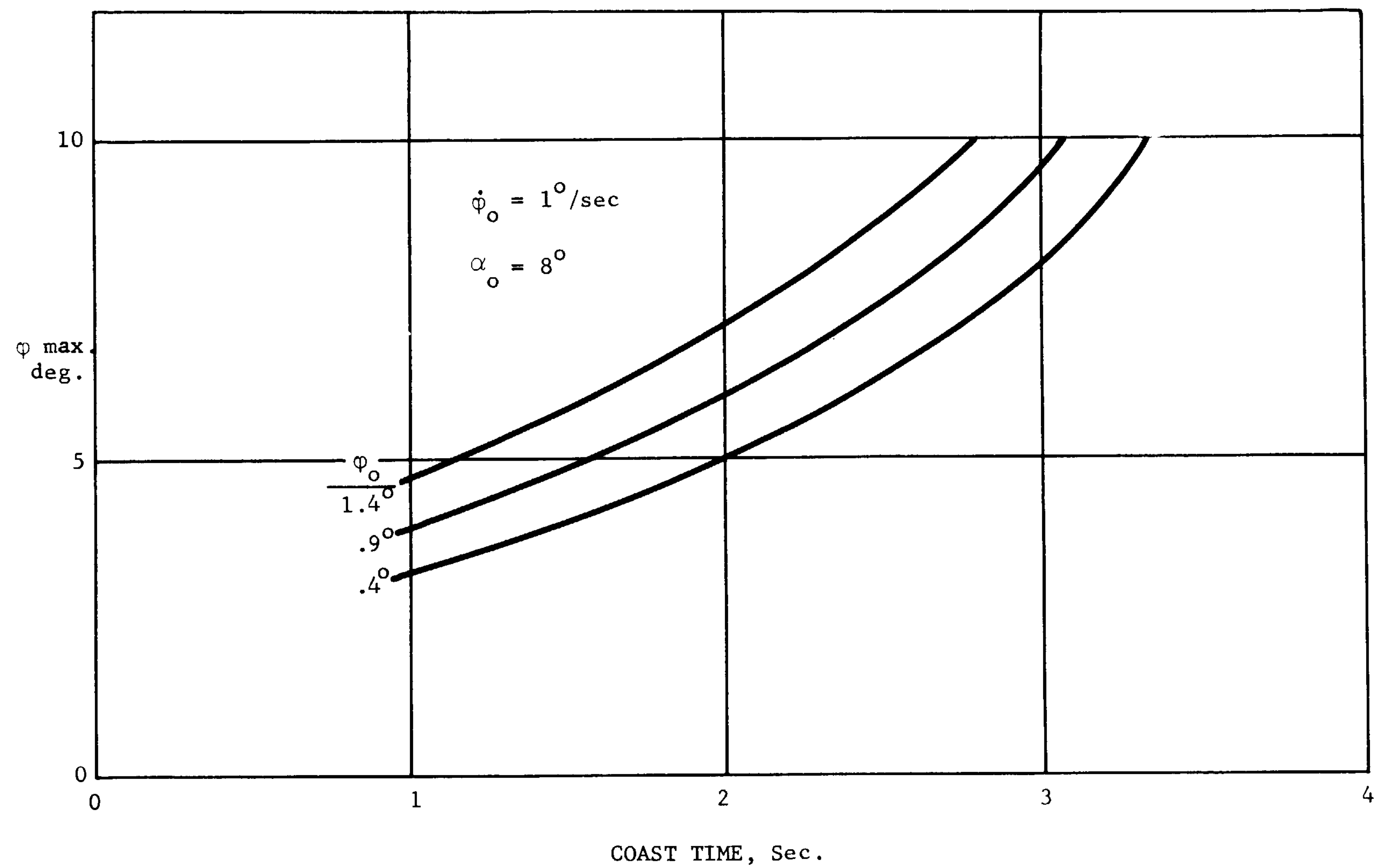

FIG. 11.3 MAXIMUM ATTITUDE DEVIATION OF S-IV AS A FUNCTION OF COAST TIME FOR SEPARATION OF S-IV FROM S-I, (5-ENGINE S-IV, 2 σ WIND)

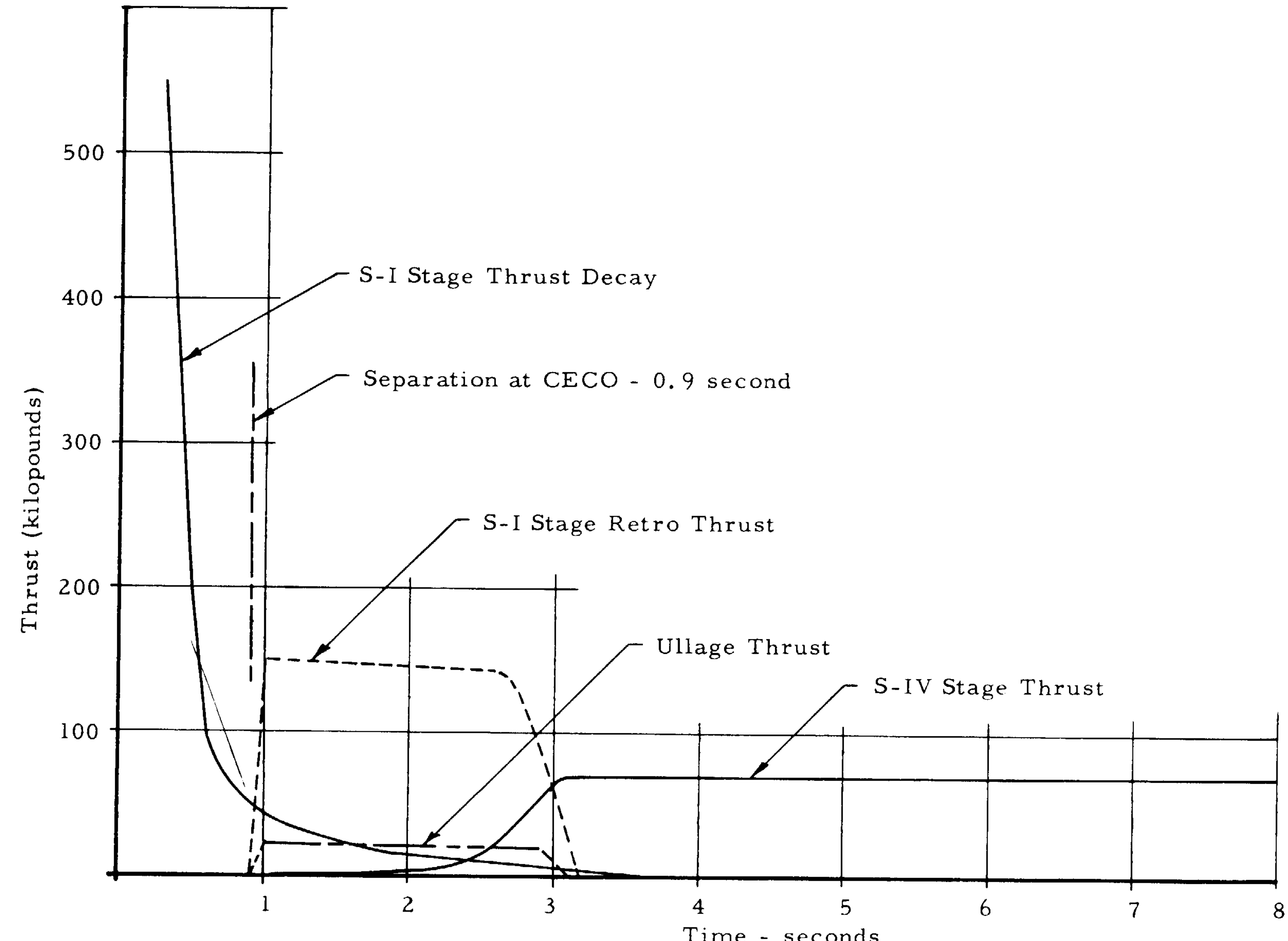

FIG. 11.4 TYPICAL THRUST-TIME HISTORY FOR SECOND STAGE SEPARATION.

(C) Table 11.1

MAJOR OPERATIONS AND RELATED TIMES

Operation	Time	Description
1	T + 3.7 sec	Liftoff
2	T + 13.7 sec	Complete vehicle roll from launch to flight azimuth Initiate tilt program Remove restrictions of only one engine shutdown capability
3	T + 153 sec	Signal from S-I stage to initiate S-IV engine chilldown Initiate S-IV engine chilldown (20 sec minimum)
4		Arm S-I cutoff
5	T + 164.3 sec	S-I fuel depletion signal S-I Stage inboard engine cutoff signal Initiate 6 second timer which gives cutoff signal of outboard engines
6	T + 170.3 sec	S-I stage outboard engine cutoff signal (expiration of 6 second timer)
7		Initiation of guidance programming for adaptive scheme
8	(Approx 1 sec after operation 6)	S-I/S-IV separation signal Ignite S-IV ullage rocket motors Ignite explosive bolts for separation Activate S-IV helium heater system for pressurization Ignite S-I retrorockets
9	(Approx 2.7 sec after operation 6)	Ignite S-IV engines

(C) Table 11.1 (Continued)

Operation	Time	Description
10	(Approx 1.7 sec after operation 9)	S-IV engines at 100% thrust
11	(Approx T + 640.4 sec)	S-IV engine cutoff signal from instrument unit

SECTION 12. (C) PERFORMANCE TRADE FACTORS

Some performance trade factors for the Saturn C-1 two-stage vehicle are given in Table 12.1. These trade factors describe the relative effect of changes in the dry weights, thrust, and specific impulse in the two stages (S-I and S-IV) on a payload which is to be injected into a circular earth satellite orbit of 100 NM altitude.

Table 12.1

PERFORMANCE TRADE FACTORS
(Two-stage Saturn C-1, Circular Orbit at 100NM)

Performance Trade Factors	STAGE S-I	STAGE S-IV
$\frac{\text{Payload Weight}}{\text{Dry Weight}}$	$-0.12 \frac{\text{lbs}}{\text{lbs}}$	$-1.0 \frac{\text{lbs}}{\text{lbs}}$
$\frac{\text{Payload Weight}}{\text{Thrust}}$	$+0.006 \frac{\text{lbs}}{\text{lbs}}$	$+0.26 \frac{\text{lbs}}{\text{lbs}}$
$\frac{\text{Payload Weight}}{\text{Specific Impulse}}$	$+180 \frac{\text{lbs}}{\text{sec}}$	$+100 \frac{\text{lbs}}{\text{sec}}$

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SECTION 13. (C) AERODYNAMICS

13.1. APOLLO "A"/SATURN C-1 VEHICLE AERODYNAMIC DATA

The present Apollo "A" Saturn C-1 vehicle has two stages, with fins on the first stage and a payload of an Apollo capsule and mission module as proposed by the Space Task Group. The configuration of the Apollo "A"/Saturn C-1 vehicle is shown in Figure 3.1. The fundamental diameters, arrangement and integration of the stages is considered firm; however, the payload portion (capsule and mission module) is open to development as the Apollo design progresses. For the present, data are presented for the Apollo/C-1 vehicle shown in Figure 3.1.

Since one of the fundamental missions of the Saturn C-1 is to provide for manned flight with a high degree of safety, the Apollo/C-1 includes the application of stabilizing fins. The four fins are attached to the eight aerodynamically faired hold-down arms that are used in conjunction with the launch complex. Various factors are involved in the determination and optimization of fin size. The maximum size allowed is governed primarily by clearance at liftoff with the fin weight penalty a secondary consideration. The actual fins were not optimized for aerodynamic properties alone but were designed in conjunction with the control output of the swivel engines and the environment that the vehicle would encounter. With four 188 K engines swiveled to 10° control angle, various amounts of wind and shear gradients can be tolerated. A highly unstable vehicle in the high dynamic pressure region of flight is severely restricted unless a very large control torque is available. This might have been designed into the system; however, this would have created another undesirable situation. That is, in case of shut down of the engines, high divergence rates would exist, a condition which is not desirable for a manned vehicle. The stability produced by the fins also removes any large restrictions that could be associated with possible unsymmetric payload designs that are not shrouded when installed on the booster.

As an added argument for a finned vehicle, the increased stability adds the potential for continued operation with one control engine out. This utilizes the maximum potential of clustering the engines. The resultant aerodynamic stability characteristics of the C-1, incorporating fins, are shown in Figure 13.1 for the first stage flight. Figure 13.2 presents the same data for the second stage.

The total drag coefficient for the first stage is given in Figure 13.3 for the power-on and power-off cases.

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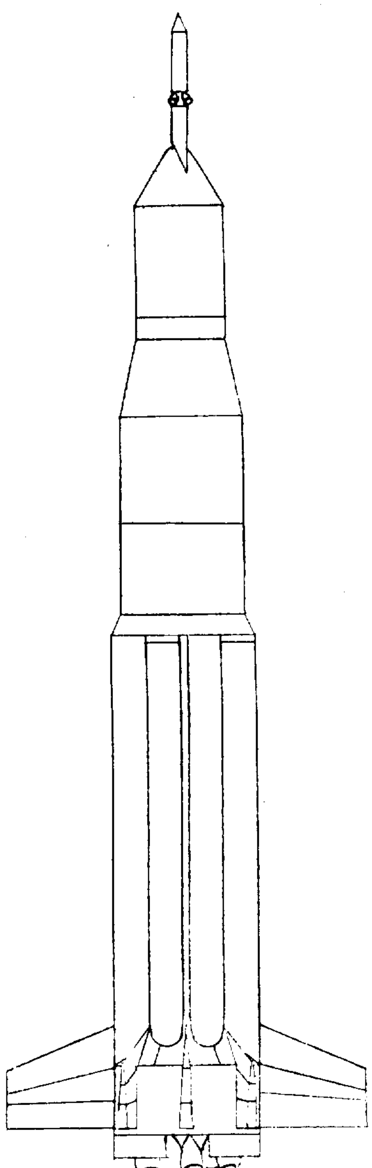

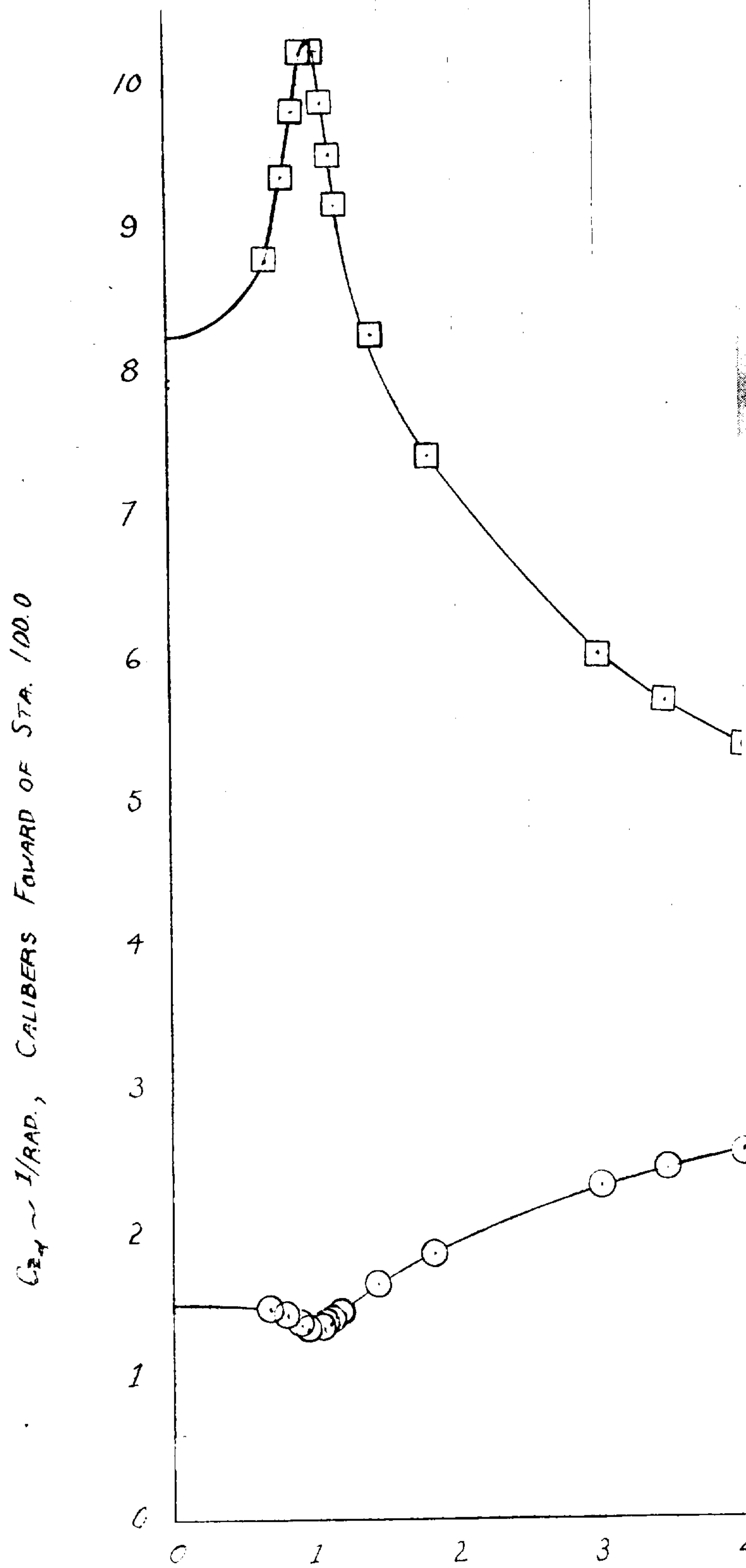

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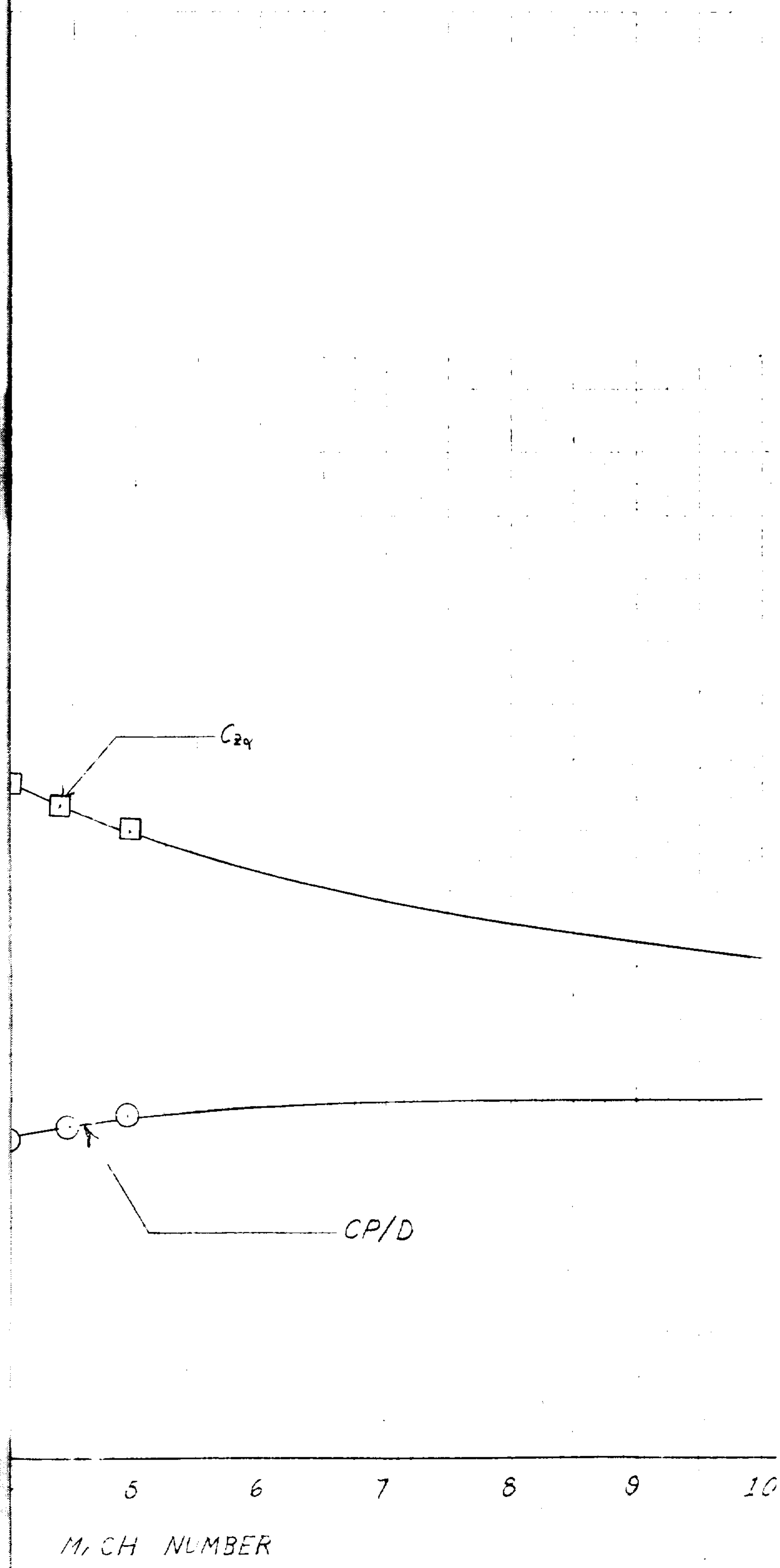
FIGURE 13.1 GRADIENT OF NORMAL FORCE CO
VERSUS MACH NUMBER FOR SATU



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ƐFFICIENT AND CENTER OF PRESSURE
ꝚN C-1/APOLLO-A VEHICLE

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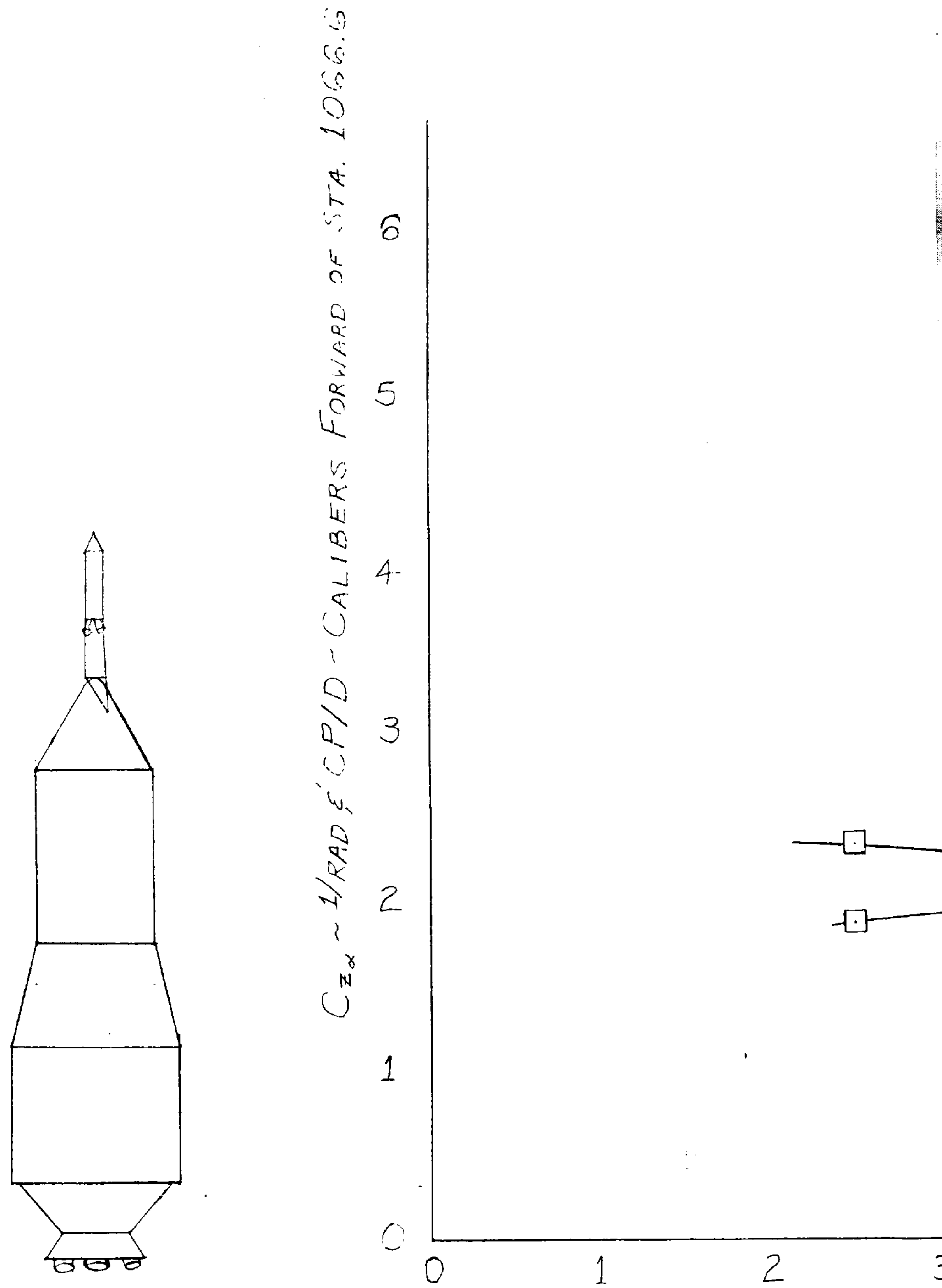

FIGURE 13.2 GRADIENT OF NORMAL FORCI
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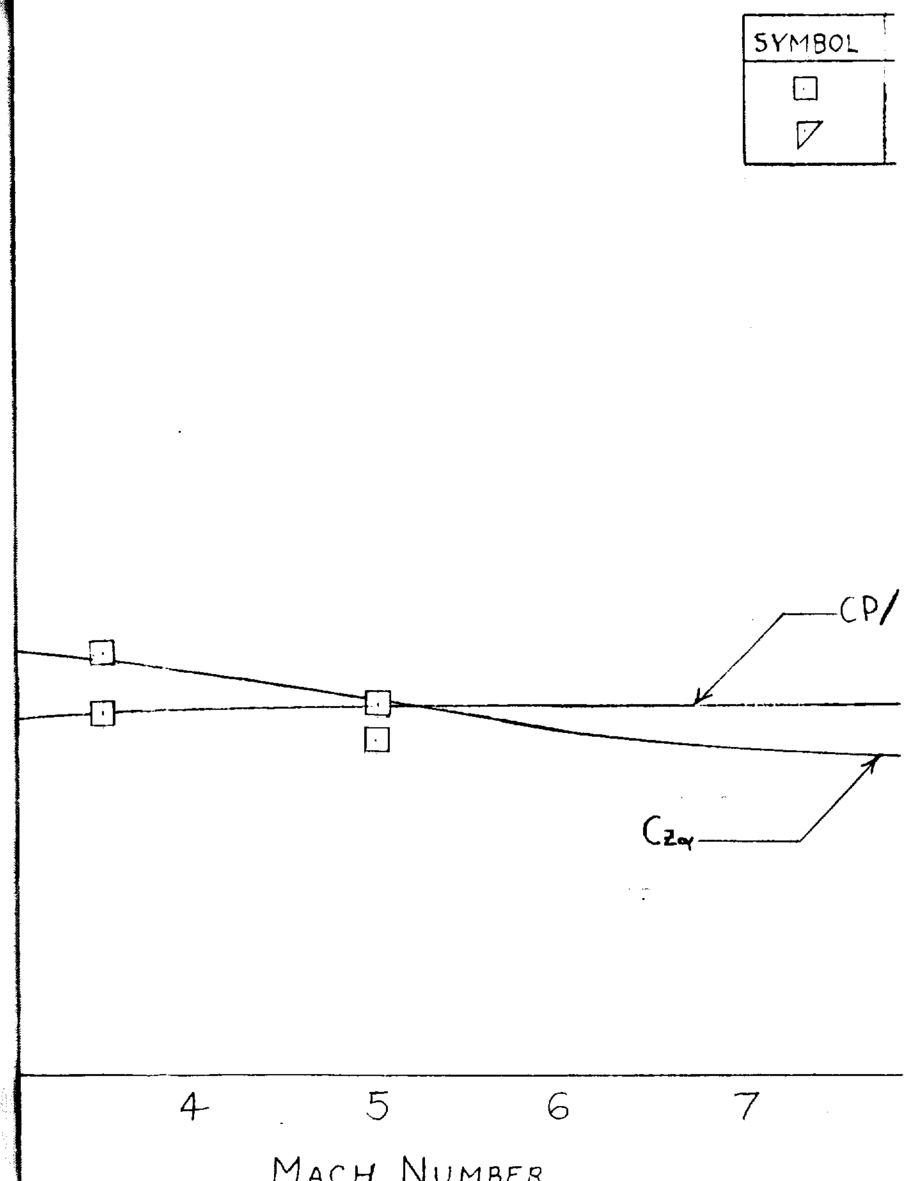
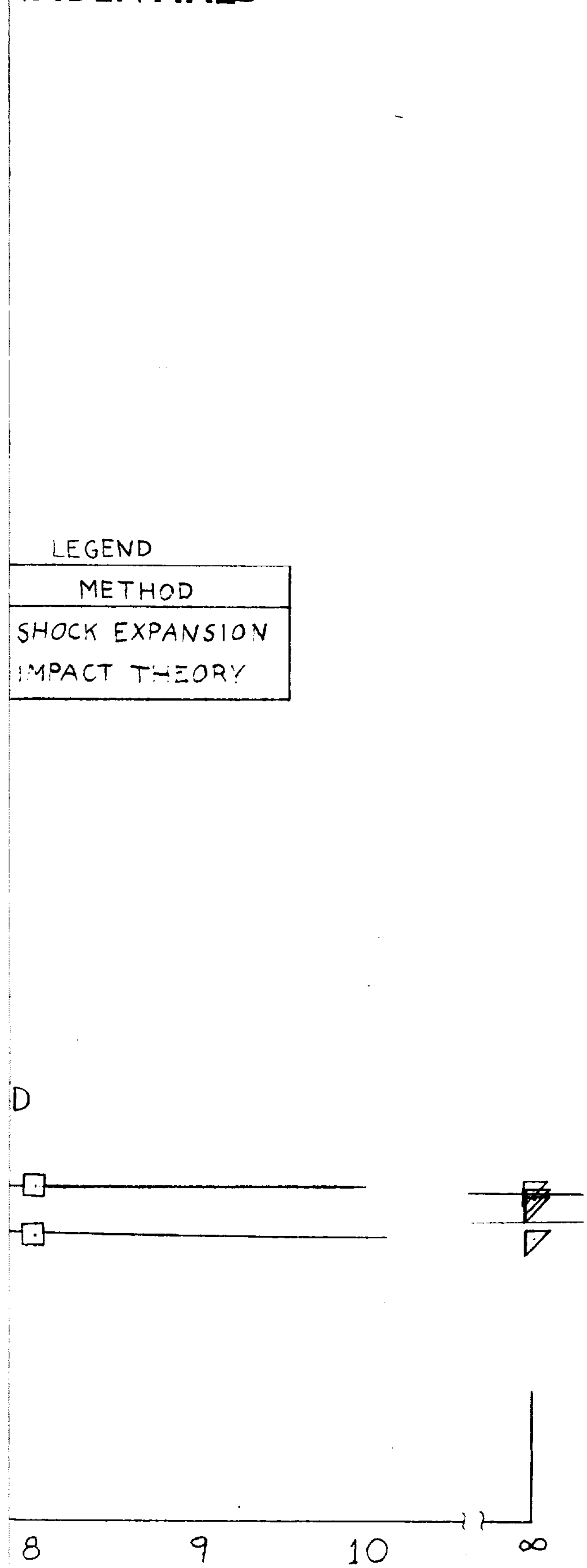


COEFFICIENT AND CENTER OF PRESSURE
-1 STAGE S-IV/APOLLO-A VEHICLE



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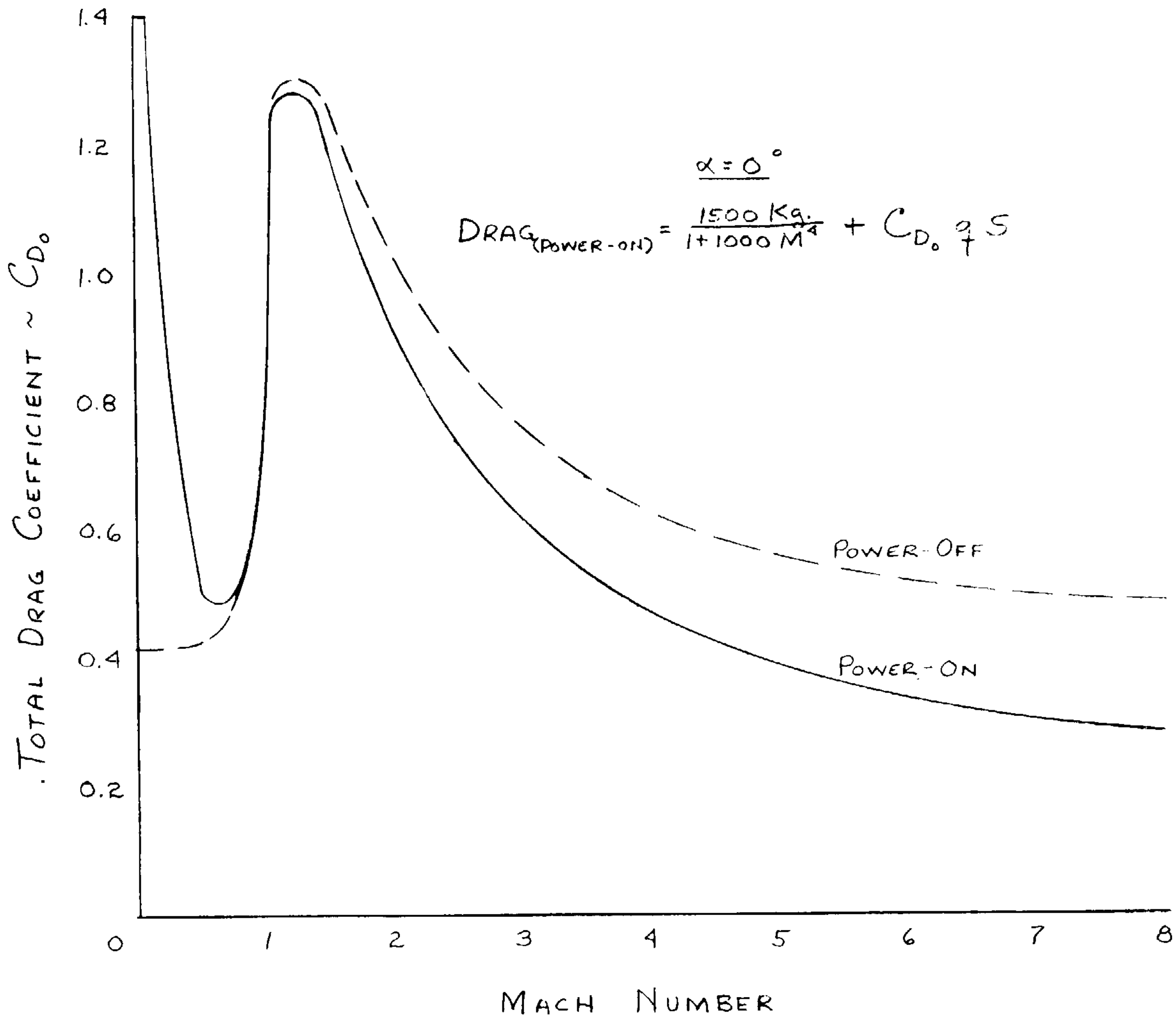

FIGURE 13.3 VARIATION OF THE TOTAL DRAG COEFFICIENT WITH MACH NUMBER FOR THE C-1/APOLLO-A VEHICLE

13.2. ESTIMATES OF THE INFLIGHT OF SATURN APOLLO "A"/SATURN C-1 VEHICLE

13.2.1. Fin Aerodynamic Heating. Heating studies have been made for the S-I stage Saturn fixed fins, utilizing a fiber reinforced plastic material (Cincinnati Testing Laboratories) (CTL)-304 or equivalent) for the outer skin supported by a 1/16 inch sheet of aluminum. Outer skin thicknesses are as follows: solid plastic leading edge (1/4 inch CTL-304), 3/32 inch CTL-304 from end of leading edge to rear spar, 1/4 inch CTL-304 from rear spar to trailing edge, 5/16 inch CTL-304 on trailing edge. See Figures 13.4 through 13.9 for skin thickness arrangements and temperature-time curves. The CTL-304 material was chosen for the fin heating studies inasmuch as its thermal properties are somewhat representative of other reinforced plastics; however, other plastics may be used for the fin material in lieu of CTL-304.

13.2.2. Fairing Between Spider Beam Ends. A metallic fairing will attain temperatures that are high enough to ignite hydrogen/oxygen propellant gases that come in contact with the hot surface. In order to minimize this potential fire hazard, a low temperature sublimer ($\approx$ 300°F) will be utilized to cover the outside surface of the metallic fairing.

13.2.3. Fairing Between Propellant Tanks. Local hot spots on the 0.032 inch thick aluminum fairing between the propellant tanks reach 980°F at S-I stage engine cutoff (Fig. 13.10).

13.2.4. Tail Section (Stations 54 to 189.5). The proposed 0.528 inch aluminum honeycomb structure in the tail region reaches a maximum outer surface temperature of about 325°F. However, in the vicinity of the fins a temperature of about 500°F can be expected due to aggravated heating resulting from wing body interference (Fig. 13.11).

13.2.5. Lower Shroud (Stations 22 to 54). Aerodynamic heating to the lower shroud was estimated for a shroud constructed of 0.04 inch stainless steel backed up by 0.4 inch of X-258. The steel portion of the shroud reaches a maximum temperature of about 770°F at S-I stage cutoff (Fig. 13.12). Hence, the steel surface will be coated with the same type of low temperature sublimer ($\approx$ 300°F) used on the fairing between the spider beam ends if oxygen and/or hydrogen propellant gases are likely to come in contact with it.

13.2.6. S-IV Stage Forward Interstage (Stations 1304.381 to 1459.596). Surface temperature history of the S-IV stage forward interstage is shown on Figure 13.13. In order to maintain a relatively cool surface, approximately 2 mm of CTL-803, or equivalent, will be applied to the exterior surface.

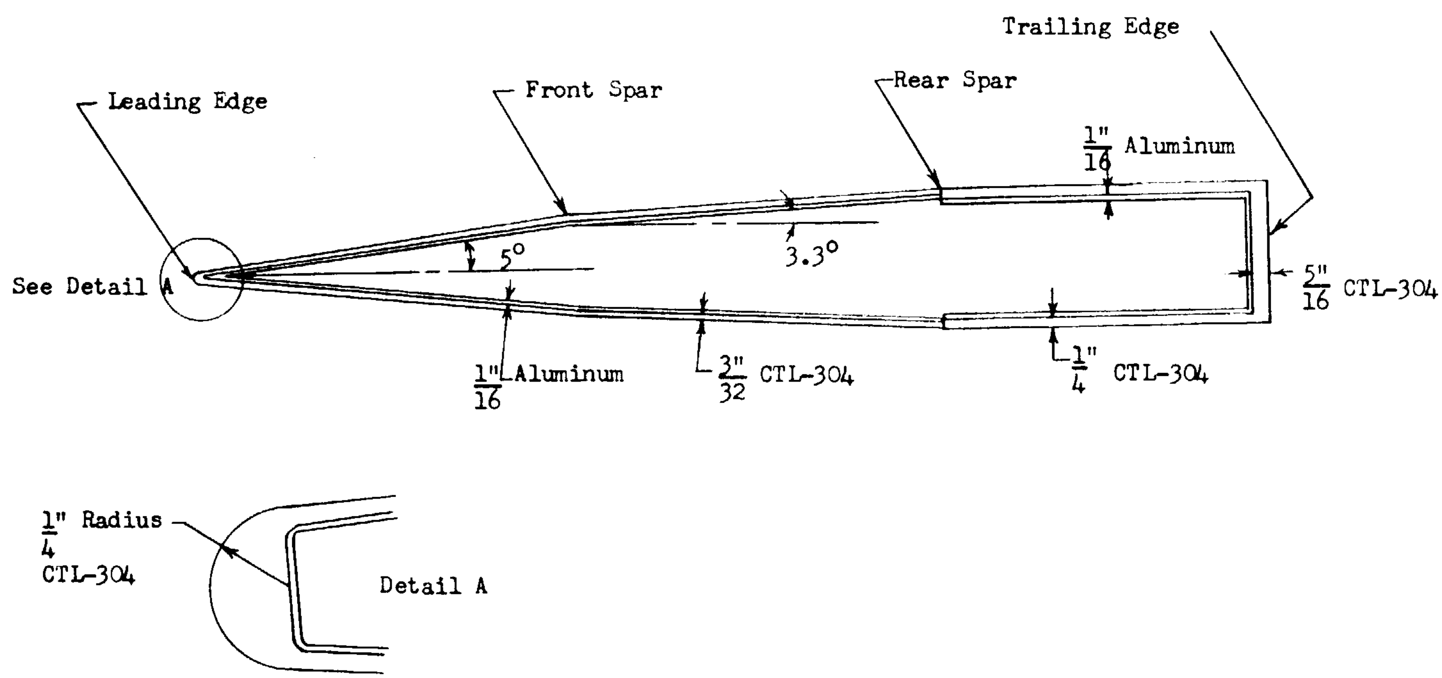

FIG. 13.4 CROSS SECTION OF THE SATURN C-1 FINS SHOWING CTL-304 AND METALLIC SKIN CONSTRUCTION

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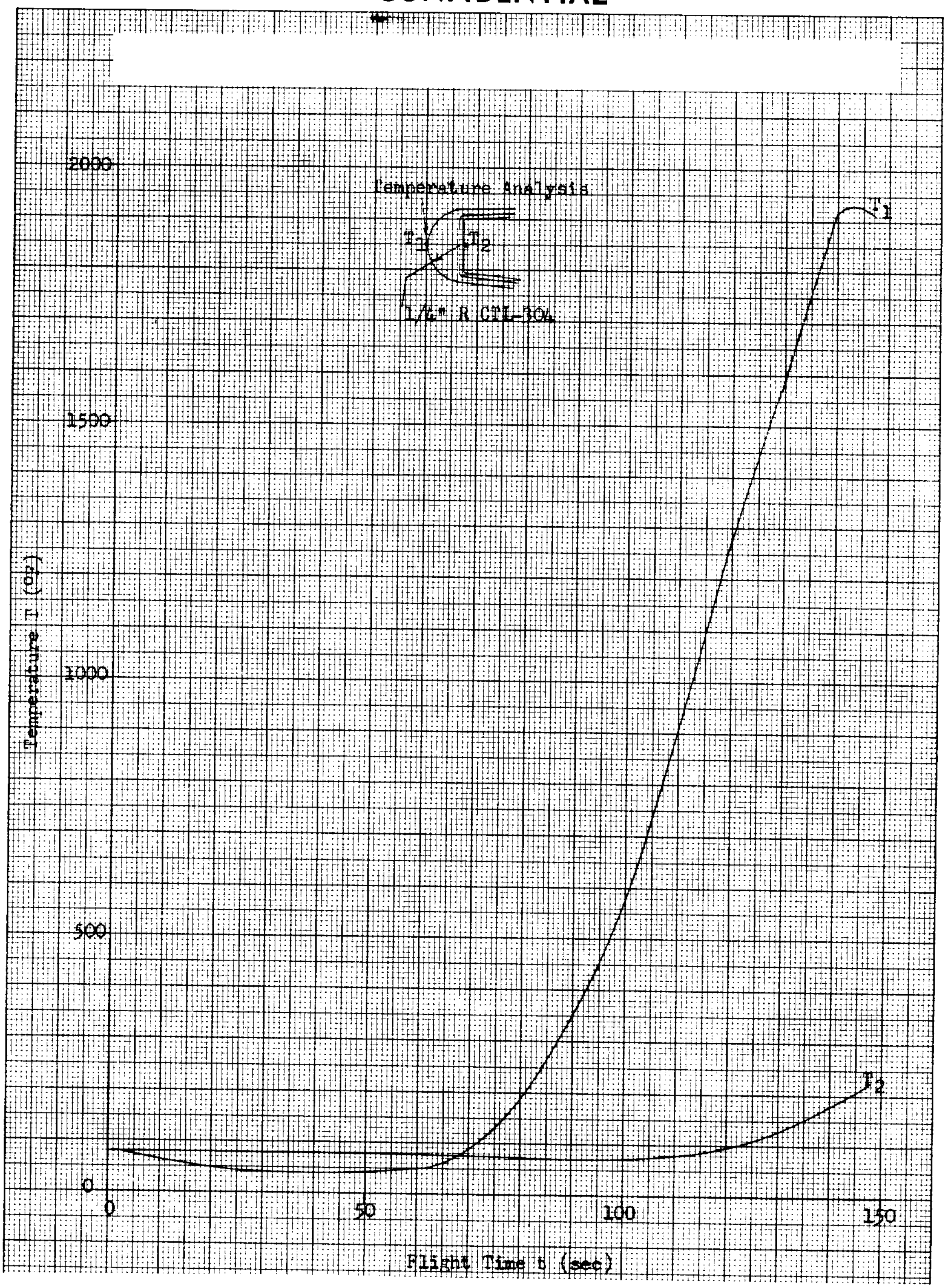

FIG. 13.5 TEMPERATURE-TIME HISTORY AT THE STAGNATION LINE FOR A REINFORCED PLASTIC MATERIAL (CTL-304) FOR SATURN C-1 FINS

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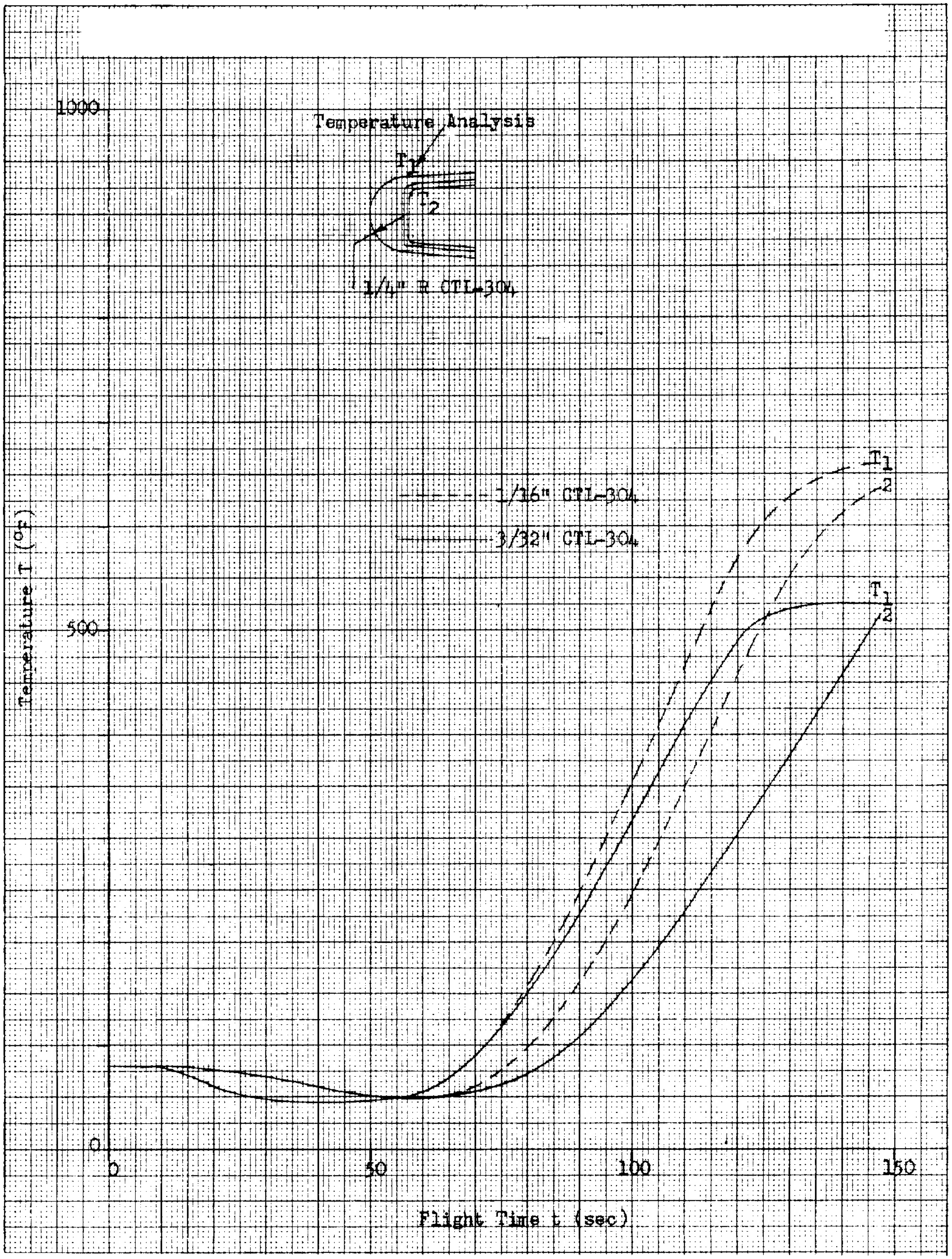

FIG. 13.6 TEMPERATURE-TIME HISTORY AT A STATION 90° FROM THE STAGNATION LINE FOR CTL-304 MATERIAL FOR SATURN C-1 FINS

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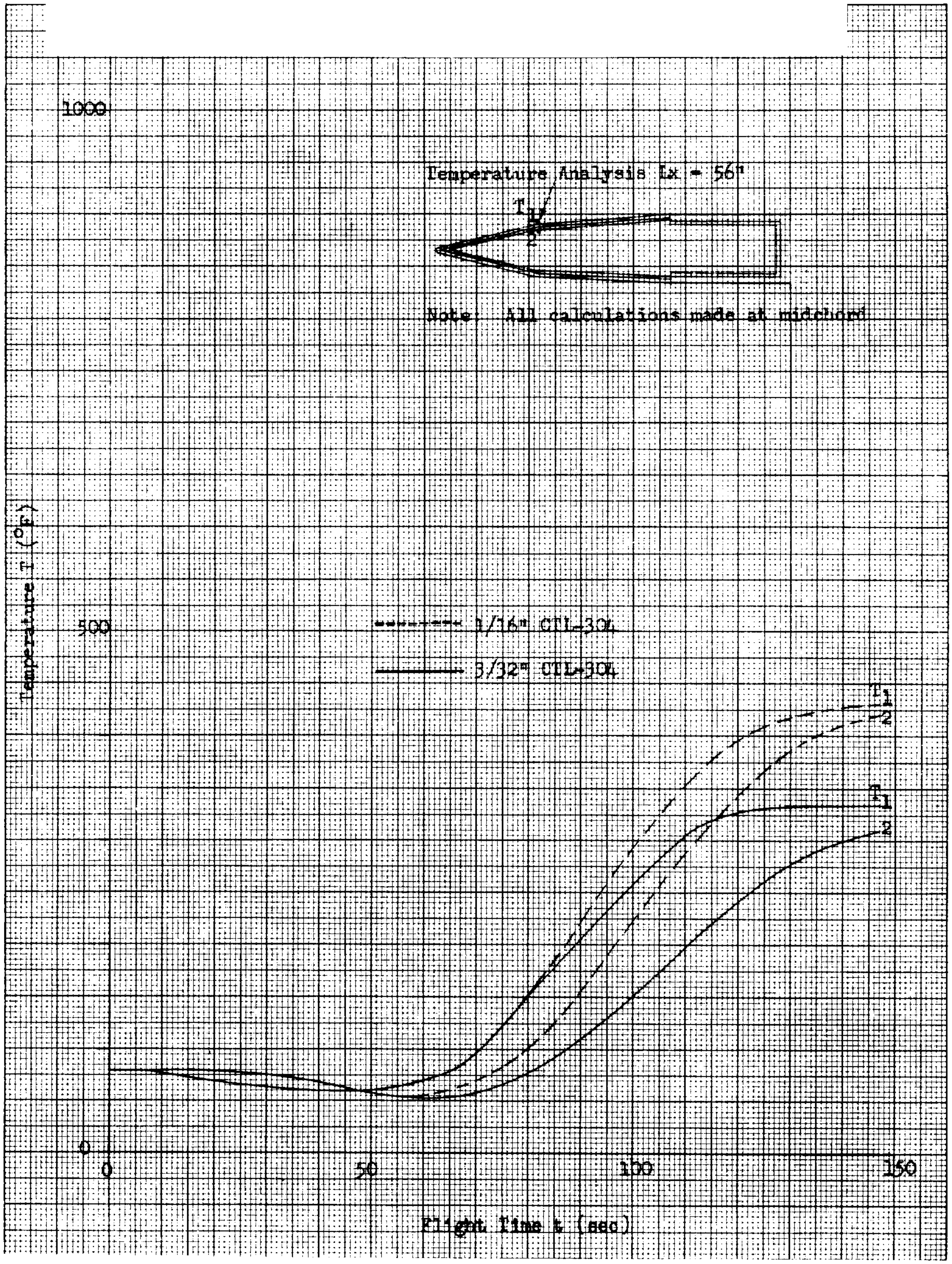

FIG. 13.7 TEMPERATURE-TIME HISTORY AT A STATION 56 INCHES FROM THE STAGNATION LINE FOR CTL-304 MATERIAL FOR SATURN C-1 FINS

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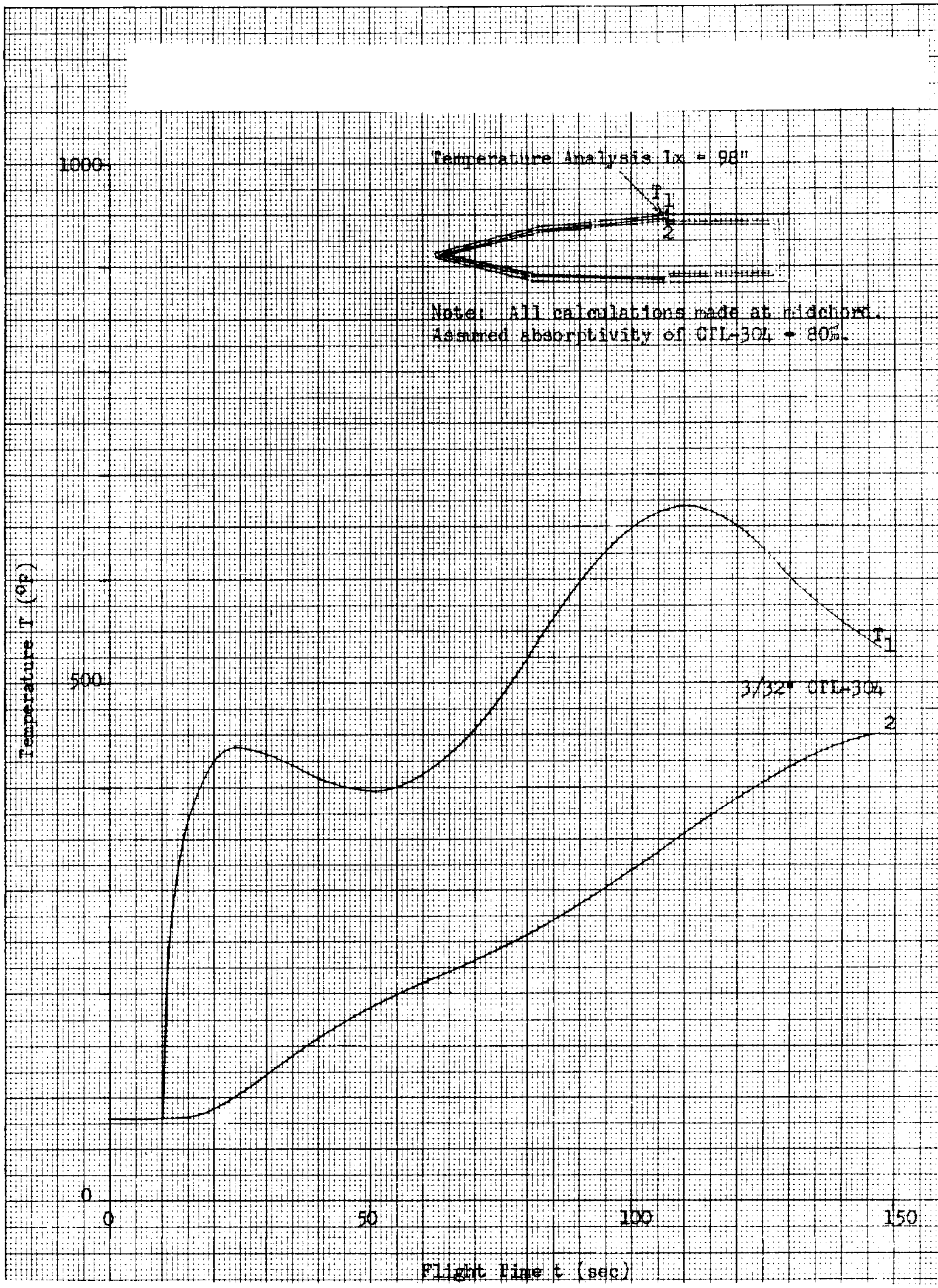

FIG. 13.8 TEMPERATURE-TIME HISTORY AT A STATION 98 INCHES FROM THE STAGNATION LINE FOR CTL-304 MATERIAL FOR SATURN C-1 FINS, INCLUDING HEATING EFFECTS OF ENGINE EXHAUST

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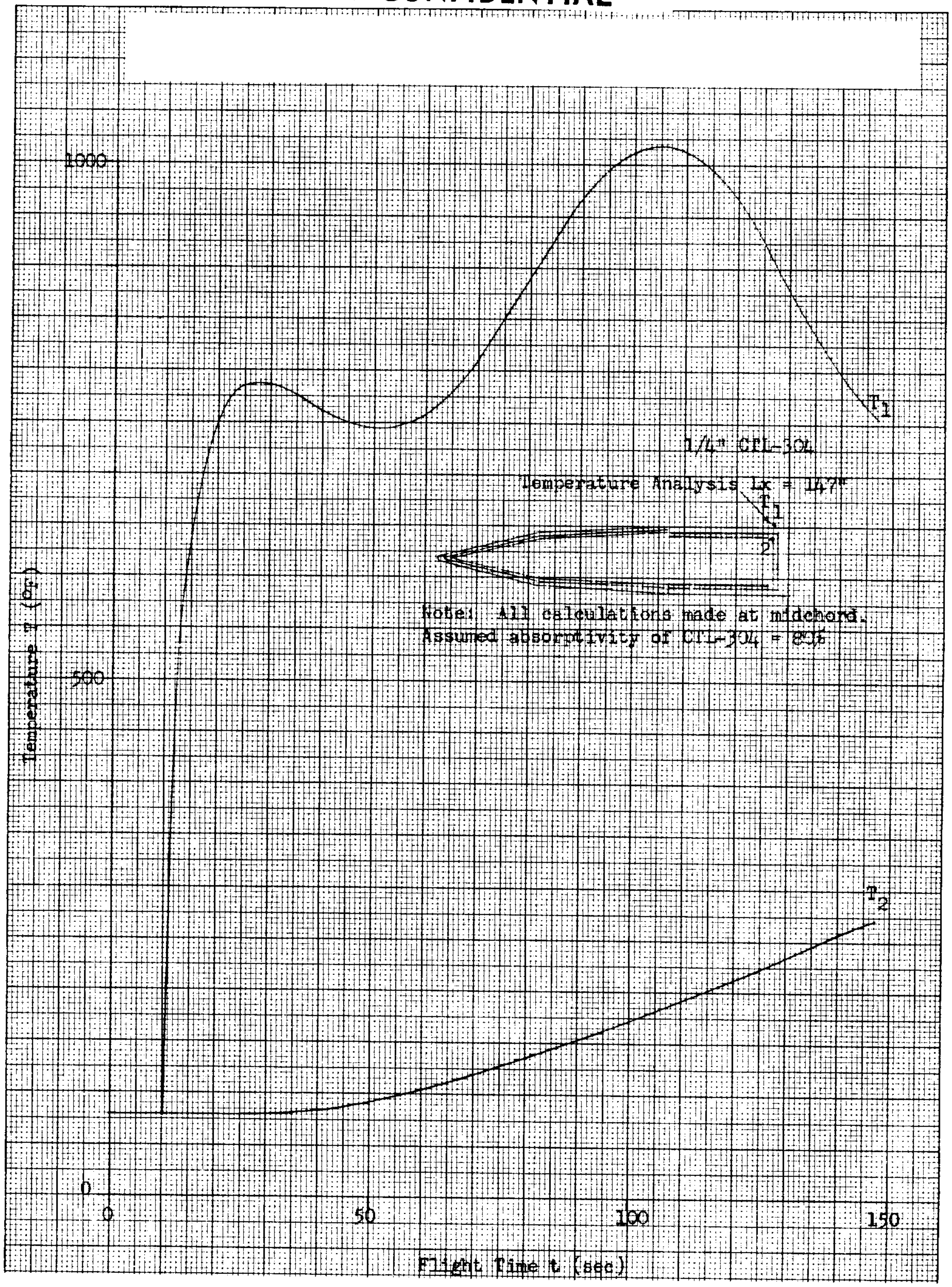

FIG. 13.9 TEMPERATURE-TIME HISTORY AT A STATION 147 INCHES FROM THE STAGNATION LINE FOR CTL-304 MATERIAL FOR SATURN C-1 FINS, INCLUDING HEATING EFFECTS OF ENGINE EXHAUST

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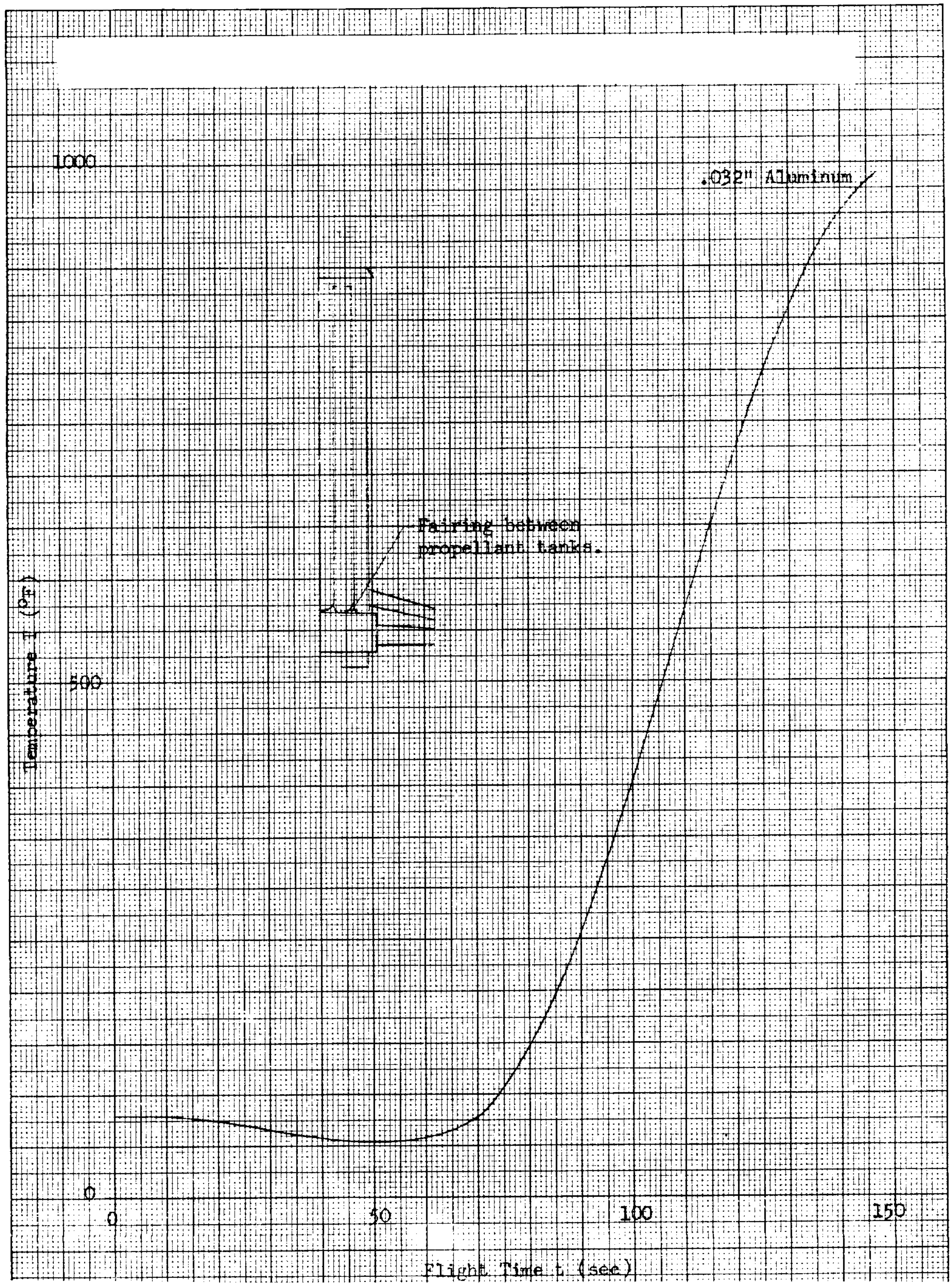

FIG. 13.10 TEMPERATURE HISTORY OF FAIRING BETWEEN PROPELLANT TANKS FOR SATURN C-1

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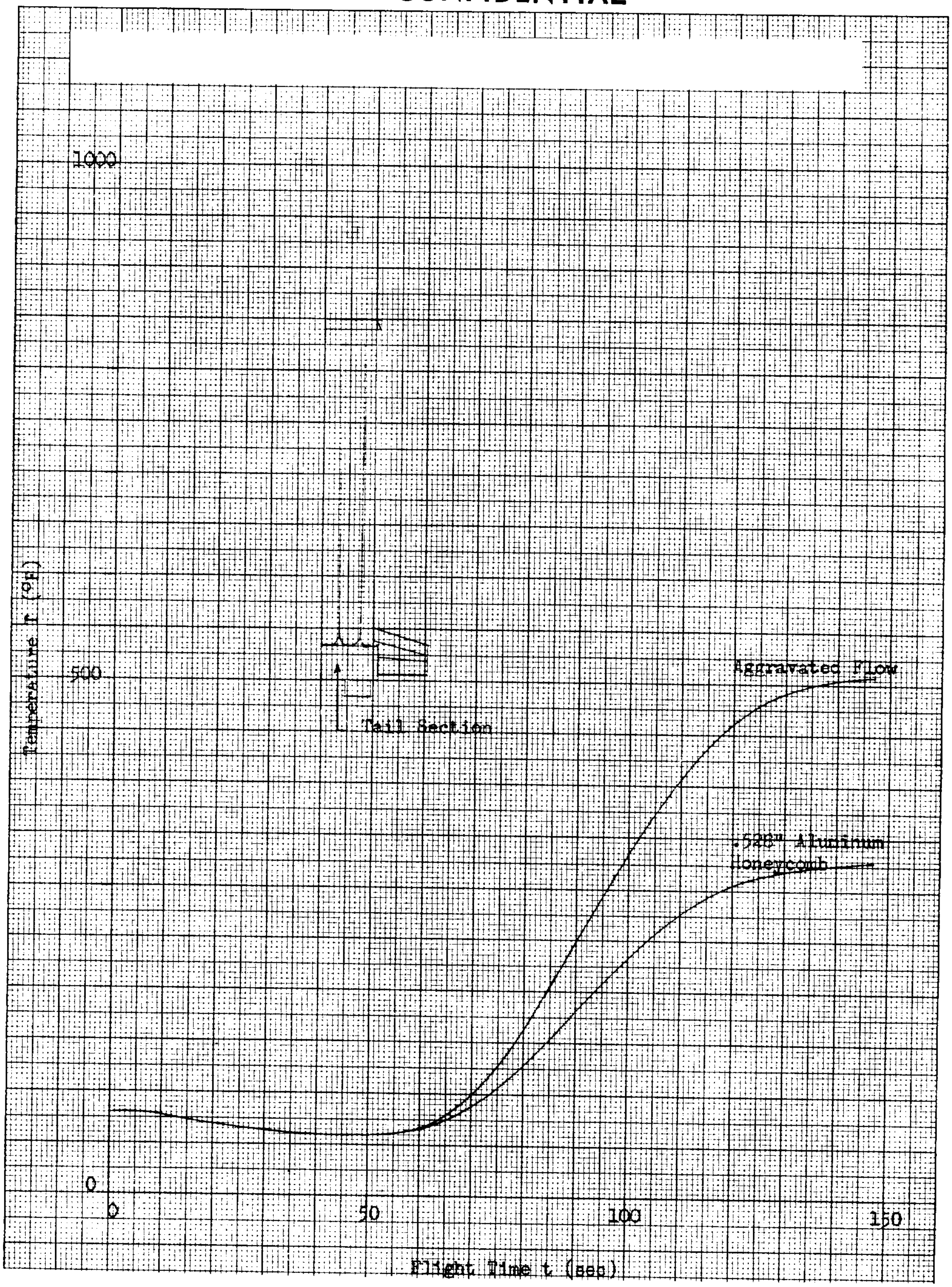

FIG. 13.11 EXTERNAL SURFACE TEMPERATURE HISTORY OF TAIL SECTION (STA. 54 TO STA. 189.5) FOR SATURN C-1

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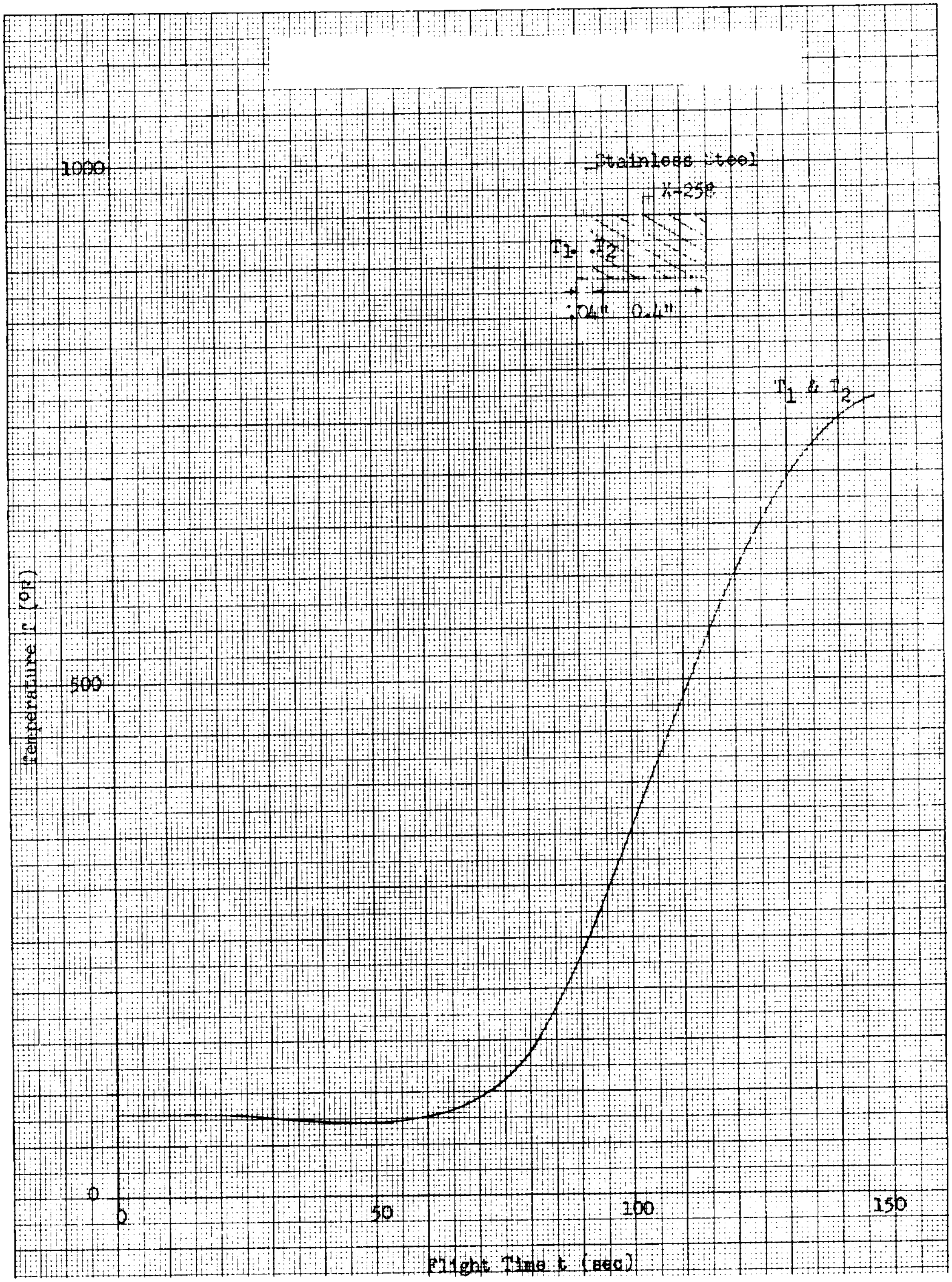

FIG. 13.12 TEMPERATURE ATTAINED BY LOWER SHROUD ON SATURN C-1, S-I STAGE DUE TO AERODYNAMIC HEATING

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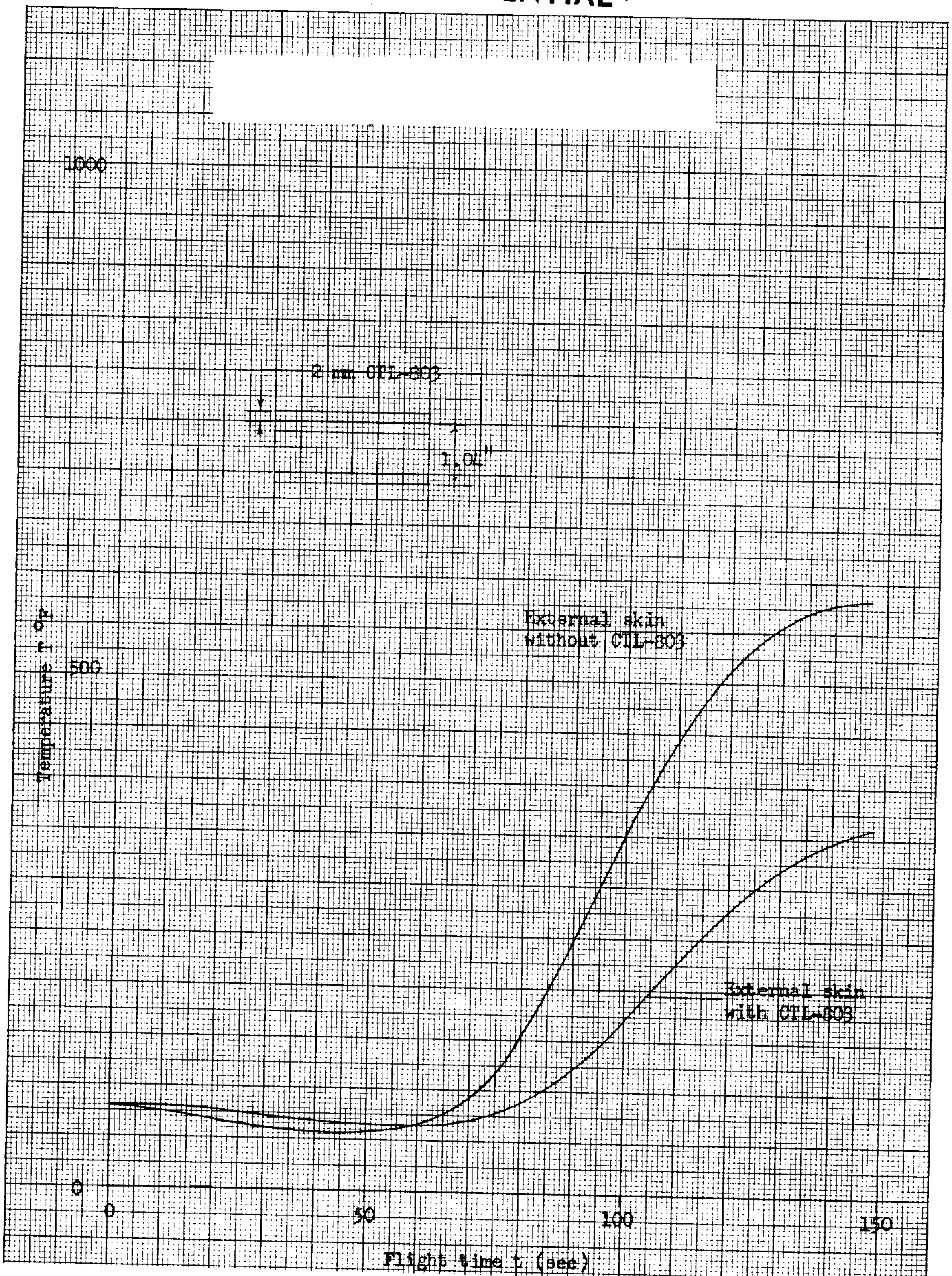

FIG. 13.13 EXTERNAL SURFACE TEMPERATURE HISTORY OF S-IV FORWARD INTERSTAGE FOR SATURN C-1/APOLLO, (STA. 1304.381 TO STA. 1459.596)

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13.3. WIND TUNNEL TESTING

The proposed aerodynamics study by wind tunnel testing of the Apollo/Saturn C-1 configuration is outlined herein. Basic aerodynamic stability tests will be conducted at the Langley Research Center's Unitary Plan Wind Tunnel, Marshall Space Flight Center's fourteen-inch trisonic facility, and various Arnold Engineering Development Center (AEDC) facilities. These basic model configuration studies will be conducted over a Mach number range from 0.70 to 8.5 to obtain static longitudinal stability characteristics.

The model to be used at the LRC Unitary Plan Tunnel facility will be 1.61 percent of full scale and fabricated of fiberglass in such a way that various configurations may be tested by interchanging various model parts to obtain the desired configuration. The other LRC model (to cover hypersonic Mach range) will be fabricated of stainless steel to a scale of about 0.8 percent.

The model to be employed in the tests at MSFC is 0.681 percent of full scale and fabricated of stainless steel and aluminum. The model parts are constructed in such a way that a number of configurations may be obtained by substituting components.

The complete wind tunnel study of the Apollo/C-1 development will encompass the following study areas:

(1) Vehicle Stability (possibly with and without hot jets)

(2) Vehicle Misalignment

(3) Fin and Stub Root Bending Moment

(4) Fore Body Load Distribution

(5) Shroud, Fin, and Stub Load Distribution

(6) Transonic Buffeting

(7) Base Heating Effects

(8) Nozzle Hinge Torques

(9) Tank Loads (Derived from SA-1 Tests)

(10) Stage Separation

(11) Booster Stability and Recovery

(12) Booster Inner Tank Heat Transfer and Gas Sampling

(13) Dynamic Characteristics

(14) On-Pad Root Bending Moments

(15) On-Pad Wind Induced Lateral Oscillations.

SECTION 14. (C) CONTROL DYNAMICS

14.1. RIGID BODY CONTROL PRINCIPLES

The most critical flight period of a Saturn C-1 flight, from the control standpoint, is at maximum dynamic pressure, where the most severe wind condition also occurs.

The wind magnitude and associated wind shear used for the vehicle studies at maximum dynamic pressure are those representative of Cape Canaveral during the month of March. The wind profile encompasses winds of the 95% probability level. This profile provides for a maximum wind of 75 m/s with associated shears of .09 $sec.^{-1}$ per 100 meters, .035 sec^{-1} per 1000 meters, .024 sec^{-1} per 2000 meters in the 10 to 14 km altitude layer.

A maximum dynamic pressure of 3745 kg/m^2 occurs at approximately 66 seconds of flight at an altitude of 11.2 km and a velocity of 455 m/s. Maximum wind velocity at this altitude is 75 m/s resulting in a wind angle-of-attack (the angle-of-attack caused by the wind alone, and not relieved by drift or altitude maneuvers) of:

$$\alpha = \frac{V_{wind}}{V_{rel}} = 9.5 \text{ degrees}$$

The well-proved drift minimum principle (DMP) has been used in computing the control gains.

The DMP forces the vehicle to nose into the wind to such an extent that the remaining lift force and the thrust component normal to the velocity vector cancel each other. This leads to a reduction of the angle-of-attack to about 4.7 degrees, and requires determination of the angle-of-attack, either by an aerodynamic indicator or by means of lateral accelerometers.

The control equation for the DMP reads:

$$\beta = a_0\varphi + a_1\dot{\varphi} + b_0\,\alpha \quad ;$$

and the moment equation is:

$$\ddot{\varphi} + C_1\alpha + C_2\beta = 0$$

To satisfy the equation, DMP requires:

$$\frac{a_o}{C_1 + b_o C_2} = \frac{F - A_x}{A_Z' C_2 - R' C_1}$$

and the undamped control frequency is determined by:

$$\omega_c^2 = (2\pi f_c)^2 = C_1 + C_2\ (a_o + b_o),$$

where

a_o, b_o, a_1, b_1, c_o	are control coefficients.
β	is the swivel angle of the engines.
φ	is the vehicle attitude angle.
α	is the angle-of-attack
C_1	is the specific restoring torque coefficient (angular acceleration per unit angle-of-attack).
C_2	is the specific control torque coefficient (angular acceleration per unit deflection of control engine).
$f_c = \frac{\omega_c}{2\pi}$	is the undamped control frequency.
F	is the thrust force.
A_x	is the axial aerodynamic force.
A_Z'	is the normal aerodynamic force derivative.
R'	is the control force derivative.

The following tabulation summarizes the control requirements of the C-1 Apollo. The control frequency is .3 cps and gains are adjusted to drift minimum.

Aerodynamic lever arm (m) = 1.96
Center of gravity from swivel point (m) = 11.75
Center of pressure from swivel point (m) = 9.79
Ratio of aerodynamic restoring coefficient to control
coefficient = .47

Four Motor Control

Steady state angle-of-attack from 2σ wind (deg) = 4.0
Steady state swivel angle from 2σ wind (deg) = 1.9
Maximum swivel angle from 2σ wind and gust (deg) = 2.6°
Maximum angle-of-attack from 2σ wind and gust (deg) = 4.6°
Maximum angular acceleration (deg/sec^2) = 1.0
Maximum lateral acceleration =(m/sec^2) = .8

Three Motor Control

Maximum swivel angle from 2σ wind and gust (deg) = 3.2°
Maximum angle-of-attack from 2σ wind and gust (deg) = 4.7°
Maximum angular acceleration (deg/sec^2) = 1.0
Maximum lateral acceleration (m/sec^2) = 1.0

14.2. ELASTIC BODY CONTROL

A preliminary control feedback stability analysis of the Saturn C-1 space vehicle was performed. The configuration studied, and shown in Figure 14.1, was a two-stage Saturn vehicle having rear fins of 340 square feet in area. Figure 14.2 presents a schematic of these fins. A booster stiffness of EI = 200 x 10^{10} lbs in^2 was used in computing the free-free uncoupled bending mode frequencies. The bending mode frequencies obtained with those assumptions are:

1st bending mode	2.08 - 2.64 cps
2nd bending mode	5.94 - 9.54 cps
3rd bending mode	8.43 - 16.92 cps
4th bending mode	15.58 - 16.92 cps

The mode frequencies of the rear fins are:

1st bending mode	17.1 cps
2nd bending mode	86.8 cps
1st torsion	41.8 cps
2nd torsion	96.1 cps

The center of gravity of the vehicle moves about 30 inches toward the rear of the vehicle during the first 50 seconds of flight time. This endangers the sloshing stability in the booster tanks slightly. From 50 seconds flight time on, the center of gravity moves about 200 inches forward, thus enhancing the sloshing situation in the booster tanks. The location of the center of instantaneous rotation moves toward the nose by 100 inches during first stage powered flight, endangering the sloshing stability in the S-IV stage.

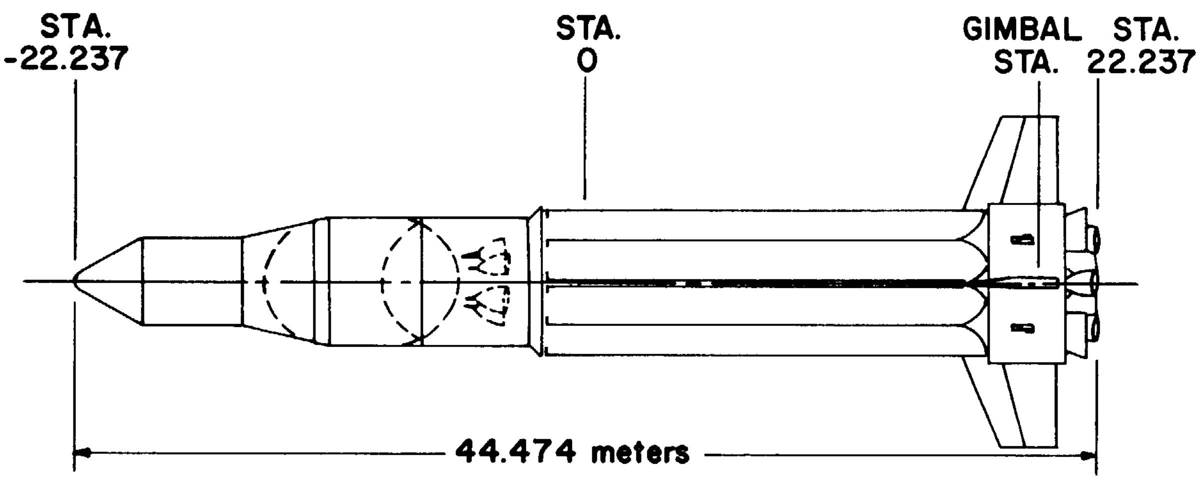

FIGURE 14.1 SATURN C-1 CONFIGURATION USED FOR CONTROL FEEDBACK STABILITY ANALYSIS

FIN CONCEPT

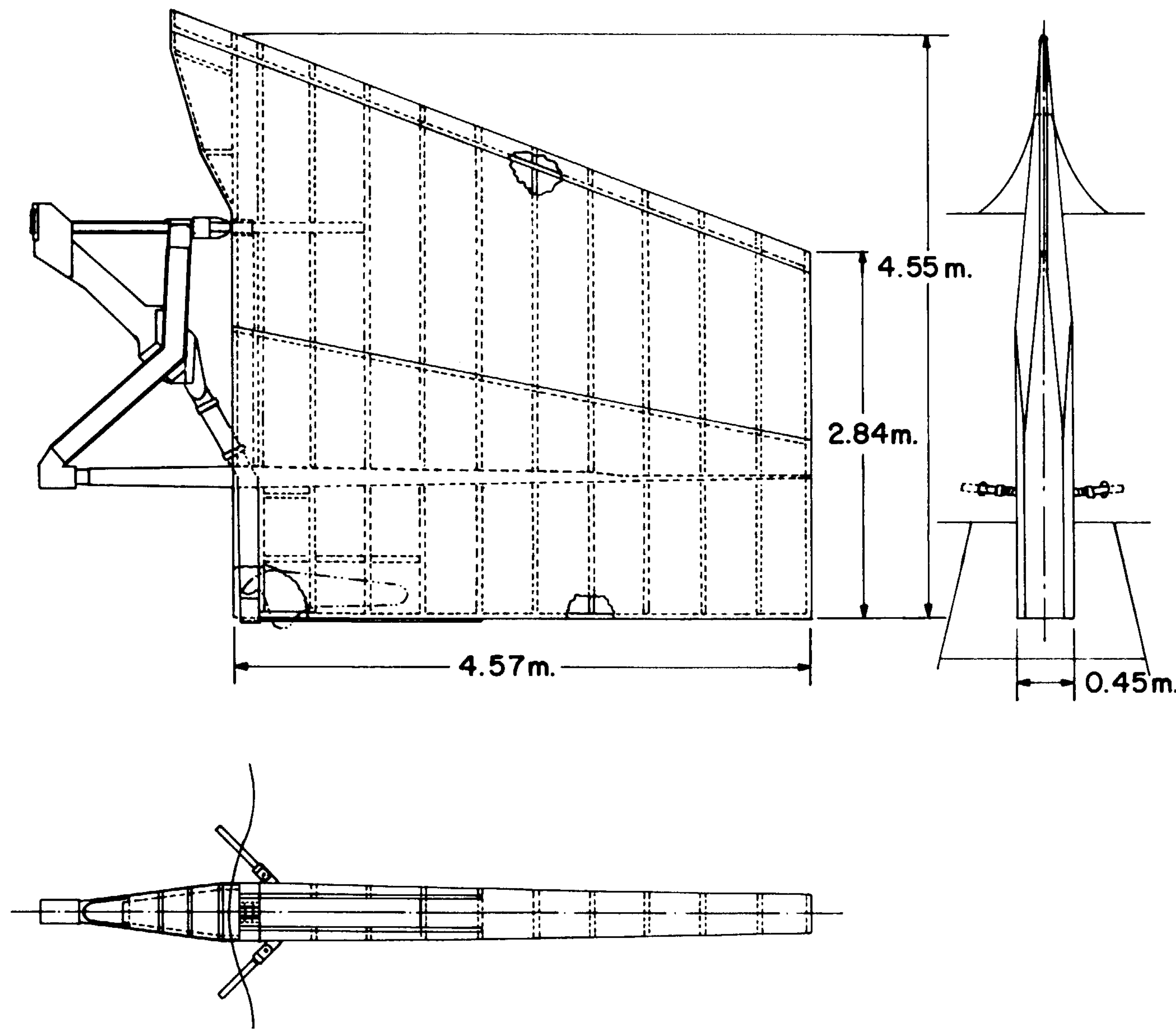

FIGURE 14.2 SCHEMATIC OF SATURN C-1 REAR FINS USED FOR CONTROL ANALYSIS

Another factor which may lead to saturation of the control system is the dynamic response of the liquid propellant in the tanks. Alleviation of this influence can be achieved in many ways and the stability of the vehicle due to propellant sloshing can be enhanced by proper choice of tank size, location, gain settings, and finally, by the introduction of baffles. In the S-I stage the influence of the tank shape (due to the clustering of nine tanks) is very favorable. The large fineness ratio $\frac{h}{a}$ of the tanks results in a small sloshing mass ratio (μ = slosh mass to vehicle mass).

Figure 14.3 gives the center of gravity, center of instantaneous rotation, sloshing mass location, sloshing mass ratio, and sloshing frequencies.

The natural frequencies of the propellant in the booster are large compared with the control frequency of 0.3 cps.

It can be seen that the S-IV stage slosh frequencies are considerably closer to the control frequency, which requires special care in a control feedback analysis. A further drawback in the S-IV stage is the tank shapes, which have a small fineness ratio and, therefore, a large slosh mass ratio. The location of the S-IV lox tank between the center of gravity and center of instantaneous rotation demands stronger baffling. A large annular baffle at a distance of about 4 to 5 inches below the free surface is necessary to obtain stability of the space vehicle due to sloshing in the S-IV stage.

A control feedback stability analysis was performed for this vehicle that included four body bending modes, two fin lateral bending modes, two fin torsional bending modes, rigid body modes of rotation and translation, and the first sloshing mode of each propellant tank. Rate gyros were used for control damping. Position and rate gyros were located at the top of the booster. No additional artificial stabilization was used. The natural frequency of the rate gyro was chosen to be 14 cps and its damping ratio $\xi_R = 0.7$. The aerodynamic forces were computed on the basis of slender body theory for the airframe within all speed ranges. The aerodynamic forces on the fins were computed from the integral equation by using the Kernel Function procedure in the the subsonic ranges. The supersonic aerodynamic forces were obtained from second order piston theory. Interference between airframe and fins was neglected.

Figure 14.4 is a block diagram of the control system and shows both a lead network and rate gyro used for control damping. Since only a rate gyro was studied, the lead network transfer function is set equal to one for this study. Figure 14.5 shows the transfer function of the amplifier-actuator system used for this investigation.

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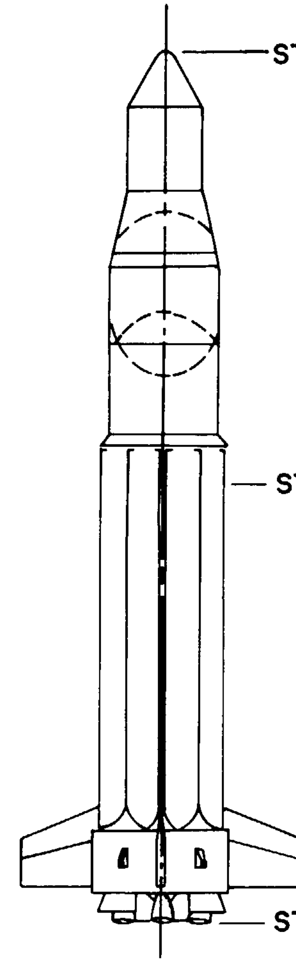
TIME	$\bar{X}_{mc}$	$\bar{X}_p$	$\bar{X}_1$	$\bar{X}_2$	$\bar{X}_3$	$\bar{X}_4$	$\bar{X}_5$
0	6.32	-9.50	-4.10	-3.45	-3.93	-11.56	-13.96
40	7.19	-11.11	-0.62	-0.06	-0.49	-12.43	-14.83
80	6.99	-12.55	3.96	4.57	3.94	-12.23	-14.63
120	3.07	-11.10	11.87	12.02	11.69	-8.31	-10.71

TIME	μ_1	μ_2	μ_3	μ_4	μ_5
0	0.0097	0.0086	0.0069	0.0535	0.0041
40	0.0128	0.0114	0.0091	0.0706	0.0055
80	0.0187	0.0166	0.0132	0.1028	0.0080
120	0.0304	0.0225	0.0213	0.1890	0.014

TIME	f_1	f_2	f_3	f_4	f_5
0	0.880	0.719	0.880	0.454	0.490
40	0.995	0.813	0.995	0.512	0.554
80	1.260	1.030	1.260	0.649	0.700
120	1.633	1.216	1.627	0.893	0.963



SYMBOLS

$\bar{X}_{mc}$ = DISTANCE TO MASS CENTER FROM GEOMETRICAL CENTER OF VEHICLE (METERS)

μ = RATIO OF SLOSHING MASS TO VEHICLE MASS

f = SLOSHING FREQUENCY

$\bar{X}_p$ = DISTANCE OF INSTANTANEOUS CENTER OF ROTATION FROM MASS CENTER OF VEHICLE (METERS)

$\bar{X}$ = DISTANCE OF SLOSHING MASS C.G. FROM MASS CENTER OF VEHICLE (METERS)

SUBSCRIPTS

1 = 70" LOX TANK
2 = 105" LOX TANK
3 = 70" FUEL TANK
4 = S-II LOX TANK
5 = S-II FUEL TANK

FIGURE 14.3 TABLE OF CENTER OF GRAVITY, CENTER OF INSTANTANEOUS ROTATION, SLOSHING MASS LOCATION, SLOSHING MASS RATIO AND SLOSHING FREQUENCIES FOR VARIOUS FLIGHT TIMES

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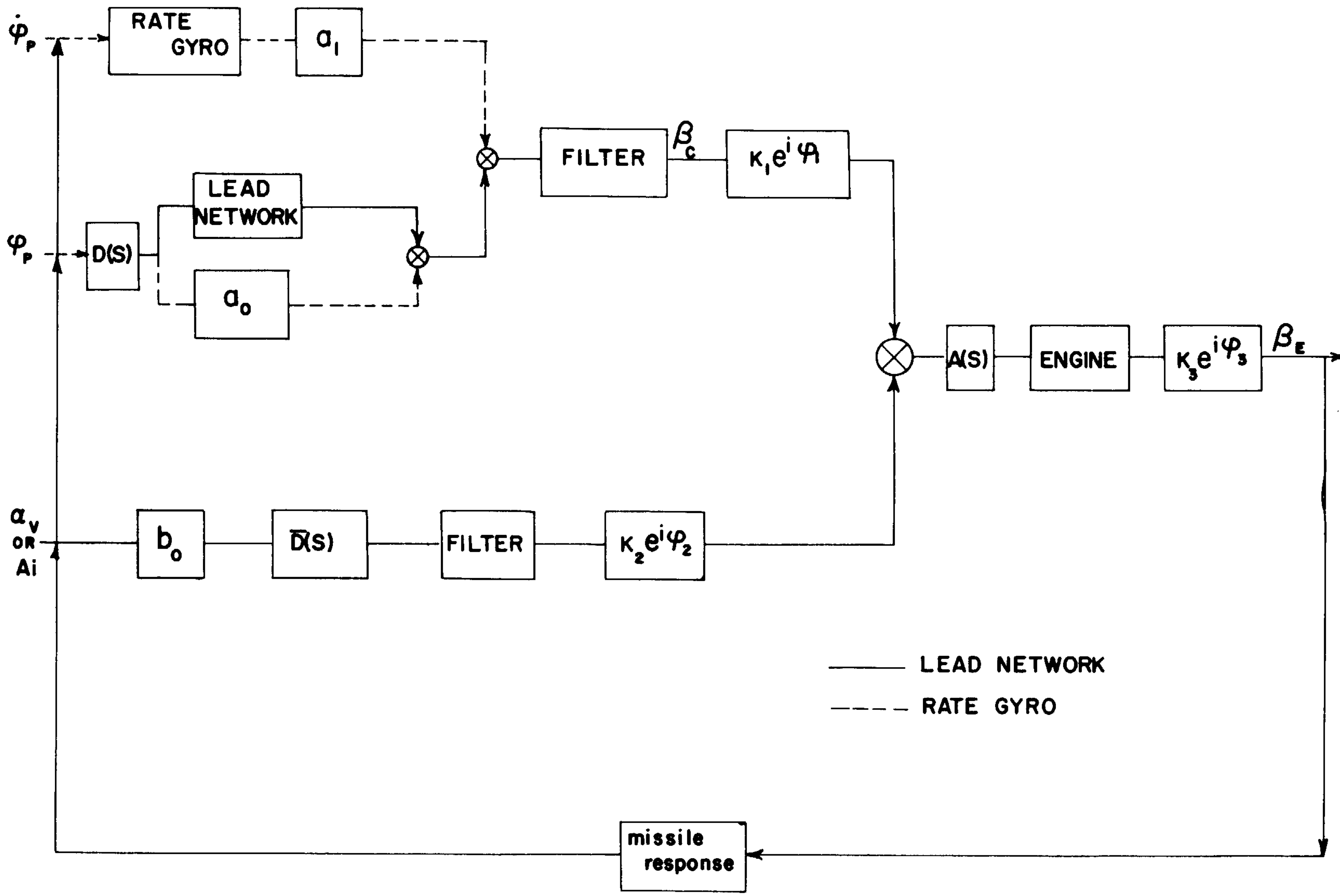

FIGURE 14.4 BLOCK DIAGRAM OF THE CONTROL SUBSYSTEM

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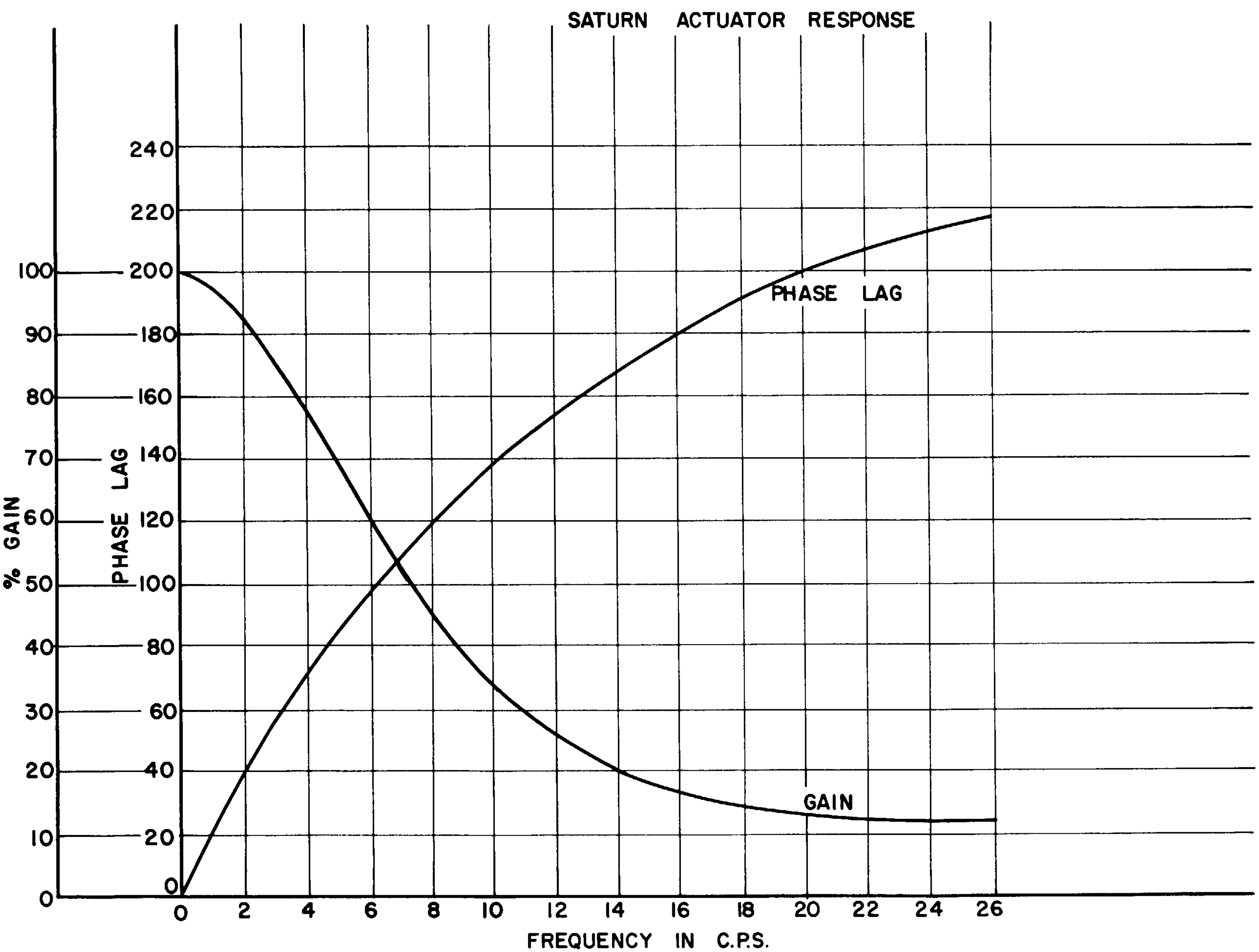

FIGURE 14.5 TRANSFER FUNCTION OF THE AMPLIFIER-ACTUATOR SYSTEM USED FOR THE INVESTIGATION

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It can be seen from the σ versus flight time plot of the sloshing roots (Fig. 14.6) that all tanks are stable if a damping of $g = .065$ is used throughout. This is equivalent to z-rings located just below the free propellant surface. It is observed that the S-IV lox tank needs more baffling at later flight time to insure good stability. As stated previously, ring baffles are needed. These would provide a damping of approximately $g = 0.13$.

Root locus plots for the four lateral bending modes are given in Figures 14.7 through 14.10. The first bending mode is unstable for all flight times, but can be stabilized by a phase shaping network. This network seems feasible since other like systems have been stabilized which had the same basic requirements (SA-1). The second bending mode is unstable, but can be stabilized either by phase shaping or the introduction of attenuation. Since an attenuation of 1 to 4 was already introduced in the study, additional attenuation of 1 to 8 would be required. This should be in the attenuation range obtainable with present passive network filters. The third bending mode could be gain-stabilized by introducing an additional attenuation of 1 to 2. Figure 14.9 at 70 and 100 seconds flight time represents this case. The fourth bending mode is stabilized for all flight times by an attenuation of 1 to 4.

It may be noted, that this investigation used the conservative structural damping factor of $\zeta = 0.01$; whereas a value of two times this magnitude can be expected. This, of course, will increase the stability.

The fin mode roots are stable for all flight times (Fig. 14.11 and 14.12). These are σ versus flight time plots. Figure 14.11 was made for an attenuation of 1 to 4 in the control loop and Figure 14.12 shows the results if an attenuation of 1 to 10 is used. No great effect is noticed from the change in attenuation of the control loop, indicating that very little feedback is observed. The main damping is apparently due to aerodynamics. The aerodynamic theory is invalid in the transonic range leading to a high jump in stability at 50 and 60 seconds flight times. After 60 seconds the Mach range of the space vehicle is well within the validity of the second order piston theory used. Figure 14.13 is a plot of the coupled angular frequency (ω) graphed versus flight time. Very little change is seen in these frequencies due to coupling.

Investigation of feedback stability in the yaw plane and in roll will also be necessary. However, for this proposal primary emphasis has been placed on the control feedback analysis in the pitch plane.

A control feedback analysis for the various stages will be performed, treating roll motion and torsion in addition to pitch motion, bending and sloshing, and treating fins as elastic members of the system. An IBM 7090 computer program is in the process of being modified to take these additional degrees of freedom. Furthermore, other control devices such as a lead network for rate damping, and accelerometers will be studied to obtain optimum stability with respect to the control modes (pitch and roll), bending torsion, and sloshing.

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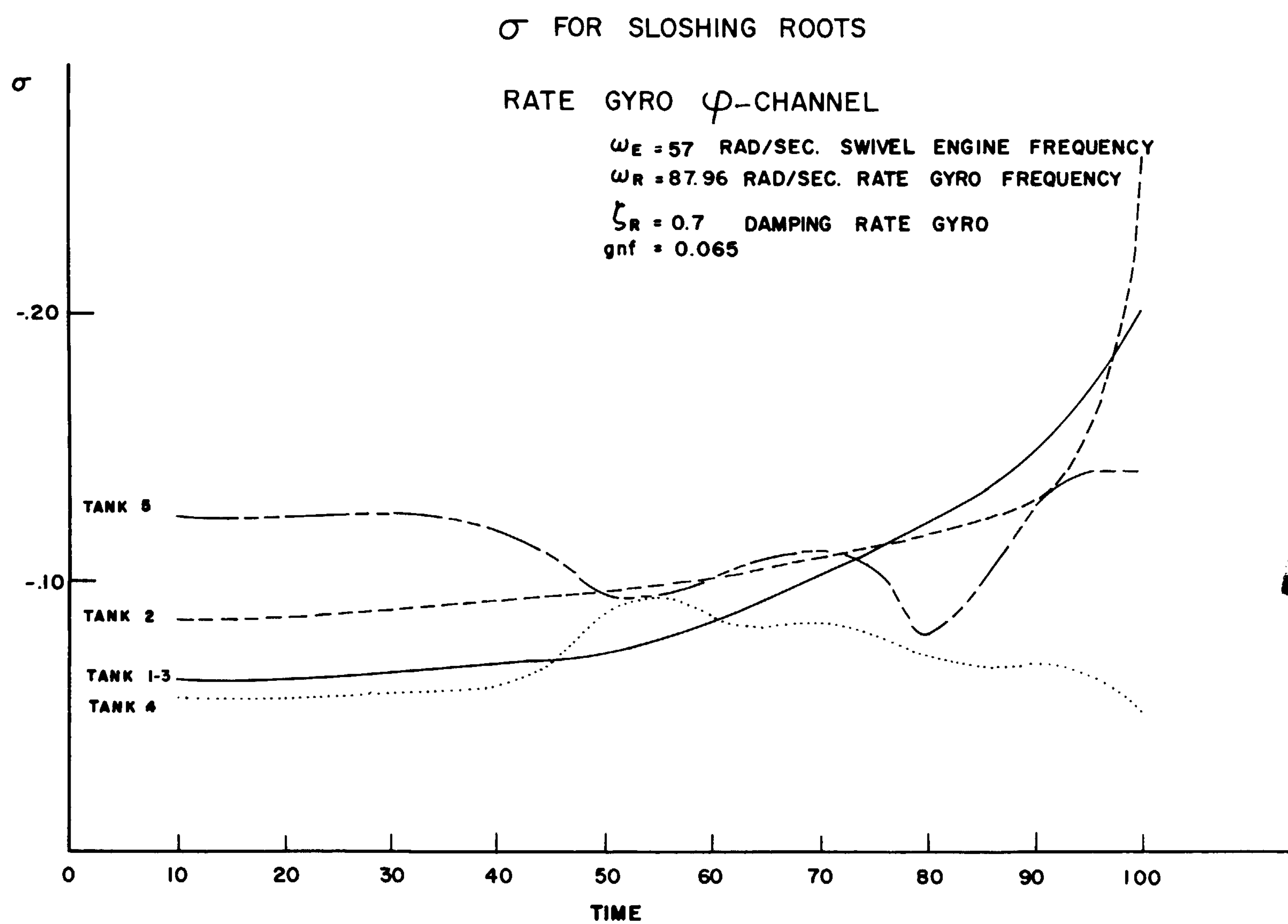

FIGURE 14.6 DAMPING RATE, σ, FOR SLOSHING ROOTS

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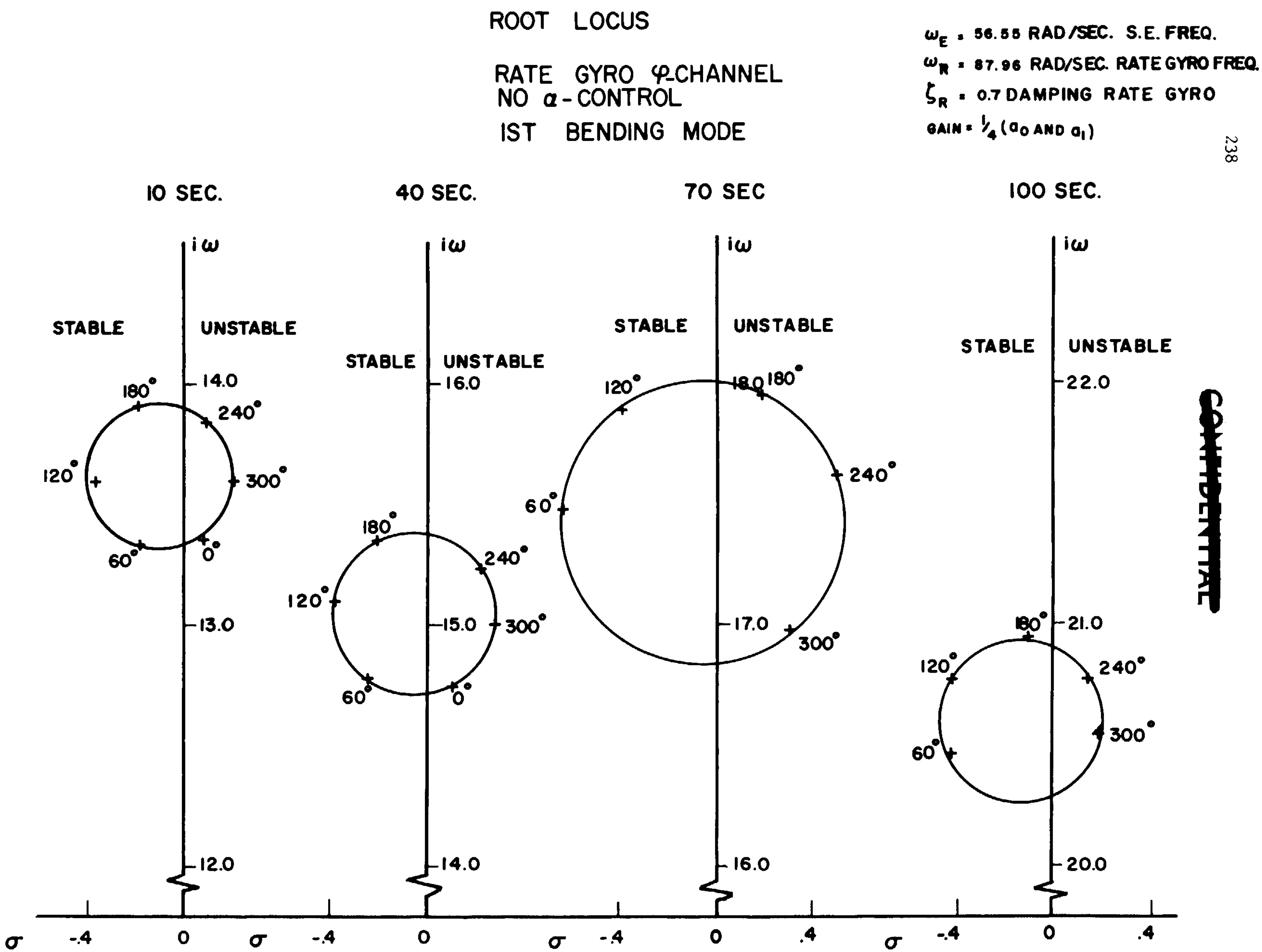

FIGURE 14.7 ROOT LOCUS FOR VEHICLE - FIRST LATERAL BENDING MODE

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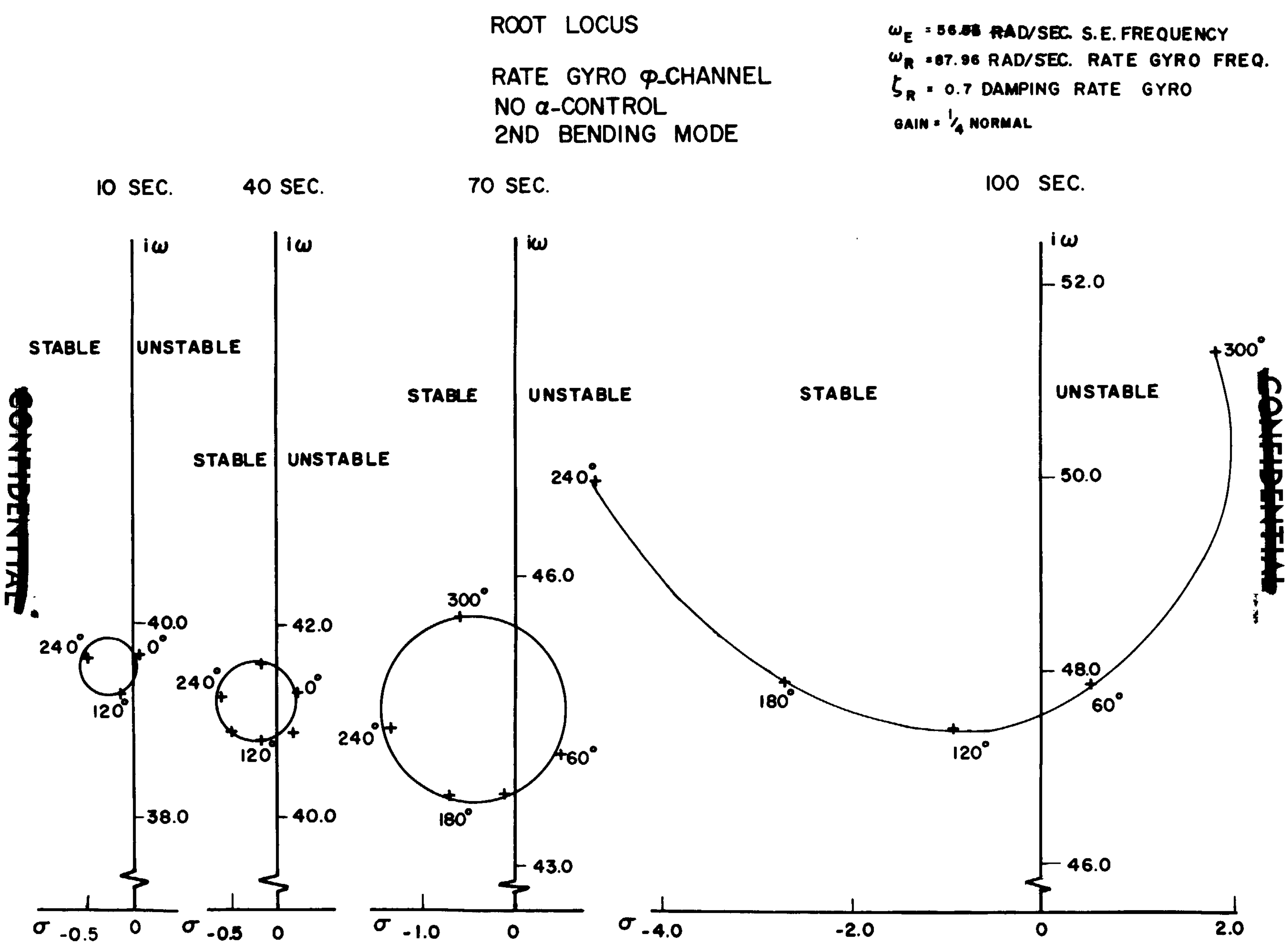

FIGURE 14.8 ROOT LOCUS FOR VEHICLE - SECOND LATERAL BENDING MODE

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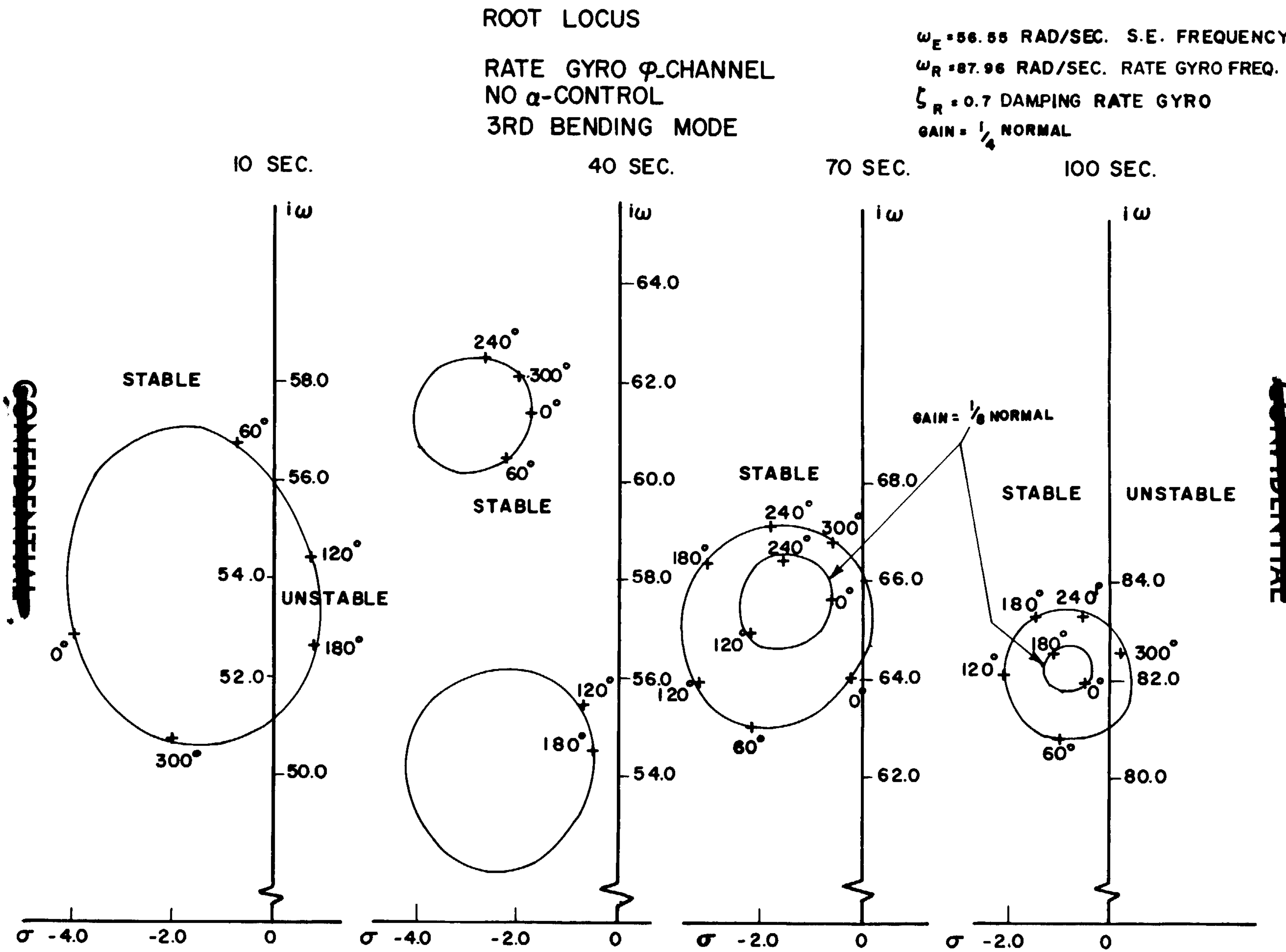

FIGURE 14.9 ROOT LOCUS FOR VEHICLE - THIRD LATERAL BENDING MODE

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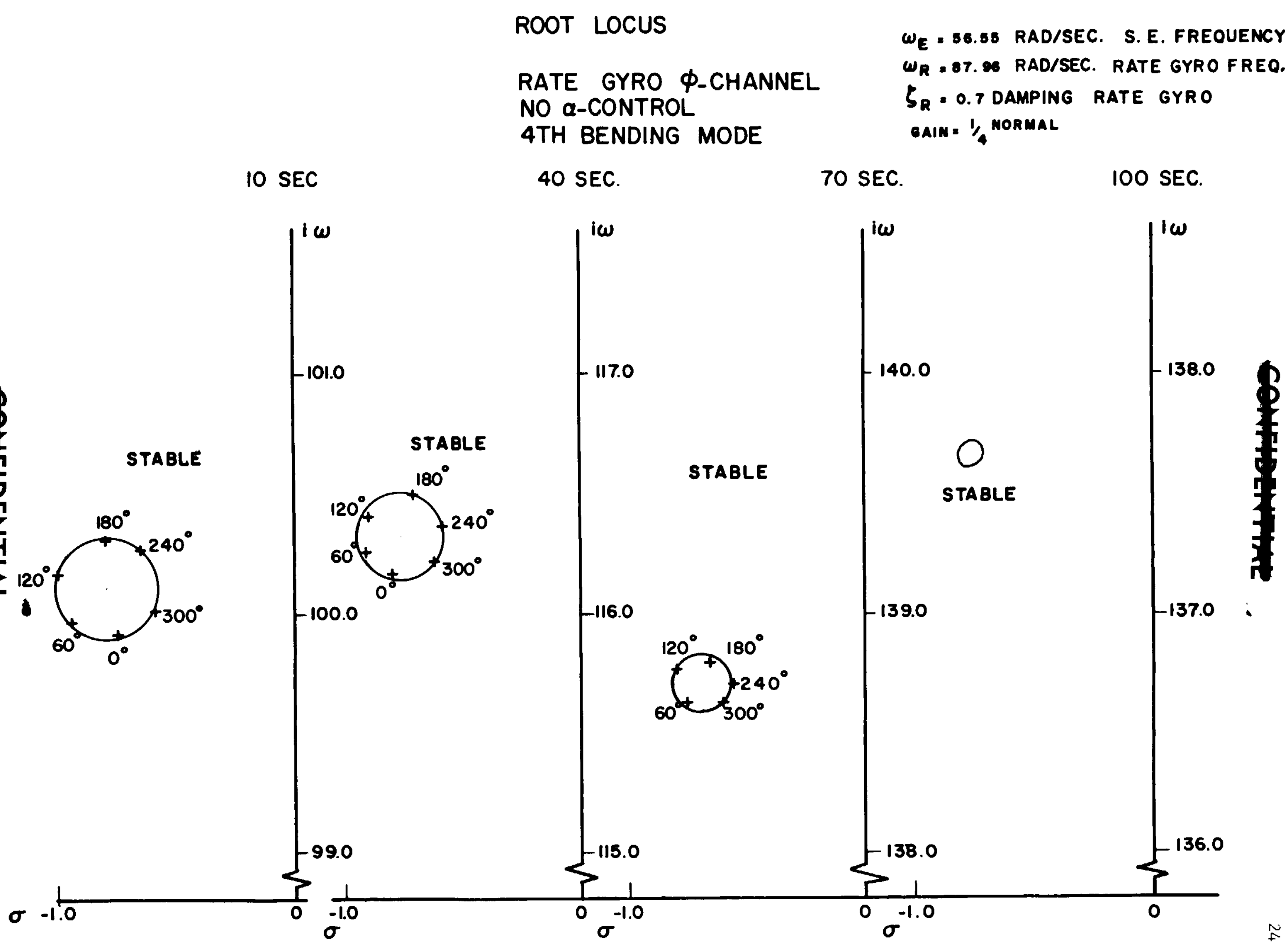

FIGURE 14.10 ROOT LOCUS FOR VEHICLE - FOURTH LATERAL BENDING MODE

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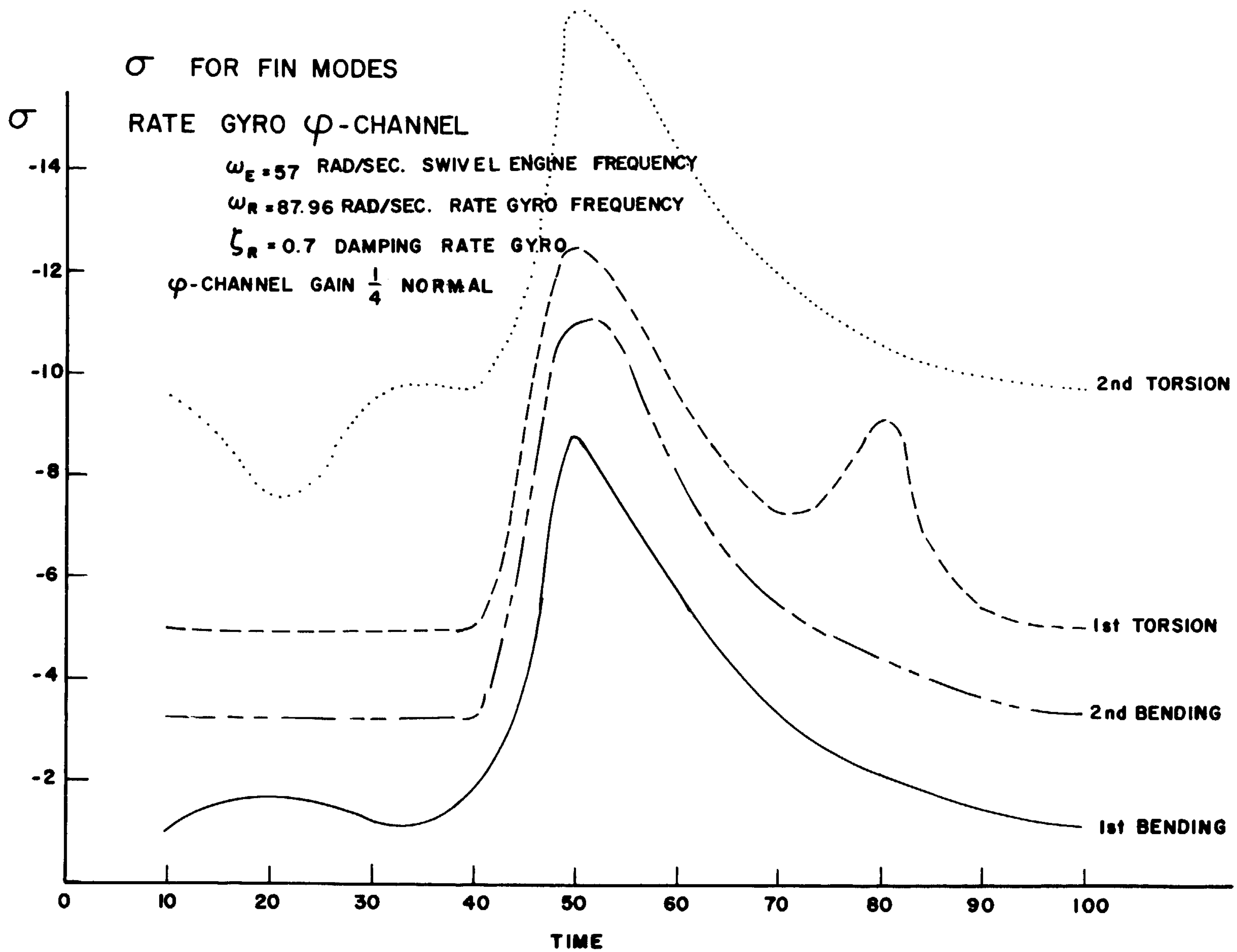

FIGURE 14.11 DAMPING RATE, σ, FOR FIN BENDING MODES, CONTROL GAINS ATTENTUATED 1 TO 4

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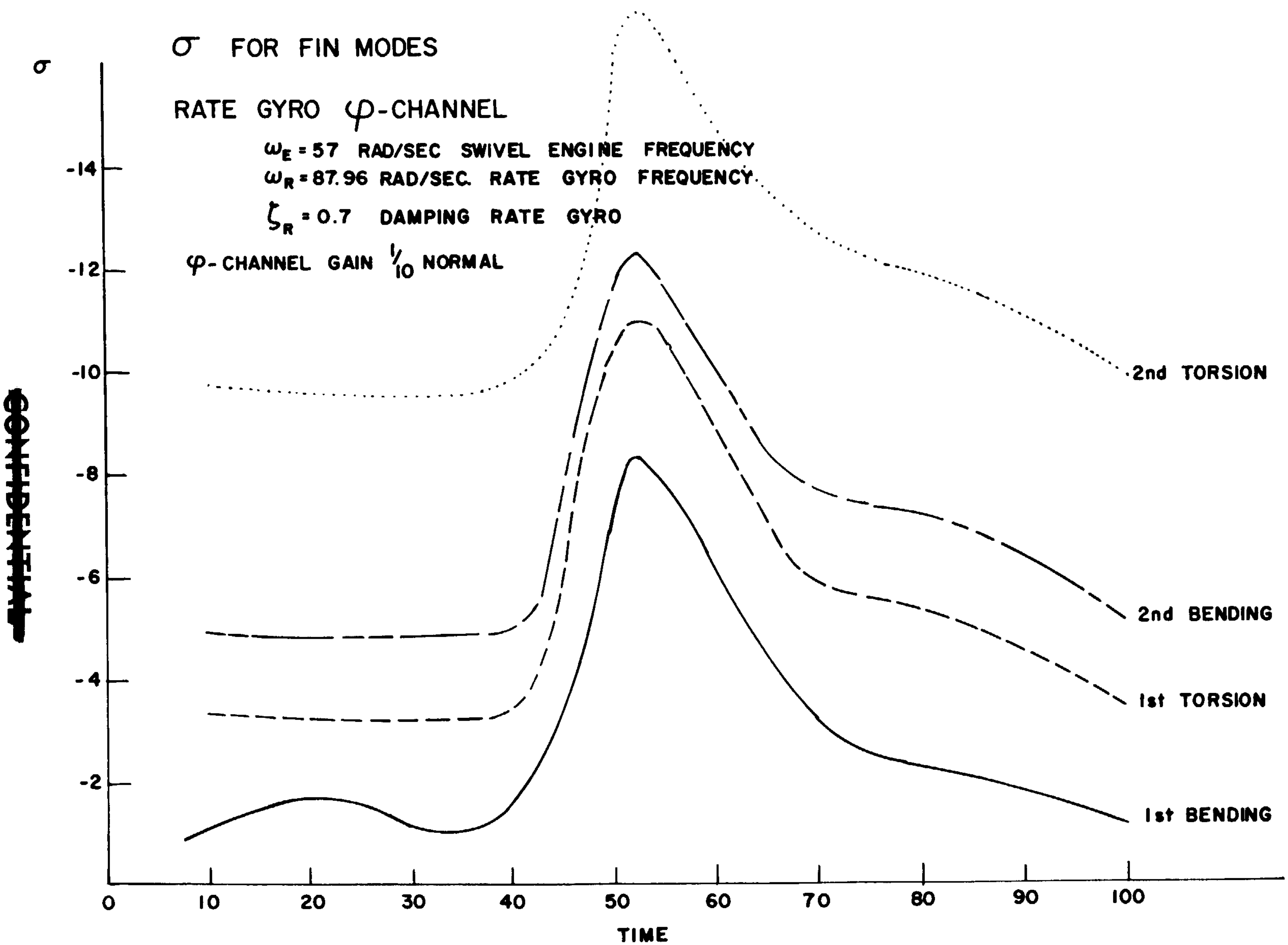

FIGURE 14.12 DAMPING RATE, σ, FOR FIN BENDING MODES, CONTROL GAINS ATTENTUATED 1 TO 10

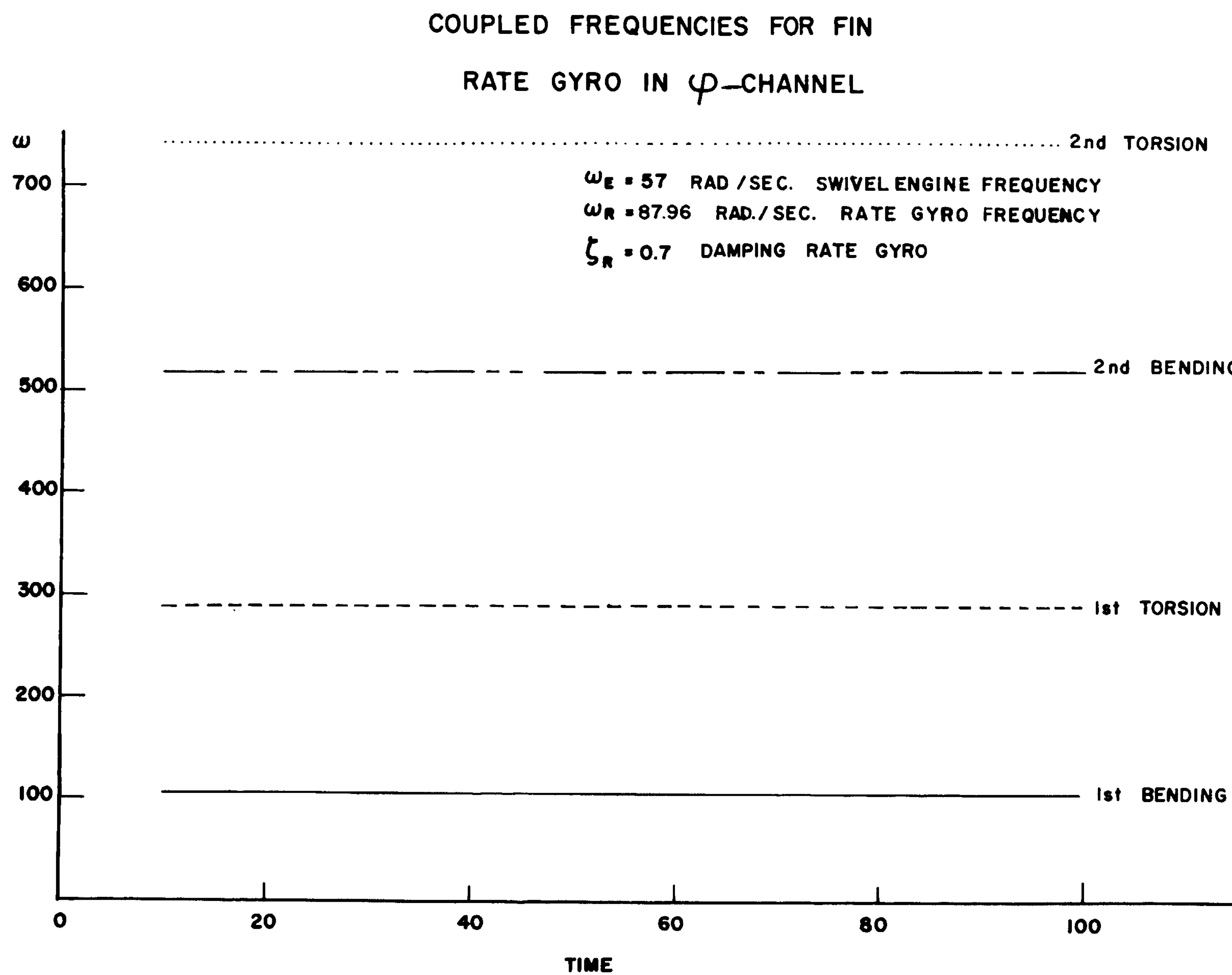

FIGURE 14.13 COUPLED ANGULAR BENDING FREQUENCIES FOR FINS

SECTION 15. (C) SPACE VEHICLE LOADS

15.1. WIND ENVIRONMENT LOADS

The Saturn C-1/Apollo "A" space vehicle has been designed to withstand the wind loads encountered at the launch site and in the different phases of the ascent.

15.1.1. Launch Site Wind Velocity Criteria. The wind velocities stated below include the effects of gusts. The given velocities are for a height of 10 feet above ground level. It should be noted that this will result in a higher peak wind speed at the upper levels of the vehicle as erected on the launch pad. Figure 9.6 provides the expected peak steady state wind speeds at altitudes from 3 to 120 meters above ground level, for the condition of 10 meters/sec at 10 feet. These values are based on wind measurements at Cape Canaveral and are applicable only for launch sites at the Cape.

With ground wind velocities up to 40 knots, the Saturn vehicle, empty or fueled, shall withstand the resulting load condition while free standing on the launch pad. During severe weather warnings, with predicted wind velocities greater than 40 and less than 65 knots, the first and second stages must be secured to the service structure and the upper stages will not be removed from the vehicle.

If predicted wind velocities are greater than 65 knots, the first and second stages will be secured to the service structure with the upper stages remaining and the fuel and lox tanks must be pressurized to a nominal stabilizing pressure of approximately 12 psig.

The Saturn C-1/Apollo "A" space vehicle has been designed for launching in a peak wind speed of 28 knots [20 knots ($\approx$ 10 m/sec) steady state plus 8 knots ($\approx$ 4 m/sec) gust] at 10 feet above ground.

15.1.2. Ascent Wind Velocity Criteria. The Saturn C-1/Apollo "A" is designed for the maximum inflight loads which occur in the maximum dynamic pressure region and where the most severe wind condition also occurs.

The maximum wind velocity at maximum dynamic pressure results in a steady state wind angle-of-attack of 9.5 degrees. The drift minimum principle (DMP) is utilized to reduce the steady state angle-of-attack to approximatley 4.7 degrees. Details of the wind induced angle-of-attack and the drift minimum principal are discussed in Section 14.

For load computations, values of the angle-of-attack are chosen to represent a flight condition that can still be corrected by the available control moment.

15.2. STRUCTURAL STRENGTH REQUIREMENTS

15.2.1. General. The following definitions and terms are used for design of the Saturn C-1 stage to establish uniform nomenclature with respect to loads, safety factors, etc. The factors of safety to be used in the design of structures, components, and ground equipment are listed herein.

15.2.2. Definitions

15.2.2.1. Factor of Safety. Factor of safety is the ratio of the design load on a structure to the maximum calculated operational load experienced by the structure under specified conditions of operation.

15.2.2.2. Yield Factor Safety. Yield factor safety is the ratio of the design yield load on a structure to the unit load.

15.2.2.3. Ultimate Factor of Safety. Ultimate factor of safety is the ratio of the design ultimate load on a structure to the limit load.

15.2.2.4. Normal Load Factor. Normal load factor is the ratio of the normal calculated operational load experienced by the structure under specified condition of operation to the design load. The normal load factor is also the inverse of the normal factor of safety.

15.2.2.5. Margin of Safety. Margin of safety is the percentage by which the allowable load or stress exceeds the design load or stress.

15.2.2.6. Limit Load. Limit load is the maximim calculated load which will be experienced by the structure under the specified conditions of operation.

15.2.2.7. Design Ultimate Load. The design ultimate load is the limit load multiplied by the ultimate factor of safety.

15.2.2.8. Design Yield Load. The design yield load is the limit load multiplied by the yield factor of safety.

15.2.2.9. Allowable Yield Load. The allowable yield load is that load which must be applied to the structure in order to cause a permanent deformation of a specified amount.

15.2.2.10. Allowable Yield Stress. The allowable yield stress is that stress at which the material exhibits a permanent deformation of 0.0020 inch per inch (0.2 percent).

15.2.2.11. Allowable Ultimate Load. The allowable ultimate load is the load at which catastrophic failure of the structure occurs. A catastrophic failure is defined as a failure producing general collapse or instability buckling.

15.2.2.12. Allowable Ultimate Stress. The allowable ultimate stress is the stress at which a material fractures or ruptures.

15.2.2.13. Operating Pressure. The operating pressure is the nominal pressure to which the components are subjected under steady state conditions in service operations. The term is also synonymous with working pressure. The maximum operating pressure is the nominal pressure plus tolerance.

15.2.2.14. Maximum Relief Valve Pressure. Maximum relief valve pressure is the pressure at which the relief valve, or vent valve, is fully open.

15.2.2.15. Proof Pressure. Proof pressure is the pressure which the component is subjected to during acceptance testing. It is equal to the maximum relief valve pressure, or maximum operating pressure, plus hydrostatic head (if applicable), multiplied by the appropriate factor of safety.

15.2.2.16. Design Yield Pressure. Design yield pressure is the proof pressure multiplied by the appropriate factor of safety.

15.2.2.17. Design Burst Pressure. Design burst pressure is the maximum relief valve pressure, or maximum operating pressure, plus hyderostatic head (if applicable), multiplied by the appropriate ultimate factor of safety.

15.2.2.18. Allowable Yield Pressure. The allowable yield pressure is the pressure at which the component material is stressed such that the material exhibits a permanent deformation of 0.0020 inch per inch, 0.2 percent.

15.2.2.19. Allowable Ultimate Pressure. Allowable ultimate pressure is the pressure at which the component material is stressed to produce complete rupture or bursting of the component. This is also the pressure that produces instability buckling when the pressure is applied externally.

15.2.2.20. Combined Stresses. Combined stresses are defined as the stresses resulting from consideration of simultaneous action of all factors such as direct loadings, thermal stresses, creep, fatigue, etc. The safety factors listed below are to be applied in cases involving such simultaneous action.

15.3. FACTORS OF SAFETY

The following safety factors are applicable to the stage and components.

Yield Factor of Safety	Ultimate Factor of Safety
1.10	1.35

These safety factors are applied in the simultaneous action of combined loads induced, including thermal stresses, stress concentration, creep, fatigue, etc.

15.3.1. Propellant Containers.

Proof Pressure: 1.10 x (Maximum relief valve pressure + hydrostatic head)

Design Yield Pressure: 1.10 x (Proof Pressure)

Burst Pressure: 1.35 x (Maximum relief valve pressure + hydrostatic head)

Structural Safety Factor (Including Surge Pressure): For container openings, flange or welded ducted connections to the tank gauge or structure, and other areas of stress concentration:

Proof Pressure 2.00
Burst Pressure 4.00

15.3.2. Mechanical Components (Safety Factors)

15.3.2.1. High Pressure Pheumatic Systems (1,500 psi and above)

15.3.2.1.1. Lines, Fittings, and Hose

Proof Pressure: 2.00 x (Maximum operating pressure)

Burst Pressure: 4.00 x (Maximum operating pressure)

15.3.2.1.2. Air Reservoirs

Proof Pressure: 1.67 x (Maximum operating pressure)

Design Yield Pressure: 1.10 x (Proof pressure)

15.3.2.1.3. Actuating Cylinders and Other Components

Proof Pressure: 1.50 x (Maximum operating pressure)

Burst Pressure: 2.50 x (Maximum operating pressure)

15.3.2.2. Low Pressure Pneumatic Systems (Propellant loading, venting, drain systems, etc.)

Proof Pressure: 1.50 x (Maximum operating pressure)

Burst Pressure: 2.50 x (Maximum operating pressure)

15.3.2.3. Tubing and Flexible Connectors (Less than 1.5 inch nominal diameter)

Proof Pressure: 2.00 x (Maximum operating pressure)

Burst Pressure: 4.00 x (Maximum operating pressure)

15.3.2.4. Miscellaneous Items

Bolts, Rivets, and Bearings	In accordance with MIL-HDBK-5
Castings, All types	Use MIL-C-6021 as a guide
Fusion and Spot Welds	Use MIL-HDBK-5 and verify by test
Resistance Seam Welds	Establish by test

15.3.3. Ground Support Equipment Factors of Safety

15.3.3.1. Pressure Vessels

(1) Yield factor of safety = 1.6 x design load

(2) Ultimate factor of safety = 4.0 x design load

15.3.3.2. Unpressurized Structures

(1) Yield factor of safety = 1.5 x design load

(2) Ultimate factor of safety = 3.0 x design load

The factors of safety above are considered to be minimum. They are to be used in addition to the considerations given to vibration magnification factors, allowable deflections and shock factors on such structural components as brackets, handling eyes or hooks, etc. Safety factors different from these herein specified shall be subject to the approval of the procuring agency.

15.4 LOADS IN THE SATURN C-1/APOLLO "A"

The predicted loads in the Saturn C-1/Apollo "A" space vehicle on the launch pad and in flight are given in the following figures:

Figures 15.1 and 15.2 show the bending moment and shear loads for the flight-ready vehicle under 28 knot wind conditions.

Figures 15.3 and 15.4 show the bending moment and shear loads for the flight-ready vehicle under 40 knot wind condition.

Figures 15.5 and 15.6 show the bending moment and shear loads for t = 66 sec ("q" maximum condition), 8 engine-operation.

Figure 15.7 shows the longitudinal force for t = 66 sec ("q" maximum condition), 8 engine operation.

Figures 15.8 and 15.9 show the bending moment and shear loads for t = 66 sec ("q" maximum condition), 7 engine-operation.

Figure 15.10 shows the longitudinal force for t = 66 sec ("q" maximum condition), 7 engine operation.

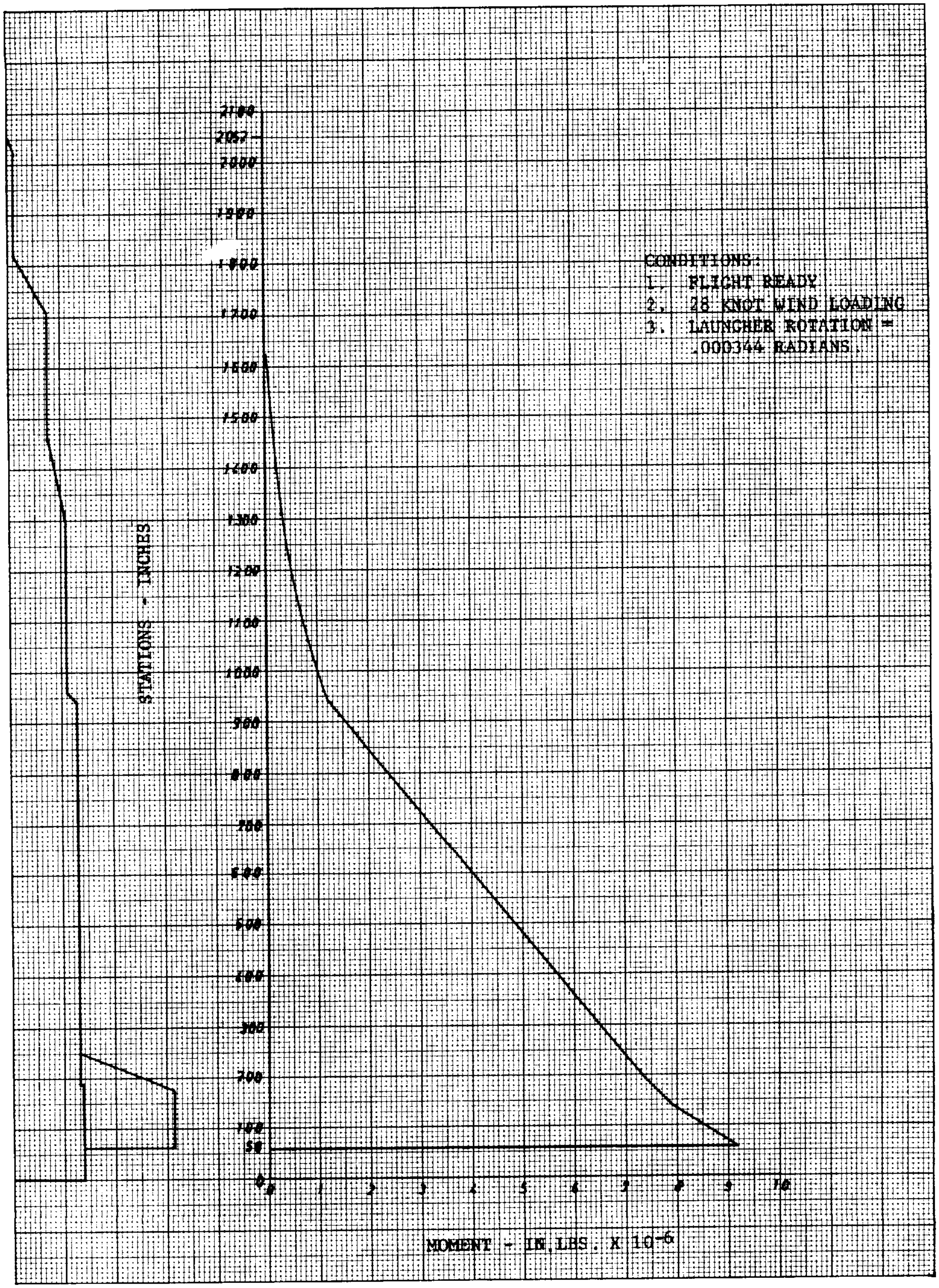

FIGURE 15.1 MOMENT VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE

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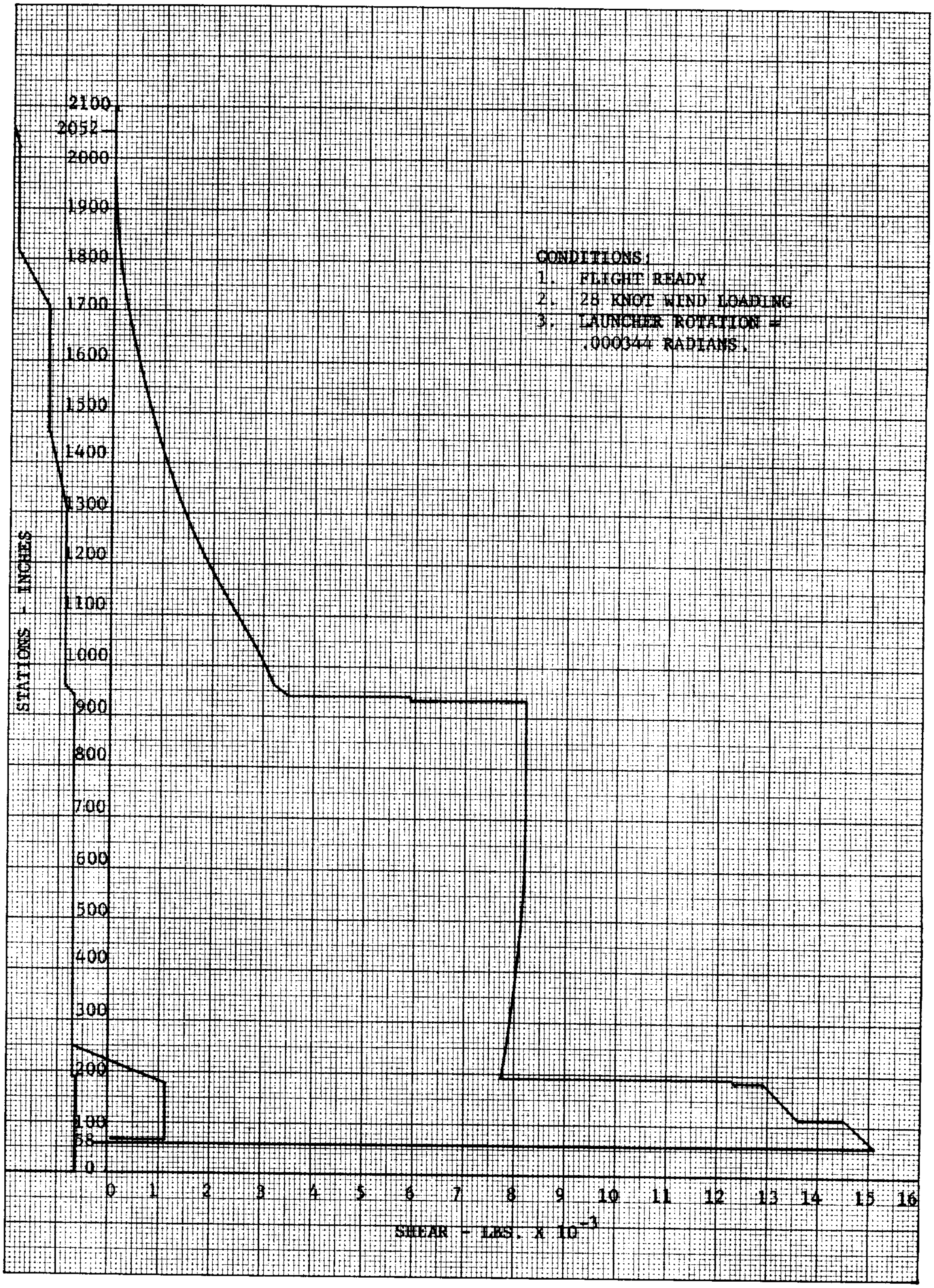

FIG. 15.2 SHEAR VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE

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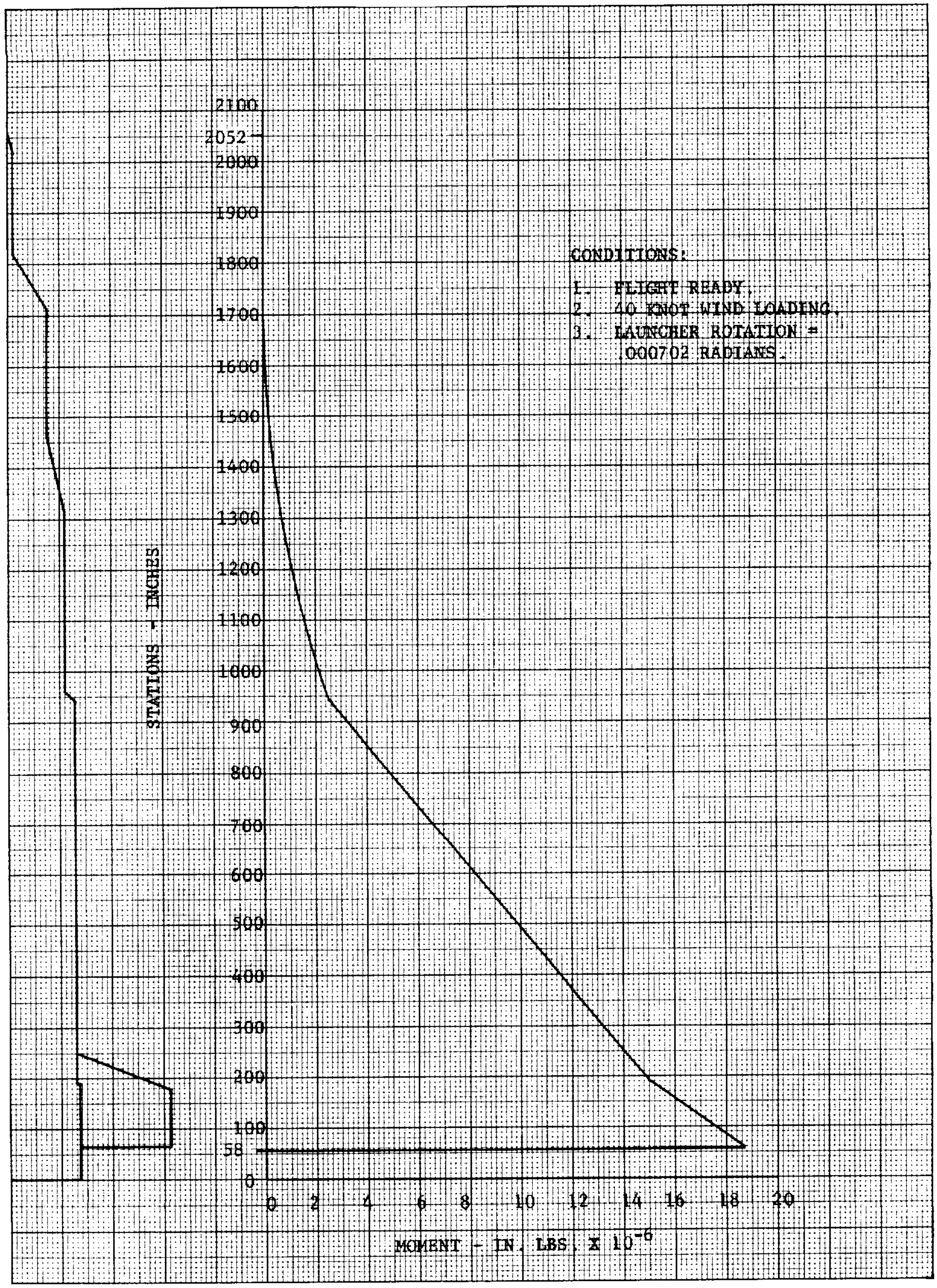

FIG. 15.3. MOMENT VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE

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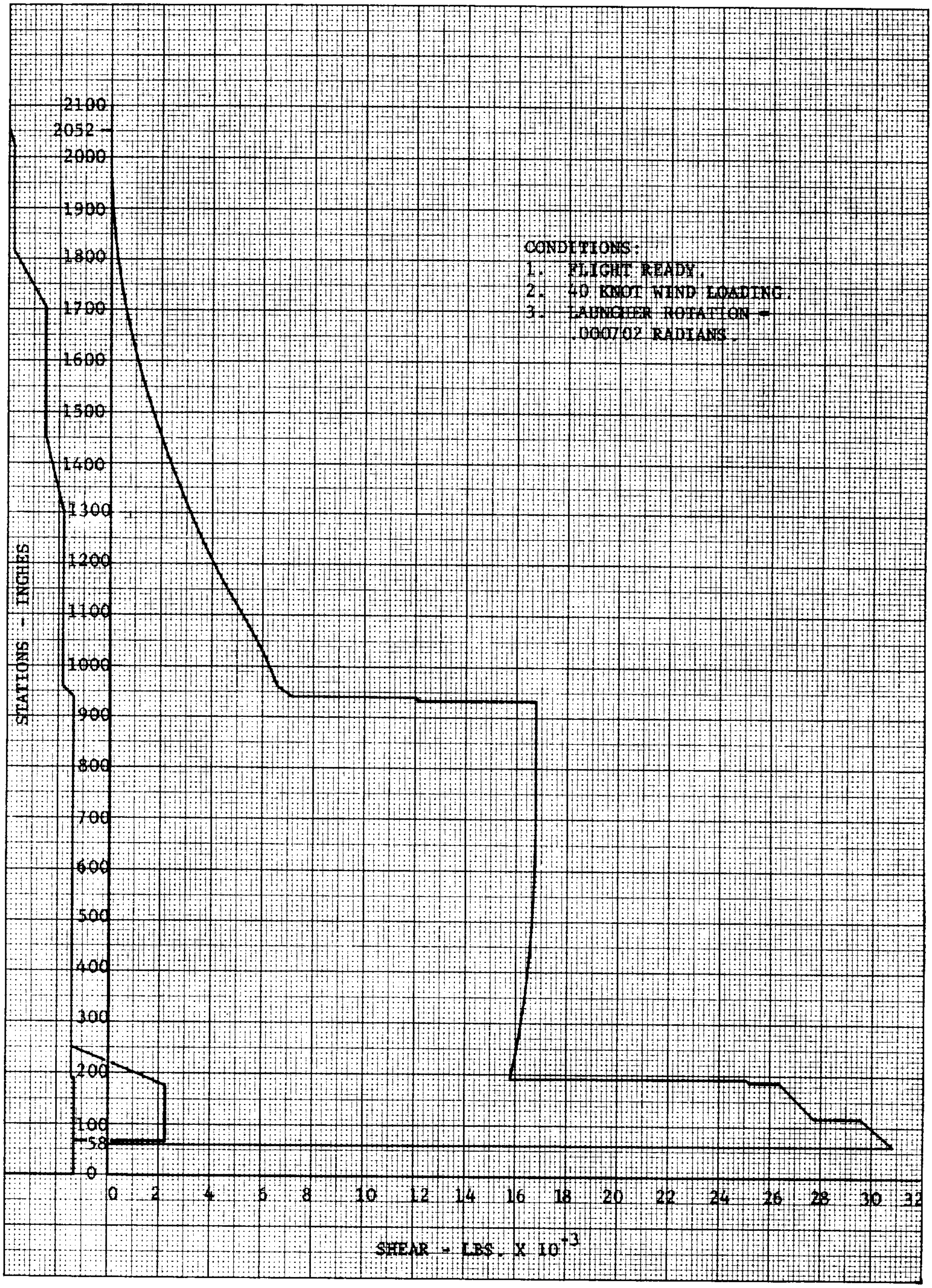

FIG. 15.4 SHEAR VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE

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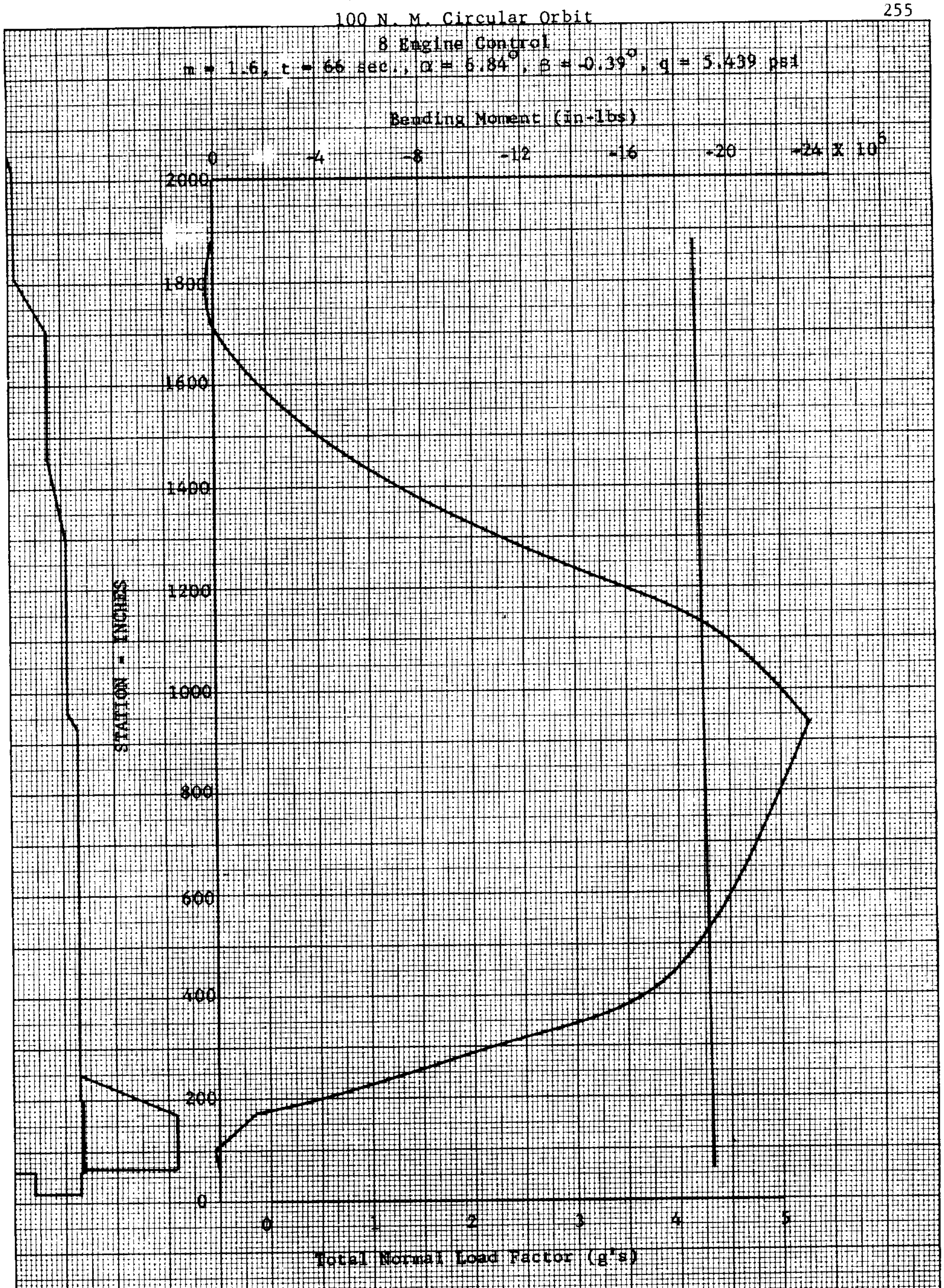

FIG. 15.5 BENDING MOMENT VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS OF 8-ENGINE OPERATION

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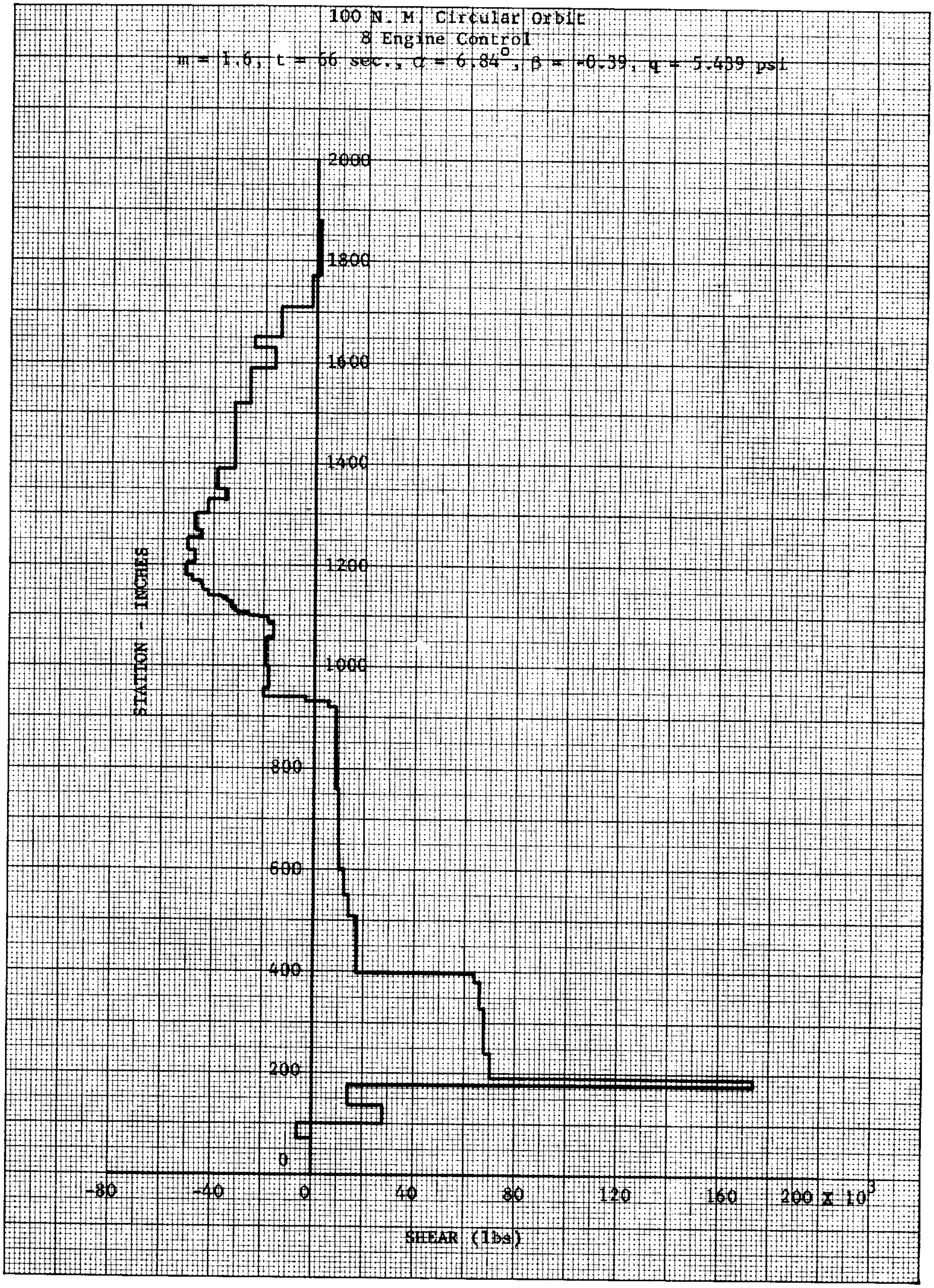

FIG. 15.6 SHEAR VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS OF 8-ENGINE OPERATION

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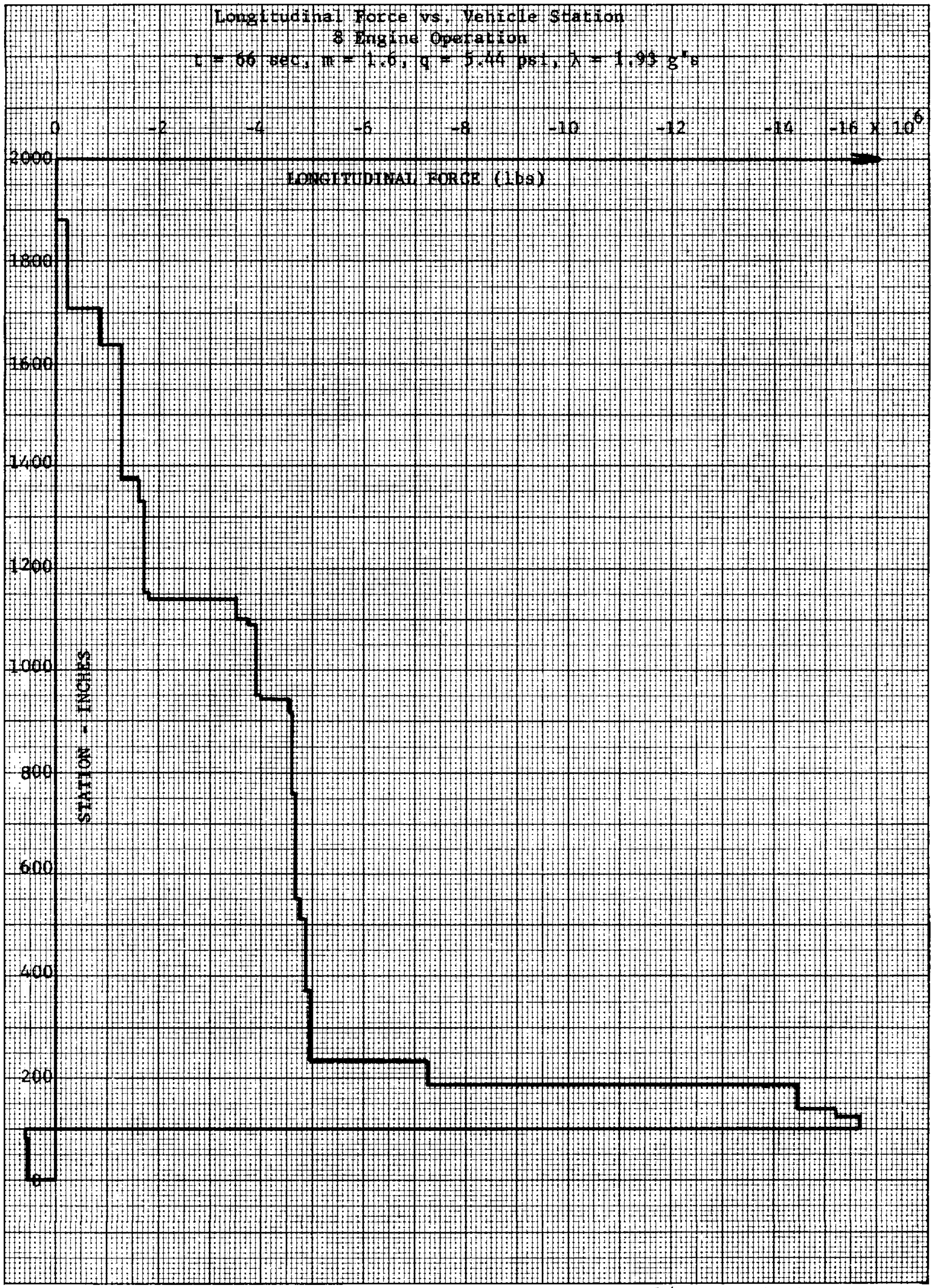

FIG. 15.7 LONGITUDINAL FORCE VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS OF 8-ENGINE OPERATION

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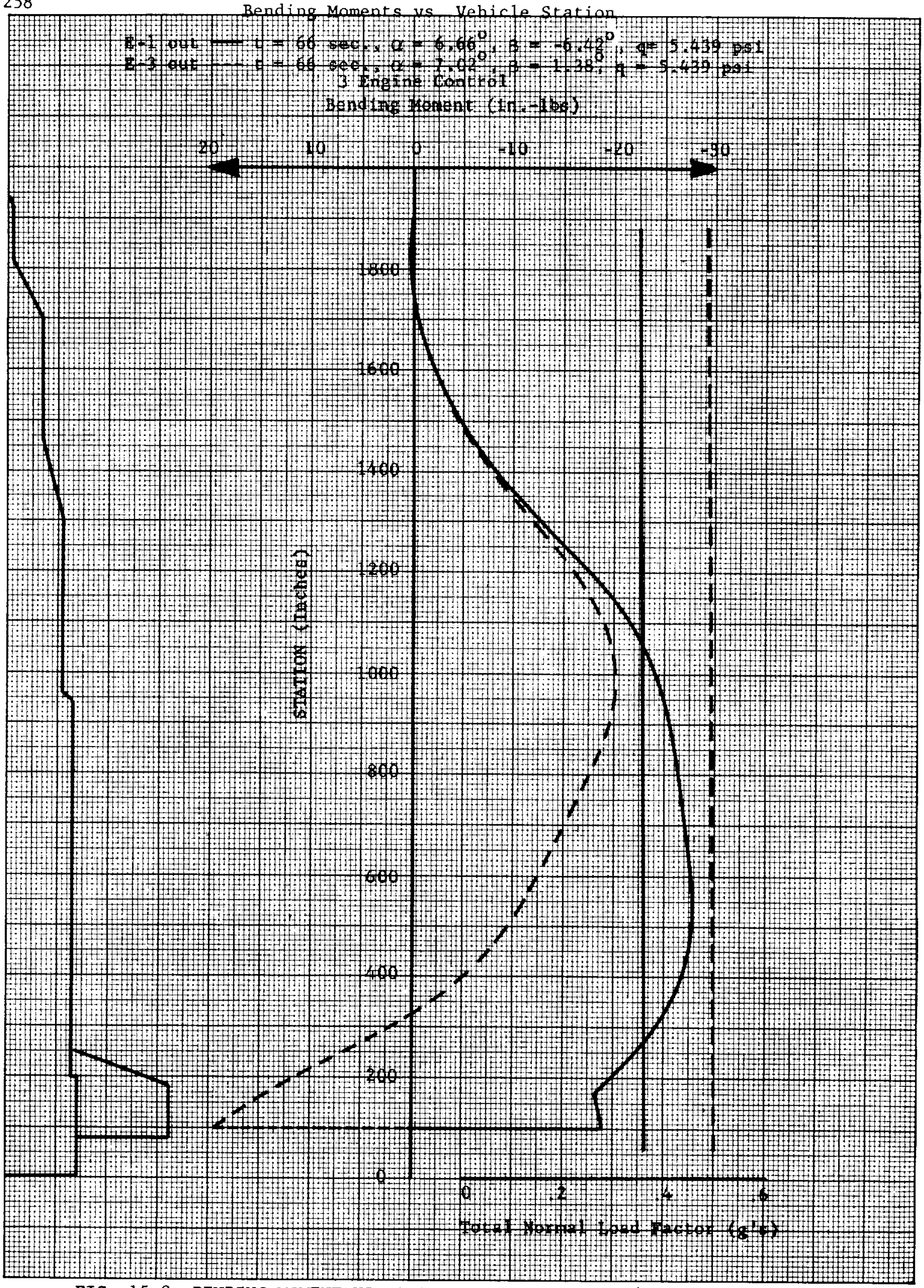

FIG. 15.8 BENDING MOMENT VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS WITH ONE CONTROL ENGINE OUT

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Shear vs. Vehicle Station

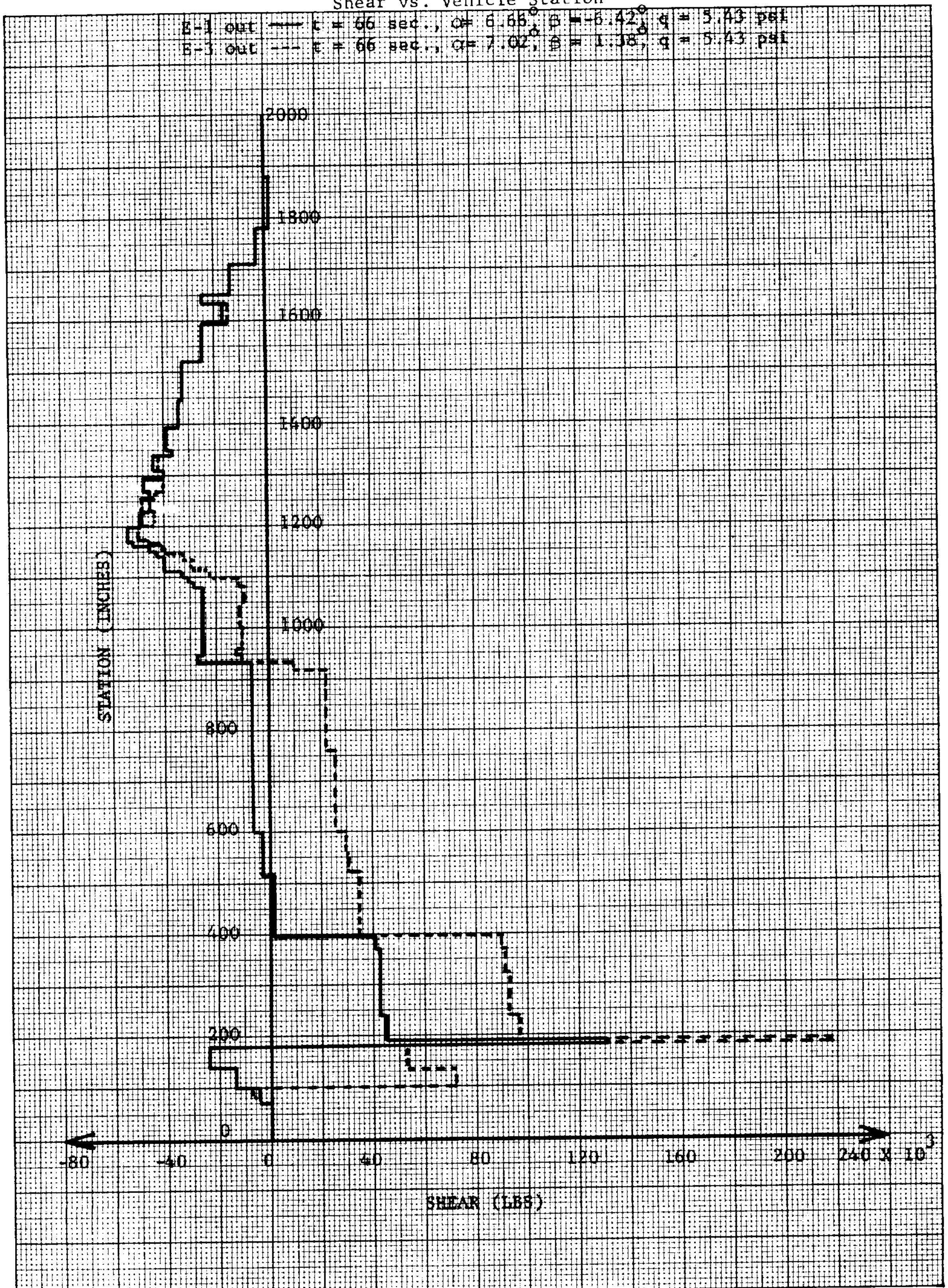

FIG. 15.9 SHEAR VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS WITH ONE CONTROL ENGINE OUT.

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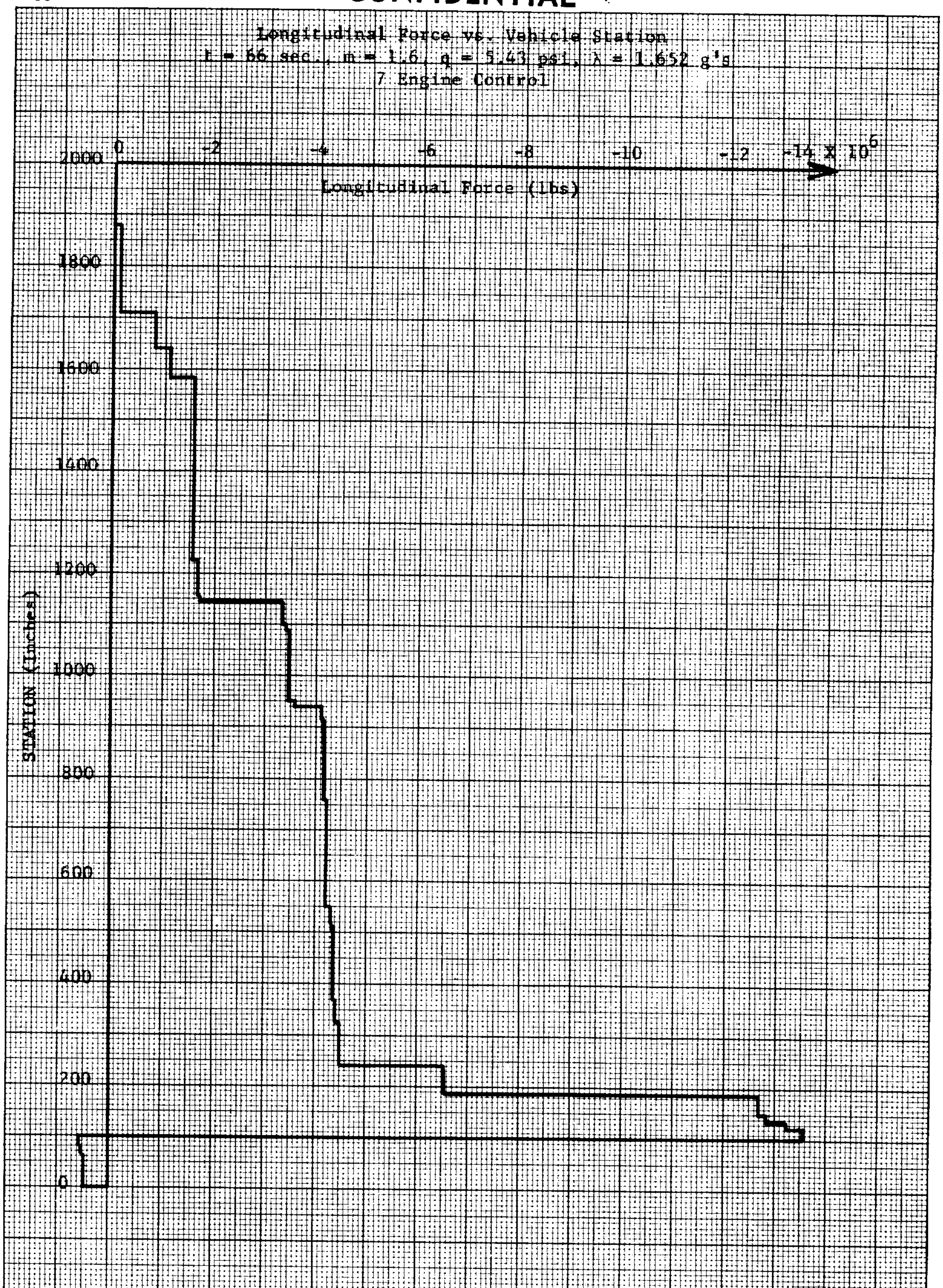

FIG. 15.10 LONGITUDINAL FORCE VS. STATION FOR SATURN C-1/APOLLO LAUNCH VEHICLE AT t = 66 SECONDS OF 7-ENGINE OPERATION

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SECTION 16. (C) DYNAMIC TESTING

16.1. DYNAMIC TEST OPERATIONS

The purpose of the dynamic test operations is to obtain some essential information on the dynamic behavior of the Saturn vehicle and its stages in full-scale configuration. The dynamic effects of active components, such as engines and propellants, on the stages and the spacecraft, and on the entire vehicle may be investigated.

For these dynamic tests the space vehicle is either erected on launcher arms (as on the launch pad) or is suspended on cables for the simulation of free flight conditions. Electromagnetic shakers, which are attached between the structure of the dynamic test stand and the space vehicle, are used to excite bending, torsional, or longitudinal modes of vibration. The shakers are attached to the space vehicle at various external stations near the antinodes of the modes of interest. By scanning a range of excitation frequencies of known amplitude and phase, and by employment of suitable sensors, it is possible to determine the significant dynamic dharacteristics of the vehicle. These characteristics may be described as shapes of bending, torsional, or longitudinal modes of vibration, as mode frequencies and their associated damping coefficients, and as transfer functions between inputs (shakers) and outputs (sensors).

16.1.1 Cantilever Modes. Tests are made on the cantilever modes with the space vehicle supported on the eight launcher arms in a fully loaded condition. Data are secured for the bending modes and for torsional modes. These modes may be excited by winds while the space vehicle is erected on the launch pad.

16.1.2. Free Flight Simulation. In order to obtain information on the dynamic behavior in simulated preflight conditions, the space vehicle is suspended at some or all of its eight launch support points in a support system consisting of: beams of the dynamic test stand, damping cylinders, spring systems, and cables that lead, with a slope of 2° off the vertical, to the launch support points (Fig. 16.1). The spring suspension system is designed with sufficient stiffness to maintain vertical stability, but it is soft enough to prevent interference or mode coupling of the lateral bending and torsion modes of the almost free space vehicle. This suspension system produces a negligible error in the test results.

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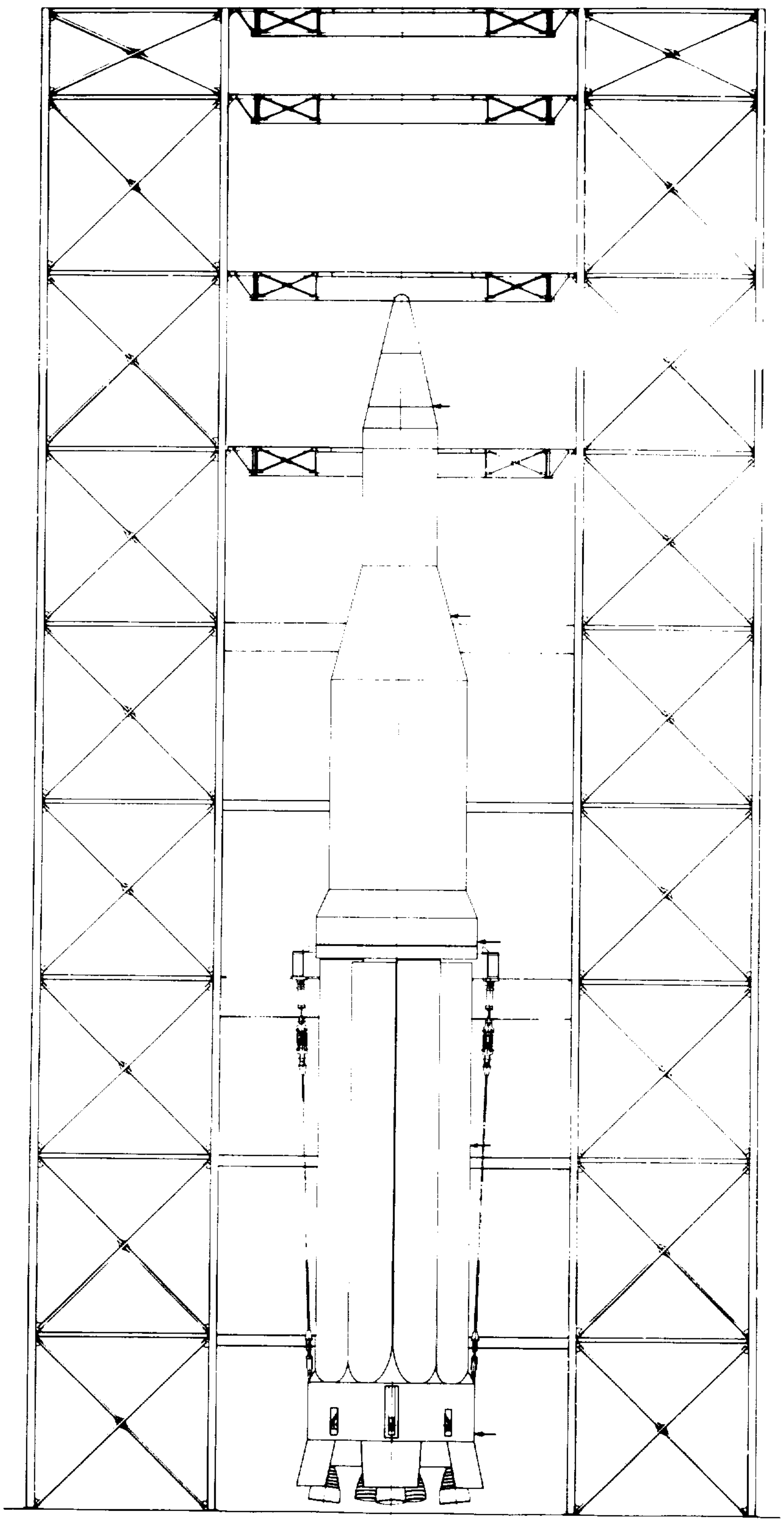

FIG. 16.1 SATURN SUSPENSION IN DYNAMIC TEST STAND

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16.1.3. The First Three Free-Free Bending Modes. These are of particular importance because of their effects on the flight control system of the space vehicle. The first two torsional modes also are of interest.

Flight conditions will be simulated, particularly for the following propellant load conditions:

(1) Liftoff (fully loaded)

(2) Maximum dynamic pressure

(3) Cutoff of first stage

(4) Ignition and cutoff of upper stages.

The completely dry space vehicle is also tested in the free-free suspension although this is not a flight condition. Conditions at other times of flight will be simulated as required.

Some of the initial vibration tests of fully and partially loaded space vehicles are being made by simulating propellants with deionized water. The temperature and time effects on the structures and on the vibrational modes caused by cryogenic propellants such as liquid oxygen and liquid hydrogen also may be determined.

Propellant sloshing and its coupling with structural vibration modes will be investigated, particularly in the nearly full condition of the flight. The determination of the sloshing effects at a later flight time by the simulation method is complicated by the lack of the proper longitudinal acceleration of the space vehicle. From the measurements, theoretical methods give good estimates of the sloshing effects throughout the flight. Sloshing modes in the vicinity of structural modes change the frequency and increase the damping of the space vehicle modes.

16.2. DYNAMIC TEST FACILITIES

The dynamic test facilities for the Saturn space vehicle are located at MSFC in Huntsville, Alabama. These facilities have been provided for the dynamic test operations which were described above.

The dynamic test stand itself provides support to the full-scale, full-weight Saturn space vehicle either on eight launcher arms (cantilever) or suspended in a simulated free flight condition, (Fig. 16.2). The steel super-structure also accommodates a number of work platforms for access to the desired stations of the vehicle. The dynamic test stand presently has a height of 204 feet (not counting the derrick) and contains about 600 tons of structural steel. This

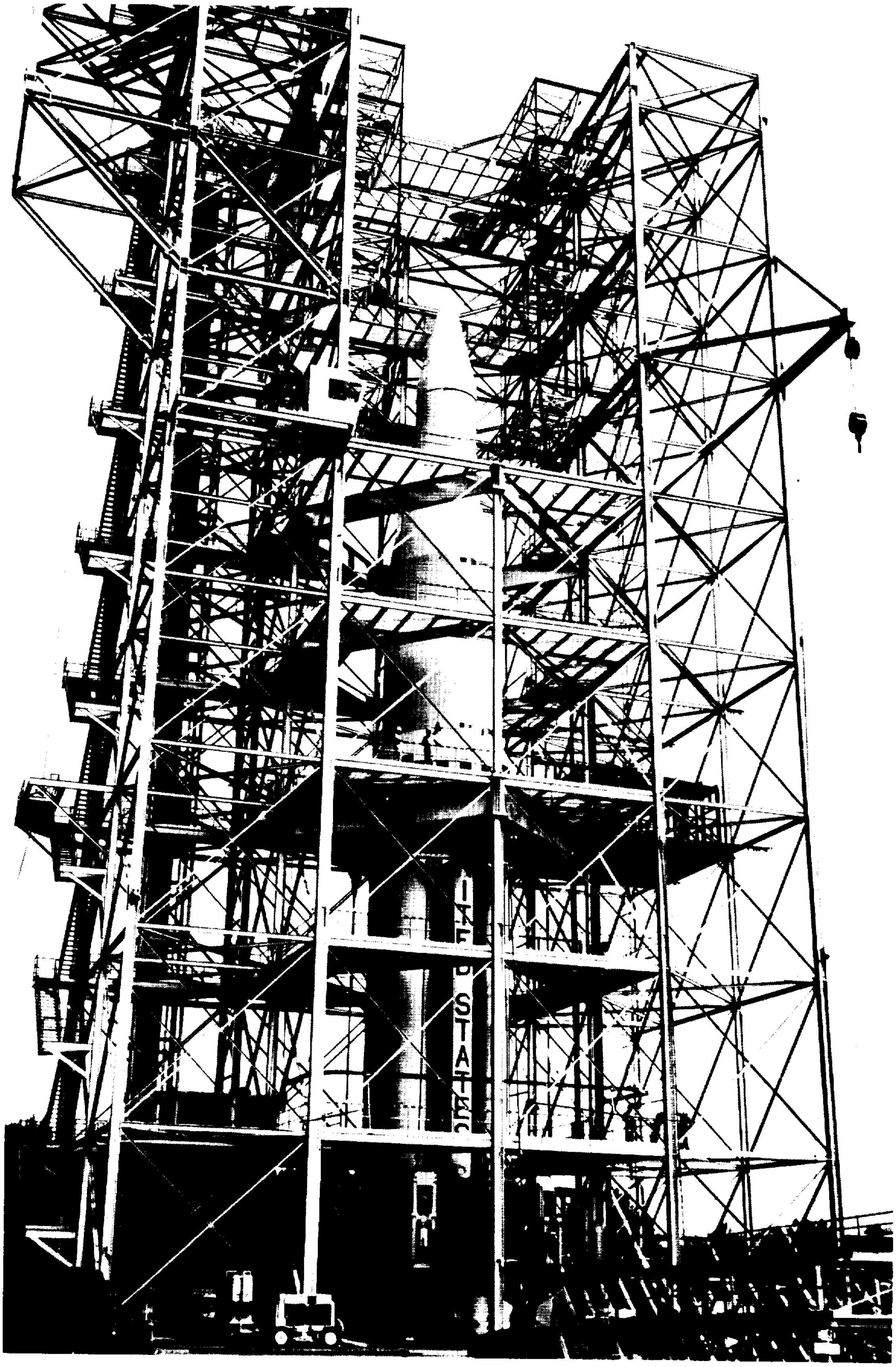

FIG. 16.2 SATURN BLOCK I VEHICLE IN DYNAMIC TEST STAND

height may be increased to about 276 feet should this become necessary in the future. The structure measures about 60 feet x 120 feet at the base. It can accommodate tail fins on the first stage of the space vehicle to a span of about 50 feet, without major modifications. The structure has an upper section of removable trusses for easier side loading of the test stand. The stiff-leg derrick on top of the dynamic test stand can handle the empty Saturn stage S-I. The suspension system has a hydraulic subsystem for lifting the entire space vehicle off the launcher arms.

Supporting facilities for the propellants the loading and unloading of the propellants, and numerous smaller equipment also have been provided in connection with the dynamic test stand. The additional electrical equipment and ground instrumentation for the dynamic tests consist of the power supply for the shakers, recorders, and oscillographs which are mounted in a trailer. The power supply (about 12 kw) for the electrodynamic shakers consists of an amplidyne generator for frequencies below 10 cps and of an electronic power supply for frequencies above 10cps.

The plans and facilities also provide for the mechanical handling and mating of the entire multistage space vehicle, as well as for tests and crew training for some launch operations such as propellant loading up to standby readiness.

16.3. DYNAMIC TEST VEHICLES

At present especially instrumented Saturn vheicles are being used in the dynamic test operations. These dynamic test vehicles are numbered by the SA-D series. The flight configuration shown in the dynamic test stand on Figure 16.1 is the dynamic test vehicle SA-D1. It has the Saturn C-1 Block I Configuration with dummy stages S-IV, S-V, and payload. The actual flight hardware also could be tested, but this is not presently planned mainly for scheduling reasons.

Dynamic tests are planned for every significant change in the flight configuration of the Saturn space vehicle. In particular, dynamic tests with prototypes of spacecraft will be performed as soon as possible in order to effectively improve the design of the final flight configuration. Although structural hardware of flight configuration is used on all tests it is possible to replace certain components (e.g., some of the complete engines) by equivalent masses with identical support points and center of gravity. Prior to the initiation of the dynamic test program of the Saturn C-1 launch vehicle R&D program will have been completed and evaluated. The results of these tests will be considered in the design of the space vehicle and its components.

SECTION 17. (C) RELIABILITY AND SAFETY

At the present time no minimum number for reliability has been specified as the point at which a space vehicle is suitable for manned flight. Rather, certain goals have been set which are deemed both necessary and a practical compromise with the ultimate desirable goal of absolute reliability.

Such goals include the minimum inherent design reliability of 0.95 with 0.90 confidence factor for stage operation and inherent subsystem design reliability of 0.99 with the same confidence.

If it can be assumed that the vehicle payload has a similar minimum inherent reliability, then an overall inherent design reliability would be approximately 0.86.

Such an assumption, however, is not entirely valid at this point for it does not include the effects of human control from a manned payload. The full effects of this on the overall system operation are yet to be determined.

17.1. RELIABILITY APPROACH

In order to obtain the requisite degree of reliability for safe and economical manned vehicle operations, the following is typical of the statements and specifications outlining stage contractor responsibility in the Saturn vehicle.

17.1.1. Reliability Engineering. The contractor shall establish methods and procedures for the planning, budgeting, and execution of an overall reliability program to insure that system reliability goals are met through assessment of the detailed aspects of program engineering, production, storage, and field use. The contractor and his major subcontractors shall perform all necessary investigations, studies, and engineering works as may be required to accomplish this objective. The requirements to meet this objective shall include, but not be limited to, the following:

17.1.1.1. Reliability Organization. The Reliability Engineering Program shall be accomplished through a Reliability Group, separately identifiable within the contractor's organizational structure. This group shall be staffed with technically competent personnel in order to advise design engineers as to potential reliability problems, plan and execute a comprehensive reliability program, have access to an adequate test laboratory, perform evaluations, and identify areas or items in need of improvement. This group shall be organizationally located within the engineering group and the person directly charged with implementation of Reliability Engineering shall report to the Project Engineering Manager who is directly responsible for all other engineering activities on the project.

17.1.1.2. Reliability General. Specifically, the contractor's reliability group shall accomplish the following requirements and expand on them as required to successfully satisfy the above stated objectives.

17.1.1.2.1. Reliability Plan. The contractor shall prepare for the approval of the procuring agency, within two months after first contract date, a reliability plan for the program which details all areas of reliability work to be accomplished. Particular emphasis should be placed on the FY-62 detail requirements. This plan shall include, but not be limited to, the estimated starting and completion dates for the various phases, as well as detailed cost analyses for the various phases expressed in manhours, materials, hardware, and remote facilities required. This plan shall include any applicable reliability plans for possible major subcontractors. The plan shall be reviewed on a yearly basis in order to reflect the latest program requirements.

17.1.1.2.2. Reliability Goals. The reliability goals shall be such that the inherent capability of the system shall be acceptable for manned usage. Consideration shall be given to both the flight hardware and the ground support equipment.

(1) The contractor shall develop a reliability mathematical model predicting the inherent design reliability of the stage system to equal or exceed a value of 0.95 with a confidence factor of 0.90. The inherent design reliability of the individual subsystems shall equal or exceed a value of 0.99 with a confidence factor of 0.90.

(2) The contractor shall develop a reliability assessment model indicating the attained reliability of the individual systems and subsystems. This assessment is intended to indicate the actual reliability of the systems as measured or determined at a given phase of the program. It is not to be confused with the inherent design reliability which cannot be measured practically.

(3) The contractor shall develop inherent design reliability goals expressed in mean-time-to-failure and safety margins. These attained values shall be correlated with the attained reliability figures.

17.1.1.2.3. Specifications. The contractor shall establish and maintain specifications indicating performance criteria and environmental exposures for the Saturn stage, and related subsystems and components. These specifications shall be subjected to continuous review to insure prompt updating as development progresses. They shall be compatible with the overall Saturn System Specifications and the individual stage specifications required by the procuring agency. Characteristics such as compatibility of materials, aging, deterioration, susceptability to natural and induced environments, as well as operational requirements shall be reviewed. The contractor reliability group shall review and approve all these specifications for correctness and check for compatibility with each other and those of related systems. All specifications are subject to review and approval by the procuring agency.

17.1.1.2.4. Design Review. The contractor shall perform an effective design review on a continuing basis. The design review group shall include qualified personnel from the manufacturing, design, reliability, and other pertinent groups, to verify that the presented design meets the performance and environmental criteria, and is capable of being manufactured with no degradation of reliability.

17.1.1.2.5. Planning and Programming. The contractor shall provide adequate planning and programming to insure that parts, components, subsystems, and systems are timely available to support the reliability test programs as outlined herein.

17.1.1.2.6. Reliability Review Conferences. The contractor shall provide attendance of qualified personnel at reliability review conferences to be held either at the contractor's facilities, the Marshall Space Flight Center, or other possible locations as may be appropriate to discuss the stage program or other possible phases of the complete Saturn system program.

17.1.1.2.7. Remedial Engineering. The contractor shall plan for, and provide when appropriate, effective remedial engineering action on all reliability problems encountered in design, manufacturing, testing, storage, and field maintenance of the stage system.

17.1.1.2.8. Studies. The contractor shall conduct studies to determine the effects caused by changes in electrical and mechanical designs on the reliability of the stage system as the development program progresses.

17.1.1.2.9. Approved Lists. The contractor shall develop and maintain lists of approved (qualified) parts, components, and subsystems which have been evaluated in whole or in part as a result of the various test programs.

17.1.1.2.10. Human Engineering. The contractor shall establish and coordinate a human engineering evaluation program to insure that the stage system is capable of being handled and operated without hazard or undue degradation of performance when subjected to its intended functions. Consideration shall be given to both flight hardware and ground support equipment. This program should begin in design, and surveillance should be continued throughout the complete life of a given system.

17.1.1.2.11. Education. The contractor shall conduct a vigorous and sustained educational program of reliability concepts for personnel directly associated with the stage system.

17.1.1.2.12. Documents. The contractor shall submit to the procuring agency, for information and approval as required, the test program proposals, test procedures, test reports, and other related documentation. The format and times for submission of these documents and others necessary to conduct or report the test program will be agreed to by the contractor and MSFC.

17.1.1.2.13. Progress Reports. The contractor shall include significant reliability accomplishments or problems in the progress reports required by the procuring agency.

17.1.1.3. Parts Qualification, Standardization, and Application. The contractor shall establish and implement a parts qualification, standardization, and application program for the parts that will comprise the Saturn stage. Consideration shall be given to both the flight hardware and the ground support equipment.

17.1.1.3.1. Studies. The contractor shall conduct studies and investigations to locate parts that have been qualified under other missile or space vehicle development programs. Maximum use of this type information shall be made to minimize duplication of effort. Proper dissemination of this information will be made within the contractor and major subcontractor organizations to issue qualified parts.

17.1.1.3.2. Parts Qualification Tests. The contractor shall initiate qualification tests on questionable parts to insure their capability for meeting minimum design requirements. Requalification of parts shall be accomplished whenever significant changes are made in design, materials, or source of supply.

17.1.1.3.3. Approved Supplier List. The contractor shall generate an approved supplier list based upon qualification testing, quality surveys, and quality assurance data.

17.1.1.3.4. Qualified Parts Lists. The contractor shall prepare Qualified Parts Lists bound in loose leaf folders using approved format. Documentation shall include all physical, functional, and environmental characteristics to describe a given item. A list of approved suppliers for each item shall be included. The items that comprise the list shall show that they meet minimum acceptable requirements dictated for the usage of the part.

17.1.1.3.5. Parts Failure Analysis. Parts that fail under test shall be analyzed and failure diagnosis reports prepared. Analysis and corrective action shall be taken as a result of parts failure reports in laboratory or field use. Data obtained from field and laboratory tests shall be coordinated with part manufacturer so that they may improve the reliability of the parts in question.

17.1.1.3.6. Parts Application. The contractor reliability group shall act as consultants to the design groups concerning the application of parts.

17.1.1.4. Components Qualification and Reliability Tests. Qualification tests (also known as design proofing tests) shall be conducted on equipment to determine that it meets the minimum design performance and operational requirements under the various combinations of service environments that may be applicable. Qualification tests shall be conducted on a sample size as approved by the procuring agency.

17.1.1.4.1. Component Reliability Tests. Reliability tests shall be a continuation of the qualification tests to determine failure points, safety margins, and life expectancy under the environmental conditions required for manufacturing, system storage, transportation, and operation. Reliability tests shall be conducted on a sample size as approved by the procuring agency. The reliability test samples may also include those which were used in the qualification tests.

17.1.1.4.2. Test Proposals. Test proposals for a given component or group of components shall be submitted to MSFC prior to the start of testing. Proposals shall include general information concerning test conditions, environmental conditions, hardware to be tested, instrumentation, special facilities, cost estimates (manhours and material), and time schedules. This document is not a detailed testing procedure.

17.1.1.4.3. Reports and Analyses. Failures and analysis of test results shall be reported. Analysis of test results shall be based on ability to correlate laboratory conditions with actual damage potential under field and induced enviroments. Results of tests shall have maximum dissemination among the contractor design organizations, affected subcontractors, NASA, and other Government Agencies.

17.1.1.4.4. Corrective Action. The contractor reliability group shall take effective steps to insure that corrective action is completed on the deficient items by the cognizant design and manufacturing organization.

17.1.1.5. System Tests. The contractor shall conduct a comprehensive systems test program to establish reliability factors, safety margins, conformance to design criteria, and to evaluate system capabilities under operational conditions.

17.1.1.5.1. Test Proposals. Test proposals shall be submitted to the procuring agency prior to conducting the tests. Proposals shall include test conditions, environmental conditions, hardware to be tested, instrumentation, special facilities, cost estimates (manhours and material), and time schedules.

17.1.1.5.2. Test Programs. The contractor shall conduct system evaluations in accordance with the approved test proposals. The system shall be evaluated to determine the effects of natural and induced environments under conditions of storage, handling, transportation, operation, and human-factors problems during these tests.

17.1.1.5.3. Re-evaluation. The system or subsystem shall be re-evaluated when significant changes are made in design, material, or source of supply.

17.1.1.5.4. Design Changes. The effect of any proposed change or modification upon the compatibility or reliability of the stage used in conjunction with the complete Saturn system shall be determined before the changes or modification are made.

17.1.1.6. Data Center. The contractor shall establish methods and procedures necessary to create a data center to aid in the development, manufacture, operation, and maintenance of a highly reliable stage for the Saturn system. The data center shall provide for a continuing capability to accept pertinent information relative to the system development, manufacture, and operation. The data center shall have sufficient flexibility for growth and control potential. The design of the data center will be such as is compatible with government and contractor data systems having similar objectives and complexity. It shall be designed to function for information necessary for design development, manufacturing, quality control, and reliability data handling and processing.

17.1.1.6.1. Capability. The data center shall have the capability to provide the data and information to generate the following documents, reports, and listings:

(1) Parts Lists -- Listing the frequency of usage and identification of parts occurring in subassemblies, assemblies, and the complete stage system to aid in determining priority of parts qualification, reliability testing, and related information

(2) Qualified Parts List

(3) Limited Life Items and change of status of these items

(4) Correlation of environmental data with test data for comparison with design and specification requirements

(5) Data on mode and cause of equipment failures while in development and manufacturing stages, and comparisons with field failure reports

(6) Document corrective action and follow-up

(7) Generate a continuous review of all areas of the launch vehicle system so that defects and failures may be grouped and problem areas pinpointed or predicted

(8) Generate a complete technical review of a given item at any time during its development, manufacture, and operation

(9) Summarizing data on deviating materials

(10) Inputs from field failure reports.

17.2. FAILURE MODES

The determination of the failure modes for each system, subsystem, and component part is an integral part of both the inherent design reliability and the attained reliability. As a part of the reliability determination of the system each stage contractor must determine both the failure modes of the individual units and their subsequent effects on associated components and stage operation. Figure 17.1 illustrates in a simplified manner this portion of the reliability approach. The failure modes of individual components of a subsystem and the subsequent effect of component failure on both the associated subsystem and stage performance are given in Table 17.1.

Based on a thorough analysis of the operational modes of all portions of the system, including failure modes and their net affects, the progressive determination of attained reliability will be made. This progressive determination, illustrated in Figures 17.2, 17.3, and 17.4. for one particular subsystem breakdown, will give a ready comparison of the achieved or demonstrated reliability and the minimum design goals.

17.3. ABORT SYSTEMS

At the present time the abort system for the Saturn vehicle is not completely defined. However, generally the stage contractor will be responsible for monitoring those parameters which indicate component or subsystem malfunction or impending failure. Such data will be

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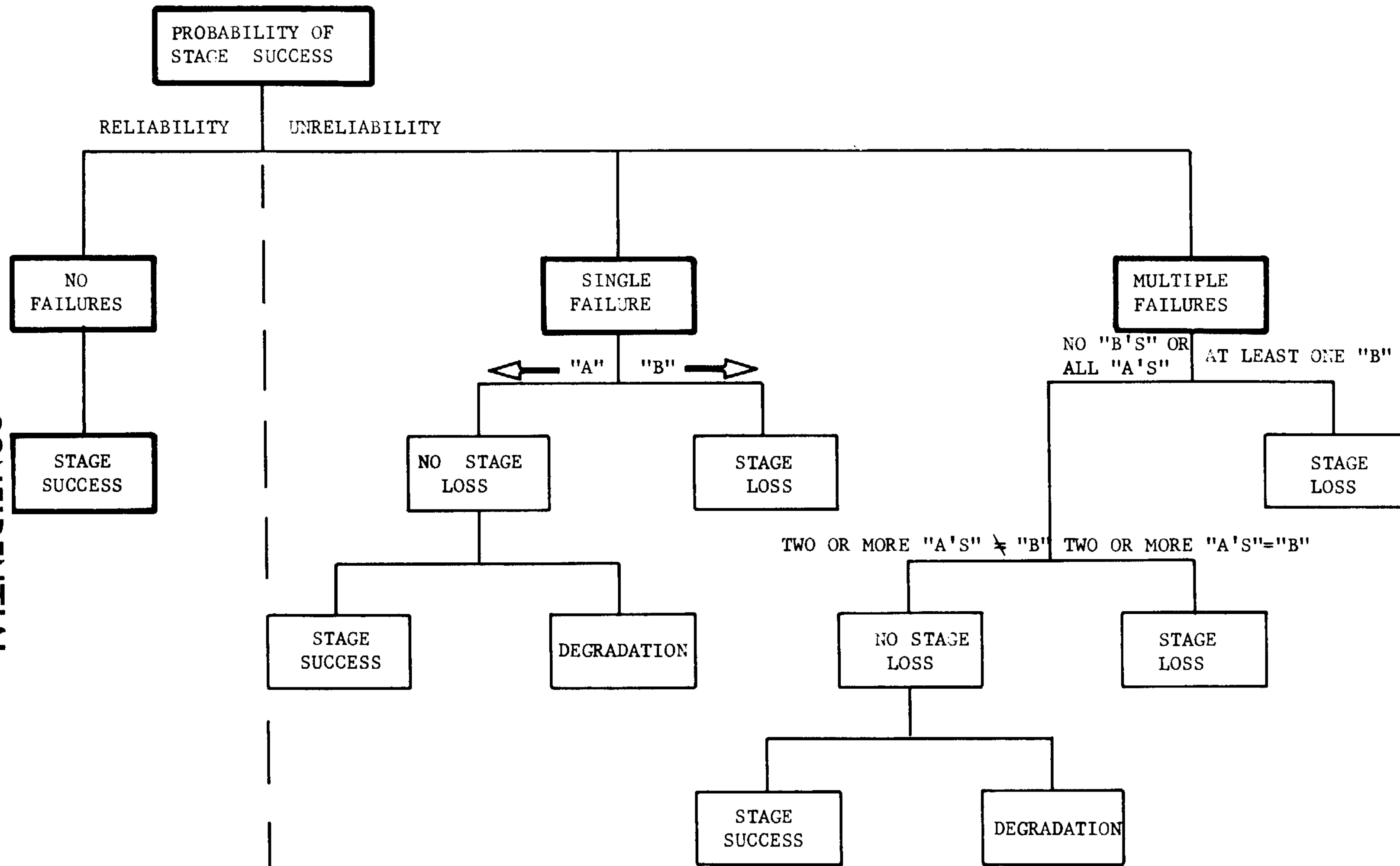

$$\text{UNRELIABILITY} = 1-r = \overbrace{\Pr\left\{\text{NO STAGE LOSS}\right\}}^{A} + \overbrace{\Pr\left\{\text{STAGE LOSS}\right\}}^{B}$$

FIG. 17.1 PROBABILITY OF STAGE SUCCESS BASED ON COMPONENT RELIABILITY

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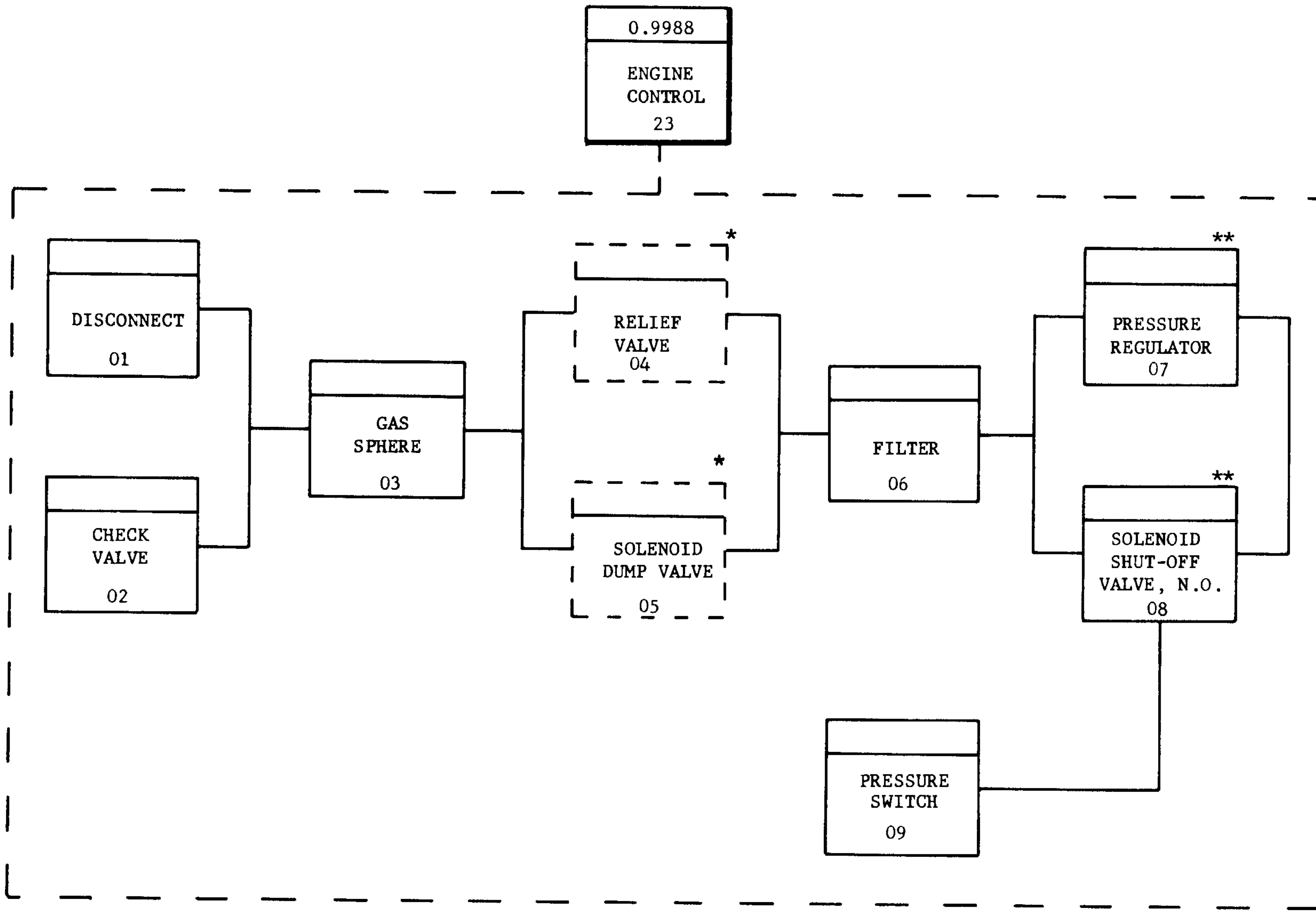

FIG. 17.2 PROGRESSIVE RELIABILITY DETERMINATION - ENGINE CONTROL UNIT

*FOR FAILURE TO OPEN ONLY
**FOR FAILURE TO CLOSE ONLY

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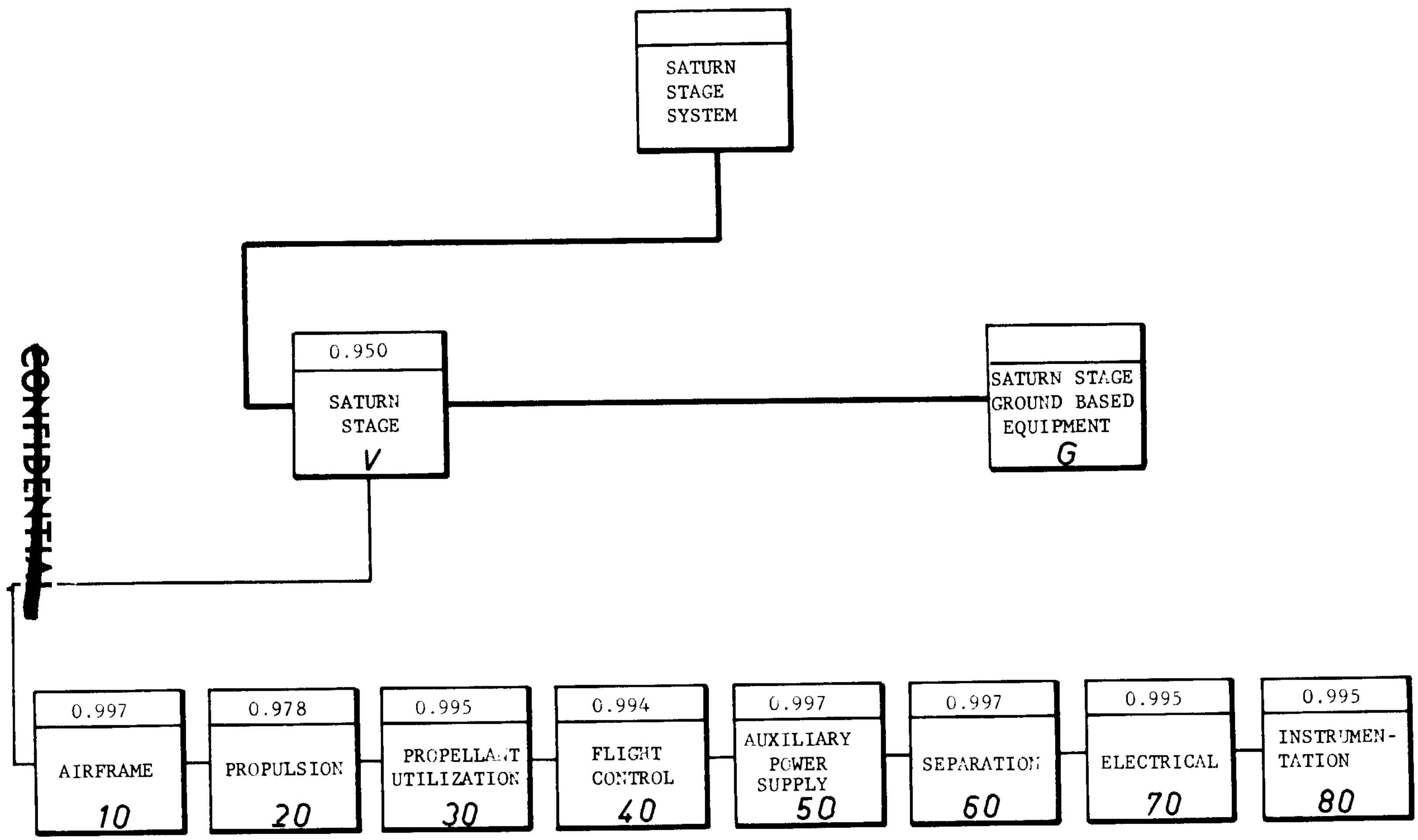

FIG. 17.3 PROGRESSIVE RELIABILITY DETERMINATION - PROPULSION SUBSYSTEM

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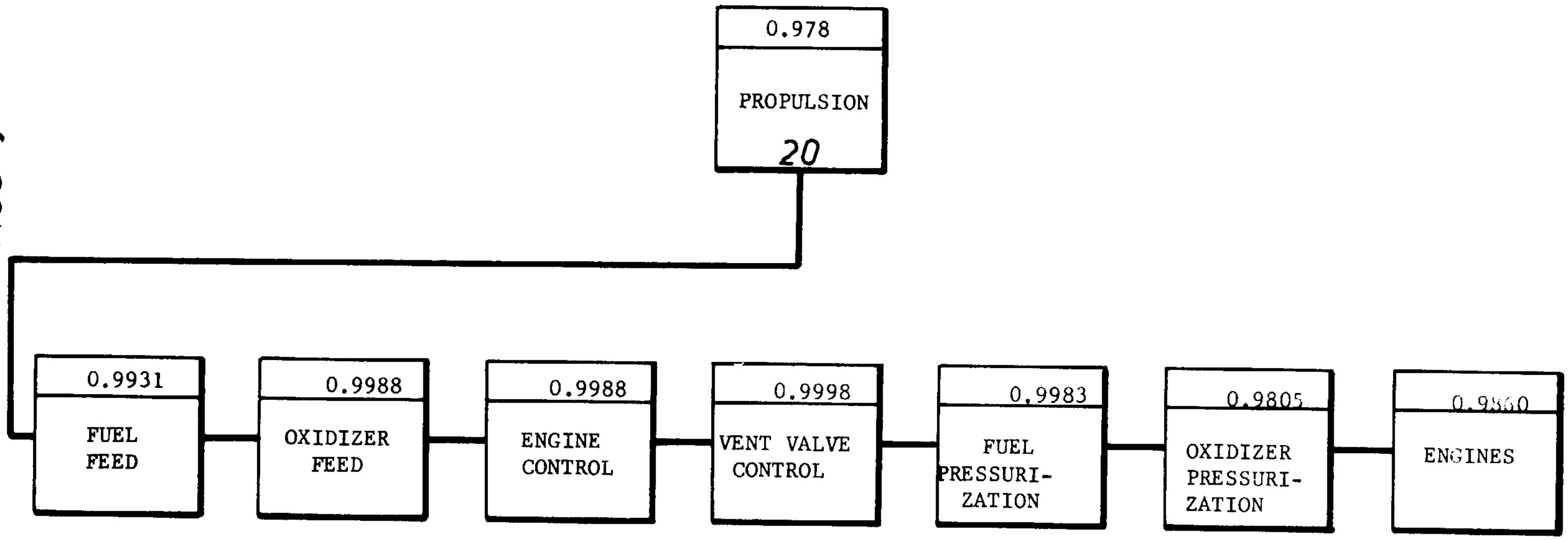

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FIG. 17.4 PROGRESSIVE RELIABILITY DETERMINATION - SATURN STAGE transmitted to the instrument unit for processing by the abort computer. In the event of an immediately impending catastrophic failure as indicated by a gross malfunction, an automatic abort sequence will be initiated. If immediate action beyond the reaction capabilities of the crew commander is not necessary, then suitable information will be indicated within the crew capsule for a command decision. The abort computer will contain provisions for variable threshold values of abort parameters. These values will depend on individual stage operation and, in some cases, may vary during stage operation.

Provisions in the abort system will be made for the crew commander to have abort initiation capability both during flight and for a portion of the time prior to first stage ignition. Such action, if taken, would automatically cause stage shutdown. In addition, ground command destruct of the vehicle will automatically initiate the abort sequence, thus insuring capsule safety.

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Table 17.1

A) Launch Countdown (T Minus Time)
B) Flight (T Plus Time)

RELIABILITY FAILURE EFFECT ANALYSIS

ENGINE CONTROL SUBSYSTEM

ITEM	FAILURE TYPE	FAILURE EFFECT ON SUB-SYSTEM PERFORMANCE	FAILURE EFFECT ON STAGE
Disconnect	1) Failure to close at disconnect. 2) Internal leakage	None; upstream check valve prevents loss of engine control gas.	A) None; redundancy provided. B) None; possible loss of stage if both items fail.
Check Valve	1) Internal leakage (failure to reseat)	None; downstream disconnect prevents loss of engine control gas.	
Gas Sphere	1) External leak at connection.	Loss of engine control gas	A) Delay in launching or scrub B) <u>POSSIBLE LOSS OF STAGE</u>.
Relief Valve	1) Internal leakage (failure to reseat)	Loss of engine control gas	A) Delay in launching or scrub B) <u>POSSIBLE LOSS OF STAGE</u>
	2) Failure to open	None; solenoid dump valve can be used to dump over-pressure	A) None; redundancy provided. B) None; not operational in flight.
Solenoid Dump Valve	1) Internal leakage (failure to reseat)	Loss of engine control gas	A) Delay in launching or scrub B) <u>POSSIBLE LOSS OF STAGE</u>

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Table 17.1 (Continued)

ITEM	FAILURE TYPE	FAILURE EFFECT ON SUB-SYSTEM PERFORMANCE	FAILURE EFFECT ON STAGE
	2) Failure to open	None; relief valve can dump over-pressure	A) None; redundancy provided. B) None; not operational in flight.
Filter	1) Implosion of element	<u>PROBABLE LOSS OF SUBSYSTEM</u> Filter particles will damage or affect the operation of all downstream components	A) Scrub B) <u>PROBABLE LOSS OF STAGE</u>
Pressure Regulator	1) Failure to open	<u>LOSS OF SUBSYSTEM</u>	A) Scrub B) <u>LOSS OF STAGE</u>
	2) Failure to Close	None; upstream solenoid valve will cycle due to action of the pressure switch	A) Scrub B) None; redundancy provided
Solenoid Shutoff Valve, Normally Open	1) Failure to close	<u>LOSS OF SUBSYSTEM</u> (operates only if upstream regulator fails to close)	A) Scrub B) <u>PROBABLE LOSS OF STAGE</u>
	2) Failure to open	<u>LOSS OF SUBSYSTEM</u> Engine control gas supply ceases	A) Scrub B) <u>LOSS OF STAGE</u>

Table 17.1 (Continued)

ITEM	FAILURE TYP	FAILURE EFFECT ON SUB-SYSTEM PERFORMANCE	FAILURE EFFECT ON STAGE
Pressure Switch	1) Failure to "make" circuit	LOSS OF SUBSYSTEM Operation of all downstream components affected due to over-pressurization	A) Delay in launching or scrub. B) PROBABLE LOSS OF STAGE
	2) Failure to "break" circuit	LOSS OF SUBSYSTEM Engine control gas supply will remain cut-off	A) Delay in launching or scrub B) LOSS OF STAGE

SECTION 18. (C) TRANSPORTATION

18.1. GENERAL

The fabrication of very large tank structures for space vehicle boosters raises difficult problems in the field of transportation. Many common and usually convenient transport facilities can be used only with some difficulty or not at all. However, it is recognized that many space vehicles and missiles of the future will incorporate large tanks and large single-chamber engines. For this reason, NASA and its contractors have been investigating the transportation and handling problems associated with the very large structures contemplated. The information gathered in such studies has been useful in formulating a logistic plan for the Saturn-Apollo system.

In general, land transport is limited by road clearances, telephone and power line crossings, special permits, and moving time restrictions in local areas. Air transport is limited by the dimensions of the load in the case of aircraft, and by the weight of the load in the case of helicopters. Transport on water presently appears to be the most attractive mode, but it is limited by the extent to which navigable bodies of water approach rocket fabrication, testing and launching facilities.

18.2. TRANSPORTATION REQUIREMENTS

The following transportation requirements exist, at this time, for the Saturn C-1 system.

(1) Saturn C-1 S-I stages will be transported from NASA, MSFC facilities to AMR Launching sites.

(2) Saturn C-1 S-I stages will be transported from a contractor facility direct to AMR launching sites.

(3) Saturn C-1 S-IV stages will be transported to the following destinations:

Santa Monica to the Sacramento Field Station
Sacramento Field Station to MSFC
Sacramento Field Station to the Atlantic Missile Range

The vehicle will be positioned in the transporter at all times while it is being transported.

Santa Monica to Sacramento Field Station

The stage will be towed from the Douglas Santa Monica Plant to the Los Angeles/Long Beach harbor, where it will be loaded aboard an ocean-going barge with a crane.

The barge will be towed to San Francisco, then up the Sacramento River to the McClellan Air Force Base wharf.

The stage will then be lifted from the barge to the road and towed by prime mover from there to the Douglas facility.

Sacramento Field Station to MSFC

The first leg of this trip is the land route to the McClellan wharf. At the wharf, the stage will be placed aboard a standard river barge for the trip to San Francisco Bay, California. In San Francisco Bay, the stage (on its transporter) will be transferred to an ocean freighter bound for New Orleans.

The freighter will proceed to New Orleans, where the stage will be transferred to a river barge for the trip up the Mississippi, Ohio, and Tennessee River to Huntsville.

Sacramento Field Station to the Atlantic Missile Range

The stage will be transported to San Francisco Bay in the same manner as for the trip to MSFC. There it will be placed aboard an ocean freighter bound for Tampa, Florida. In Tampa, the stage will be transferred to another river barge and towed through the Okeechobee inland water route to the Atlantic Coast, where it will proceed up the intercoastal water way to Cape Canaveral.

For the requirements listed above, equipment and facilities have to be provided to allow transportation of Saturn system stages with a guarantee of the best combination of economy, speed, immunity to seasonal factors, and protection of the cargo from physical and environmental hazards.

18.3. THE MODES OF TRANSPORTATION

18.3.1. Land Transportation. Land transportation of Saturn stages is limited, due to the size of the Saturn stage diameters, (S-I, 265 inch, S-IV, 220 inch), to transportation on MSFC arsenal roads between facilities and dock site, between contractor facilities and their dock sites and between AMR dock sites and AMR launching sites.

18.3.1.1. S-I Stage Transporter. The Saturn S-I stage transporter, as shown in Figure 18.1, will be used for land transportation. The detachable S-I stage fins employ holding mechanisms, which are replaced with dummy fixtures during assembly, transportation and static testing. The dummy fixtures provide attachment points which are coincident in location with the outrigger attachment points on a standard NASA Saturn S-I stage.

The land transporter is designed to provide complete maneuverability for the S-I stage at the required locations. The transporter consists of an assembly fixture for cradling the S-I stage, and two supporting dollies. It is towed by a prime mover. Each of the dollies is composed of a frame and running gear assembly, drawbar, operator's seat and a steering and braking system. The steering mechanism is an electro-hydraulic system. Steering of the transporter is accomplished by an operator on each of the two dollies.

A storage system, consisting of four electrically operated screw jacks, has been designed for lifting the Saturn S-I stage and the carrying assembly fixture off the dollies. This will allow the dollies to be repaired or used for other movements, thereby decreasing the required number of dollies and cost necessary to handle the S-I stages during high density firings at the launch site.

A jack assembly must be placed under each outrigger jack pad assembly (which is attached to a corner (Fig. 18.2) of the assembly fixture). The jacks are then raised to allow dollies to be removed from under the assembly fixture. Figure 18.3 shows the fixture assembly supported by jack assemblies.

18.3.1.2. S-IV Stage Transportation. There are two basic modes of land transport: highway, and railroad. Specialized items of transport equipment may be designed which would be capable of long distance moving of the Stage S-IV over the nation's roads, but the use of these roads imposes serious restrictions. The large tank sizes would require the use of at least two highway lanes, and hence, movement must conform to house-moving type of restrictions. In addition, the overall minimum transport height (in excess of 20 feet) limits the number of usable underpasses and requires the lifting or relocation of power lines and telephone cables. Clearly, any long distance highway transportation of Stage S-IV will be a costly and tedious process and should be minimized.

Railroad flatcars cannot be used at all since railroad routes are generally limited to a 12-foot width, and to no more than 14 feet even under special conditions. A picture illustrating the S-IV land transporter is shown in Figure 18.4.

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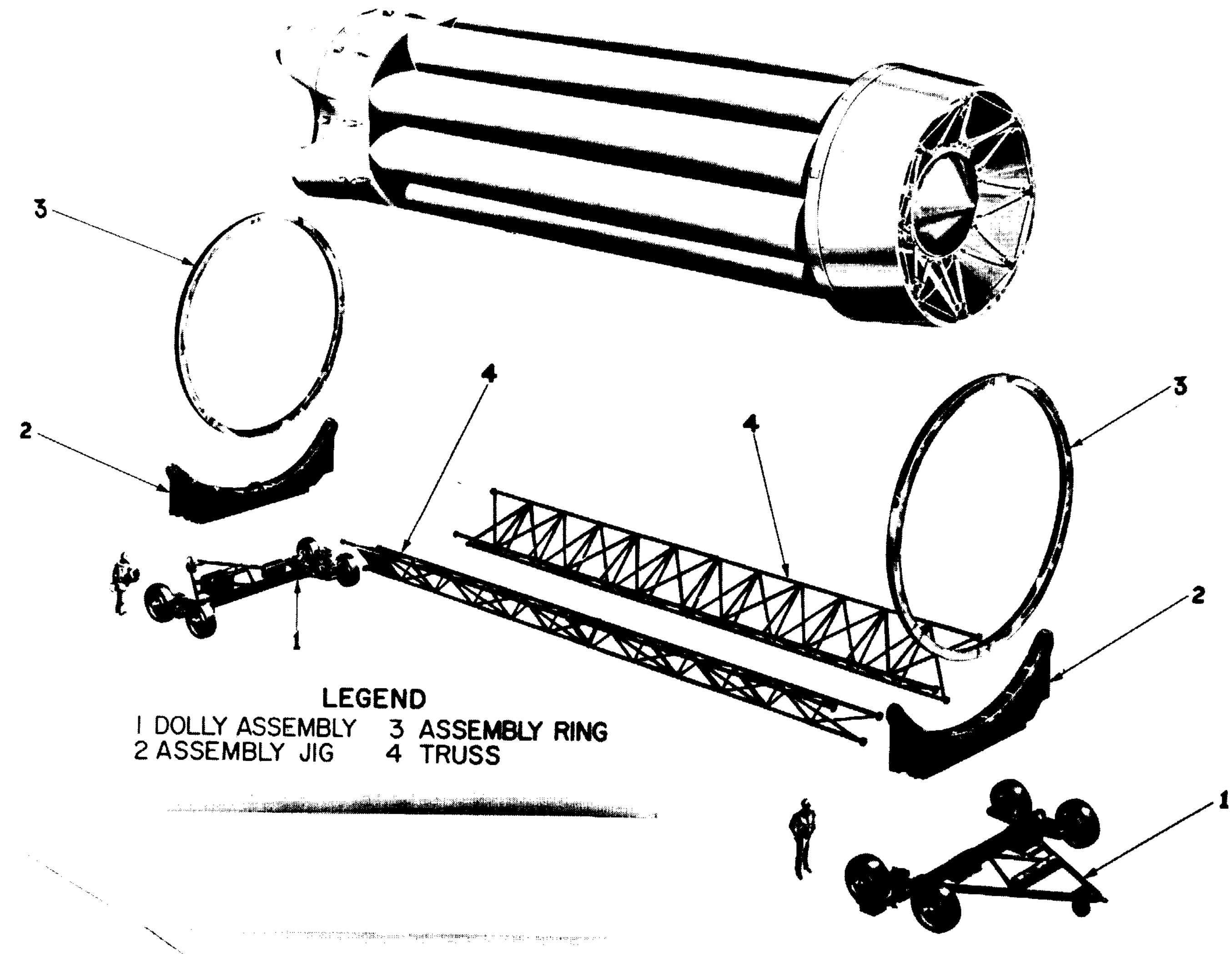

FIG. 18.1 SATURN S-I STAGE TRANSPORTER

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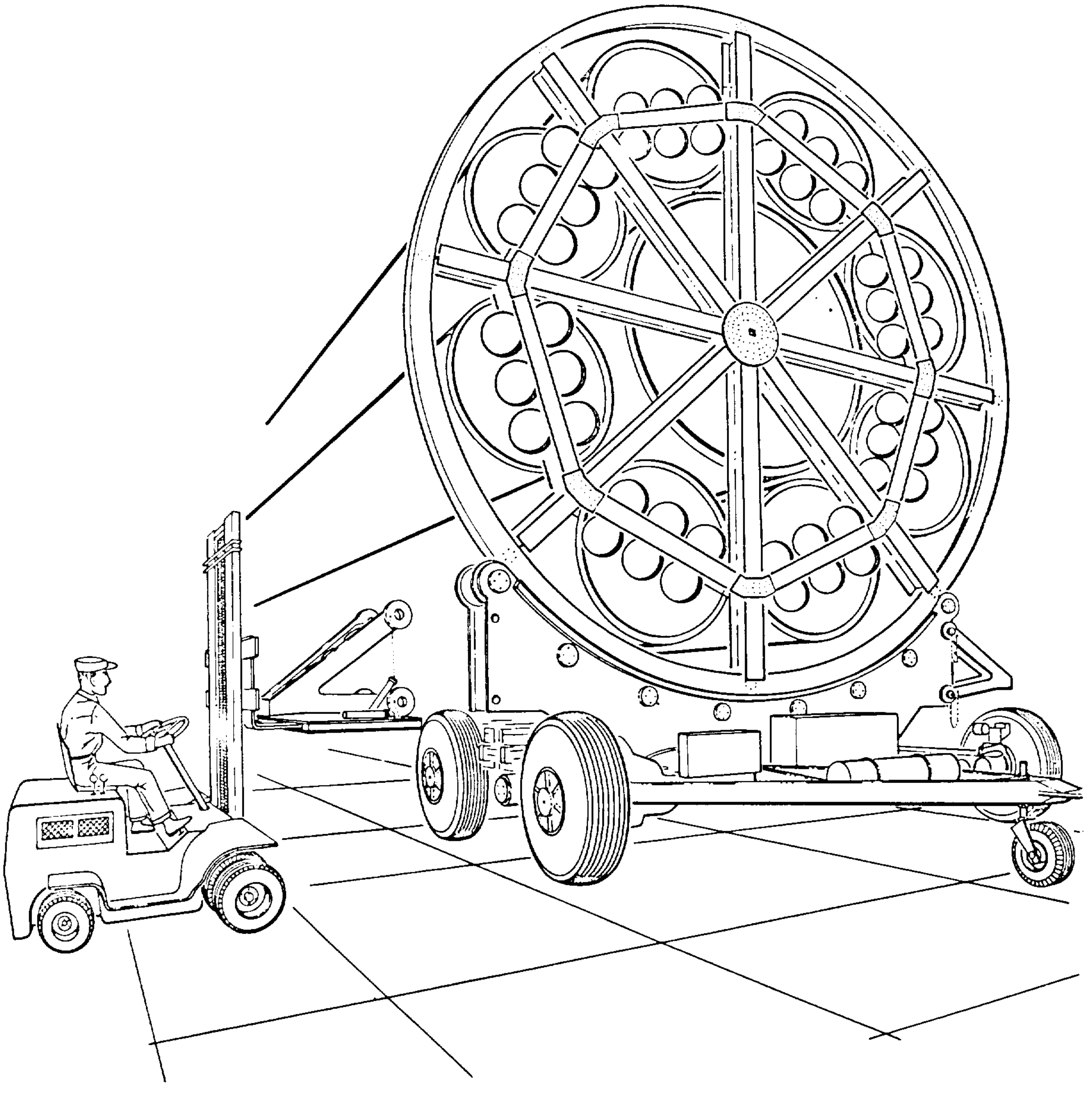

FIG. 18.2. ATTACHING JACK PAD ASSEMBLY

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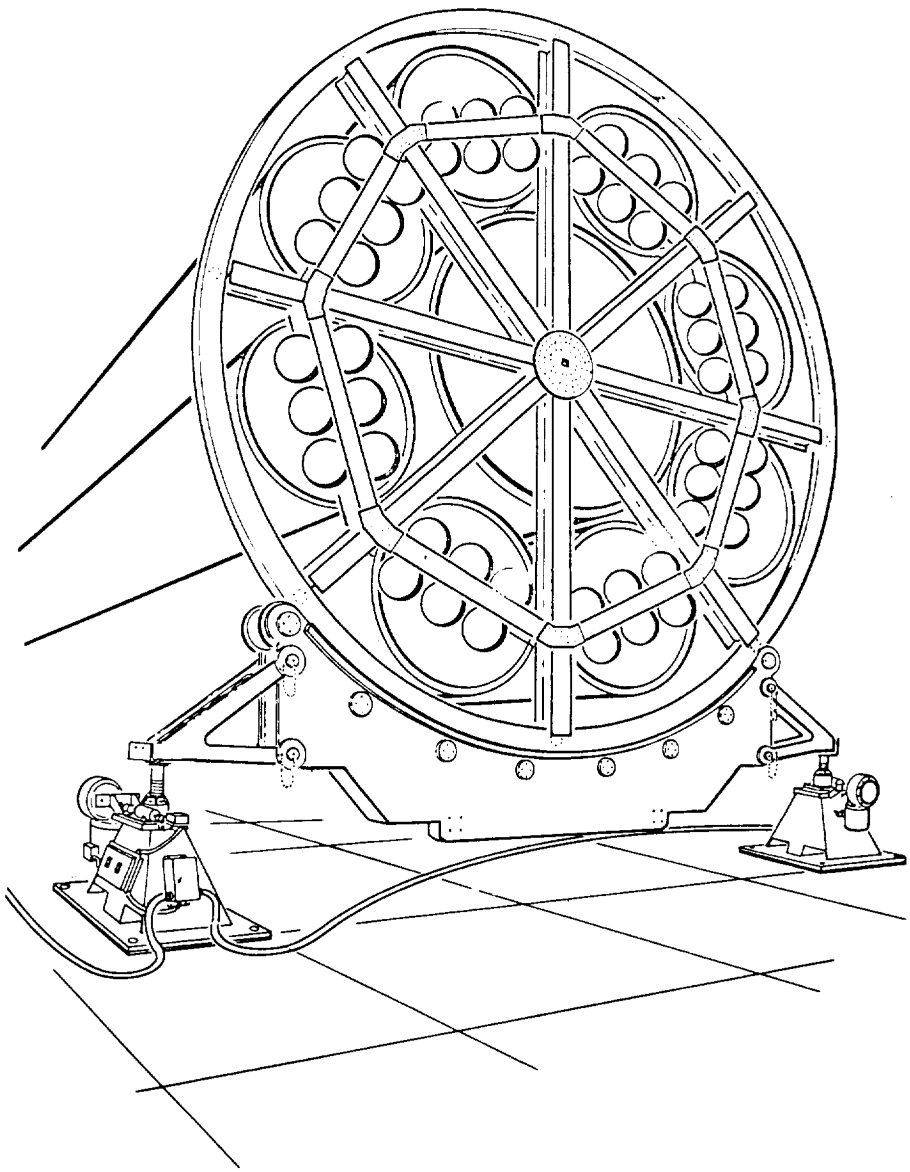

FIG. 18.3. FIXTURE ASSEMBLY SUPPORTED BY JACK ASSEMBLIES

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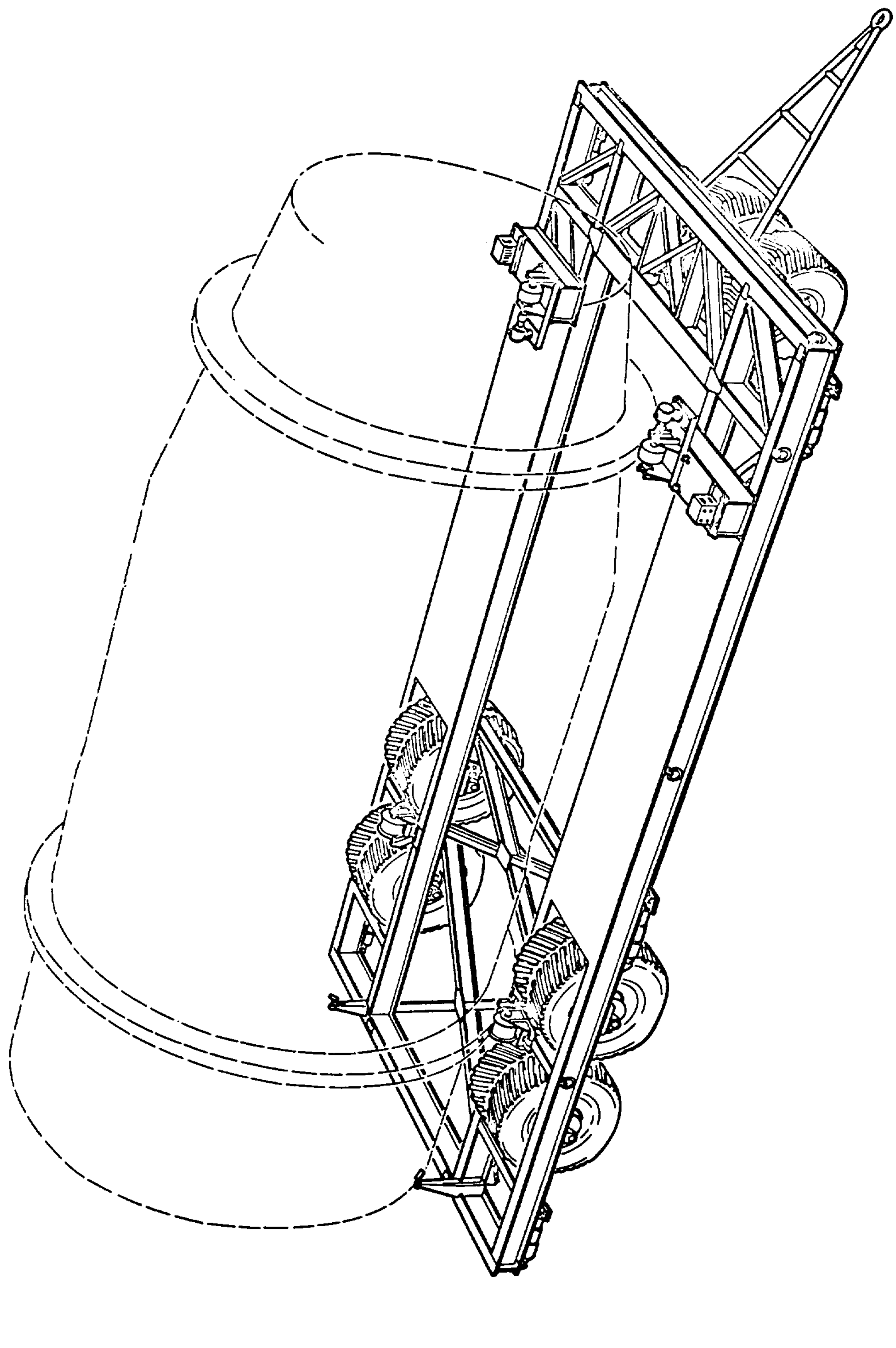

FIG. 18.4. TRANSPORTER ASSEMBLY (MODEL-144) FOR STAGE S-IV

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18.3.1.3. S-IV Transporter

18.3.1.3.1. Description. The transporter is a towed vehicle which provides support and mobility to the S-IV stage. It consists of a built-up base frame mounted on a six-wheeled under-carriage, arranged to provide a three point suspension system, two points at the rear, and a single steerable point at the front. Low pressure Terra-Tires obviate need for a sprung suspension and are adequate for the required highway speeds of five to ten miles per hour. Disc brakes will be incorporated in all four rear wheels; separate parking brakes are provided on two rear wheels. A 360 degree roll system is an integral part of the mounting system. Powered roll, accomplished by two electric motors, drives the front rollers. Two idler rollers are mounted over the rear wheels. The stage may be locked in any roll position. Trunnion points are provided at the rear of the transporter to permit vertical erection of the stage. During vertical erection, the transporter will be stabilized by two outriggers and six mechanical jacks. The transporter utilizes towbar type towing and incorporates tiedown rings for water transport. It has adequate clearance lights for night operation and will have standard electrical and hydraulic line connections to the prime mover. The transporter can be easily disassembled for road or air shipment.

18.3.1.3.2. Function. The transporter provides support, mobility, and shock isolation to the S-IV vehicle at all times except when the stage is in a test stand or mating fixture. It will be the basic component of a protective package that will include the transport handling kit and the transport protective kit.

18.3.1.3.3. Dimensions. The approximate dimensions of the transporter are: height 147 inches; length, less towbar, 495 inches; width 264 inches; length, including towbar, 619 inches; and weight 22,000 pounds. Dimensions of the transporter, including the S-IV stage, will be: height 270 inches, width 264 inches, and weight 37,000 pounds.

18.3.1.4. S-IV Transport Handling Kit.

18.3.1.4.1. Description. The transport handling kit will consist of all handling hardware required to place the S-IV stage on the transporter during all phases of transport operations. These components will be installed and used with the S-IV stage during all operations except for specific testing and checkout procedures. The transport handling kit is shown in Figure 18.5.

18.3.1.4.2. Function. The function of the transport handling kit will be to provide for the mounting and handling of the S-IV stage on the transporter during all ground and water transport operations.

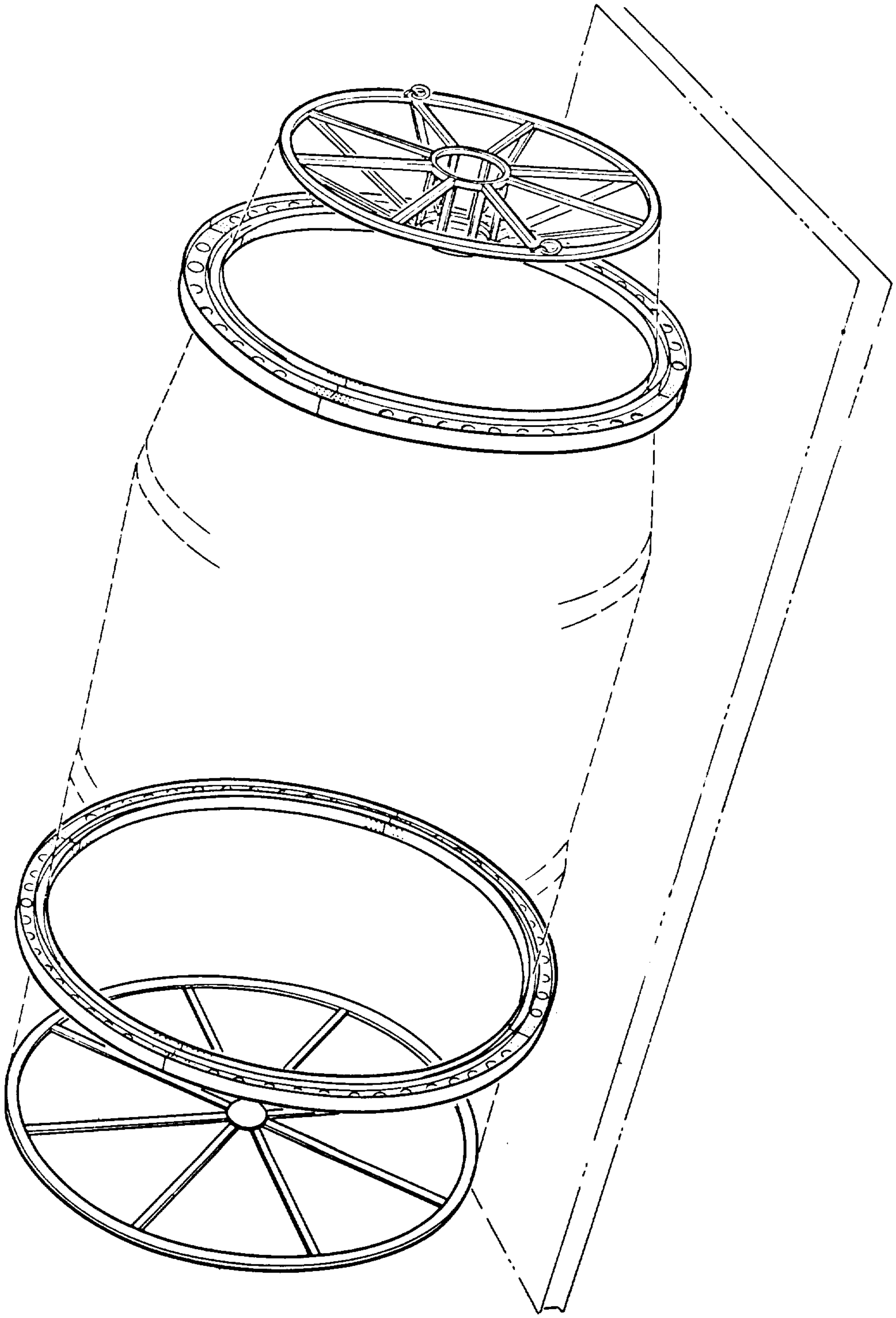

FIG. 18.5. TRANSPORT HANDLING KIT (MODEL-146)
(TENTATIVE)

18.3.1.4.3. Components and Dimensions. The transport handling kit will consist of the following components:

(1) Forward hoist ring. This will be a truss type ring attached to the forward end of the forward interstage. The ring will have fittings to allow for vertical erection and hoisting of the complete S-IV stage without deformation of the stage shape and hole pattern. It will have provisions for attaching the environmental covers and will be modified to accommodate the shorter forward interstage.

(2) Forward roll ring. This ring will be a segmented metal ring and will provide for the support of the forward section of the S-IV stage on the transporter. The ring will be bolted to the S-IV stage at the eight panel joints of the forward interstage structure. It will provide for the 360 degree roll of the S-IV stage on the transporter. Provisions will be made for the attachment of the environmental cover. The approximate dimensions will be: diameter 210 inches, width 7 inches, and weight 750 pounds.

(3) Aft roll ring. This will be a ring similar to the forward roll ring except that it will be located on the aft interstage structure at the engine mounting plane. The approximate dimensions will be: diameter 244 inches, width 7 inches, and weight 850 pounds.

(4) Aft trunnion ring. This will be a truss type ring which is bolted to the aft end of the interstage. It will incorporate pins to fit the trunnion fittings of the transporter. This ring will maintain the stage shape and hole pattern. Provisions will be made for the attachment of the environmental covers. The approximate dimensions will be: diameter 222 inches, width 8 inches, and weight 500 pounds.

(5) Engine support structures. This will be a tubular type structure which will support the weight of the engines during transport. It will be attached to the engine mount bulkhead and to the aft roll ring through the interstage joint attachments. Approximate weight will be 200 pounds.

(6) Hoisting beams. The hoist beams will provide for hoisting required during all ground and water transport operations. The beams will be capable of hoisting the stage on and off the transport vehicle and hoisting the loaded transport vehicle on and off the freighter at the dock.

18.3.1.5. Transport Protective and Tiedown Kit

18.3.1.5.1. Description. The protective and tiedown kit will consist of all the hardware necessary to provide environmental protection during all phases of transport and to provide for tiedown during the water transport phase. The protective covers will be easily packed for air and ground shipment to facilitate re-use.

18.3.1.5.2. Function. The function of this kit is to afford complete environmental protection during all phases of transport and to provide tiedown devices for shipboard tiedown during water transport. The kit is shown in Figure 18.6.

18.3.1.5.3. Components. The transport protective and tiedown kit consists of the following:

(1) Body protective cover. The stage will be protected from impact and abrasive damage and all environmental conditions by a double walled inflatable cover. The cover will be attached to the forward and aft closing rings and forward and aft roll rings. The cover will be easily repairable in the field and can be folded for storage or shipment by ground or air for re-use.

(2) Forward end cover. This will be an impregnated nylon or glass fabric dust cover which will be attached to the forward hoist ring. The cover will protect the interior of the forward interstage.

(3) Aft end cover. This will be similar to the forward end cover and will be attached to the aft trunnion ring.

(4) Engine nozzle covers and seals. Suitable covers will be used to protect the engines from damage. All vent openings, access holes, and umbilical receptacles will be sealed with suitable easily removable and replaceable protective material prior to shipment.

(5) Tiedowns. Suitable cables and shock mounts will be provided to adequately tie down the transporter and stage during all phases of water transport.

18.3.1.6. Handling and Access Kit. The following items will be included in a handling and access kit: engine handling fixture, forward and aft section access kit, stage support fixture, forward end cover, and tank section access kit.

18.3.1.7. Engine Handling Fixture. The engine handling fixture is a device permitting the engines to be changed when the stage is in the horizontal position on the transporter. A modification will be required to allow engine change in the vertical position on the battleship stand.

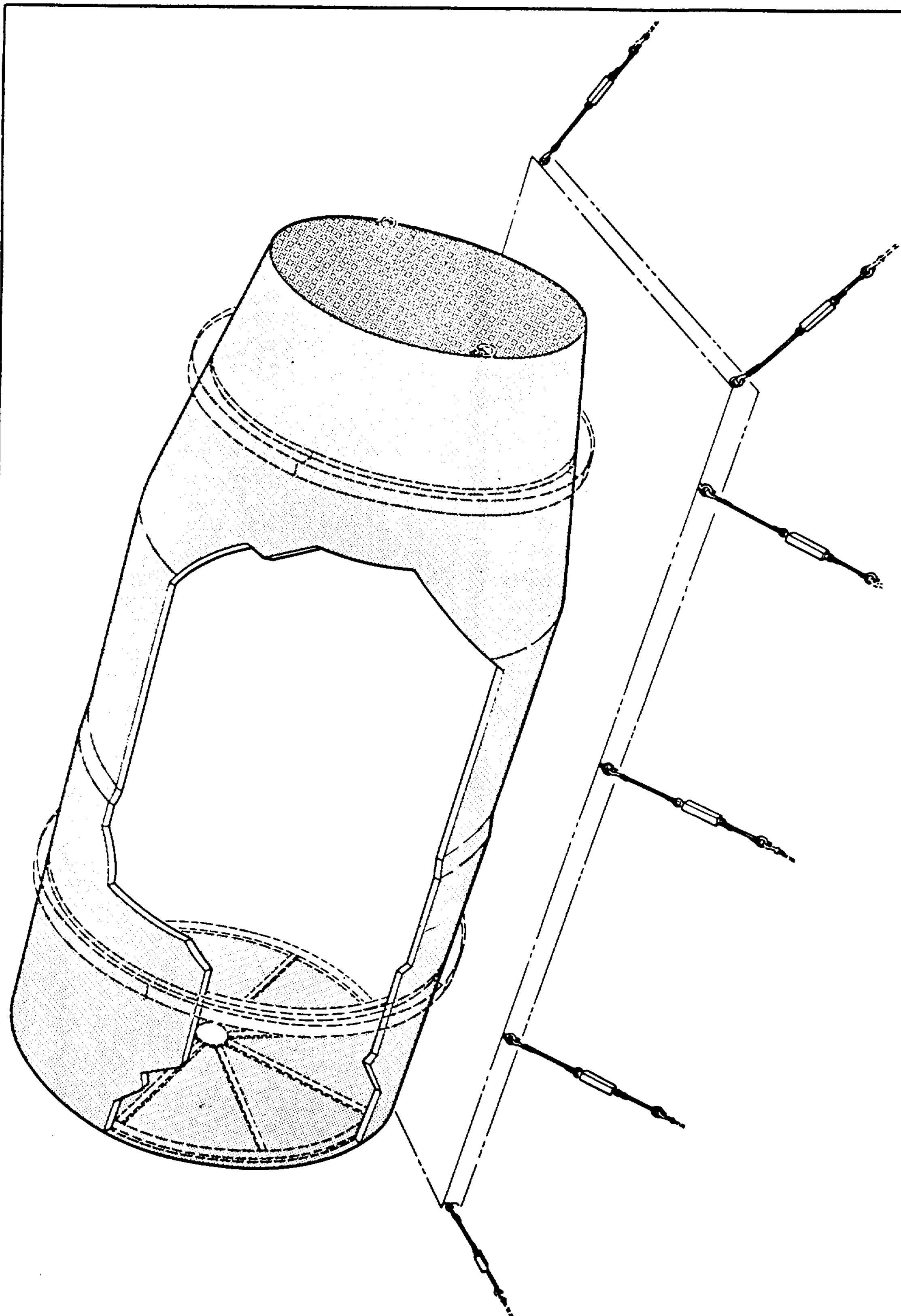

FIG. 18.6. TRANSPORT PROTECTIVE AND TIEDOWN KIT (MODEL-147)
(TENTATIVE)

18.3.1.8. Forward and Aft Section Access Kit. The forward section access kit consists of inflatable sections which are placed over the forward tank dome, and provide a work platform when the stage is in the vertical position. These covers also protect the forward dome from damage from falling objects. This kit will also contain other items such as platforms brackets, and attaching hardware, required to provide access and work areas for all maintenance required in the forward interstage area while stage is in either horizontal or vertical position.

The aft section access kit consists of all work platforms, brackets, ladders, and attaching hardware, to provide internal maintenance access in the hydrogen tank while the stage is in the vertical position.

18.3.1.9. Stage Support Fixture. The stage support fixture is a simple structure to support the stage in the horizontal position for storage in the hangars. This frees the transporter to be shipped back to Santa Monica for re-use.

18.3.1.10. Forward End Cover. The forward interstage end cover is a solid cover to keep rain and other elements out of the forward interstage area while the stage is in the test stand without the upper stages installed.

18.3.2. Water Transportation

18.3.2.1. Tug-Towed Barge. An ocean-going barge was selected in August 1959 as the means for transporting the Saturn S-I stage from the MSFC dock facilities to the dock facilities at Cape Canaveral, Florida. The first of these barges, the Palaemon, was commissioned in December 1960 at MSFC. The barge is a flat-deck type, equipped with end-opening doors for roll-on/roll-off operations and removable upper hatch covers for S-I stage removal by crane (Fig. 18.7).

There are several types of vessels which may be used for over-water transportation of the Saturn S-IV stage. Conventional freighters or cargo vessels will be used for the long ocean trips, such as from San Francisco to New Orleans. An LSD is a type of vessel which is also a possibility for ocean transport of the Saturn S-IV stage. Barges will be used for short hauls up rivers where the draft is limited, and for beach loadings where harbor facilities are not available. Suitable barges can be rented for beach loading and short ocean hauls. One particular barge investigated for this use (owned by the Pacific Tow and Salvage Company of Long Beach, California) is 34 feet by 204 feet and draws about $3\frac{1}{2}$ feet of water when empty , or about one inch more with a 15-ton load, and the S-IV stage is easily accommodated onboard. The barge is equipped with sea anchors and winches by means of which

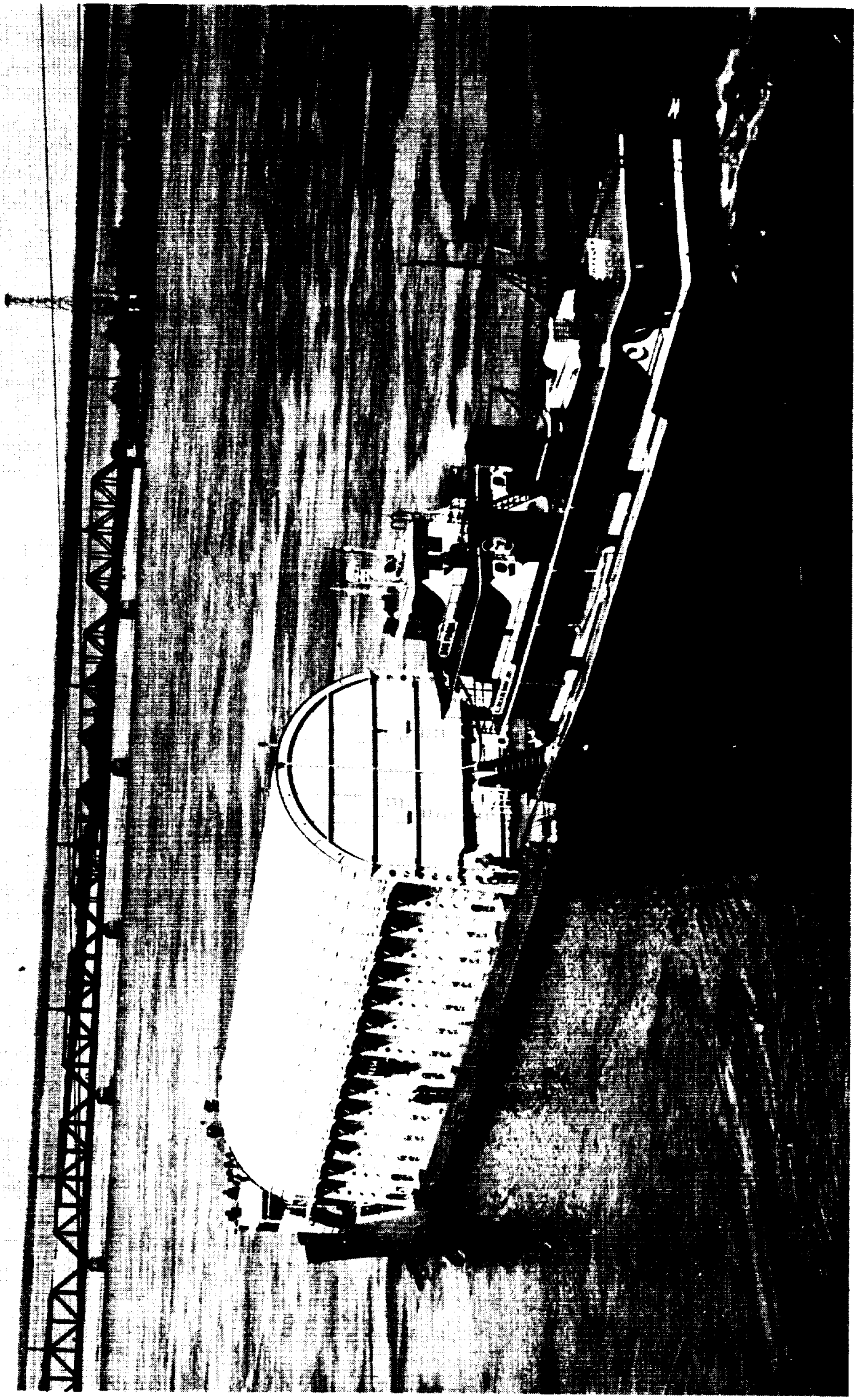

FIG. 18.7 STAGE S-I TRANSPORTATION BARGE

it can pull itself on and off the beach. The loading ramp width may be altered to accommodate the S-IV stage, which is winch-loaded onto the barge.

In a later paragraph a detailed route plan is presented for the use of barges in moving the S-IV stage from Santa Monica to Sacramento and from New Orleans to Huntsville. For river trips, special river barges will be used since they are generally smaller, simpler, and less expensive than the seagoing barge.

For long distance water transport, freighters provide better protection, greater speed, and lower costs per mile than seagoing barges. Any victory-class freighter can handle the S-IV stage as standard deck cargo. The use of existing ships, routes, and standard shipping rates is anticipated for the basic S-IV stage since delivery schedules do not require more than standard transit times. This system is readily expanded to meet any delivery dates imposed by the Apollo program.

18.3.2.2. Self-Propelled Barge. A self-propelled barge (Fig. 18.8) suitable for transport of the S-I and S-IV is a possibility. The proposed barge would have length, width, and draft dimensions compatible with coastwise and inland waterway requirements. The barge would be propelled by twin diesel engines totaling 2000 horsepower with an estimated speed of 15 knots. A crew of at least 21 men would be required. During the initial part of the program when a very low launch density will exist, only the captain, first mate, chief engineer, and steward would need to be retained on a permanent basis and operating crews could be hired locally as required.

Approximately three weeks would be required to negotiate the estimated 5000 mile trip from the West Coast to AMR. The barge would be compatible with existing dock facilities, assuming a roll-on/roll-off operation. The most attractive features of the barge are, a low total expenditure and a proved "state-of-the-art".

The principal disadvantage to a barge mode of transportation would be the vehicles effectively in the "pipe-line" at any one time. Since it is estimated that approximately three weeks would be required for transit, it would appear that production schedules could be arranged accordingly since, regardless of the transportation time required, AMR facilities could only launch a certain program rate.

18.3.2.3. Water Transport Operations. To complete the transportation system between the road network and waterways, a dock facility is available both at Redstone Arsenal (RSA) and Cape Canaveral. A roll-on/roll-off concept evolved as the most feasible loading and

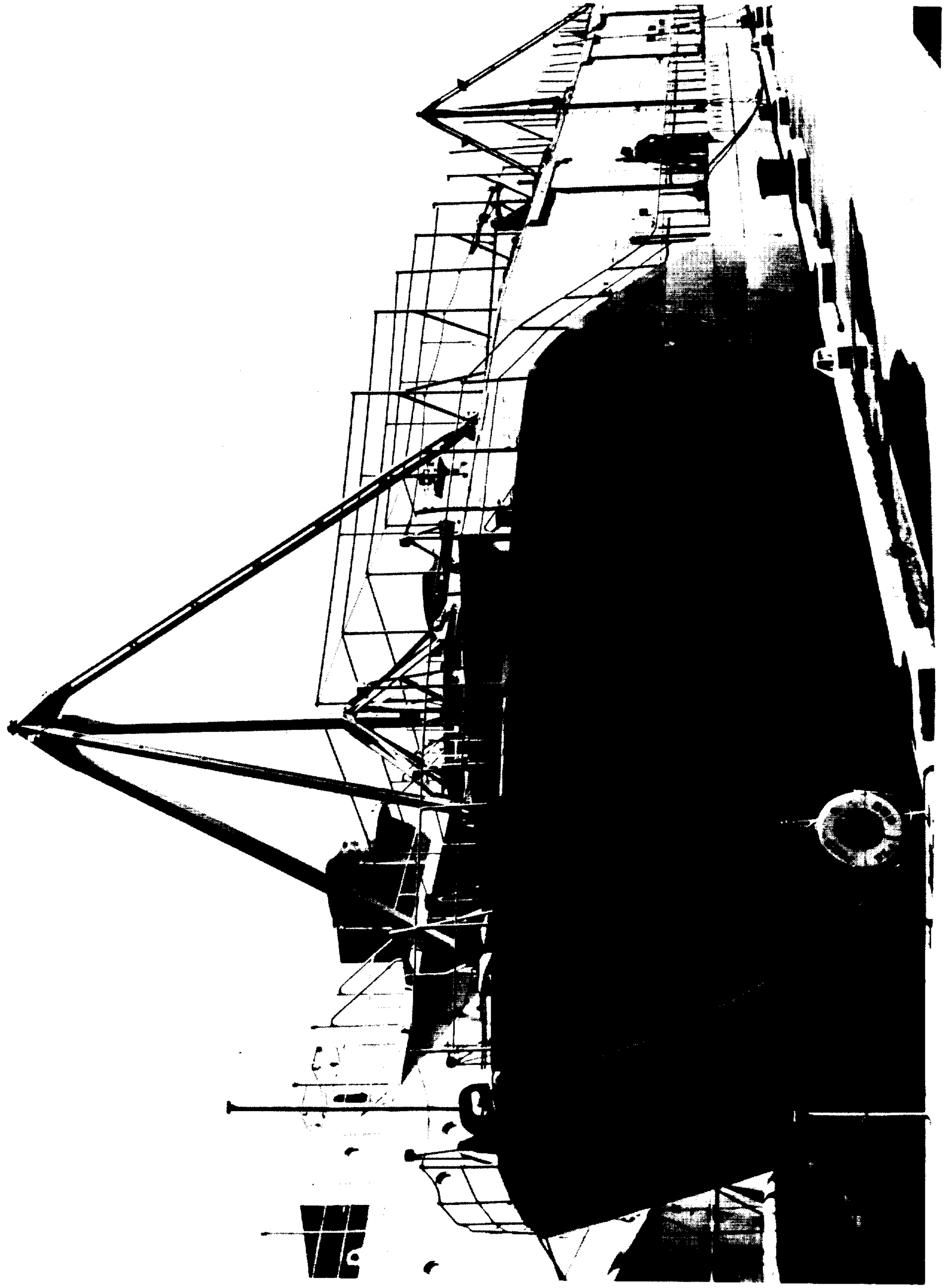

FIG. 18.8. SELF PROPELLED BARGE DURING CONSTRUCTION

unloading system. A barge loading facility has been built at Redstone which facilitates the road to barge movement. Since the Tennessee River level has a seasonal fluctuation and a variable current, a dredge slip is provided at Redstone. The barge is capable of being ballasted at either end to augment the variation in river level. In this way, a river height fluctuation of $8\frac{1}{2}$ feet can be accommodated. The situation at the Cape is much simpler, since the water level is stable within six inches, with practically no current. The dock facility at the Cape is similar to that at Redstone with the exception that a slip is not required.

Prior to arrival ot the booster at the RSA dock site, the barge will be secured in loading position. Also, the steel ramps which span from dock to barge will be set in place and other necessary loading preparations will be made.

Upon arrival of the booster at the RSA dock site, the prime mover will be disconnected and removed from the immediate area. A double-drum continuous-cable winch is then attached to the transporter. A portable winch control box is provided for more reliable and closer control over movement. The two transporter drivers, the winch control-men, and two or three lookouts will then proceed to maneuver the loaded transporter onto the barge. As the forward wheels of the transporter cross onto the barge, reballasting will be required before the rear wheels are allowed to cross. This necessitates an approximate ten minute delay. The booster is then positioned in the storage compartment in conformance to guide lines painted on the deck. Chocking and tie-down will then proceed immediately.

Once the Saturn booster is safely tied down within the barge, instrumentation and preservation equipment for water transportation must immediately be attached. This includes:

(1) Explosive sensors

(2) Humidity indicators

(3) Pressure indicators

(4) Temperature indicators

(5) Vibration and other shock test equipment

(6) Additional breathing equipment to lox and fuel systems.

It is estimated that six to eight hours will be required to complete the barge loading.

Barge unloading at the Cape is similar to the loading operation at RSA, but simplified due to only a six inch fluctuation in water level as compared to about $8\frac{1}{2}$ foot fluctuation on the Tennessee River. Very little ballasting will be required to position the barge for the unloading operation.

The rear winch of the M-26 tractor prime mover (CCMTA facility) will be employed to unload the booster from the barge. When this is accomplished, the winch will be disconnected and the tractor will then be used as a towing vehicle for road transportation to Complex 34.

There is only one water route available from RSA to New Orleans, Louisiana, as shown in Figure 18.9. This route follows the Tennessee, Ohio, and Mississippi Rivers to New Orleans. From this point to the Cape, three routes are available, but the most advantageous at the present appears to be as follows: the Gulf Intercoastal Waterway from New Orleans east for a distance of 350 miles to St. Georges Sound. The Gulf of Mexico is then traversed for a distance of 281 miles to San Carlos Bay, 20 miles southwest of Fort Myers, Florida. From this point, Okeechobee Waterway is followed across Florida for 147 miles to Stuart, Florida. From there, the Florida Intercoastal Waterway leads for 138 miles to the Canaveral Barge Canal.

The total distance from RSA to CCMTA is approximately 2200 miles which will require an estimated three weeks to traverse. Only 281 miles are considered sea miles. This route is never more than 50 miles from shore and a few hours from shelter.

18.3.3 Air Transportation. A number of unsolicited proposals have been made to George C. Marshall Space Flight Center by aircraft companies to use aircraft for transporation of Saturn system stages. Some of these proposals offer the utilization of powered aircraft by modification of already existing transport aircraft or the fabrication of new type transport aircraft tailor-made for the transportation of Saturn system stages. Other proposals recommend the fabrication of glider type aircraft carrying the stages and being towed by powered aircraft.

Following an unsolicited proposal by Douglas Aircraft, the George C. Marshall Space Flight Center awarded a contract to Douglas to make a detailed transportation study including the transportation of S-I and S-IV stages.

18.3.3.1. Transport of Saturn C-1 Stages by C-133 Aircraft. The configuration conceived for transporting the S-IV stage is shown in Figure 18.10. The S-IV stage is mounted over the aircraft spar area on a cradle supported by a welded steel truss assembly. To reduce aerodynamic drag levels to acceptable values, sectionalized fairings would be attached to existing end plates on the missile.

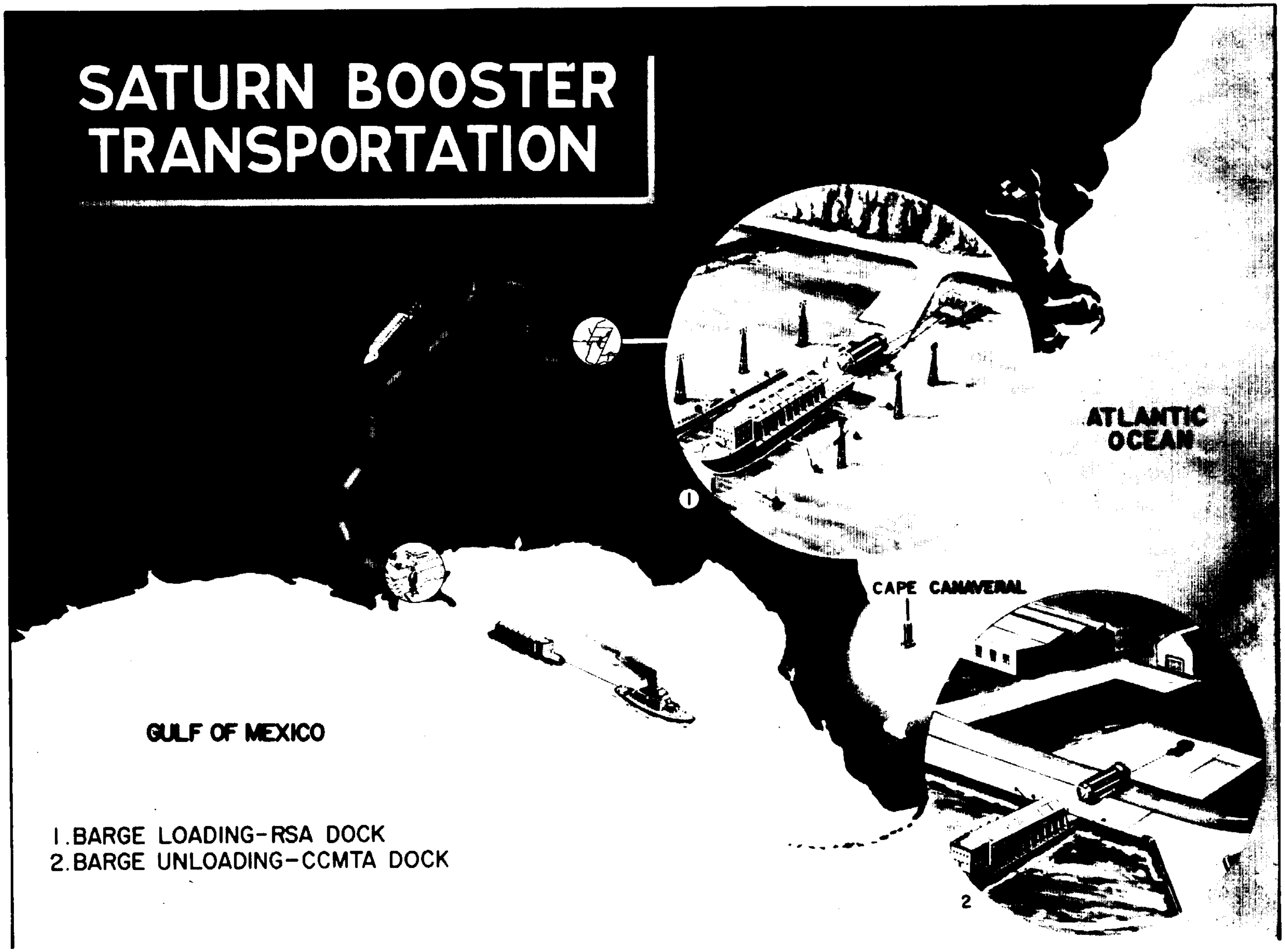

FIG. 18.9. SATURN STAGE S-I TRANSPORTATION

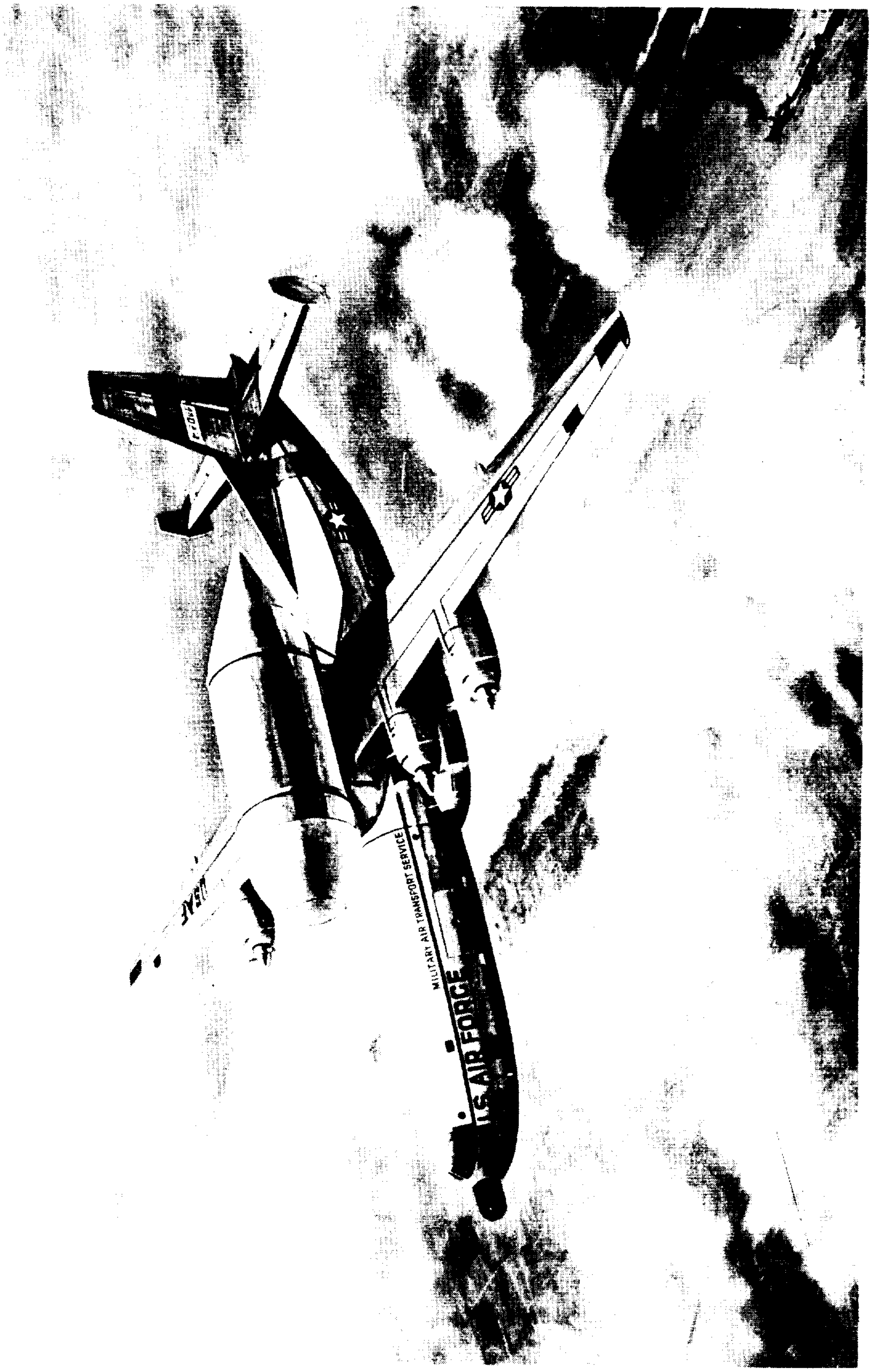

FIG. 18.10. VIEW OF AIRCRAFT "PIGGY BACK" TRANSPORT OF STAGE S-IV

To compensate for loss in directional stability, small fixed vertical stabilizers would be installed at the tips of the horizontal covering of molded fiberglass. Calculated performance of this configuration, presented in Figure 18.11, demonstrates adequate performance to meet the S-IV stage transportation requirements.

In studying the configuration involving the S-I, it became apparent that considerable effort was required in fairing out the end areas of the tanks to reduce buffet and drag problems to an acceptable level. This led to the possibility of providing a faired container in which the S-I stage could be placed for transportation. Although this system would incur additional weight penalty, it would enjoy distinct advantages in not requiring attachment of fairings to the stage and would also decrease the degree of exposure during transport. In addition, this configuration would provide a more universal system in that the basic container could accommodate one S-I stage, one S-II stage, or two S-IV stages.

The obvious problem area of the various proposed configurations is that of effect of the air flow around the capsule on the tail surfaces of the C-133. This, then, is the area of prime concern in the wind tunnel model tests, and configuration variations have been prepared to investigate potential solutions. The basic configuration is shown in Figure 18.12. In addition, tests have been made of a configuration wherein twin vertical tails replace the existing single vertical. Should this prove to be necessary, it then becomes feasible to install the S-I with the engine section forward which results in a simpler installation on the C-133B (Fig. 18.13). In this event, however, the overall program cost would increase due to the requirement for design and test of a complete new empennage assembly. The estimated performance for the system is shown in Figure 18.14 and demonstrates adequate capability to meet the S-I transport mission requirements. Thus far, air transportation of the S-IV stage only has received serious consideration.

18.3.3.2. Transport by C-130A Aircraft with Glider. Preliminary proposal studies have been made by Temco Aircraft Corporation, Dallas, Texas, in which a C-130A is used as the tractor aircraft to tow a Temco-developed glider (Fig. 18.15).

This concept was proposed by Temco after a study of all transportation modes, considering cost and operational requirements. It is realized on a transcontinental flight of 2500 nautical miles that either two fueling landings or mid-air refuelings would be required. Modification costs to the tractor aircraft are included.

The principal advantage of this concept is that one tractor aircraft can serve more than one glider, allowing for ground loading

NOTES:
1. ENGINE SPEC: P&W A 3541-B
2. NO AIRBLEED
3. 2% EXHAUST DUCT LOSS
4. 85% RAM RECOVERY
5. DATA BASIS
   a. BASIC C-133 DRAG-FLIGHT TEST
   b. SATURN DRAG – AEW W.T. TEST
6. OPERATING WEIGHT: 124,985 LB
7. MATS DOMESTIC RESERVES
   a. 1.5 HR HOLD AT DESTINATION
   b. 100 N MI ALTERNATE

MISSION DISTANCE
1. SANTA MONICA – MATHER
2. MATHER – HUNTSVILLE
3. MATHER – CAPE CANAVERAL
   MISSION WEIGHTS INCLUDES THE MISSILE TRAILER

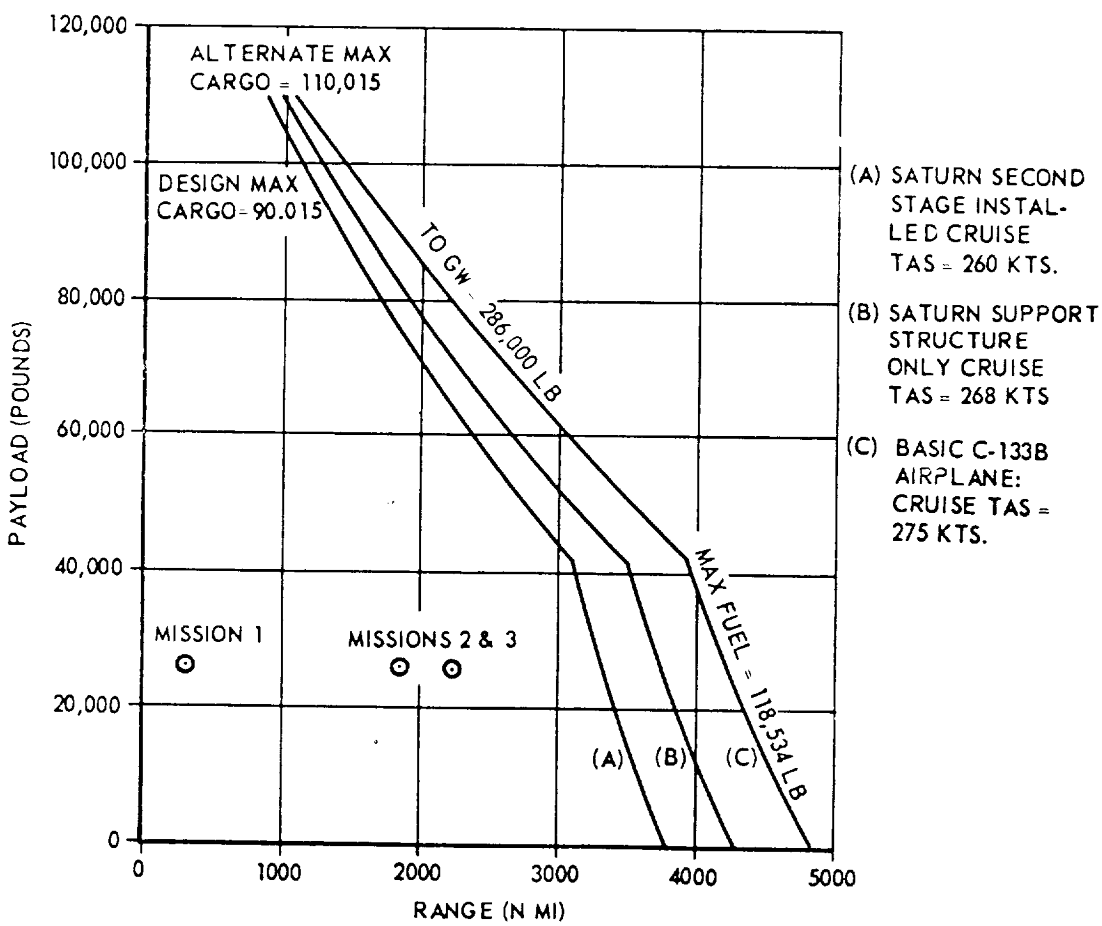

FIG. 18.11. RANGE PERFORMANCE–FOUR P & W T34-P-9 WENG–ICAO STANDARD DAY

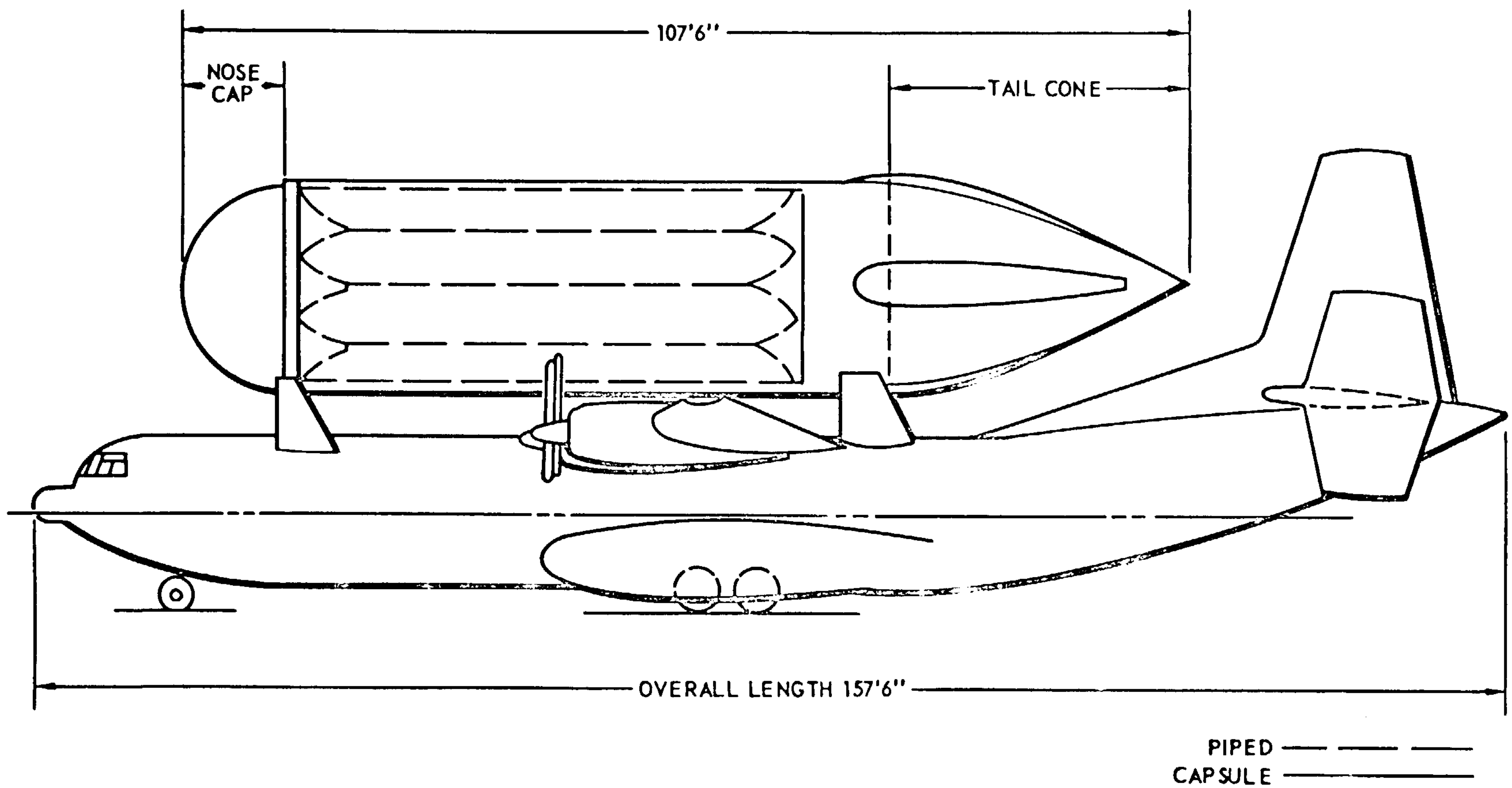

FIG. 18.12 DRAWING OF C-133 WITH STAGE S-I PIGGY BACK

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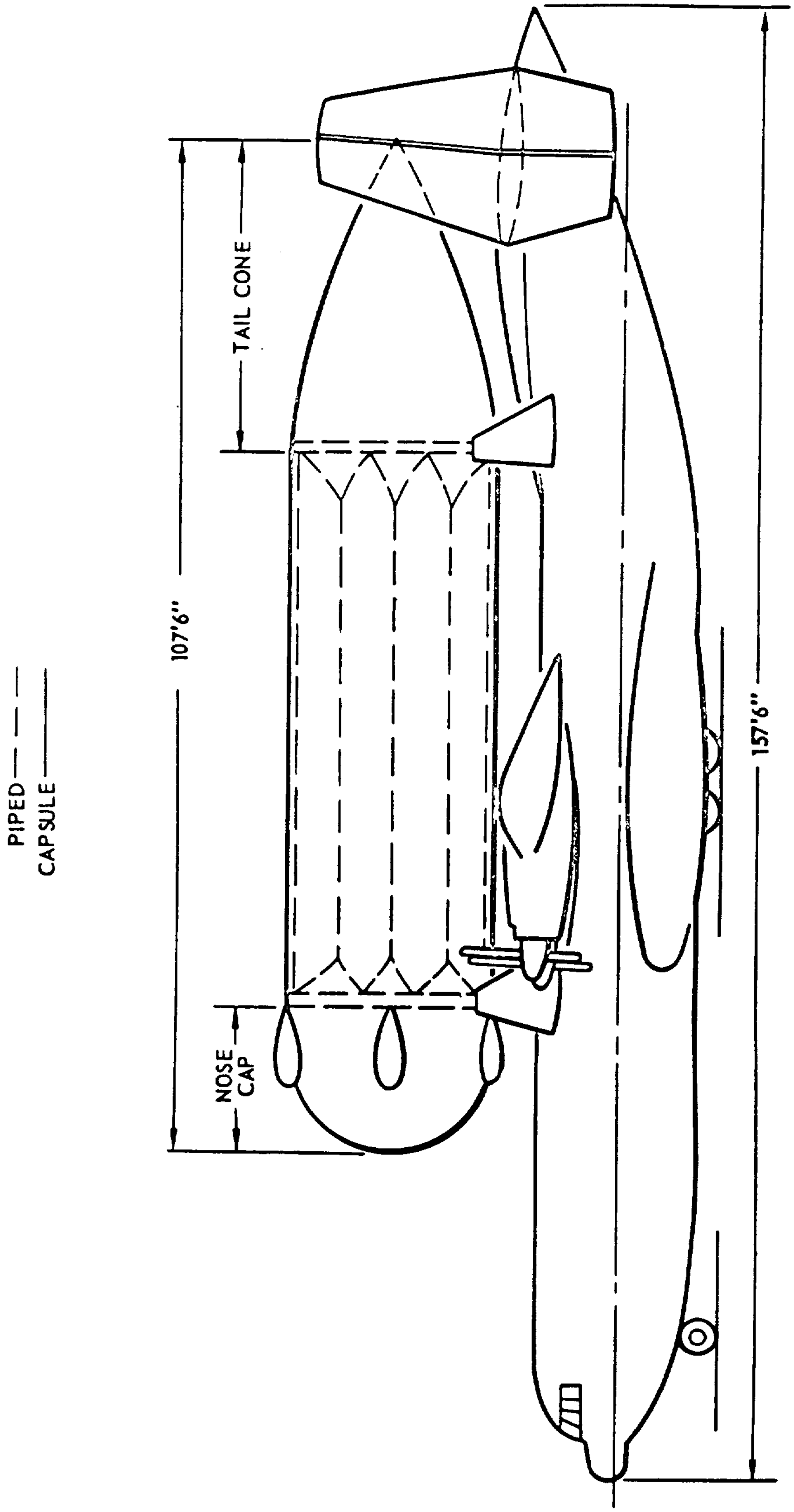

FIG. 18.13 DRAWING OF C-133 CARRYING STAGE S-I PIGGY BACK (ALTERNATE)

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MISSION: HUNTSVILLE TO CAPE CANAVERAL (555 N MI)

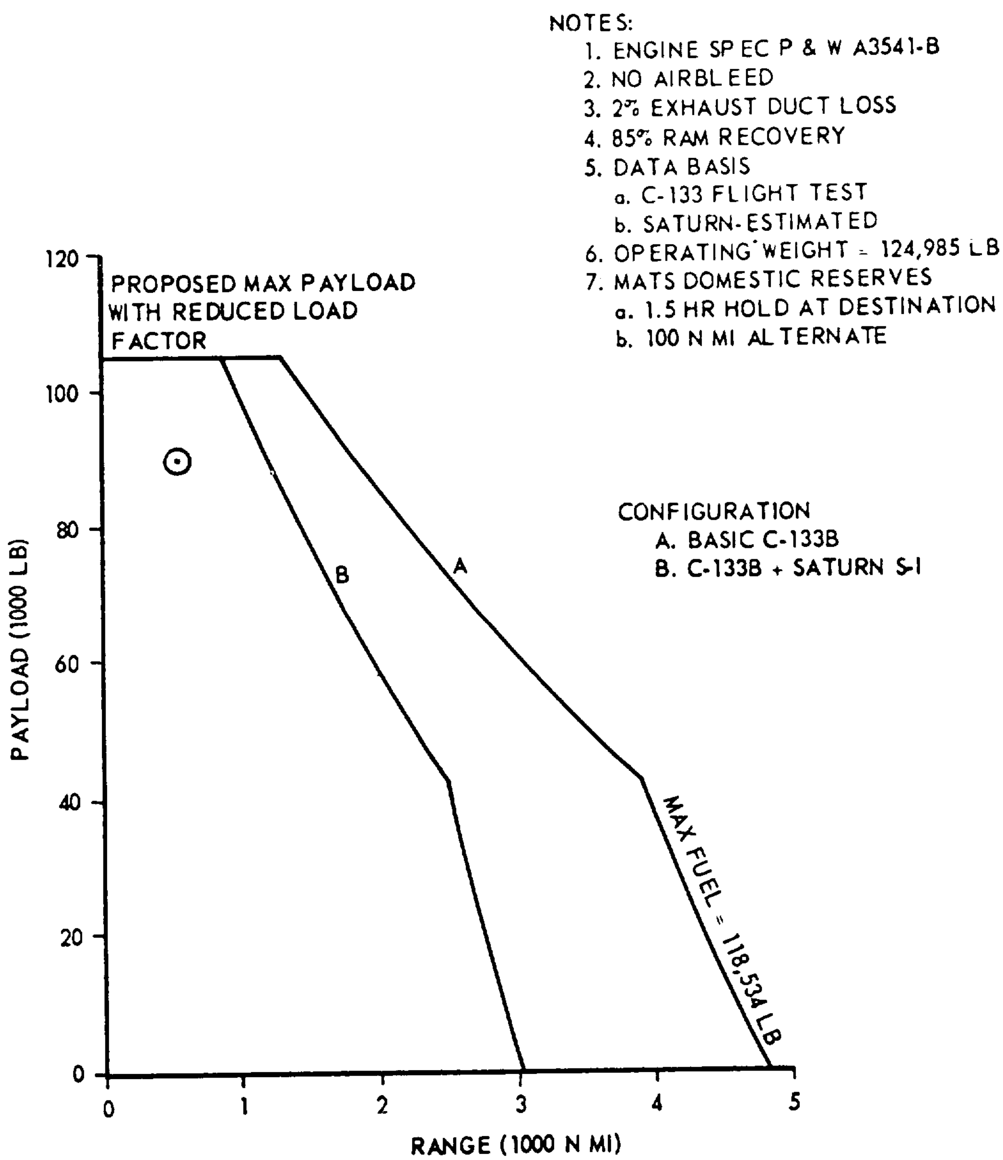

FIG. 18.14 RANGE PERFORMANCE OF MODEL C-133B AIRCRAFT WITH SATURN S-I STAGE INSTALLED (ICAO STANDARD DAY)

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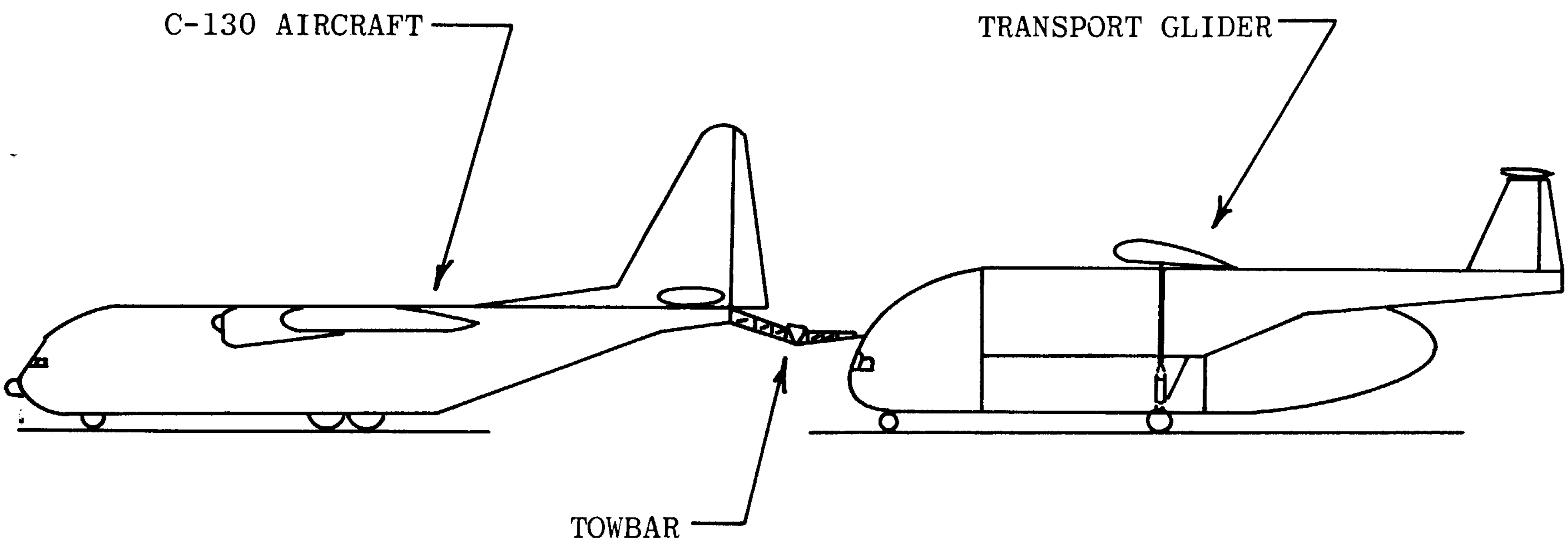

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FIG. 18.15 TOWING ARRANGEMENT, SATURN S-I STAGE time and glider maintenance. It should be emphasized that in the high launch density programs, this aircraft would have a "use factor" equivalent of commercial airlines operation with relief crews available. The operation of tractor aircraft and glider is a proved reality. This particular concept would use a semirigid hydraulically-dampened tow bar during flight. The tractor aircraft can both land and take off with glider attached. Night and moderately inclement weather appear not to hinder operation.

Transportation time for one trip is approximately 21 hours. This time is slightly more than C-133B trip time.

SECTION 19. (C) LAUNCH OPERATIONS AND FACILITIES

The NASA Launch Operations Directorate (LOD) (Fig. 19.1) a part of the George C. Marshall Space Flight Center, serves as the NASA point of coordination for the preparation and submission of all requirements for launch support at the Atlantic Missile Range (AMR) and the Pacific Missile Range (PMR). The coordination with the AMR for all NASA launch operations is provided through the NASA Test Support Office. The LOD provides standard facility and resources support for all NASA projects, including projects Delta and Mercury. The technical management of the Delta vehicle activities at AMR, and the Mercury launch, recovery, and worldwide support requirements are excepted from the LOD's responsibilities. The LOD also serves as the central NASA activity at AMR and PMR with general responsibility for all phases of NASA launch operations including however, only such activities for Delta and Mercury as are specifically assigned. Within the LOD, an Office of Flight Missions is assigned the special functions of local representative of the Director, Office of Space Flight Programs (OSFP) and of providing formal contact between LOD and several flight mission groups using the AMR.

The Launch Facilities and Support Equipment Office of the Launch Operations Directorate is located in Huntsville, and has cognizance over all Ground Support Equipment (GSE) for the Saturn vehicle system. Its primary missions are:

(1) To design, develop, and/or provide technical supervision of design and development of vehicle launch facilities, support facilities, accessories, and recovery systems to meet technical logistical mission requirements including mechanical and electrical components, propellant service storage equipment, handling, transportation, and preservation equipment, support building, and related payload equipment.

(2) To provide an engineering staff with full responsibility on assigned projects in the field of launch and support facilities and firing accessories.

(3) To maintain liaison with Flight Test and Research Center, Corps of Engineers, other associated divisions of MSFC, and contractors in exercising authorized technical supervision to assure conformance with contract requirements and schedules.

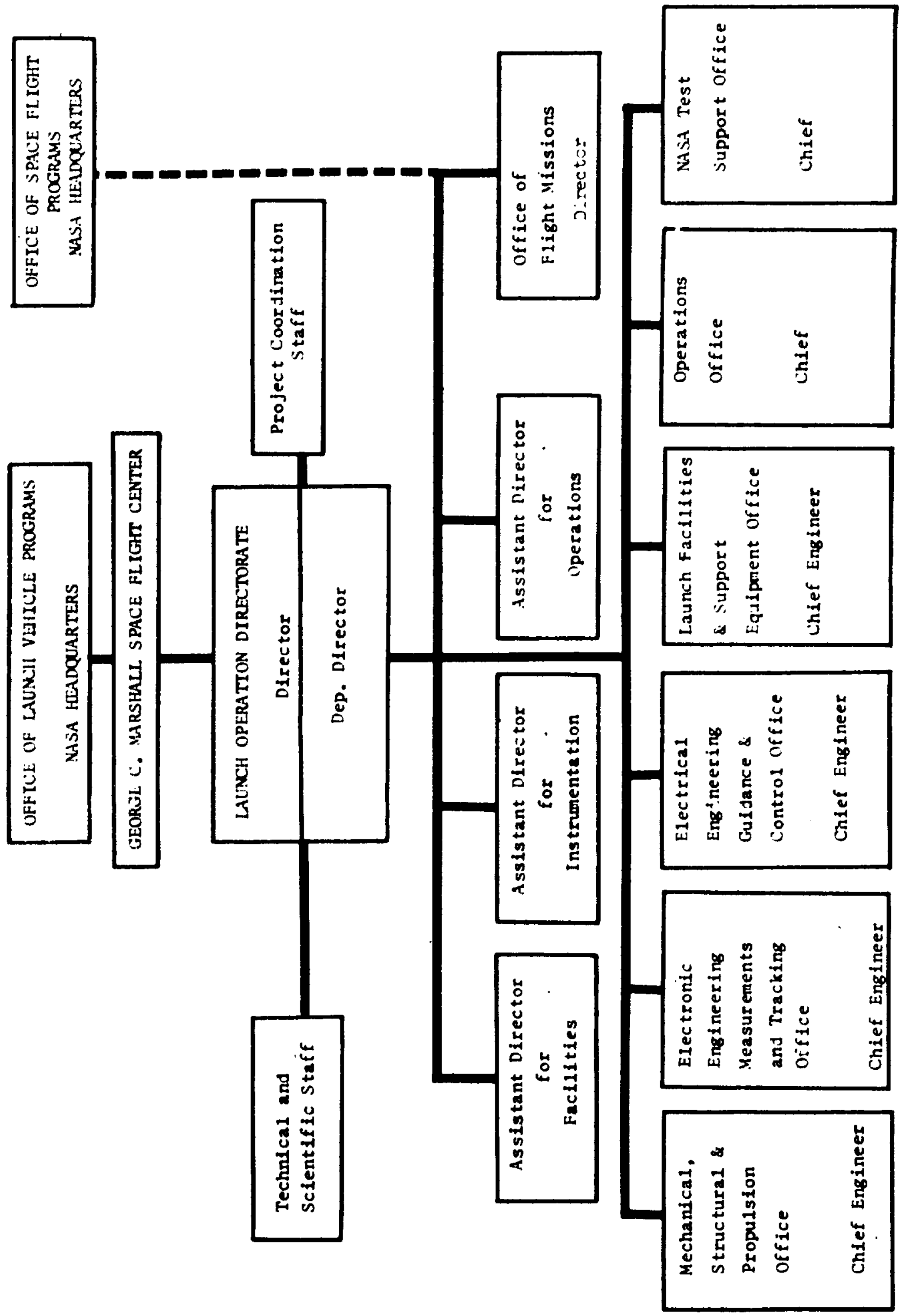

FIG. 19.1 NASA FIELD ORGANIZATION FOR VEHICLE LAUNCH OPERATIONS.

19.1. SATURN/APOLLO LAUNCH OPERATIONS AT AMR

19.1.1. Test Coordination. The Chief Test Coordinator and his staff in LOD are responsible for design, detailing, coordination, and execution of overall tests, launch procedures, and schedules. It will also be the Chief Test Coordinator's responsibility to integrate all checks, tests, operational, and safety requirements of the Apollo capsule into the overall test procedures. For this purpose, early contact will be made with the responsible test conductor for the Apollo capsule and a statement of his requirements will be requested. The final procedures agreed upon will be the result of a combined effort of the LOD test coordinator and the Apollo capsule test conductor.

19.1.2. Testing Techniques. The test philosophy applied to the Saturn vehicle will be essentially the same as that successfully used in the Redstone, Jupiter, and Pershing programs. The procedures follow a sequence beginning with visual inspection and testing of vital components in the vehicle, if possible, or in the laboratory if necessary. These steps are followed by subsystem testing, various special purpose overall tests, and terminate in a final simulated flight test, after which all systems are secured against any status changes. For the Redstone and Jupiter vehicles conventional manual and automatic techniques in the ground support equipment have applied. However, the Saturn ground support equipment (GSE) is being designed to eventually become a fully automatic system using digital techniques controlled by a digital computer. The necessity for a digital computer in the GSE is primarily dictated by the use of a vehicle-borne digital guidance computer in the Saturn system. As other subsystems in the vehicle progress to digital techniques and testing procedures and operational sequences become standard, computer control of these systems will be introduced. It is anticipated that, with the Apollo firings, automatic testing and operation will be in an advanced state, and partial or full incorporation of the automatic sequence concept must be considered for capsule testing.

19.1.3. Prelaunch Operations and Test Phases. At the present time the following prelaunch operations and test phases can be visualized. The Saturn Stage S-I will be erected on the launch site upon arrival at Cape Canaveral and receive a complete checkout of all its components and systems in sequence as mentioned above. To check the control system, it will be necessary to have the instrument unit containing the guidance and control package available at the pad and connected to the S-I control system. If this should interfere with a simultaneous checkout of the S-IV stage, which will take place in the special assembly building, a spare instrument unit or a guidance and control simulator could be used to check the control system of the S-I stage at the pad.

The S-IV stage will be checked out and tested in the special assembly building upon arrival at the Cape by the S-IV stage contractor personnel and their responsible test conductor.

Upon completion of individual capsule and stage tests, a preliminary mating of the S-IV stage and the instrument unit with the Apollo spacecraft will be performed in the special assembly building to ascertain compatibility of mechanical and electrical connections. This mating may be necessary only for the initial launches. Upon completion of stage testing, the S-IV stage will be mated with the S-I stage at the launch site and comprehensive overall testing of the launch vehicle may be performed before the Apollo spacecraft is installed.

After spacecraft installation, overall and simulated flight tests will be made to ascertain whether proper and reliable functioning of the abort and other integrated systems has been achieved.

19.1.4. Preliminary Sequence of Apollo "A"/Saturn C-1 Launch Operations. A preliminary sequence of Apollo "A"/Saturn C-1 Launch Operations, beginning with the arrival of vehicle components at Cape Canaveral, AMR, is given in Table 19.1. The table is carried through checkout, launch and flight evaluation. A more detailed description of the sequence between ignition start and lift-off is given in Figure 19.2. It should be understood that continuously additional details of the checkout of the stages and the launch vehicle will be come available. These details and modifications will largely depend on the experience gained with the early Saturn vehicles, and also on the specific Apollo spacecraft requirements. The checkout, test and launch procedures for Saturn will be adapted to the Apollo requirements as established at that time. Detailed procedures for each individual vehicle and its particular mission will be established and coordinated with the AMR. A complete coordinated countdown for the Apollo "A"/Saturn C-1 vehicle may not be available before the summer of 1962.

19.1.5. Launch Capabilities. It is planned to use the Saturn vertical launch facilities VLF 34 and VLF 37 for the Apollo "A"/Saturn C-1 launch operations. These launch facilities are located at Cape Canaveral, AMR, Florida (Fig. 19.3). For planning purposes, it is assumed that VLF 34 will have an initial launch capability of four vehicles per year, based upon the following:

Vehicle checkout time on pad	2 Months
Pad rehabilitation and preparation	1 Month
TOTAL	3 Months

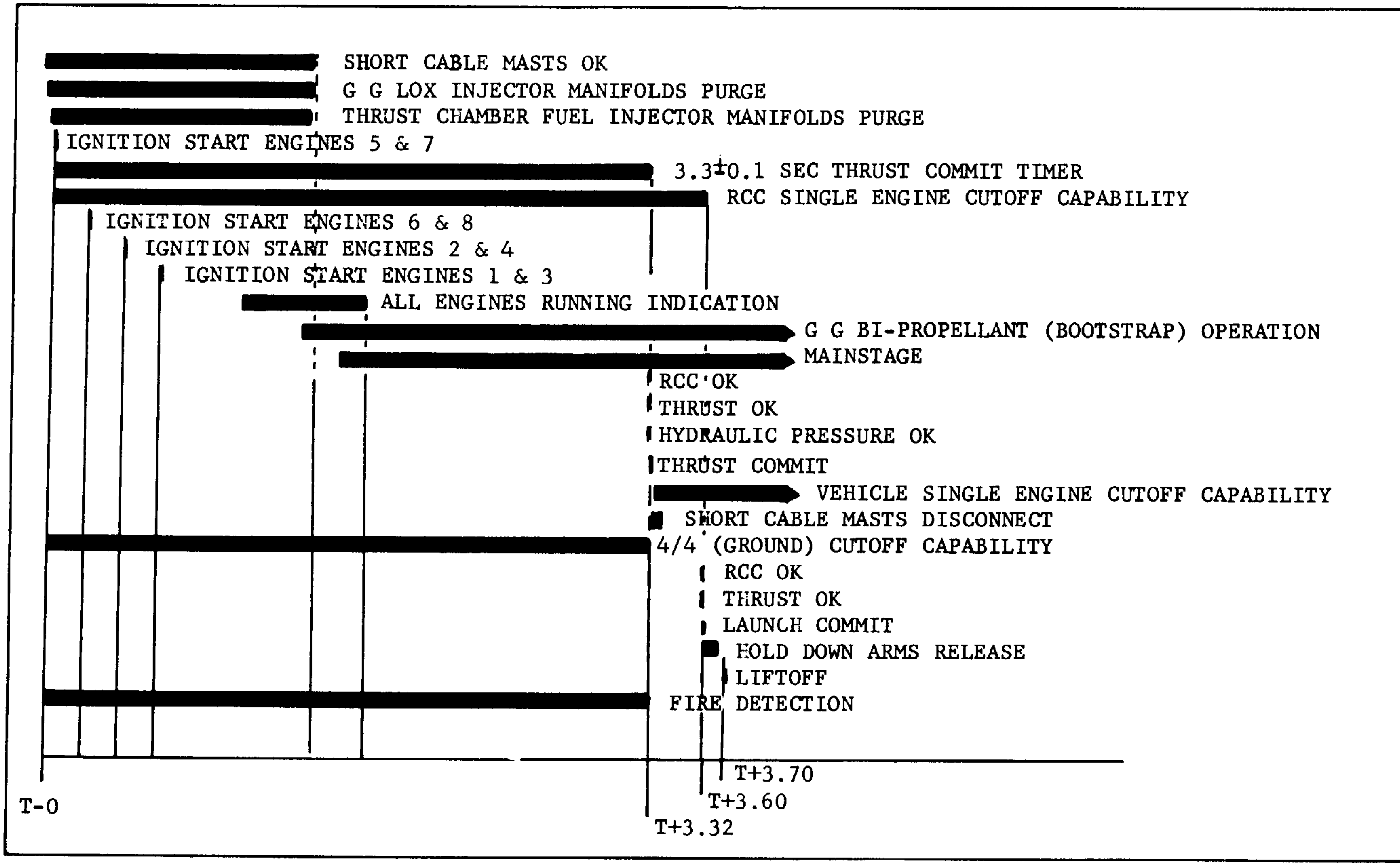

FIG. 19.2 PRESENT SEQUENCE BETWEEN IGNITION START AND LIFT-OFF FOR THE SATURN C-1.

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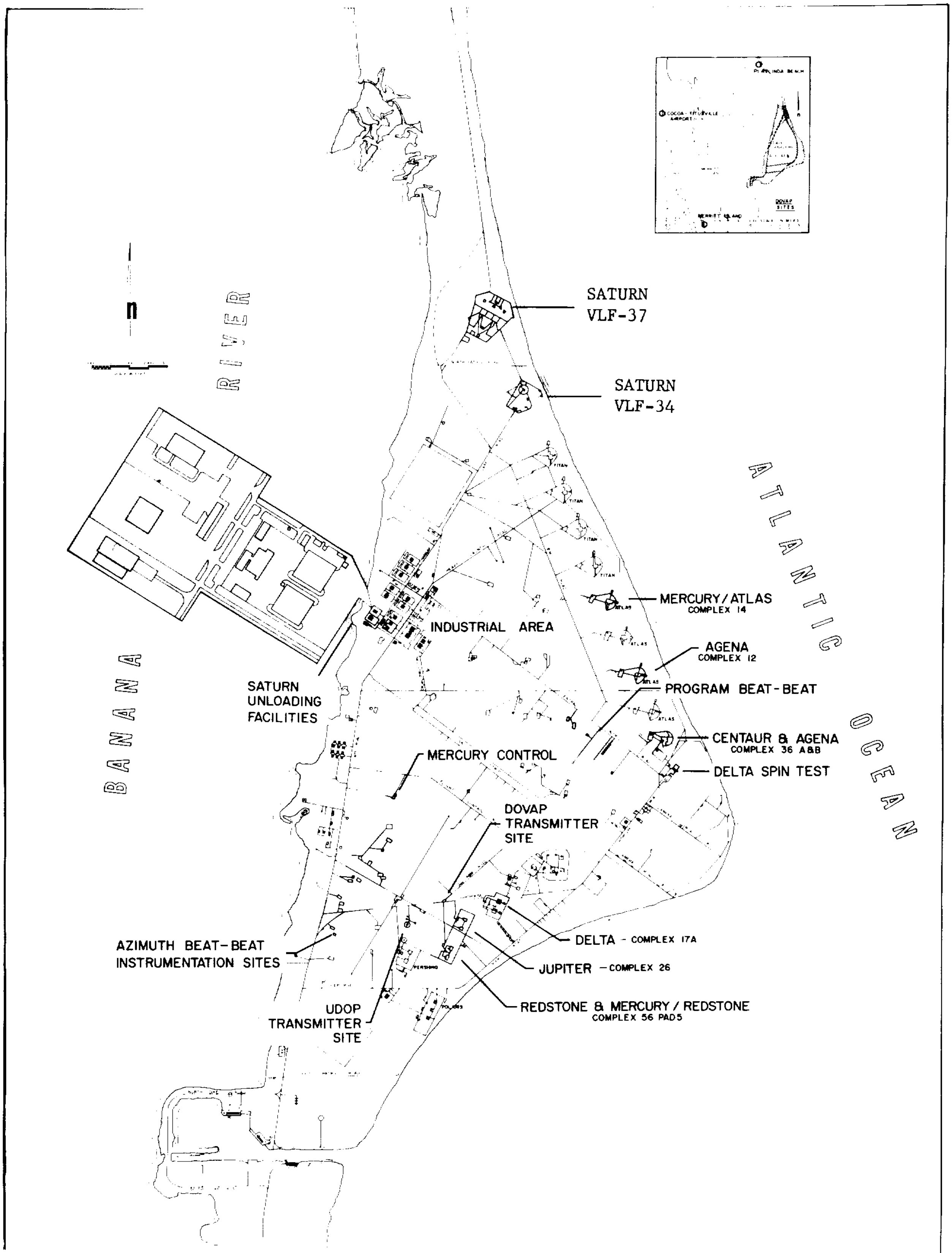

FIG. 19.3 VLF-34 AND VLF-37, AREA LAYOUT AT CAPE CANAVERAL, FLORIDA

(C) Table 19.1

PRELIMINARY SEQUENCE OF APOLLO "A"/SATURN C-1
LAUNCH OPERATIONS

TIME	EVENT
T - 17 Weeks	Stage S-IV Arrives at AMR
T - 3 Months	Previous Launch Begin Launch Pad Rehabilitation
T - 13 Weeks	Instrument Unit Ready for Horizontal Checkout and Mating Tests in Special Assembly Bldg at AMR
T - 13 Weeks	Spacecraft Ready for Horizontal check-out and Mating Tests in Special Assembly Bldg at AMR
T - 9 Weeks	Stage S-I Arrives at AMR
T - 2 Months	Launch Pad Rehabilitation Complete
T -	Launcher Alignment Complete Launcher Angular and Radial Measurements Complete Launcher Ready for S-I Erection
T -	Service Structure in Position at Pad
T -	Gantry Crane of Service Structure in Position for Erection of S - I
T -	S-I Lifted from Transporter S-I Lowered to Launcher Supports Begin S-I Alignment
T -	Make S-I Connections to GSE
T -	Complete S-I Leak Tests

Table 19.1 (Continued)

TIME	EVENT
T - 4.5 Weeks	Stage S-I Checkout on Pad Complete S-I Ready for Erection of S-IV
T - 4.5 Weeks	Complete Horizontal Checkout and Mating Tests of S-IV, Instrument Unit, and Spacecraft in Special Assembly Bldg at AMR
T - 4.4 Weeks	Erect S-IV on S-I with the Aid of the Service Structure Begin S-IV Alignment
T -	Make S-IV Connections to S-I and GSE
T - 4.2 Weeks	Erect Instrument Unit on Stage S-IV. Begin Alignment of Instrument Unit
T -	Make Instrument Unit Connections to S-IV and GSE
T - 4 Weeks	Erect Spacecraft on Launch Vehicle. Begin Alignment of Spacecraft
T -	Make Spacecraft Connections to Instrument Unit and GSE Erection of Space Vehicle Complete Begin Space Vehicle Launch Checkout
T -	Load RP-1 Fuel in S-I
T - 2.5 Hrs.	Remove Service Structure from Pad
T -	Load Cryogenic Propellants: LH_2 in S-IV, and Lox in S-I and S-IV
T - 30 Min. (Approx.)	Insert Spacecraft Crew via Umbilical Tower

Table 19.1 (Continued)

TIME	EVENT
T - 6 Min. (Approx.)	Preparation of Vehicle Complete S-I Ready to Proceed S-IV Ready to Proceed Instrument Unit Ready to Proceed Spacecraft Ready to Proceed Initiation of Automatic Sequence for S-I Firing Command Initiate Fuel Tank Pressurization Initiate S-I Boat Tail Purge Initiate He Bubbling in S-I Lox Lines S-IV Ready for Automatic Sequence Complete He Bubbling of S-I Lox Lines Initiate Pressurization of S-I Lox Tanks Indication of: (A) S-I Lox Tank Pressurization Completion (B) S-IV Flight Ready Signal (C) Instrument Unit Flight Ready Signal (D) Spacecraft Flight Ready Signal
T - 35 Seconds (Approx.)	Initiate Power Transfer from Ground to Vehicle
T*	Ignition Start of S-I Engines (In Pairs of 2 at 100 Millisecond Intervals)
T + 1.35 Seconds	S-I Mainstage (All 8 Engines)
T + 3.3 Seconds	Check All Systems, Incl: (A) Combustion Instability (B) Hydraulic Pressure (Pump Outlet) (C) Fire Indications (D) Fuel Pressure (Pump Outlet)

Table 19.1 (Continued)

TIME	EVENT
T + 3.6 Seconds	Launch Commit Initiate Release of 8 Holddown Arms
T + 3.7 Seconds (Trajectory Time Zero)	Liftoff** Holddown Arms Release All Umbilicals Release
T + 640 Seconds (Approx.)	S-IV Engine Cutoff Signal
T + 24 Hours	Quick Look Flight Evaluation
T + 3 Weeks	Preliminary Flight Test Report
T + 6 Weeks	Final Flight Test Report

NOTES

* For a more detailed illustration of the sequence between ignition start and liftoff see Figure 19.2.

** For a more detailed flight sequence see Section 11.1.

During the two month vehicle checkout time, checkout procedures for the S-I stage, the S-IV stage and the Apollo spacecraft will have to be integrated. The S-IV stage and the Apollo spacecraft should arrive at the special assembly building sufficiently in advance of the launch date for final horizontal checkout of the S-IV and mating tests prior to erection at the pad. It is estimated at this time, that the S-IV will be erected approximately 4-1/2 weeks after S-I erection.

VLF-37, with two pads, will have a capability of six launches per year, if the same vehicle checkout and pad preparation times are assumed. Pad rehabilitation on one pad can overlap vehicle erection on the other, but two vehicles never occupy adjacent pads of the same complex at the same time. For this reason, the vehicle checkout time dictates the overall firing rate.

An increase to eight launches per year for VLF-37 and to five per year for VLF-34 can be achieved with a reduction of vehicle checkout time to 6 weeks. With the incorporation of digital checkout techniques for the vehicle and spacecraft, this reduced checkout time appears feasible.

19.1.6. Apollo Flight Crew Insertion. Specifications pertaining to the specific Apollo requirements for crew insertion are not available at present. However, it is anticipated that crew loading will necessarily be via the umbilical tower, since the service structure will be removed earlier in the countdown procedure. Figure 19.4 is illustrative of this operation at the launching facility.

19.1.7. Normal and Emergency Egress Procedures for Apollo Flight Crews. To date, no requirements or specifications have been received against which adequate planning for egress procedures could be accomplished. This section is, therefore, in anticipation of forthcoming requirements based upon Mercury experience. The Mercury program has emphasized the need for adequate emergency procedures for the protection of astronauts and supporting ground personnel. The premise which developed from the Mercury program was simply: "Provide the maximum safety to the astronauts with a minimum of risk to the rescue squad." Basically, two methods for emergency egress may be considered.

Method 1 may be considered an expansion of egress procedures developed for Project Mercury-Redstone. These procedures evolved as an improvisation, because of limitations dictated by use of available facilities, equipment, and the nature of the spacecraft design. Mercury-Redstone procedures must, therefore, not be considered an optimum. Method 1 utilizes special fire fighting equipment, specially trained crews, emergency provisions on the umbilical tower and structure, and adjustment of the countdown. In the establishment of procedures such

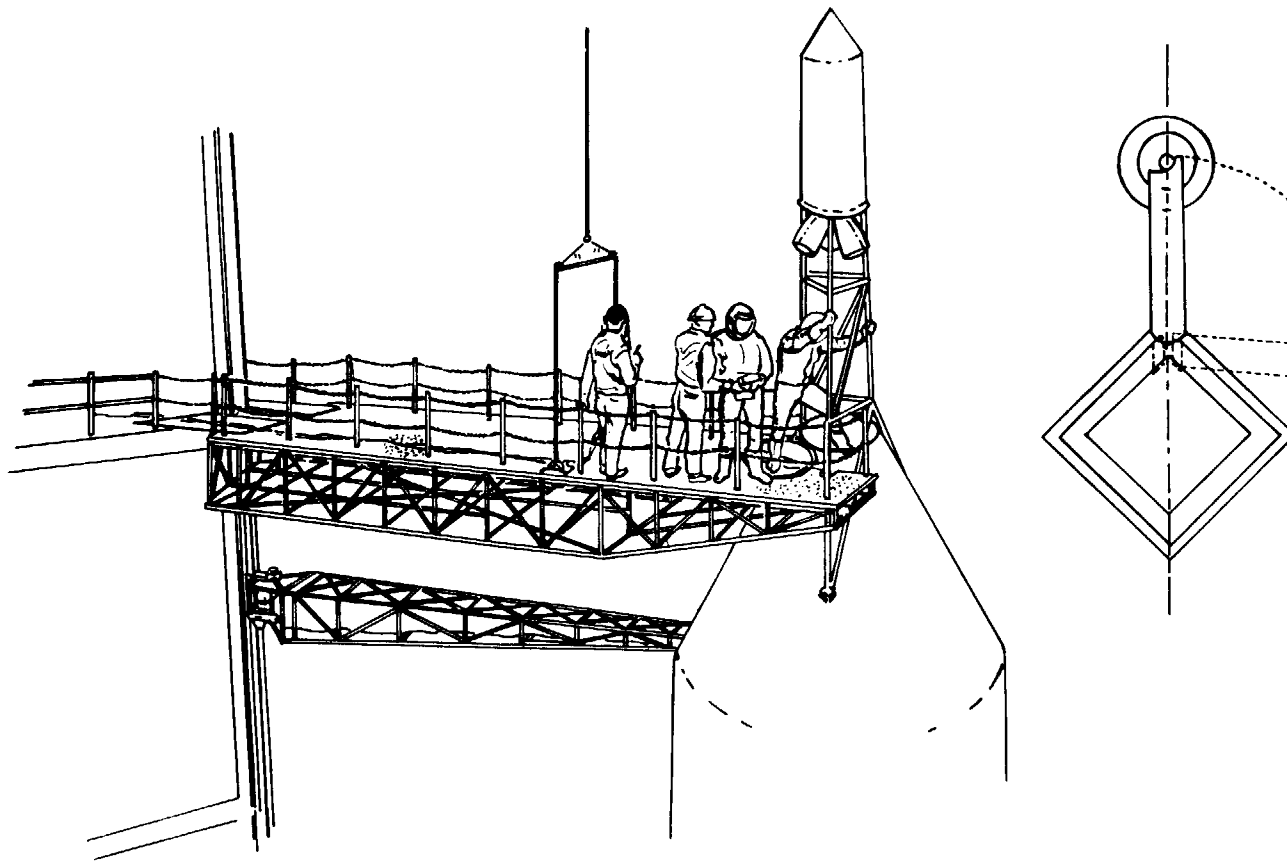

FIG. 19.4 TYPICAL CREW INSERTION PLATFORM ON UMBILICAL TOWER.

as these, the total man-seconds of exposure time during an emergency egress procedure is normally high. If this method is to be used in the case of Saturn, design modifications to the structure, pad area, and/or umbilical tower will be involved. Time may restrict the design and construction of any sophisticated emergency egress provisions, and the umbilical tower will probably have to be modified to accomplish this. The required facilities are a function of spacecraft design which is not sufficiently firm for a facilities design analysis at this time. The following major factors should now be considered to implement this method:

(1) Chutes, lifelines, or an express elevator from the hatch to a safe area with minimum exposure time to a hazardous space vehicle should be provided.

(2) Countdown procedures should be aligned to eliminate the necessity for an insertion crew, if embarkation occurs after propellant loading.

(3) Sophisticated fire fighting equipment should be permanently installed and remotely controlled.

(4) A complex communications net to all supporting functions should be provided.

(5) "Safing" methods should be provided for any ordnance aboard the entire space vehicle.

Method 2 would utilize the capsule escape system for all emergencies wherein the astronauts may be endangered. This approach requires that situations be preconceived wherein a normal egress and an emergency egress are suitably and distinctly defined. A normal egress may be generally defined as evacuation of the spacecraft by the 3 member crew in a non-critical period of time without additional exposure to hazard. An emergency egress may be defined as a hazardous condition for the crew which requires immediate escape and which would be further aggravated by an evacuation past the hazardous area. This method would utilize the escape propulsion system of the spacecraft at any time from final propellant loading through powered flight in the atmosphere.

Implementation of method 2 necessitates that certain design criteria be incorporated in spacecraft and launch vehicle. Some major design considerations to implement this approach are generally as follows:

(1) An automatic "reverse count" to include such items as "safing" the destruct system, disarming pyrotechnics, closing propellant vents, etc.

(2) Blast and fire protection from the escape system for the remaining parts of the space vehicle and nearby facilities.

(3) Maximum automation of the countdown of both spacecraft and launch vehicle.

(4) Orientation of the countdown to embark the crew via the umbilical tower after the completion of propellant loading, and as late in the countdown as possible without utilization of an insertion crew.

Method 2 also requires consideration in spacecraft design to:

(1) Provide a more comfortable escape "G" level.

(2) Provide some control over the impact point by the flight crew or launch control center.

(3) Provide sufficient altitude to assure that time is available to the flight crew for verifying recovery equipment functioning.

(4) Scheduling countdown to eliminate the need for a ground insertion crew after fuel loading.

(5) Provide compatibility of the spacecraft system with the automated countdown and "reverse" countdown.

(6) Provide for quick opening of the hatch by the flight crew.

(7) Provide safe water and land impact capability.

It is recommended that method 2 be utilized. Emergency egress procedures can thus be eliminated without jeopardy to the flight or ground crew by utilizing already available media. Implementation of this method will require numerous design considerations for spacecraft - launch vehicle and launch facility, which must immediately be resolved by joint effort. It is, therefore, essential that guide lines be established for whatever method is used. A decision is required at an early date, so that design provisions may be incorporated and flown in the launch vehicle to establish a good confidence level prior to the first manned flight.

19.2. VERTICAL LAUNCH FACILITY (VLF) NO. 34

19.2.1. Area Layout of VLF-34 (Figs. 19.5 and 19.6). The general layout has the launch control center 1,050 feet from the launch pad. The service structure rails presently extend 600 feet from the launch pad. The addition of hydrogen upper stages increases the damage potential; therefore, the 600-foot rails may have to be extended to as much as 1,200 feet to protect the service tower from a pad explosion.

LEGEND.
1. LAUNCH PEDESTAL
2. SERVICE STRUCTURE
3. LOX STORAGE
4. FUEL LINES
5. SKIM BASIN
6. LAUNCH CONTROL CENTER
7. OPERATIONS SUPPORT BUILDING
8. SITE OF VLF-37

FIG. 19.5 OVERALL VIEW OF VLF-34 WITH SITE OF VLF-37 IN BACKGROUND.

FIG. 19.6 VLF-34, PAD AREA AND SERVICE STRUCTURE

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19.2.2. Launch Control Center. The launch control center is an igloo-type design for protection of personnel and instrumentation during launching. The dome is constructed of reinforced concrete, 5 feet thick, covered with 7 feet of fill dirt and an additional 4 inches of concrete to provide protection against extremely high blast pressures. Control instrumentation, measuring racks, and firing consoles for actual launch operations are housed in the launch control center. Visual observation is provided by closed circuit TV and by two periscopes protruding from the top of the dome.

19.2.3. The Saturn Service Structure (Figs. 19.7 and 19.8). The Saturn service structure, required to accomplish vertical mating of the Saturn launch vehicle and to provide work platforms for the checkout and service operation, is an inverted U-shape, rigid-box truss-frame design. The center top box truss, with the bridge runway hoisting equipment, is approximately 130 feet long, 70 feet wide, and 50 feet deep. It rests on two box truss columns of 70 x 37 foot across sections, each of which rests on a two-story base section, 37 feet wide and 70 feet long. The total structure is 310 feet high, 130 feet wide, and 70 feet long at the base.

A bridge crane, with vertically positionable runway trusses, will have individually controlled 40- and 60-ton hoists with a 245-foot hook height. The structure is also provided with six fixed platforms within the tower legs. Each platform has 790 square feet of floor area and six enclosed retractable platforms with 814 square feet of floor area each, and is designed for a capacity of 12 personnel and 600 pounds of equipment on each half. Vertical adjustment of these platforms is provided. The structure is also provided with two personnel- and one freight elevator. Two 25-foot high base sections contain engineering and laboratory space and power equipment required for self-locomotion of the tower on a dual-track railway into the parking position.

For the handling of larger vehicle configurations, this structure may be extended to a maximum hook height of 281 feet and an overall height of 350 feet.

19.2.4. Umbilical Tower. The umbilical tower is used to support and service the umbilical arms. The umbilical swing arms extend horizontally toward the vehicle in the same vertical plane as the diagonal legs of the tower and are capable of swinging 135 degrees to either side from the vertical pivot of the tower leg.

In addition to supporting and servicing the umbilical arms, the umbilical tower supports and houses the equipment required to service the Saturn C-1 upper stages prior to launch. Included are service lines such as electrical cables, distributors, pneumatic lines, cryogenic main

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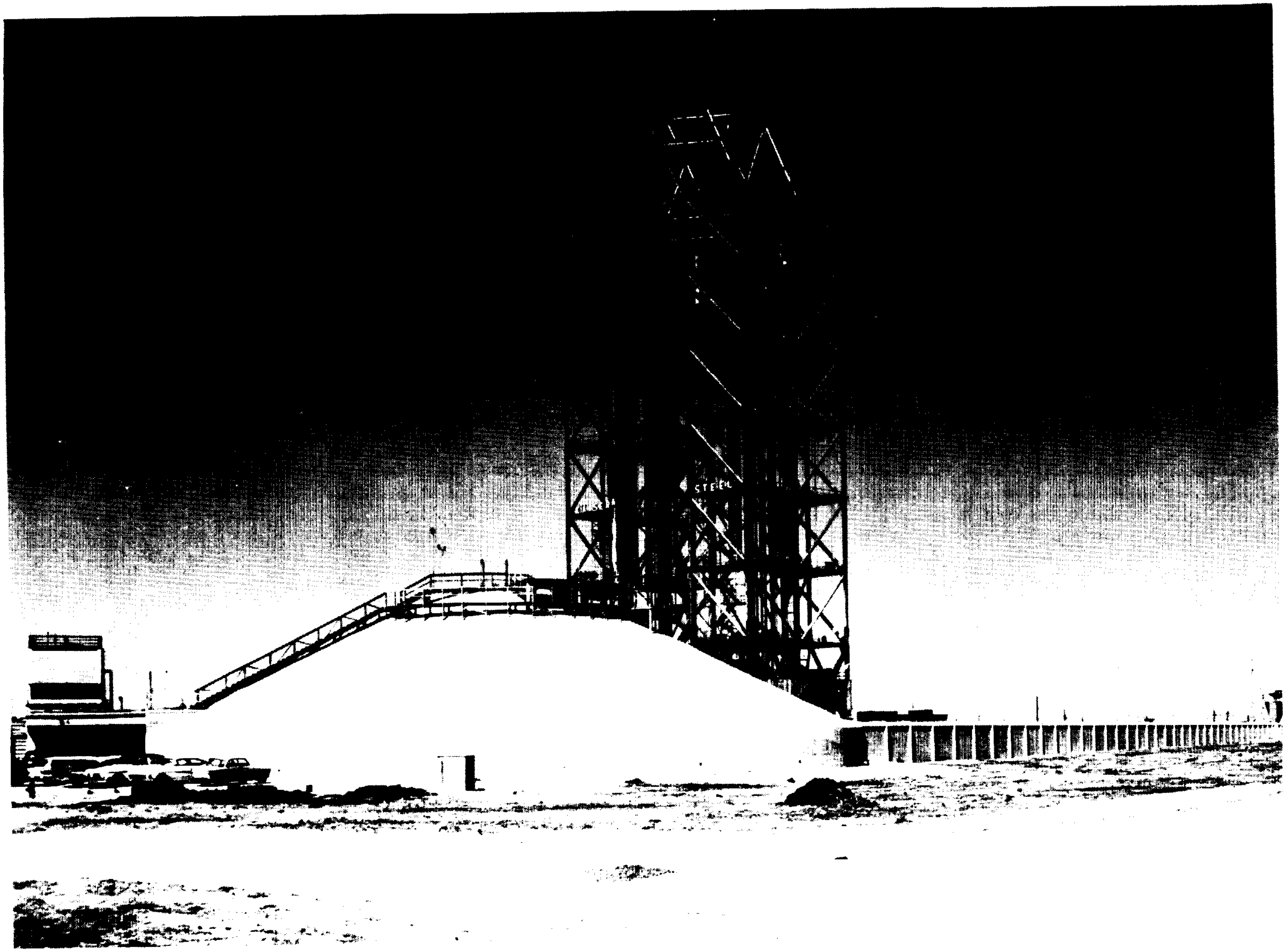

FIG. 19.7 SATURN SERVICE STRUCTURE, VLF-34.

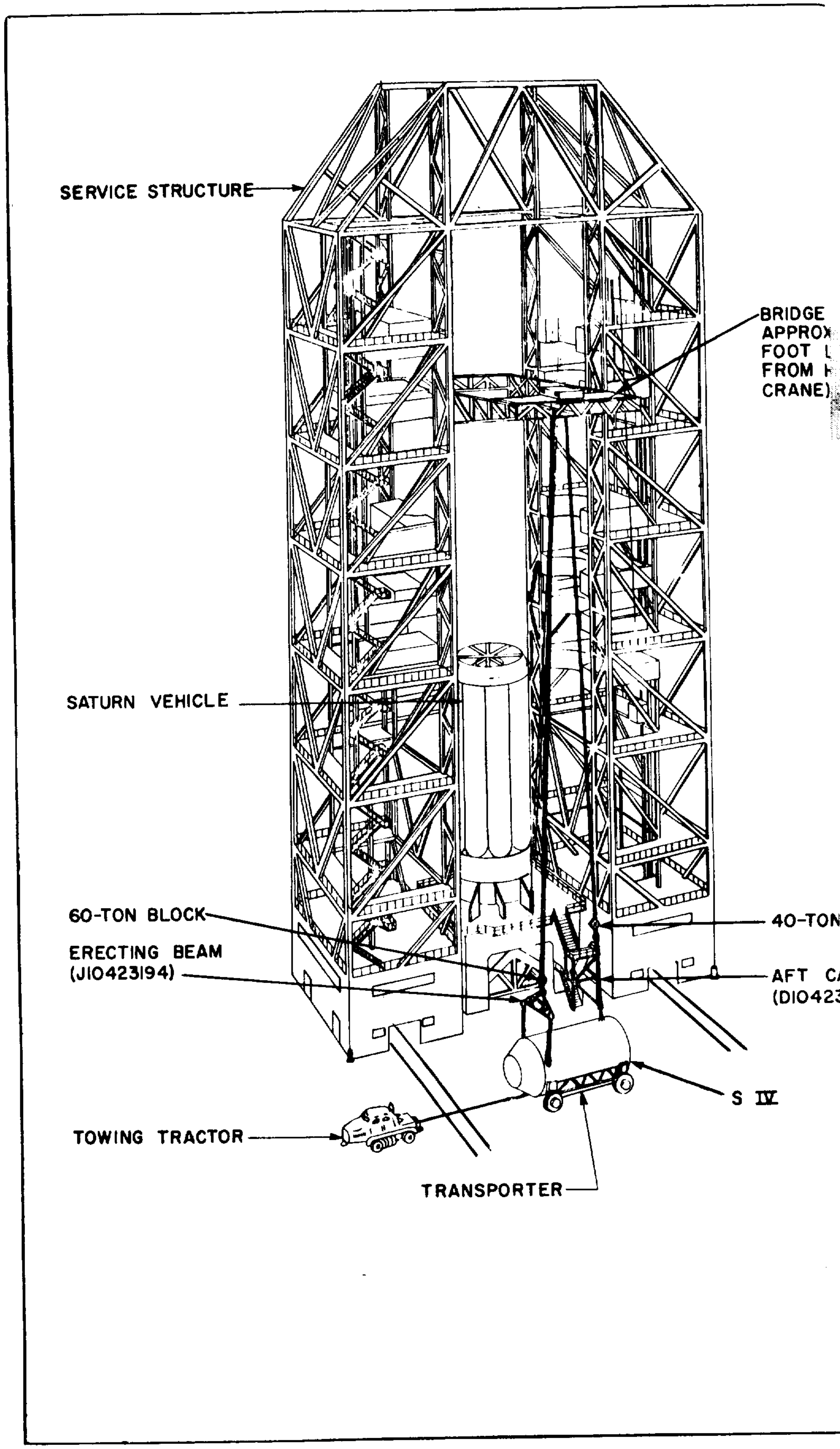

FIG. 19.8 TYPICAL ERECTION OF UPPER STAGES WITH SATURN

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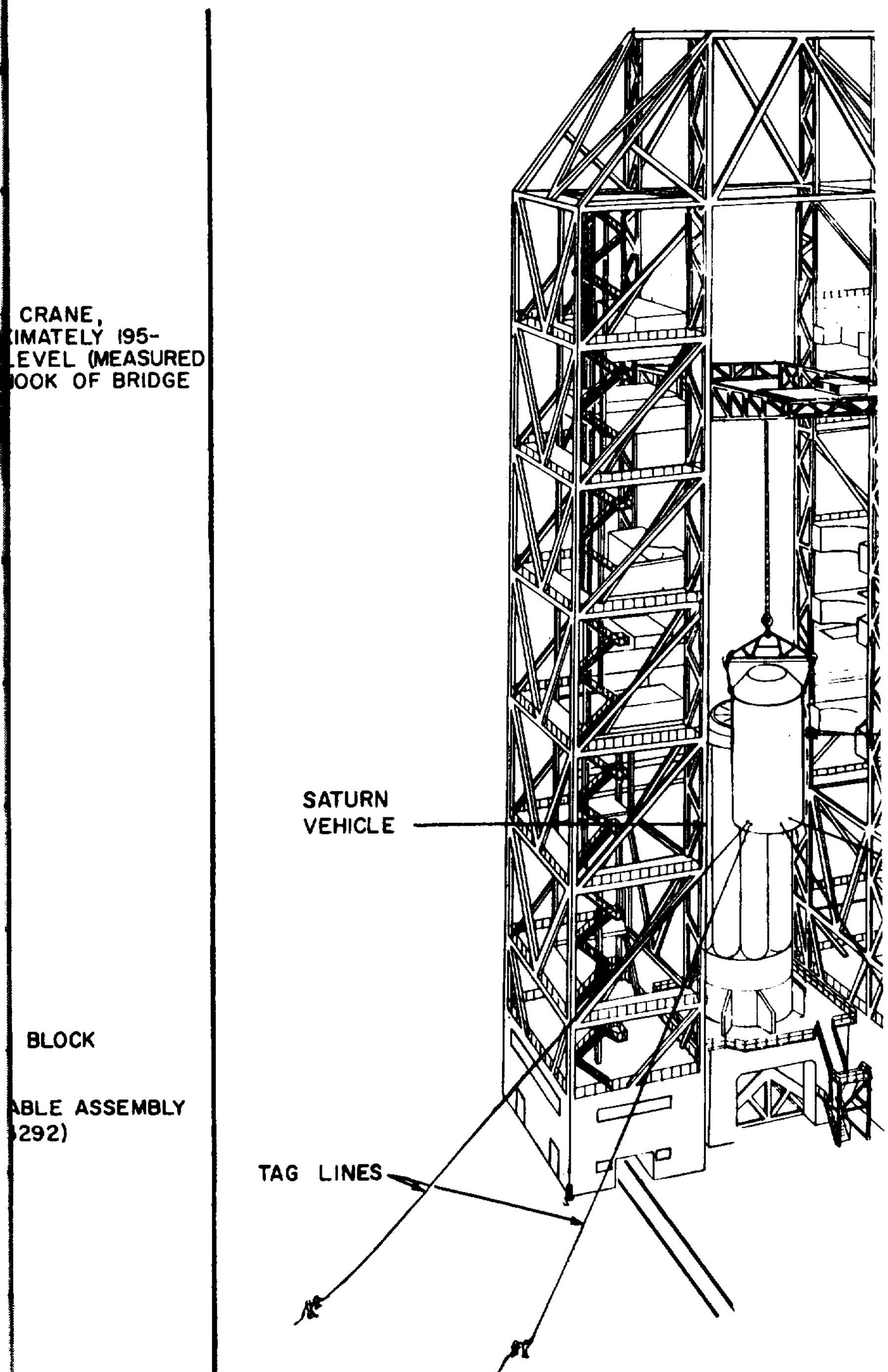

SERVICE STRUCTURE, VLF-34.

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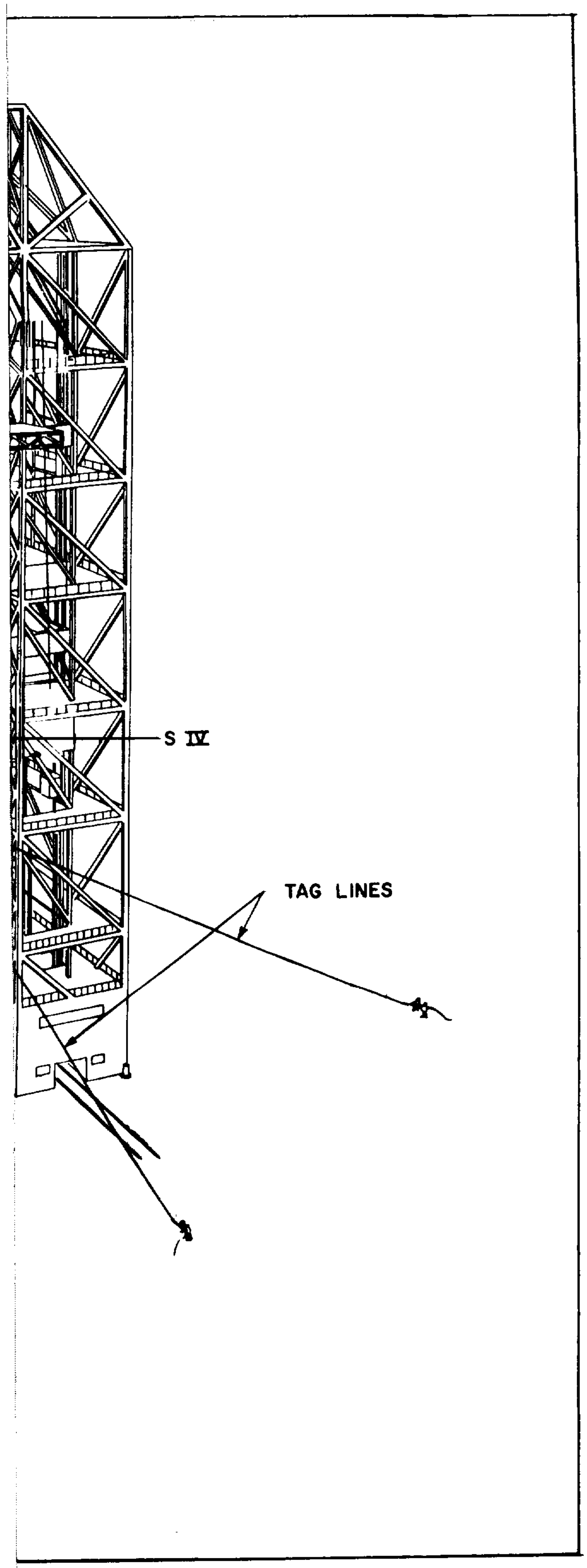
S IV
TAG LINES

fill and replenishing lines, propellant valve complexes, helium heat exchangers, pneumatic checkout consoles, mechanical refrigeration environmental conditioning ducts, and hydrogen vent stacks.

The first 27 feet of the tower is nearing completion. The remainder will be added after launch of SA-4. The upper stages are dummies on the first four flights. The C-1 umbilical tower for VLF-34 will be a 240-foot structure. An extension to 270 feet is included in the design.

Included are lightning protection, electrical troughs, distributors and feeder lines. The tower will be equipped with a safety ladder and fully decked service platforms at each 20-foot interval starting at approximately 50 feet above ground level. A jib boom will be provided at the top of the tower to facilitate handling of electrical lines, electrical cables, umbilical arms, and associated ground checkout consoles required on the tower. An elevator for carrying passengers and small items of hardware to the various service platforms will be provided. The operational wind velocity for the umbilical tower will be the same as specified for the service structure.

19.2.5. <u>High-Pressure Gas Facility (Fig. 19.9)</u>. The high-pressure gas facility is located near the launch control center. This blast-protected facility will contain the storage battery, manifold system, pneumatic distributors, valving, etc., necessary to accommodate pressurization requirements such as supply and charge of the vehicle storage spheres for in-flight gas consumption, and will be used for checkout, purges, and pressurization for the vehicle and in the functional operation of the ground support equipment. The battery will be connected to each launch pad through the launch pad platform automatic ground control (AGC) room. The gas facility will also house the compressor-converter equipment used to charge the storage battery. Gaseous helium will be stored and boosted to 6,000 psig by booster compressors. This facility will be interconnected to VLF-37.

19.2.6. <u>Propellant Systems</u>

19.2.6.1. Liquid Hydrogen System. The capacity will be adequate (approximately 125,000 gallons) for the Saturn C-1 configuration. The storage tanks will be equipped with vaporization equipment for self-pressurization, and a vacuum pump for storage tank purging and inerting sequence. Hydrogen will be pressure-transferred through vacuum insulated lines routed up the umbilical tower to the various stages, each of which will utilize an individual tanking computer for precise tankage control and replenishing during standby prior to launch. This system will be remotely controlled and monitored from the launch control center during launch operations.

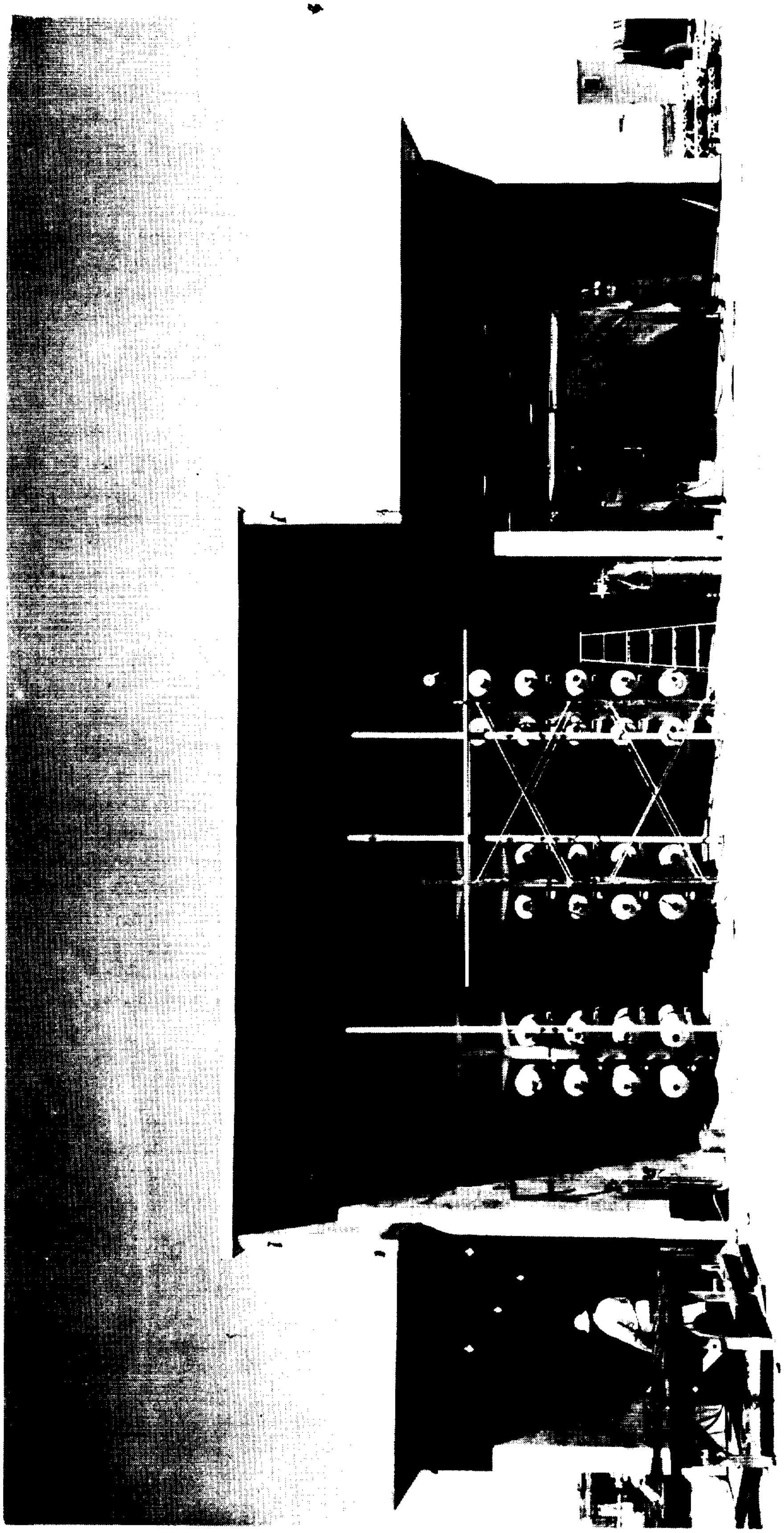

FIG. 19.9 HIGH PRESSURE GAS FACILITY, VLF-34.

19.2.6.2. Liquid Oxygen System (Fig. 19.10). The storage capacity will be adequate (one 125,000-gallon sphere and one 13,000-gallon replenishing tank) for the Saturn C-1 configuration. The storage tanks will be equipped with vaporization equipment for self-pressurization. The liquid oxygen storage tanks will be insulated to sustain an evaporation loss of less than 0.2 percent per day. Liquid oxygen will be pump-transferred through uninsulated lines routed up the umbilical tower to the various stages each of which will utilize an individual tanking computer for precise tankage control and replenishing during standby prior to launch. This system will be remotely controlled and monitored from the launch control center during launch operations.

19.2.6.3. RP-1 System (Fig. 19.11). The RP-1 fuel facility has a storage capacity of approximately 60,000 gallons. The transfer system will facilitate fuel transfer at 2,000 gpm. The storage system will include a filter-separator unit and associated plumbing to insure proper filtration of the fuel and to minimize the entrained water content of RP-1. The fuel transfer system is automated and is initiated and controlled from the launch control center fuel loading panel.

19.2.7. Ground Support Equipment (GSE). Included will be all those items of GSE necessary for system interfaces in the area of propellant loading, propellant replenishing, umbilical swing arms and controls, and erection, handling and transportation equipment. Typical GSE required includes propellant loading equipment, leak and functional checkout equipment, pneumatic high-pressure distribution systems, servicing and checkout equipment, access work platforms and servicing ladders, associated engine removal equipment, firing accessories (short cable mast, main propellant fill masts, replenishing couplings, retrorocket installation equipment), environmental conditioning equipment, and all complex interconnecting control cabling.

19.2.8. Launch Pedestal. The launch pedestal for the Block I Saturn is shown in Figures 19.12 and 19.13. The pedestal will be modified for the Block II Saturn C-1 with fins. The concrete pedestal is capable of supporting and retaining the Saturn booster and associated upper stages. Bolted on top of the pedestal are eight holddown arms, retaining, as well as supporting, the Saturn until proper ignition has been achieved with the eight booster engines. These arms are fabricated of steel and are automatically controlled during the launch sequence.

19.2.9. Deflector. The rail-mounted, two-way blast deflector is of steel construction. Its purpose during launch is to deflect the engine flame into controlled directions. A spare deflector can be easily moved into position to replace a deflector damaged during launching.

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FIG. 19.10 LOX STORAGE TANK, VLF-34.

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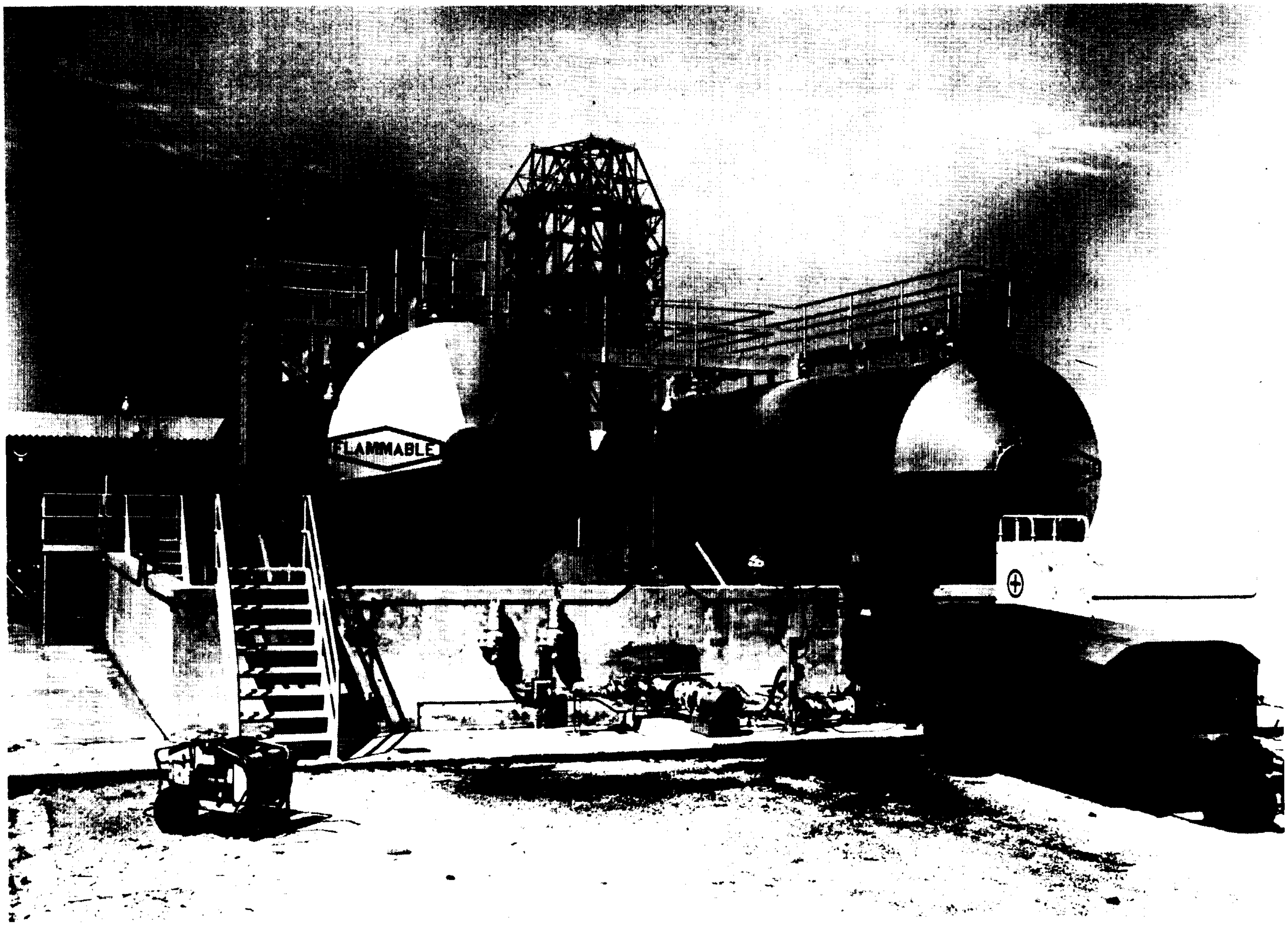

FIG. 19.11 RP-1 STORAGE FACILITY, VLF-34.

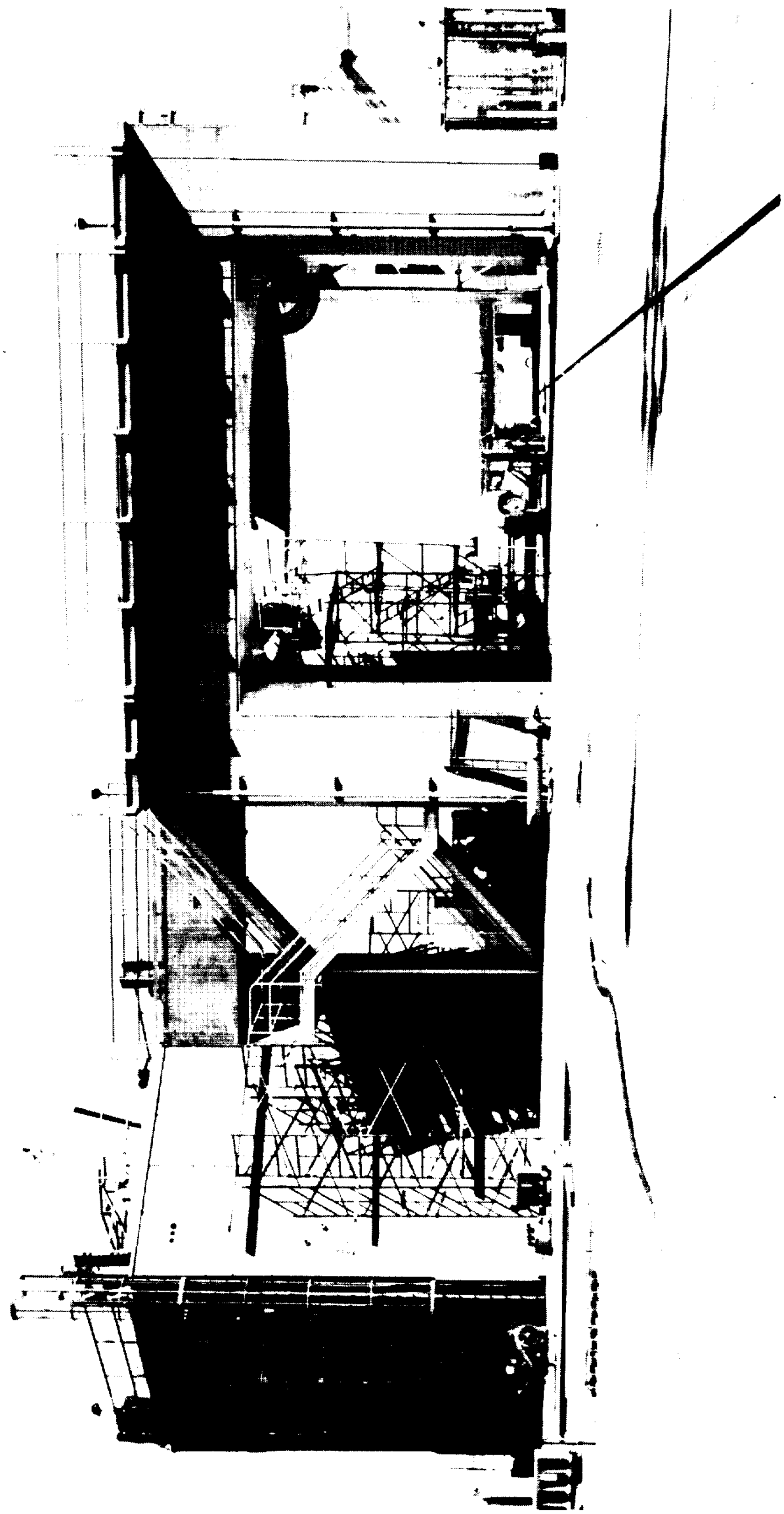

FIG. 19.12 LAUNCH PEDESTAL, VLF-34

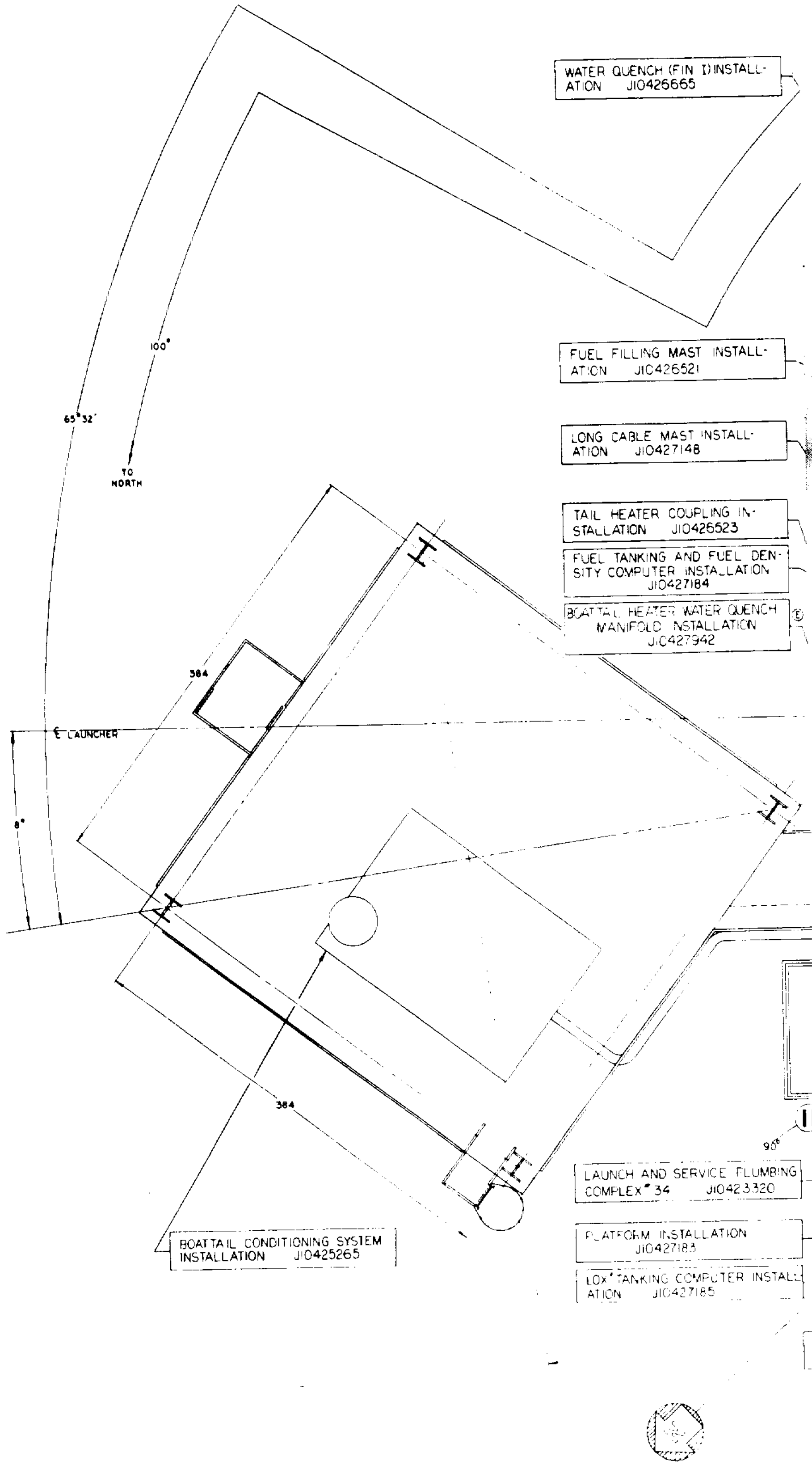

FIG. 19.13 LAUNCH PEDESTAL FOR BLOCK I SATU

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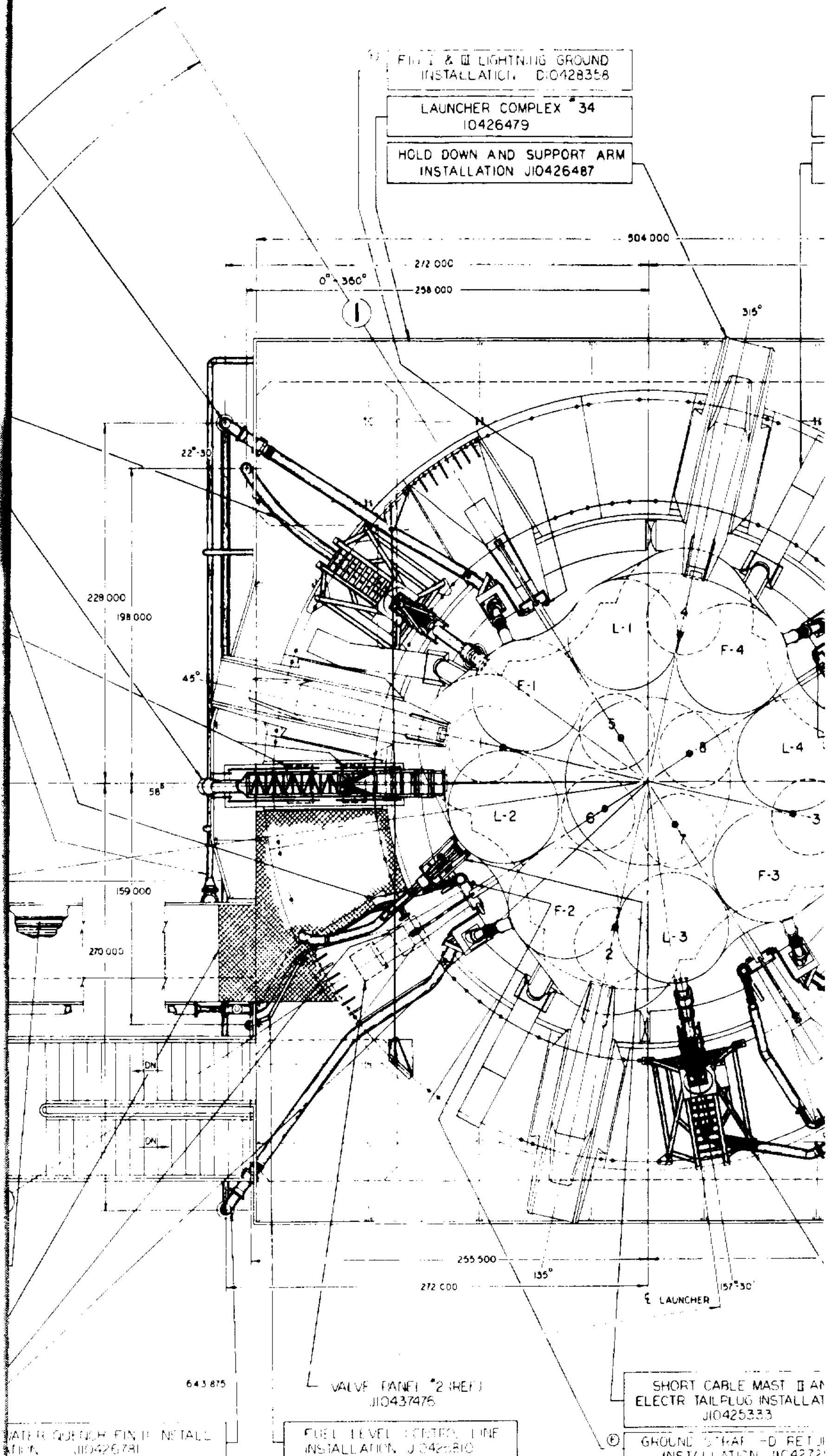
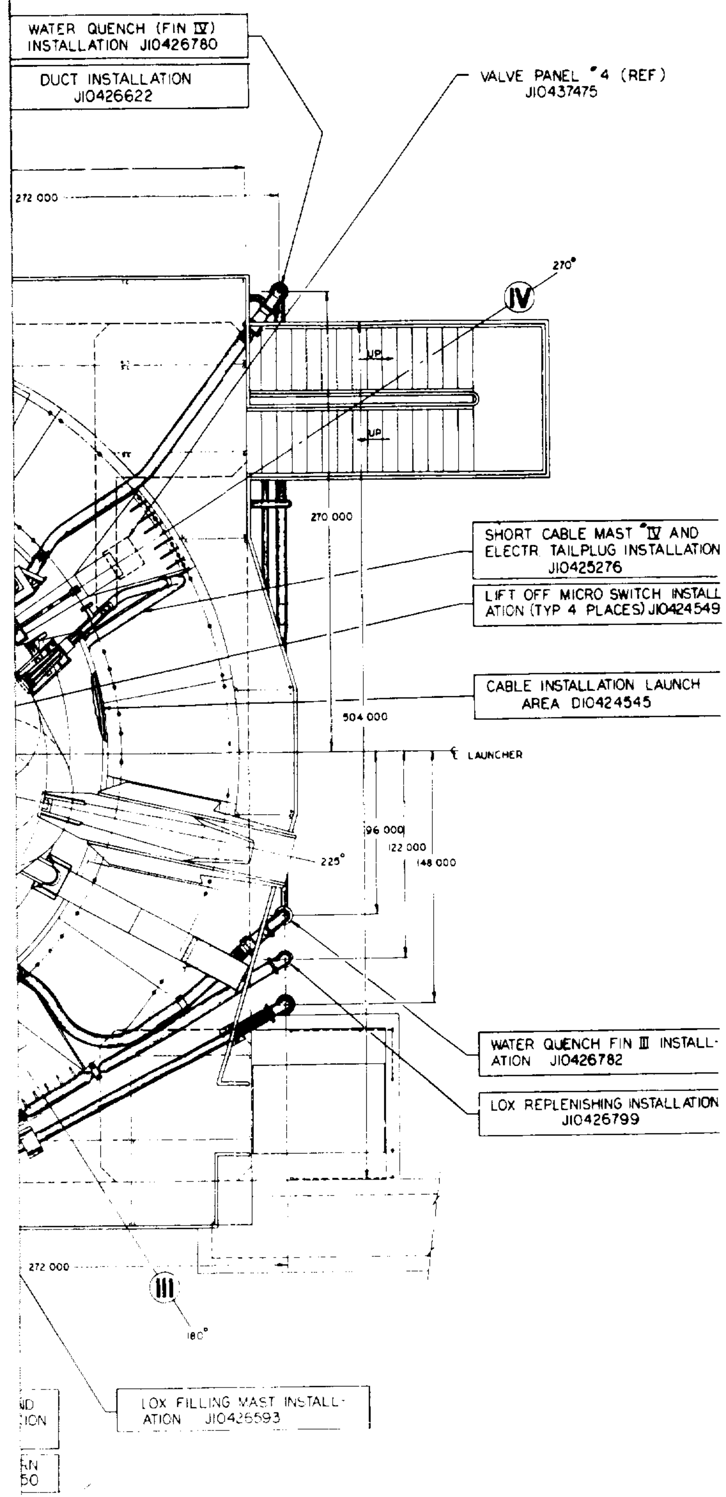


RN WITHOUT FINS, VLF-34, PLAN VIEW

FOLDOUT FRAME 2


WATER QUENCH (FIN IV) INSTALLATION J10426780
DUCT INSTALLATION J10426622
VALVE PANEL #4 (REF) J10437475
272 000
270°
IV
UP
UP
270 000
SHORT CABLE MAST #IV AND ELECTR TAILPLUG INSTALLATION J10425276
LIFT OFF MICRO SWITCH INSTALLATION (TYP 4 PLACES) J10424549
CABLE INSTALLATION LAUNCH AREA D10424545
504 000
℄ LAUNCHER
96 000
122 000
148 000
225°
WATER QUENCH FIN III INSTALLATION J10426782
LOX REPLENISHING INSTALLATION J10426799
272 000
III
180°
LOX FILLING MAST INSTALLATION J10426593

19.3. VERTICAL LAUNCH FACILITY (VLF) NO. 37

Since the GSE and facilities under design for VLF-37 are also to be used for the Saturn/Apollo "A" vehicle, the following information on VLF-37 is included. Initial construction for VLF-37 will be for the Saturn C-1 configuration.

19.3.1. Area Layout of VLF-37 (Fig. 19.14). The general layout envisions a circular launch control center 1,050 feet from the center of the pad. Pads would be separated by 1,200 feet. This separation distance is required to provide protection from a vehicle explosion to the service structure while in the parked position on the adjacent pad.

19.3.2. Launch Control Center. Construction of the launch control center is essentially the same as that for VLF-34. The basic floor plans (Figs. 19.15 and 19.16) indicate equipment layout and placement of the test conductors consoles. Closed circuit TV and three periscopes provide visual observation during testing and launch operations.

19.3.3. Service Structure (Fig. 19.17). The service structure is a mobile structure operating on rails, and is capable of servicing a launch pad platform located at each end of the rail. Essentially, the service structure is a trapezoidal-shaped structure extending to a total height of 300 feet, which can be increased.

Hoisting and erection equipment in the form of a stiff-leg derrick is mounted to one side on the top of the service structure. The stiff-leg derrick has a capacity of 60 tons at an 80-foot working radius plus a 10-ton auxiliary hook and 10-foot extension of the main boom. Hoisting machinery will be located in the base of the structure.

Two combined passenger and freight elevators of 5,000-pound capacity each will service all fixed platforms and movable service platforms.

The Saturn vehicle, when embraced by the service structure, will be protected, with the exception of the S-I booster, by 50-foot high split "silo" enclosures. There will be a minimum of 10 such service platforms provided. The access to the service platforms will be from individually adjusted service platform landings. Hoisting machinery for the platform landings will be housed in the base of the structure.

The service structure rails between the launch pads, together with anchor foundations and devices at each launch pad position, are required to anchor the structure and remove the loads from the wheels during an extended stay. The service structure is stable in winds up to 40 knots while moving with the "silo" enclosures retracted. The structure will withstand hurricane winds up to 125 mph while anchored and with the "silo" protecting the vehicle on the launch pad platform.

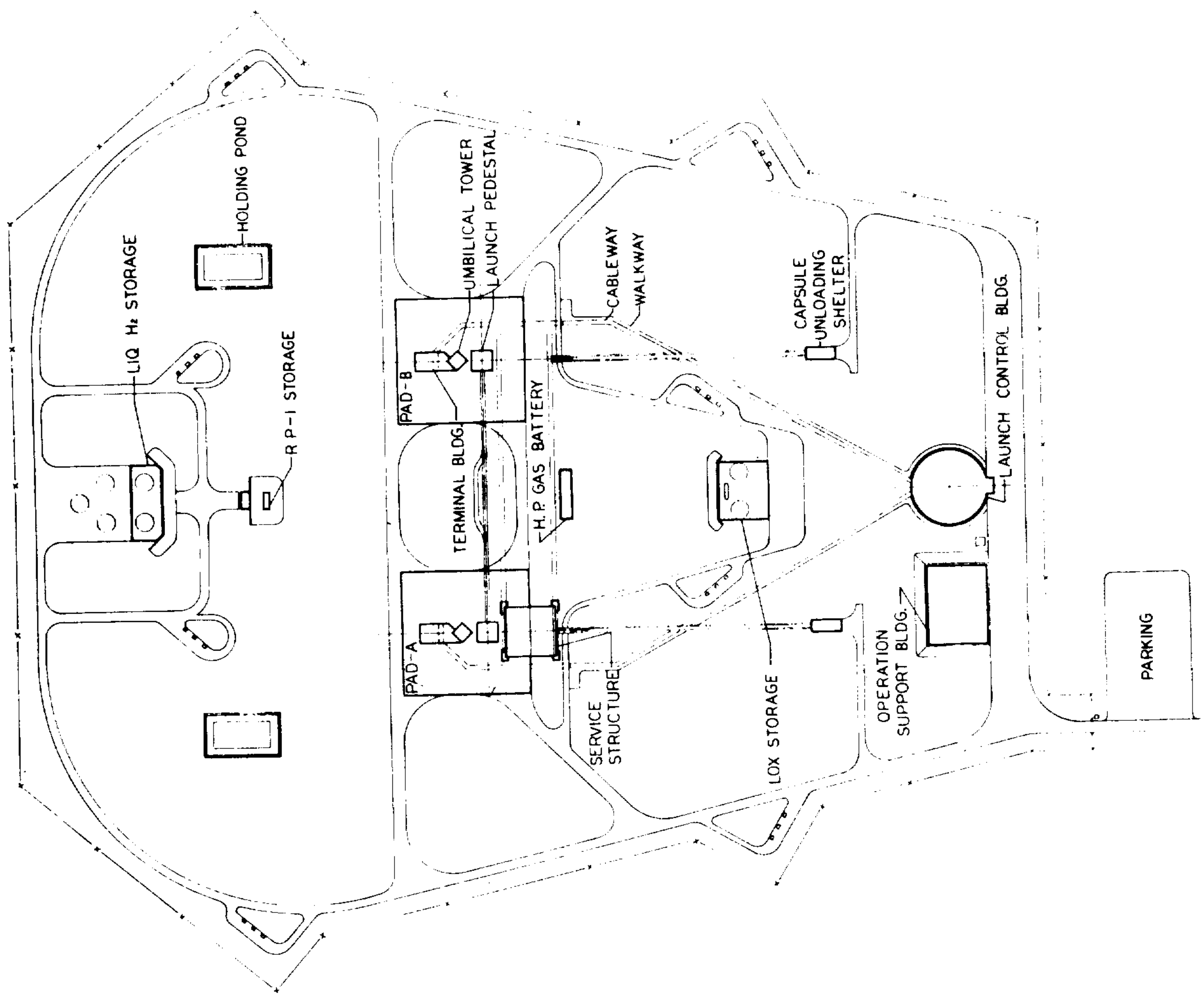
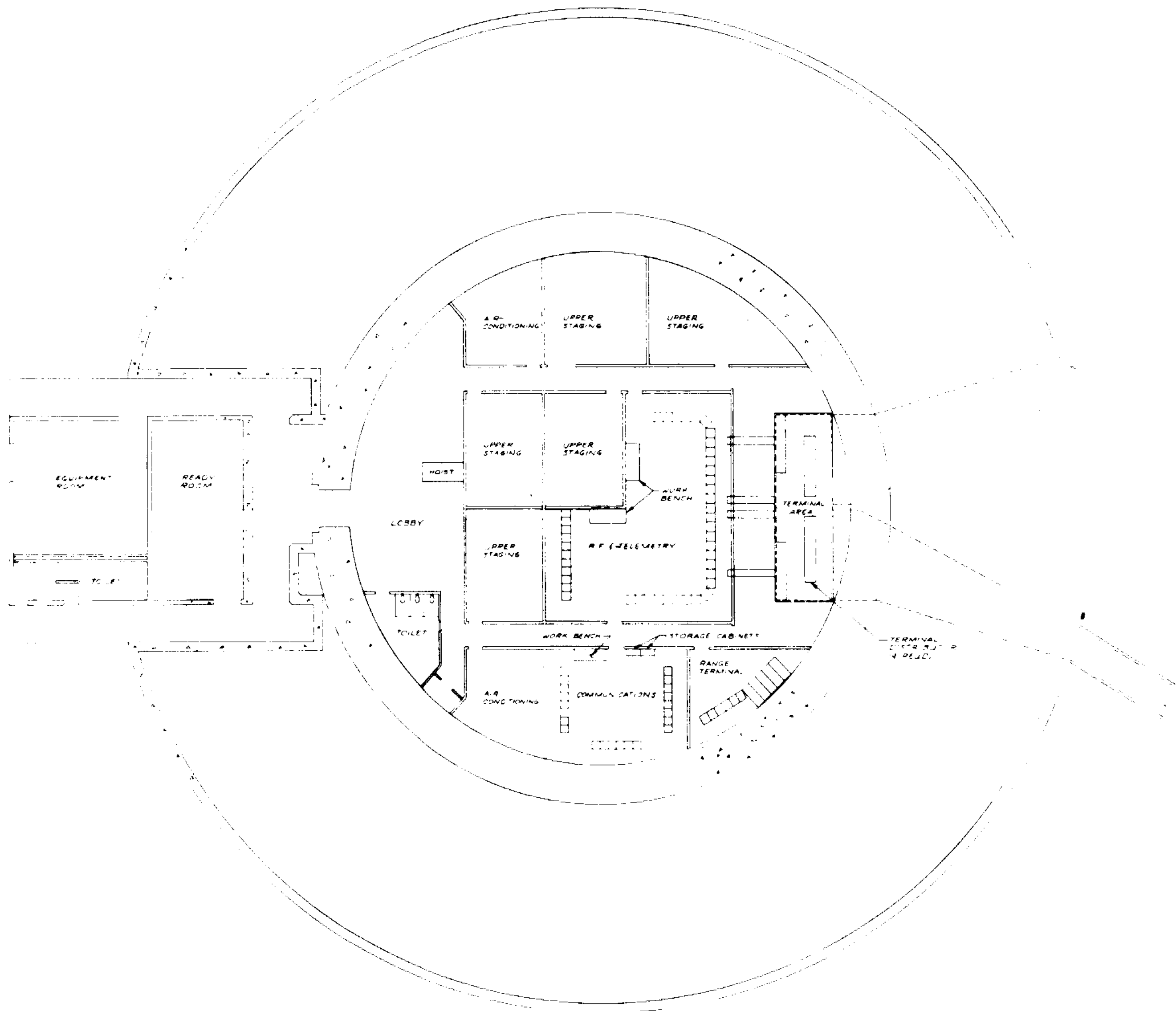


FIG. 19.14 VLF-37, AREA LAYOUT.



FIRST FLOOR PLAN

FIG. 19.15 FIRST FLOOR PLAN OF LAUNCH CONTROL CENTER, VLF-37.

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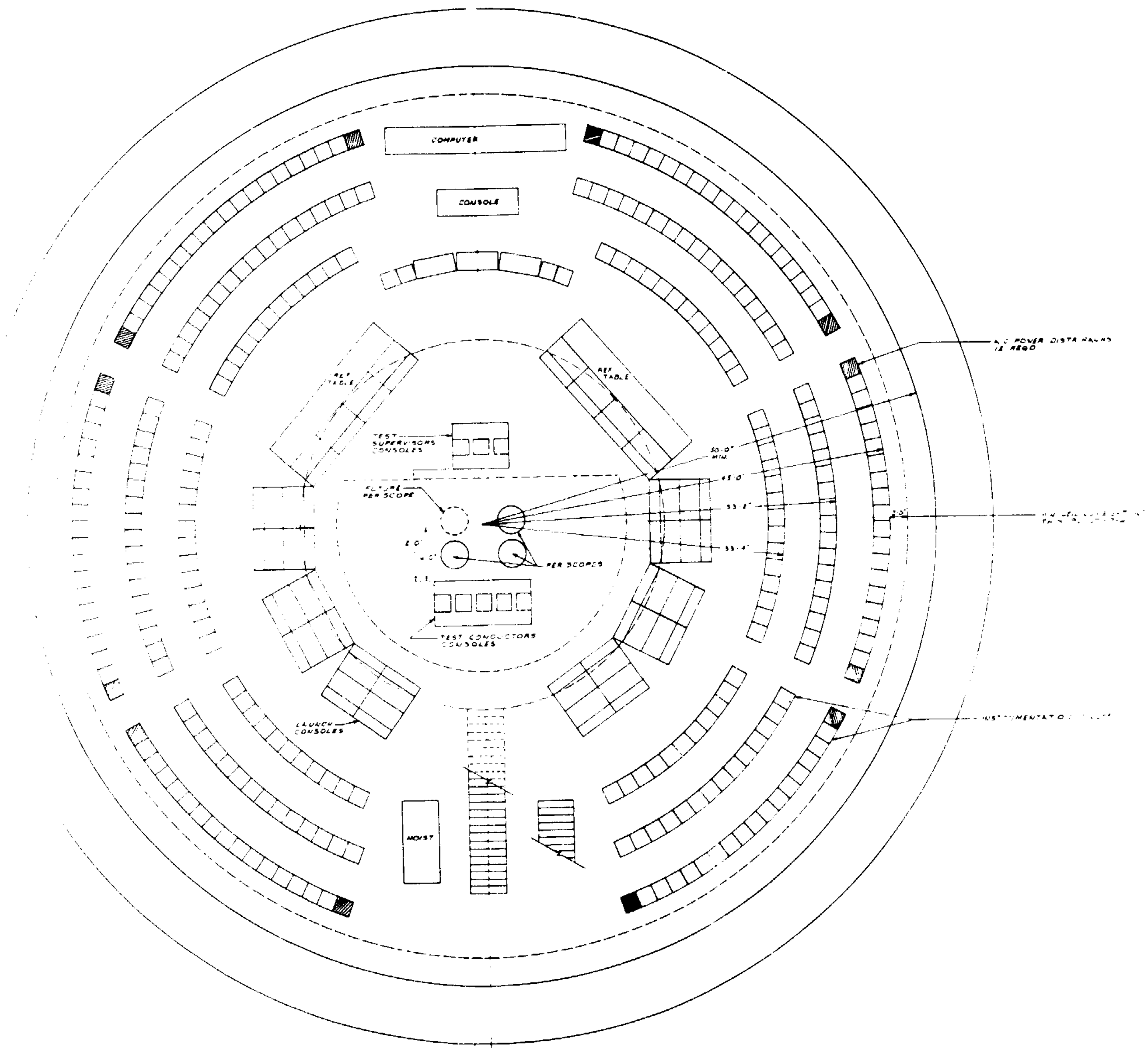

SECOND FLOOR PLAN

FIG. 19.16 SECOND FLOOR PLAN OF LAUNCH CONTROL CENTER, VLF-37.

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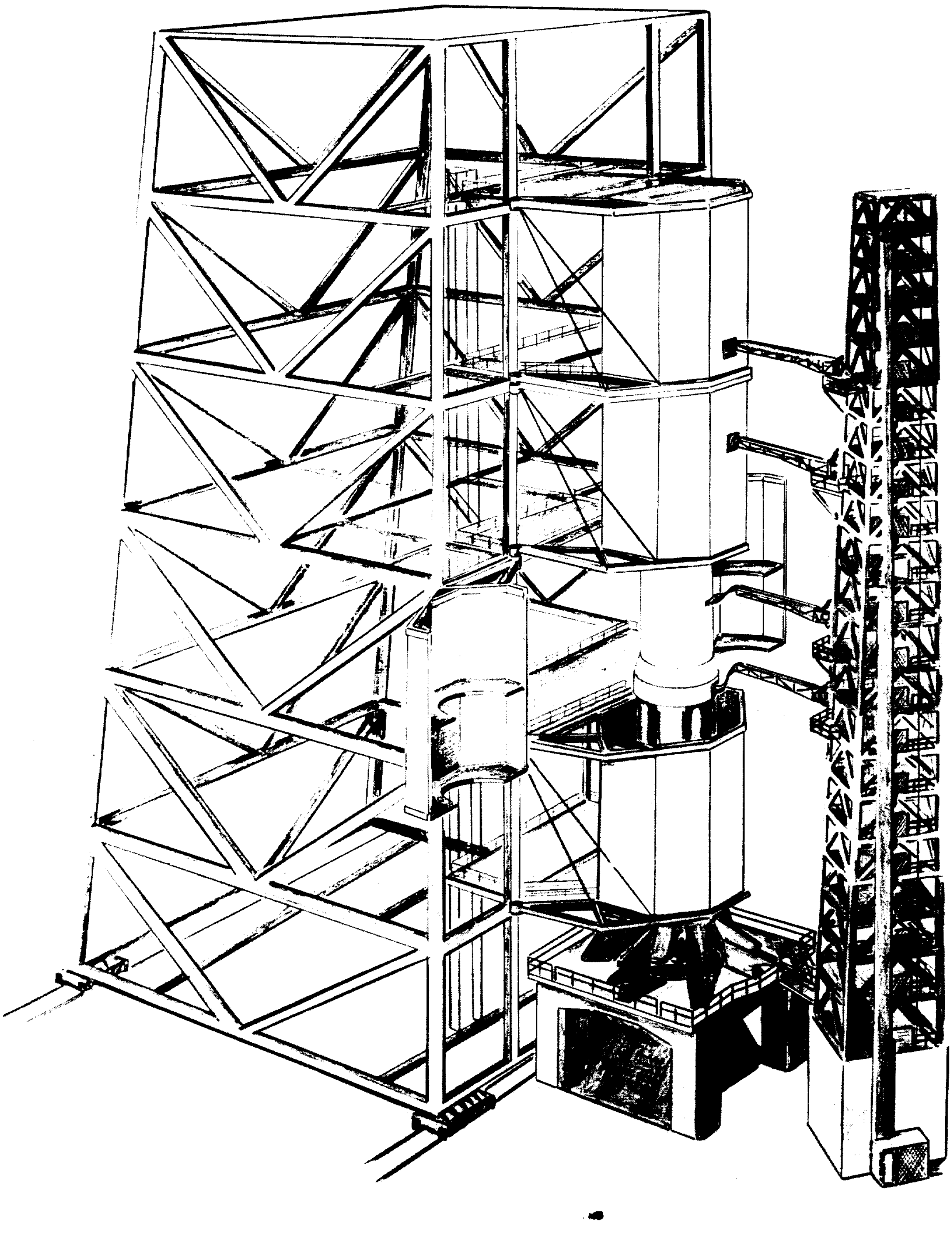

FIG. 19.17 SATURN SERVICE STRUCTURE. VLF-37.

19.3.4. <u>Umbilical Tower (Figs. 19.18 and 19.19)</u>. The umbilical tower is used to support and service the umbilical arms. The umbilical swing arms extend horizontally toward the vehicle in the same vertical plane as the diagonal legs of the tower and are capable of swinging 135 degrees to either side from the vertical pivot of the tower leg.

In addition to supporting and servicing the umbilical arms, the umbilical tower supports and houses the equipment required to service the Saturn upper stages prior to launch. Included are service lines such as electrical cables, distributors, pneumatic lines, cryogenic main fill and replenishing lines, propellant valve complexes, helium heat exchangers, pneumatic checkout consoles, mechanical refrigeration environmental conditioning ducts, and hydrogen vent stacks. The umbilical tower is a 320-foot high structure with a 32-feet square base. From the 35-foot level to the full height, the tower columns slope inward to 14 feet, 1-1/2 inches square at the top. Included is lightning protection, electrical troughs, distributors, and feeder lines. The tower is equipped with a safety ladder and fully-decked service platforms at each 20-foot interval starting at approximately 50 feet above ground level. A 3,000-pound capacity jib boom will be provided at the top of the tower to facilitate handling of electrical lines, electrical cables, umbilical arms, and associated ground checkout consoles required on the tower. A 3,000-pound capacity elevator for carrying passengers and small items of hardware to the various service platform is provided.

An integral part of the umbilical tower is the automatic ground support station (Fig. 19.20) at the base of the tower. This reinforced concrete building consists of two sections. The two-floor section will contain power transfer equipment, air conditioning equipment, and instrumentation while the three-floor section will serve as the umbilical tower support and contain additional instrumentation, various consoles, and test equipment. The basement of the building will serve as an entrance area for various cabling, conduits, and high pressure lines.

19.3.5. <u>High Pressure Gas Facility</u>. The high-pressure gas facility is located midway between the pads. This blast-protected gas facility will contain the storage battery, manifold system, pneumatic distributors, valving, etc., necessary to accommodate pressurization requirements such as supply and charge of the vehicle storage spheres for in-flight gas consumption, and will be used for checkout, purges, and pressurization for the vehicle as well as in the functional operation of the ground support equipment. The battery will be connected to each launch pad through the launch pad automatic ground control (AGC) room.

A central compressor-converter facility for VLF-37 will house a storage battery for nitrogen gas at 6,000 psig with a 6,300 cubic-foot volume of water and helium at 6,000 psig with a 1,600 cubic-foot volume of water. The location will be 3,800 feet from VLF-37, but interconnected to VLF-34.

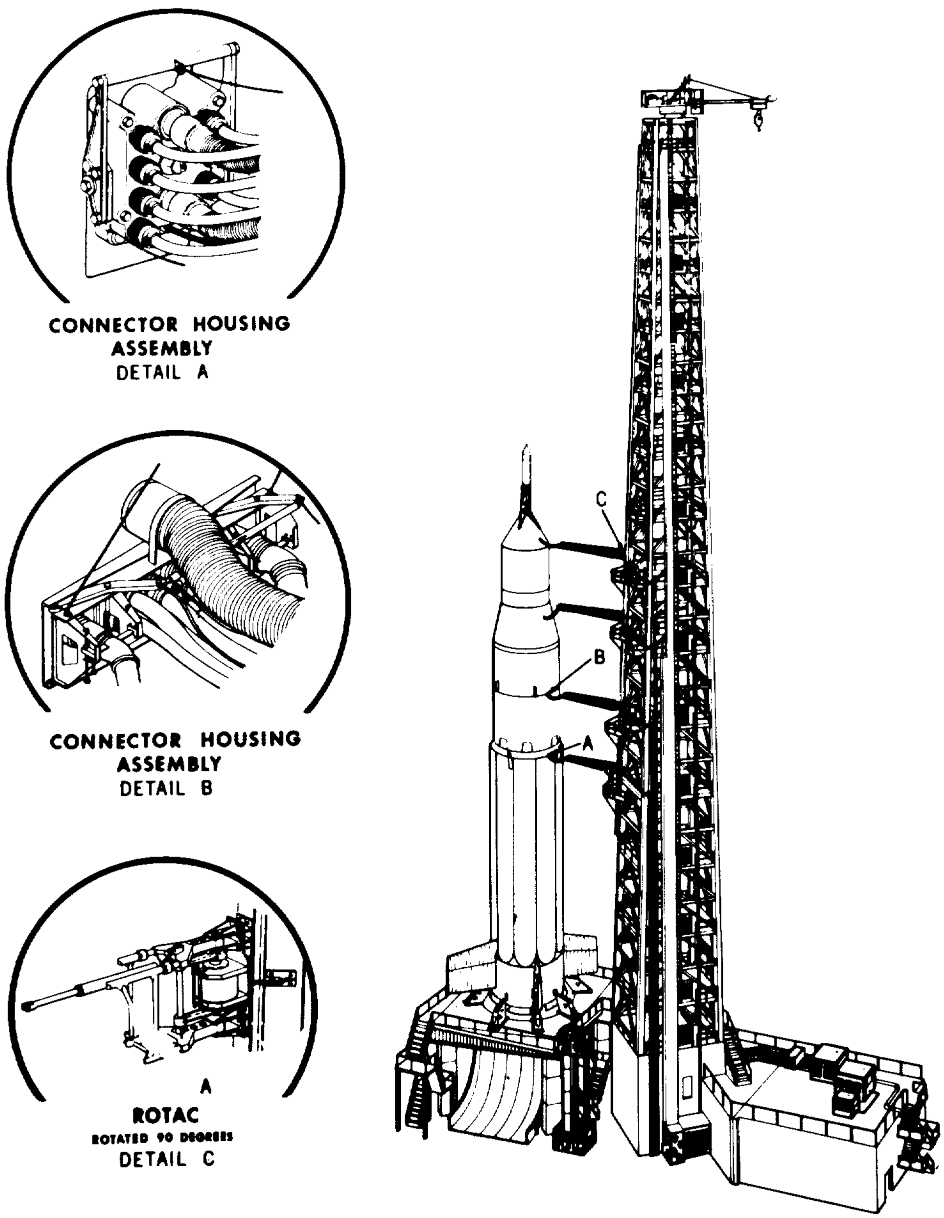

FIG. 19.18 SATURN C-1 WITH UMBILICAL TOWER

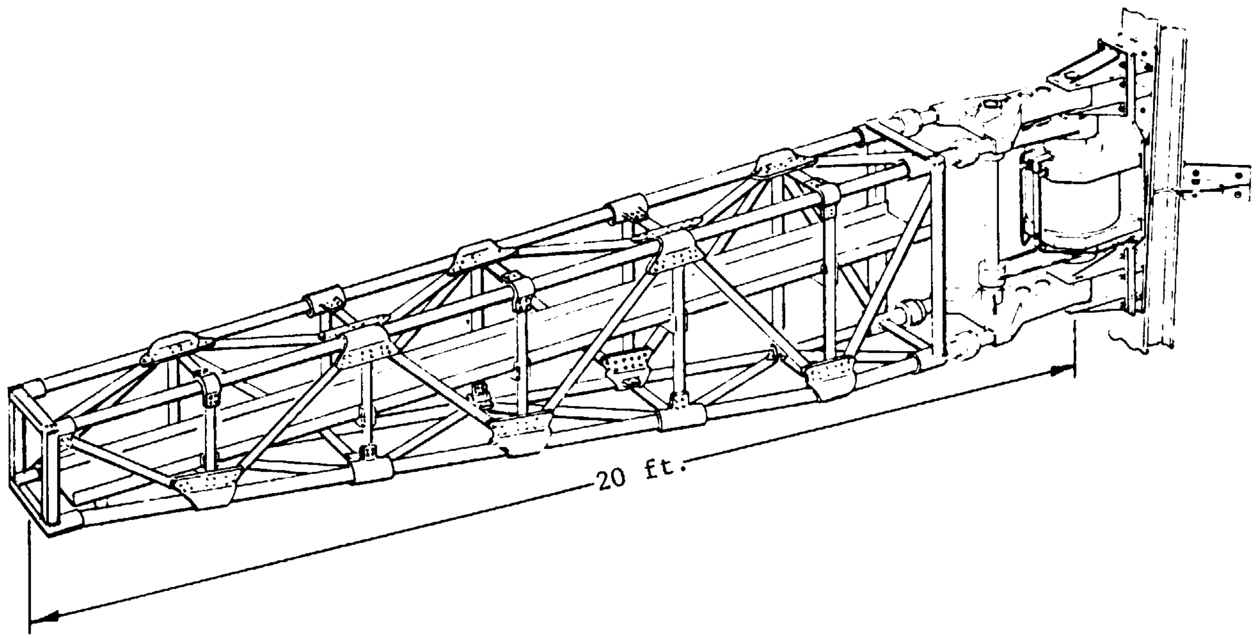

FIG. 19.19 TYPICAL UMBILICAL TOWER SERVICE ARM, VLF-37.

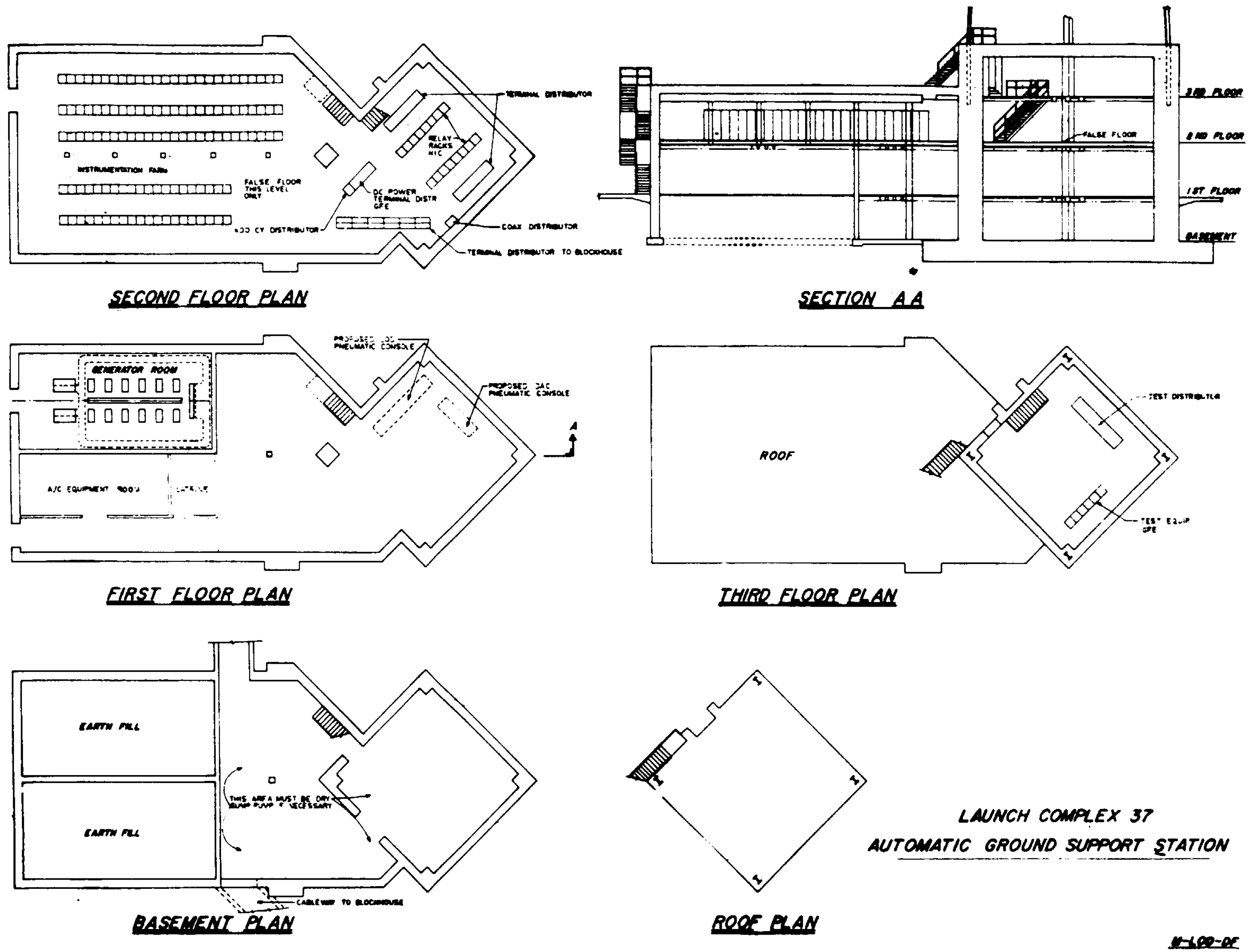

FIG. 19.20 FLOOR PLANS OF AUTOMATIC GROUND SUPPORT STATION, VLF-37.

19.3.6. Propellant Systems

19.3.6.1. Liquid Hydrogen System. The storage tanks will be equipped with vaporization equipment for self-pressurization, and a vacuum pump for storage tank purging and inerting sequence. Hydrogen will be pressure-transferred through vacuum insulated lines routed up the umbilical tower to the various stages, each of which will utilize an individual tanking computer for precise tankage control and replenishing during standby prior to launch. This system will be remotely controlled and monitored from the launch control center during launch operations.

19.3.6.2. Liquid Oxygen System. The storage tanks will be equipped with vaporization equipment for self-pressurization. The liquid oxygen storage tanks will be insulated to sustain an evaporation loss of less than 0.2 percent per day. Liquid oxygen will be pump-transferred through uninsulated lines routed up the umbilical tower to the various stages, each of which will utilize an individual tanking computer for precise tankage control and replenishing during standby prior to launch. This system will be remotely controlled and monitored from the launch control center during launch operations.

19.3.6.3. RP-1 System. The RP-1 fuel facility has a storage capacity of approximately 45,000 gallons. The transfer system will facilitate fuel transfer at 2,000 gpm. The storage system will include a filter separator unit and associated plumbing to insure proper filtration of fuel and to minimize the entrained water content of RP-1. The fuel transfer system is automated and is initiated and controlled from the launch control center fuel loading panel.

19.3.7. Ground Support Equipment (GSE). Included will be all those items of GSE necessary for system interfaces in the area of propellant loading, propellant replenishing, umbilical swing arms and controls, and erection, handling and transportation equipment. Typical GSE required includes propellant loading equipment, leak and functional checkout equipment, pneumatic high pressure distribution systems servicing and checkout equipment, access work platforms and servicing ladders, associated engine removal equipment, firing accessories (short cable mast, main propellant fill masts, replenishing couplings, retrorocket installation equipment), environmental conditioning equipment, and all complex interconnecting control cabling.

19.3.8. Launch Pedestal. The VLF-37 launch pedestal is a four-legged steel table, 35 feet high and 55 feet square, which forms a base for eight support arms, four of which are also holddown arms.

The top of the pedestal is pierced by a 32-foot diameter hole, the centerline coinciding with the vertical axis of the pedestal. The eight support arms are spaced equally around the periphery of this hole on top of the pedestal in such a way as to be symmetrically located with relation to the legs of the pedestal.

The main body of the structure consists of four legs supporting a box girder system, which is 7 feet deep. The outside webs of the box girders from an approximately 47-foot square. The inside webs of the box girders follow approximately the periphery of the 32-foot hole.

The four legs are anchored at their bases to the concrete foundation. Each leg is uniformly tapered from the 28-foot height to the base, the leg being smaller at the base.

The main body of the structure is oriented 9-1/2 degrees askew in relation to the axis connecting pads A and B. Triangular platforms are added to the top of the pedestal to extend the work space to a 55-foot square surface.

The launch pedestal is designed for a 3,000,000 pound thrust vehicle with dynamic factors of 1.6 upwards and 2.0 downwards. The material used for the launcher is mild steel since the launch pedestal stiffness is more critical than the relatively low stress level.

The launcher water cooling manifold system is included in the design of the launch pedestal.

19.3.9. Launch Support and Holddown Arms. The launcher has eight support arms (Fig. 19.21) which are dual purpose support and holddown arms located 45 degrees apart.

The arms support a portion of the dead weight of the vehicle and retain the vehicle on the launch pad during build-up of full thrust. These arms are of a box-type construction made up to 7/8-inch thick steel plates welded together in a submerged arc welding process. The arms are designed to withstand down loads, up loads, and dynamic rebound loads imposed by the vehicle. Wind loads, engine misalignment, shock and vibration loads were also considered in the design, and the structure was designed with a safety factor of 2 based on the yield point of the materials of construction.

Necessary access openings are provided to bring tubing and electrical wiring into each arm. A large opening with a steel door is provided in the rear of the arms for entry in order to perform required service and maintenance functions.

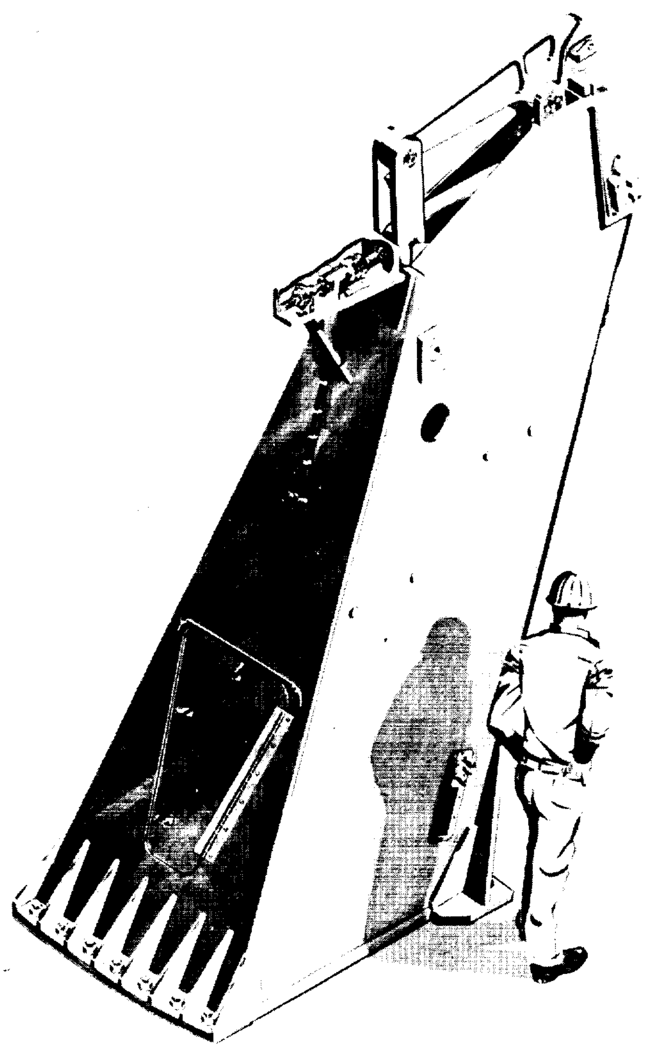

FIG. 19.21 LAUNCH PEDESTAL SUPPORT AND HOLDDOWN ARM, VLF-37.

Two of the arms serve as housings for the valve boxes. The arms also may be used to support other auxiliary launching components such as the short cable mast, etc.

Vertical adjustment necessary to provide proper leveling of the vehicle on the launch pad has been provided in the holddown arms by means of an arrangement of sliding wedges and shims in the upper portion of the arm.

The vehicle is retained on the launch pad under full thrust by a mechanical device similar to a nut cracker. The lower jaw of the device is designed to sustain the down loads and the upper jaw is designed to withstand the thrust loads of the vehicle. The upper jaw of the holddown is connected to a pre-loaded linkage which is held in a constant load condition under the surveillance of a double bridge load cell. This linkage is retained by a pneumatically actuated ball-lock separator.

The command to launch triggers the solenoid-actuated pneumatic valves in the holddown arms which releases the ball-lock separators. This allows the retaining linkage to collapse which permits the upper jaws to spring away from the vehicle, thus releasing the vehicle for free flight under full thrust.

19.3.10. Water Torus. Design of the water torus for the VLF-37 launcher has been incorporated into the steel launcher structure. This design will be similar to the one now planned for VLF-34. The purpose of the water torus is to prevent overheating of the engines due to after-burning effects in case of vehicle cutoff. The after-burning may cause considerable damage due to the flow and burning of approximately 6,000 gallons of propellant after the cutoff sequence is complete.

The water torus consists of a 20-inch pipe that is fabricated to form two 180-degree halves on a 150-3/4 inch radius. This ring contains two types of nozzles. V-Jet nozzles are located on top of the ring and both V-Jet and adjustable fog nozzles are located on the bottom.

The torus ring receives its supply of water from two 20-inch industrial water pipes which connect to the torus ring on the opposite sides of the ring. The water supply of 12,000 gpm is controlled by two 12-inch remote-control cone valves which are located beneath the top of the launcher. The cone valves are set to open in 5 seconds and close in 30 seconds using live water pressures of 125 psi total dynamic head (TDH).

The torus ring and flushing water control are both tapped from the same industrial water line and the systems are automated with vehicle cutoff. Either or both systems may be controlled by the Range Safety Officer.

19.3.11. Deflector. The VLF-37 Saturn dry launcher deflector (Fig. 19.22) is an eight-jet deflector of a symmetrical two-directional design. The relation of the eight-engine pattern of the Saturn to the deflector ridge is fixed. The direction of escape of the flame is horizontal.

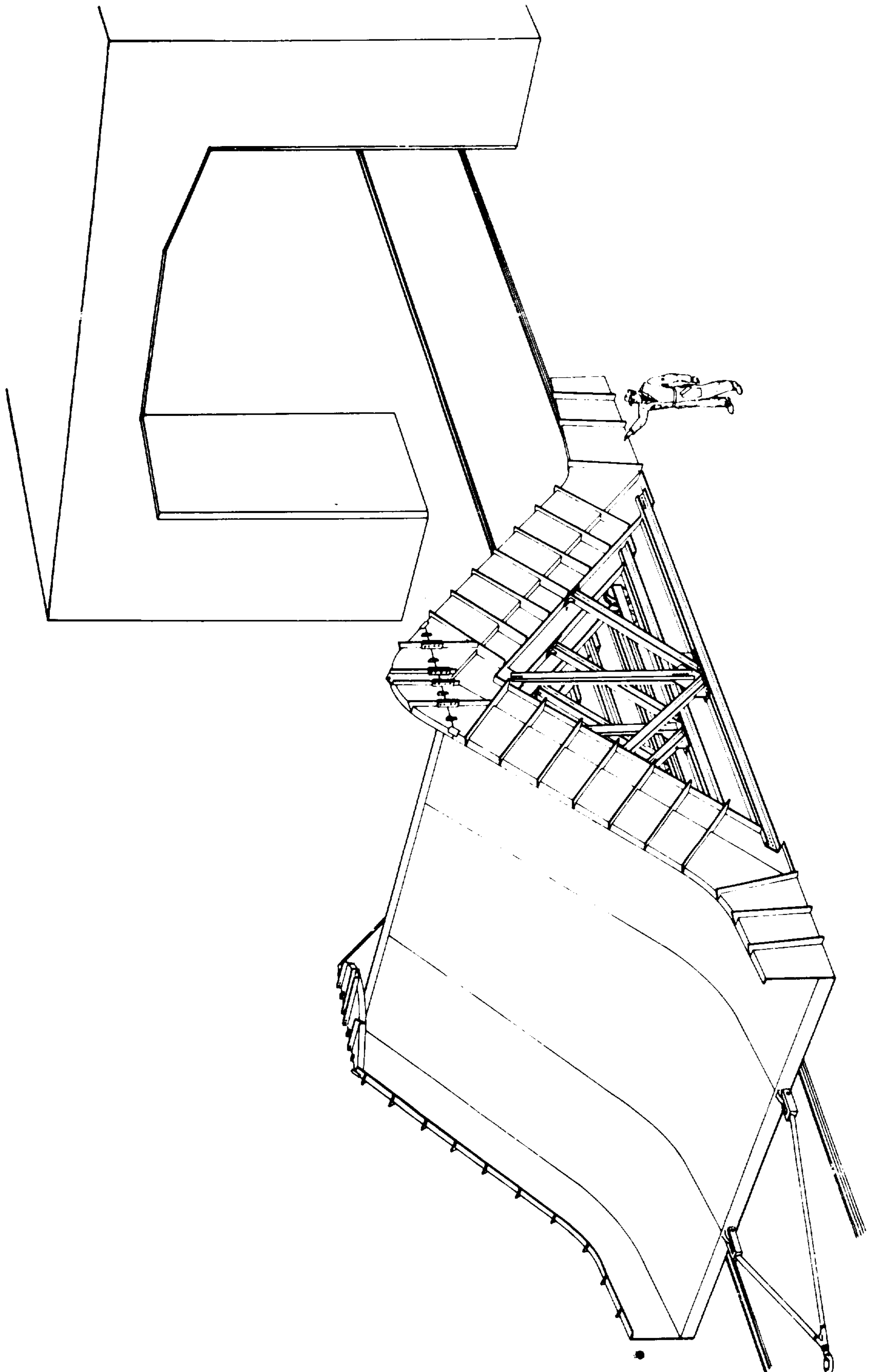

FIG. 19.22 SATURN LAUNCHER DEFLECTOR, VLF-37

SECTION 20. (C) FLIGHT EVALUATION

20.1. CONCEPT

The Saturn post flight evaluation will be performed in two principal phases. The first phase is termed "Quick Look" and is the responsibility of the MSFC's Launch Operations Directorate located at Cape Canaveral. This phase will normally be completed within 24 hours after launch. The second phase, the Early Flight Evaluation, will be directed and conducted by the Saturn Flight Evaluation Working Group at MSFC, Huntsville.

The basic concept of the flight evaluation is to integrate the analysis and evaluation efforts of MSFC and all stage contractors at an early time and to the highest possible degree. The specific tasks of the Early Flight Evaluation are defined as follows:

(1) Complete analysis of mission objectives

(2) Recognition of all problem areas

(3) Establish actual trajectory of vehicle system

(4) Complete qualitative analysis of malfunctions and unexpected vehicle behavior, including vehicle events prior to liftoff

(5) Limited quantitative analysis of problem areas

(6) Arrive at unified results for all involved parties

(7) Communicate recognized deficiencies to appropriate divisions for corrective action

(8) Recognize the need for additional instrumentation as indicated by the flight test

(9) Issue preliminary report as result of integrated evaluation effort.

20.2 IMPLEMENTATION OF CONCEPT

The stage contractors submit all standard flight test data requirements necessary to accomplish the stage flight evaluation.

When setting up the evaluation procedures, the contractors must assure compatibility of data reduction equipment and procedures with existing MSFC methods. Units, coordinate system, etc., to be used are specified by MSFC.

Special liaison groups of the contractors' evaluation personnel establish the necessary communication for the MSFC flight evaluation. The liaison groups consist of an adequate number of highly qualified personnel and are required to be present at MSFC for a short period of time after each firing to participate in the vehicle early flight evaluation. Data reduction, computation, and similar services required by the liaison group are supplied by the contractors.

The contractors' early evaluation results will be fed directly and individually to the MSFC flight evaluation meetings in Huntsville, Alabama. The early results will contain all graphic and tabulated material necessary to explain the observed flight occurrences.

The preliminary flight test report stresses system malfunctions rather than complete test documentation. Target date for completion of this report is approximately three weeks after launch.

Marshall Space Flight Center also establishes one unique master three degree-of-freedom post-flight trajectory for all stages of the Saturn vehicle.

After completion of the preliminary flight test report, the contractors will publish a final flight test report on their stage containing all reduced data, system analysis, test article history, and applicable illustrations. These reports will be an expansion of, and in basic agreement with, the preliminary flight test report. Target date for publishing the final reports is six weeks after launch.

The contractors also will deliver to MSFC a 1:24 scale model of their stages to be used during the post flight evaluation. The models will be delivered no later than two months prior to the first launch. The models are to be cut away to show all major components and measurement locations. These measurement locations are identified by color or number codes. In every major detail a model is exactly like the flight vehicle at the time of test flight and, therefore, the models are constructed to facilitate updating vehicle configuration and flight test instrumentation changes as they occur. The contractors are performing this updating task.

SECTION 21. (C) SCHEDULES AND LEAD TIMES

In order to facilitate complete space vehicle scheduling exercises, typical leadtimes for Saturn C-1 systems are given in Tables 21.1 to 21.4. These leadtimes are based on the scheduled launch date and are given for the completion of the events unless otherwise noted. The lead times given are taken from present schedules for the Saturn C-1 Block II vehicles. For those Apollo "A"/Saturn C-1 vehicles which are following Block II, it is expected that these lead times may be reduced somewhat by the effects of one or more of the following improvements:

(1) Omission of captive (static) test firings for some of the stages and combination of the present prestatic and poststatic checkouts into one preshipment checkout

(2) Increased utilization of the faster air transportation instead of the presently used water transportation of large items

(3) Omission of horizontal space vehicle checkout and mating tests in the special assembly building at AMR; Cape Canaveral, and use of immediate vertical assembly and launch checkout of the space vehicle only on the pad.

Table 21.1 TYPICAL LEAD TIMES FOR STAGE S-I

EVENT	LEAD TIME* (Weeks)
Order Long Lead Time Items	T -120
Begin Fabrication	T -79
Fabrication	T -57
Assembly	T -36
Checkout (Prestatic)	T -30
Captive (Static) Test Firing	T -24
Checkout (Poststatic)	T -12
Ship to AMR, Erect on Pad	T -9
S-I Checkout on Pad	T -4.5
Erect S-IV, Instrument Unit, and Spacecraft on Pad	T -4
Space Vehicle Checkout and Launch	T
Final Flight Evaluation	T +6

*NOTE: Lead times are referred to the launch date and are for the completion of the event, unless otherwise noted.

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Table 21.2 TYPICAL LEAD TIMES FOR STAGE S-IV

EVENT	LEAD TIME * (Weeks)
Order Long Lead Time Items	T -102
Begin Fabrication	T -70
Fabrication	T -54
Assembly	T -35
Checkout (Prestatic)	T -28
Ship to Sacramento	T -26
Captive (Static) Test Firing	T -20.5
Ship to AMR	T -17
Horizontal Checkout and Mating Tests in Special Assembly Building	T -4.5
Erect S-IV on S-I	T - 4.4
Space Vehicle Checkout and Launch	T
Final Flight Evaluation	T +6

*NOTE: Lead times are referred to the launch date and are for the completion of the event, unless otherwise noted.

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Table 21.3 TYPICAL LEAD TIMES FOR INSTRUMENT UNIT

EVENT	LEAD TIME* (Weeks)
Order Long Lead Time Items	T -96
Delivery of Long Lead Time Items	T -43
Fabrication of Structure	T -30
Installation of Equipment	T -22
Checkout	T -15
Ship to AMR	T -13
Horizontal Checkout and Mating Tests in Special Assembly Building	T -4.5
Erect Instrument Unit on S-IV	T -4.2
Space Vehicle Checkout and Launch	T
Final Flight Evaluation	T +6

*NOTE: Lead times are referred to the launch date and are for the completion of the event, unless otherwise noted.

Table 21.4 TYPICAL LEAD TIMES FOR SPACECRAFT

EVENT	LEAD TIME * (Weeks)
Ready for Horizontal Checkout and Mating Tests in Special Assembly Building	T - 13
Horizontal Checkout and Mating Tests in Special Assembly Building	T - 4.5
Erect Spacecraft on Launch Vehicle	T - 4
Space Vehicle Checkout and Launch	T
Final Flight Evaluation	T + 6

*NOTE: Lead times are referred to the launch date and are for the completion of the event, unless otherwise noted.

APPROVAL MPR-M-SAT-61-5

THE APOLLO "A"/SATURN C-1
LAUNCH VEHICLE SYSTEM

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