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(U) Study of Utilizing Apollo for the MOL Mission

Volume II: Subsystem Studies - Applied Mechanics Division

11 JANUARY 1965

Prepared by
THE APPLIED MECHANICS DIVISION

Prepared for COMMANDER SPACE SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
LOS ANGELES AIR FORCE STATION
Los Angeles, California

EL SEGUNDO TECHNICAL OPERATIONS • AEROSPACE CORPORATION
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Report No.
TOR-469(5510-41)-1,
Volume II

A64-13546

(U) STUDY OF UTILIZING APOLLO FOR THE MOL MISSION

VOLUME II: SUBSYSTEM STUDIES - APPLIED MECHANICS DIVISION

Prepared by

Applied Mechanics Division

EL SEGUNDO TECHNICAL OPERATIONS
AEROSPACE CORPORATION
El Segundo, California

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(U) STUDY OF UTILIZING APOLLO FOR THE MOL MISSION
VOLUME II: SUBSYSTEM STUDIES - APPLIED MECHANICS DIVISION

Prepared by Applied Mechanics Division

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System Design

The information in a Technical Operating Report is developed for a particular program and is therefore not necessarily of broader technical applicability.

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[1] See individual sections for specific tables of contents and lists.

[2] See TOR-469(5510-41)-1, Volume III, Section 5 for additional Power Subsystems (power requirements) material.

FOREWORD

This report is one of three volumes, which jointly document the results of studies performed by the Aerospace Corporation, with co-operation of the Air Force/SST, to evaluate the feasibility of utilizing the Apollo spacecraft to perform the Air Force MOL mission. The volumes comprising the results of the Apollo/MOL study are:

TOR-469(5510-41)-1, Vol I	Summary
TOR-469(5510-41)-1, Vol II	Subsystem Studies - Applied Mechanics Division
TOR-469(5510-41)-1, Vol III	Subsystem Studies - Electronics Division

Volume I briefly describes the Apollo spacecraft and the Apollo elements considered for the MOL program, summarizes the results of Apollo/MOL configuration studies utilizing the Titan III-C and the Saturn IB launch vehicles, and presents the overall conclusions reached. A brief summary of the subsystem studies and the potential growth capability of Apollo/MOL Saturn IB is presented in the second part of the volume.

The recommendations in Volume I are based upon analysis of overall system considerations and in a few cases may not completely reflect the recommendations associated with the detail subsystem studies reported in Volumes II and III.

Volume II is a compilation of documents which details the results of subsystem studies performed in the applied mechanics area to evaluate the Apollo/MOL requirements and to develop the tradeoffs required to make subsystem recommendations. The studies included in Volume II are in the areas of:

Design	Performance
Weights	Test Operations
Experiments Integration	Reliability
Crew Time Allocation	Fluid Mechanics
Power Systems	Propulsion
Life Support	Solid Mechanics

Volume III is a compilation of documents which details the results of subsystem studies performed in electronics and related areas in support of the Apollo/MOL study. The areas of study included in Volume III are:

Attitude Control Subsystems
Communication Subsystems
Guidance and Navigation Subsystems
Photo-Optical Subsystems
Power Subsystems (Power Requirements)

INTRODUCTION

This volume includes studies pertaining to the feasibility of the Apollo/MOL concept, comparisons of Apollo/MOL with Gemini/MOL, and investigations of increased Apollo/MOL mission duration. In general these studies encompass the technical disciplines within the responsibility of the Applied Mechanics Division. The possibility of using Titan IIIC as a launch vehicle is investigated, and a number of variations in the basic Apollo command/service module package are studied in connection with the Saturn IB launch vehicle. The experiments and system requirements are those used for the Gemini/MOL at the time of these studies, which is September 1964.

The principal purposes of these studies then, are to determine the significant problems in adapting the Apollo spacecraft and supporting equipment to the MOL mission, the number of flights required to perform all of the MOL experiments using 30-day missions, and whether missions of 60 or 120 days are feasible. If there are no significant problems in adapting Apollo/MOL, and if fewer flights are required than with Gemini/MOL, then the systems are to be considered roughly competitive, subject to a cost comparison.

The studies include the formulation of preliminary designs of several configurations and the analysis of those configurations which appear promising. Estimates of required power and life support equipment, and the necessary modifications to the Apollo system to provide that equipment, are made to fit the various design concepts. Power and life support systems are chosen for both two and three-man crews for mission durations of 30, 60, and 120 days.

On the basis of the preliminary designs, equipment selections and other modifications to the basic Apollo system, estimates of system weights are made. Performance analyses are also carried out to determine the payload that can be placed into orbit. The payload capability and over-all system weight are then compared to determine the weight available for experiments in each configuration. Also, crew time allocations are estimated on the basis of previous studies for MORL, XMAS, and Gemini/MOL. Space availability is estimated directly from the preliminary design drawings. Using all of these estimated data, and following the same constraints that are used in the Gemini/MOL studies, the experiments are allocated among as many flights as are needed to perform them all. This number of flights provides a direct comparison with the Gemini/MOL concept. To make this comparison more meaningful, the over-all system reliability of Apollo/MOL is estimated and compared with that of Gemini/MOL.

In order to determine whether significant problems exist, a number of specialized areas were investigated, including structures, propulsion, aerodynamics, and operational considerations. These tasks were addressed to specific problems such as ascent vibration and wind loading effects, the suitability of propellants, abort feasibility, ascent and re-entry capability, the availability of ground stations, and the security of communications.

SECTION 1

DESIGN

N. G. Ivanoff

SUMMARY

Several preliminary layouts and descriptions of Apollo/MOL configurations for each launch vehicle are presented. One drawing shows the Apollo/MOL as launched by the Titan IIIC and three other drawings show progressively improved configurations as launched by the Saturn IB booster.

The Apollo/MOL/Titan IIIC configuration, due to weight limitations, has no pressurized laboratory and can carry only a portion of the experiments carried by the Gemini/MOL/Titan IIIC configuration; however, experiments would be conducted from within the Apollo capsule, which is not possible in the Gemini.

The preferred configuration of the Apollo/MOL as launched by the Saturn IB consists of the Apollo command module and a combined laboratory/service module. All of the MOL experiments, from a weight and volume standpoint, can be integrated within the laboratory on any one flight. The docking and airlock system of crew transfer, as used for the NASA Apollo lunar mission, was accepted as feasible in this study. However, several iterations on methods of crew transfer for the Apollo/MOL/Saturn IB configurations were performed and are included.

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ILLUSTRATIONS

1. DESIGN

1.1 INTRODUCTION

The present Apollo/MOL program study is a brief investigation to determine the feasibility of the Apollo capsule as a replacement for the Gemini B capsule, both as the crew conveyance and re-entry vehicle because of its larger size and potential capability. The study considers the use of the Titan IIIC and the Saturn IB as launch vehicles using the present docking and air-lock system of the Apollo lunar vehicle (References 1-3 and 144).

Several arrangements of the Apollo/MOL/Titan IIIC vehicle system were considered as possible candidate configurations to satisfy the MOL program requirements. Due to the limitation of the boost capability of the Titan IIIC launch vehicle, the resulting arrangement consists of only a modified Apollo command module and an unpressurized service module shown in Figure 1-12 (Drawing ES-0152-004). Several arrangements of the Apollo/MOL / Saturn IB vehicle system were also considered as possible candidate configurations to satisfy the MOL program requirements; these are shown in Figures 1-1 through 1-4. An arrangement designed specifically for the Apollo/MOL program is shown in Figure 1-3 and in more detail in Figure 1-11 (Drawing ES-0152-003), and is discussed in Section 1.3.4. Various methods of crew transfer from the Apollo command module to the laboratory module were also considered and are shown in Figures 1-5 through 1-8.

1.2 BASIC APOLLO/MOL VEHICLE GROUND RULES

The ground rules applicable to this study are the same as the present MOL ground rules and are listed as the minimum vehicle requirements:

a. Thirty-day orbit duration

b. Two-man crew

c. Integral launch

d. Shirt-sleeve environment

d. Test and experiment capacity commensurate with MOL package (Reference 161)

[redacted]

f. Rendezvous, docking and transfer provisions

g. AMR launch

h. Low orbit: 150 - 250 nautical miles

i. Minimum change to the Apollo-system

Several major Apollo subsystems will have to be modified and integrated into the over-all vehicle system to accomplish the Apollo/MOL mission. These are listed as follows:

a. Life support system

b. Environmental control system

c. Power supply system

d. Water and waste management system

e. Attitude control system

1.3 APOLLO/MOL CONFIGURATION CONCEPTS

1.3.1 Apollo/MOL/Titan IIIC Configuration

The Apollo/MOL configuration, launched by the Titan IIIC booster and shown in Figure 1-12 was selected for study from several candidate configurations. The MOL experiments (Reference 1-1) are shown integrated into the Apollo command module and the service module which also serves as the adaptor to the Titan IIIC transtage. Due to the limited payload capability of the Titan IIIIC booster (Section 2.1), only a portion of the required MOL experiments can be carried for each mission flight; also, because of weight limitations, no pressurized laboratory module can be carried into orbit for an integral launch, thus requiring the integration of the experiments, which must be handled directly by the crew, into the Apollo command module. The experiments and related equipments which can be operated remotely are integrated within the service module.

The integration of the MOL experiments into the Apollo capsule would require more than the minimum Apollo modifications required by the ground rules. Although an effort has been made to locate the various experimental equipments within the capsule, so as not to interfere with the existing Apollo systems, many structural changes would have to be made. In case of the optical equipment, for example to mount the pointing and tracking scope through the capsule

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wall, a pressure seal would be required within a structural well to allow the P/T scope to be stored during ascent and extended in orbit for observation. Provision also must be made for sealing the external movable head which rotates through an angle of 360 degrees. An aerodynamic fairing to cover the scope would also be required both to reduce drag and to protect the optical surfaces of the instrument.

Similar provisions would be required for the periscope/star theodolite optical instrument. Also, the geodetic camera which observes both ground targets and a celestial reference requires optical ports which must be sealed around the instrument peripheries in the basic Apollo structure.

The air-lock hatch and crew transfer tube of the Apollo capsule would have to be modified by the addition of an internal telescoping extension tube having a second air-lock hatch to allow a single crew member to exit and enter the capsule without complete depressurization of the capsule. The astronauts maneuvering unit (AMU) could be stored in the crew transfer tube and put on after leaving the capsule.

The remote maneuvering unit (RMU) is located in the open end of the service module and is ejected from the vehicle into space where it is operated remotely from within the command module. No consideration has been made for recovery of the RMU after use because of difficulty in its refurbishment since the RMU as presently conceived cannot be brought into the pressurized capsule for servicing.

The various displays are located on the left hand side of the pointing and tracking scope and are linked electronically to the various experiments mounted within the Apollo capsule and the service module. The radiometer is modified to be remotely operated so that the filters for various wavelength radiations are changed or rotated by a selector mechanism operated from the command module. The radiometer and the laser ranging unit are slaved to the pointing and tracking scope so that all three are looking at the same target when observations are made.

[redacted]

The various supporting optical or electronic systems and antennae are packaged within the service module and are deployed when needed to make observations or to pick up radiations.

The various major subsystems are located within the service module and are connected to the Apollo capsule through an umbilical connector. The life support system requires the recycling of the capsule air through lithium hydroxide cartridges for the removal of carbon dioxide. These cartridges are located within the service module because of both weight and volume limitations of the Apollo capsule.

The Apollo heat shield is modified for an earth orbit re-entry and the attachment of a retro-package consisting of six XM-85 solid propellant rocket motors, any five of which will provide a sufficient retrograde velocity to initiate re-entry. Among the major problems limiting the experimental capability of the Apollo command module are the recovery and abort systems weight restriction. Although the weight limitations permit only a restricted number of experiments to be carried per flight (Section 3), on the Apollo/MOL/Titan IIIC configuration, at least one advantage the Apollo capsule has over the Gemini B capsule, for example, is the capability of conducting some experiments without leaving the Apollo capsule.

1.3.2 Apollo/MOL/Saturn IB Configuration Using the Geometry of NAA Extended Mission Apollo Concept II

A preliminary layout, Figure 1-9 (Drawing ES-0152-001), was made of the Apollo/MOL on the Saturn IB launch vehicle with all the MOL experiments integrated into the laboratory module using the geometry of the North American Aviation Extended Mission Apollo Concept II (Reference 1-2). Reference 1-2 is a study which had been performed by NAA on the use of the Apollo vehicle as part of an integrally launched space laboratory with mission durations of 120 days or more, using a resupply technique and crew rotation. The resulting configuration, using the geometry of NAA Concept II, was based on the minimum modification of the Apollo command module and of the service module. Therefore, the heat shield of the Apollo vehicle is left unmodified from that required for re-entry from a lunar orbit, but a retro-package of six XM-85 solid propellant motors was incorporated with the heat shield of the Apollo command module for effecting a re-entry from an earth orbit. The service module retained the same basic structure but was modified to allow for the incorporation of the following: the retro-package, the LEM rocket engine of 10.5 K instead of the SM engine of 21.9 K thrust, the propellant tanks of smaller volume, the life support system, the environmental control system, the fuel cell power system, and the reaction jet attitude control system for CM/SM repositioning maneuvers.

The NAA Extended Mission Apollo Concept II laboratory module for six men was duplicated in the arrangement of the floors, docking cones, umbilical connectors, etc., and the MOL experiments were appropriately located to suit the mission requirements. Similar to the NAA Concept II, one compartment of the laboratory module nearest the command module, was designated for rest, feeding, personal hygiene, and recreation of the crew. The other compartment was designated for the location of all of the MOL experiments. An attempt was made to locate the experiments and the instruments in such a manner that related experiments could be observed from adjacent displays and instruments. Some thought also was given to the disposition of the weight of various pieces of equipment to result in a reasonable center of gravity location.

The two compartments are connected by a tunnel which serves both as a structural support for the large diameter floor panels and as a means of easy transfer from one compartment to another by the use of hand holds. Two sets of hand rails are provided in each compartment to facilitate crew motion in a weightless environment. A track for a restraining mechanism is provided in the "floor" of each compartment so that the astronaut can keep properly oriented. Seats with restraining straps are provided to maintain a suitable frame of reference for operation of various display consoles. The seats can move on tracks and are adjustable for the best seating location.

The incorporation of the MOL experiments requires some modification of the basic laboratory module of Concept II. The pointing and tracking scope requires the placement of the movable head portion externally to the laboratory wall and must allow for 360 degrees of rotation through a pressure seal. An aerodynamic fairing is required over the exposed scope during the injection into orbit after which it is jettisoned. A similar provision is required for the periscope/star theodolite optical system. A new separate air-lock compartment of 40-inch diameter and 75-inch length is required to conduct the "extravehicular system" experiments and operate the remote maneuvering unit. This compartment reduces the repressurization gas requirements for performing the experiments.

There are numerous optical ports and small sealed compartments required in the laboratory wall for the experiments. Each of the experiments and related equipment will have to be integrated structurally, mechanically, and electronically to meet the environmental effects imposed by the mission. Some items such as the parabolic and flat antennae are placed within the unpressurized section of the laboratory module and are deployed for operation.

The Apollo/MOL configuration based on the NAA Concept II geometry was laid out without regard to the payload weight limitation of the Saturn IB booster (Section 2. 2). The resulting laboratory module exceeds the payload weight limitation of the Saturn and is considerably in excess of the volume requirements for the MOL mission for two crew members. It is representative, however, of the type of laboratory that could be used for a larger crew when both the mission requirements and payload capabilities are increased.

The NAA Concept II was designed for docking of the combined command and service modules on the laboratory module to effect crew transfer after a repositioning maneuver and requires making two umbilical connections adjacent to the crew transfer tunnel. This means that the supply lines must be carried from the service module through the Apollo command module and to the umbilicals. The making of umbilical connections in addition to the docking procedure unduly complicates the system design and adds certain weight penalties. The NAA Concept II requires the docking of a second Apollo command and service module before the first command/service modules can separate for return of crew members from orbit. If this is not done, the laboratory module would stop functioning because the environmental control and life support systems supplied by the service module would not be available.

It is clear that the Apollo/MOL configuration based on the NAA Concept II is best suited for extended missions requiring resupply and crew rotation.

1.3.3 Apollo/MOL/Saturn IB Configuration Modified NAA Concept II

The Apollo/MOL configuration, using an arrangement similar to the NAA Extended Mission Apollo Concept II, but modified for two crew members instead of six by reducing the laboratory volume, is shown in Figure 1-10 (Drawing ES-0152-002).

The modified configuration has an Apollo command module modified for an earth orbit and re-entry and a modified service module attached to a single compartment laboratory module. The heat shield of the Apollo vehicle is modified for an earth orbit re-entry instead of a lunar orbit re-entry and consequently is nearly 600 pounds lighter. A retro-package of six XM-85 solid propellant motors, as in the previous configuration, is used for effecting re-entry. The service module was shortened by 62 inches and was modified to incorporate the same subsystems as before.

The single laboratory compartment is arranged to combine areas for rest, feeding, personal hygiene, recreation, and the MOL experiments. The experiments and instruments are located so that related experiment displays could be observed from one position. The aft docking cone and air-lock was replaced by an air-lock compartment 40 inches in diameter and 75 inches long for conducting the "extravehicular" experiments and for operating the remote maneuvering unit.

A single hand rail is provided around the interior of the compartment to facilitate crew motion in the weightless environment. Again, a track for a restraining mechanism is provided in the "floor" of the compartment and adjustable seats on tracks are provided with restraining straps.

The pointing and tracking scope, the periscope/star theodolite, the deployable antennae and various other experiments were integrated into the laboratory similarly to the larger laboratory shown in Figure 1-9.

The modified configuration is designed for a single docked command/service module which supplies the environmental control and life support. The laboratory again stops functioning after separation because of its dependence on the command/service module for support. There is some capability for growth and mission extension but it requires enlarging the service module to carry more supplies or the use of a resupply technique and some modification such as adding the second docking cone and air-lock.

In the case of both Concept II configurations, to achieve the final circularization of the orbit, the following sequence is performed during the 45-minute coast period after the Saturn IVB stage burnout:

a. Separation of the command and service modules as a unit from the laboratory module

b. Repositioning maneuver (180-degree turn around) of the command/service module unit

c. Docking of command/service module unit on the laboratory module and making two umbilical connections

d. Separation of the Saturn IVB stage from the laboratory module

e. Complete 180-degree turn around of all three combined stages

f. Firing of the 10.5 K LEM engine at apogee of orbit

1.3.4 Apollo/MOL/Saturn IB Designed Specifically to MOL Requirements

A preliminary layout of an Apollo/MOL configuration with all the MOL experiments integrated into a laboratory module is shown in Figure 1-11. The Apollo/MOL configuration consists of the Apollo command module with a short adaptor attached to the combined laboratory/service module. This configuration has a single laboratory compartment of approximately the same size and arrangement as the modified Concept II configuration (Figure 1-10). The "extravehicular" air-lock is relocated to allow for the placement of the LEM 10. 5 K rocket engine on the center directly aft of the laboratory module bulkhead.

The separate service module required for both the NAA Concept II and the NAA optimized Concept II is eliminated and is combined with the laboratory module. The subsystems, the propulsion system, and the experiments are identical with the NAA modified Concept II but are rearranged and relocated within the space external to the laboratory to suit the aft mounted propulsion system. This arrangement allows the rocket motor to be operated to circularize into the final orbit at apogee without requiring the repositioning maneuver of the command and service modules during the coast period as in the case of both Concept II configurations.

The Apollo/MOL configuration with the subsystems integrated into the laboratory module requires only one repositioning maneuver after attaining final orbit with no definite time limit imposed for performing the repositioning maneuver. Since the subsystems are rearranged, the umbilical connections to provide environmental control and life support to the laboratory as used in Concept II are unnecessary. The crew can shorten the mission duration at any time and separate from the laboratory module without interrupting the functioning of the laboratory since the subsystems are self-contained.

This Apollo/MOL/Saturn IB configuration has a growth capability to accommodate larger crews and to extend mission duration.

1.4 PERTURBATION OF THE APOLLO/MOL CONFIGURATION

1.4.1 Configuration Arrangements and Laboratory Size

The laboratory volume can be varied to suit: the mission requirements, the crew size, the MOL experiment requirements, and the payload capability of the launch vehicle. The arrangement of the Apollo/MOL configuration can also be varied as shown in Figures 1-1, 1-2, and 1-3. Figure 1-4 shows the same arrangement as Figure 1-3, but the laboratory size is decreased to a cylindrical habitable compartment 150 inches internal diameter and of varying height, resulting in a volume of 1230 cubic feet.

1.4.2 Methods of Crew Transfer

The method of crew transfer has been considered in terms of a docking procedure and entry through the use of air-locks for which the Apollo vehicle was designed (References 1-3 and 1-4). Other methods of crew transfer are shown in Figures 1-5 through 1-8 as perturbations of Figure 1-4. Figure 1-5 shows a modified Apollo capsule and retro package with the central portion of the heat shield hinged out of the way to allow for a pressurized crew transfer tunnel to connect the laboratory with the Apollo capsule. This configuration would require an air-lock hatch both in the floor of the capsule and possibly in the laboratory bulkhead, which would be sealed before separation. The sealing of the hinged heat shield would be a technological problem to be solved before safe re-entry could be effected. The laboratory would need an air-lock hatch only in case of extended missions requiring resupply or crew rotation.

Figure 1-6 shows a modified Apollo capsule with a pressurized inflatable tunnel which connects the side hatch of the Apollo vehicle with the side wall of the laboratory module. A hatch in each vehicle, at the entrance to the tunnel, would be required to make a satisfactory design. The laboratory could be modified for extended mission and crew rotation by adding a central docking cone and an air-lock hatch. It is doubtful that the pressurized tunnel can be suitable for crew transfer after the separation of the first Apollo capsule unless a rear docking technique is combined with a method of securing the tunnel to the side hatch of subsequent Apollo capsules.

Figure 1-7 shows a modified Apollo capsule supported on trunnions from the laboratory module. Through the use of a flexible control wire or by radio link, the trunnion mechanism can be operated to separate the Apollo capsule from the launch supports of the laboratory module a sufficient distance to allow the mechanism to rotate the Apollo capsule 180 degrees and draw it into the air-lock hatch of the laboratory module. Crew transfer is effected through the air-lock hatches.

The mechanism can be precisely controlled and programmed to operate at the pressing of a single button. The addition of the trunnion mechanism would be a weight penalty but would insure a safe and a highly reliable system for docking and crew transfer. Weight could be saved by designing the mechanism to accomplish the rotation of the Apollo capsule at a slow rate and by cushioning against resilient material on stopping and on drawing together.

Figure 1-8 shows a modified Apollo capsule in which crew transfer is effected by the "extravehicular" system. The crew leaving the Apollo capsule moves by means of hand holds to the laboratory and enters through a tunnel serving as an air-lock.

1.4.3 Extension of Mission Duration

Extending the mission beyond the 30-day requirement involves both the physiological and endurance capability of the crew members and the quantity of supplies aboard the laboratory or service module. The crew members could likely endure being confined in a laboratory provided their duty cycle, recreation period, and rest periods are well balanced. The unknown aspect is the physiological and psychological effect of weightlessness for extended periods. It may be that even 30 days is too long a period to remain weightless, thus requiring crew rotation or periods of "artificial gravity" treatment to rejuvenate the crew.

The increased supply requirements can be handled by making the life support, power supply, attitude control systems, etc., of sufficient capacity to be ample for the duration of the mission or a resupply technique could also be used. By using the resupply technique, the original launch weight could remain the same as for shorter duration missions, but launching, rendezvous and resupply procedures would be required for the mission duration.

[redacted]

One aspect to be considered in extending mission duration is the effect of the "hard-vacuum" on the parachute recovery system of the Apollo vehicle. Since the parachutes consist of nylon fibers and certain plasticisers, evaporation effects on these constituents could result in recovery system failure. It may be necessary for extended missions to repackage the entire recovery system within pressure tight cannisters to overcome vacuum effects, to redesign the entire recovery system, or to use some other material less affected by vacuum.

1.4.4 Addition of Artificial Gravity Capability

There are many schemes for inducing artificial gravity, but they all require some means of rotating the crew or the entire station. The laboratory module could be modified to contain a large centrifuge below the "floor" of the laboratory or without much modification, a small centrifuge can be used in the laboratory which could be stored out of the way until needed. The type of machine and the requirements for artificial gravity have not been established and hence will not be considered until the need for artificial gravity is clarified.

[redacted]

1.5 CONCLUSIONS

The following conclusions have been derived from this study:

1. The Apollo/MOL/Titan IIIC configuration is limited both in volumetric and payload weight capability in carrying experiments. Experiments P-1, P-3, P-6, P-11, and P-12 must be carried entirely within the capsule; portions of experiments P-2, P-4, P-7, P-8, and P-10 may be carried in the unpressurized area. Some value, however, can be attributed to this configuration because the astronauts do not have to leave the capsule to perform the experiments.

2. The Apollo/MOL/Titan IIIC configuration, because of the limited weight and space available for experiments, would require more flights than the Apollo/MOL/Saturn 1B configuration to cover the range of MOL experiments.

3. All of the Apollo/MOL/Saturn 1B configurations appear satisfactory in volume capability for containing all of the MOL experiments in one flight.

4. The growth potential and mission duration capability of the Apollo/MOL is dependent on the launch vehicle payload capability and on updating of the information on space effects.

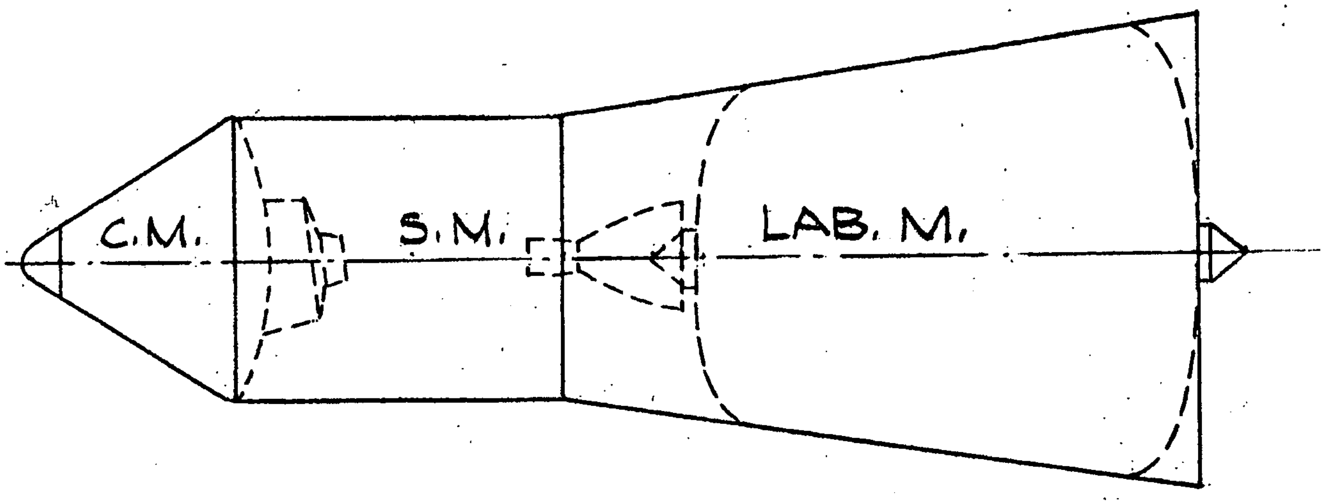

Figure 1-1. Minimum Modification Command Module, Service Module and Laboratory Module Using NAA Concept II Geometry

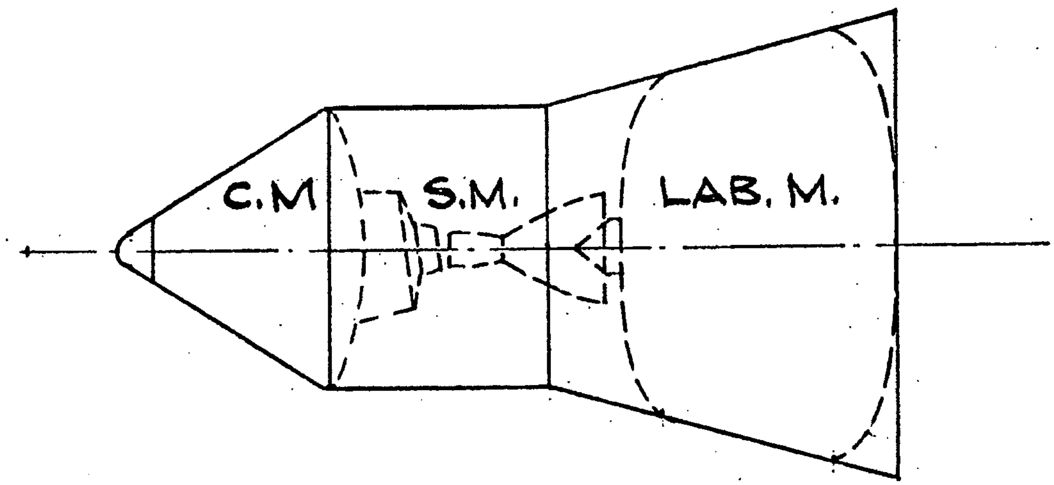

Figure 1-2. Command Module, Service Module and Laboratory Module (Modified NAA Concept II)

[redacted]

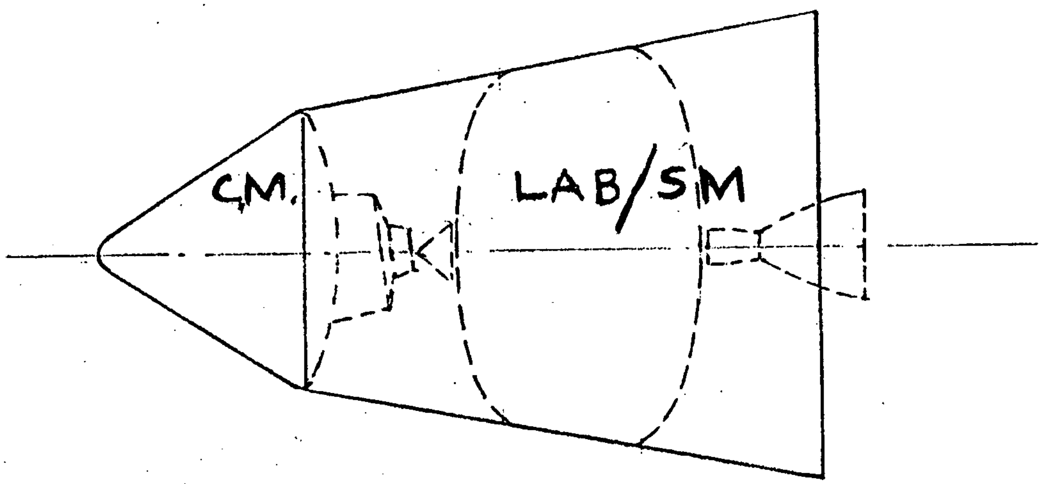

Figure 1-3. Command Module and Combined Laboratory/Service Module with the Propulsion System Aft

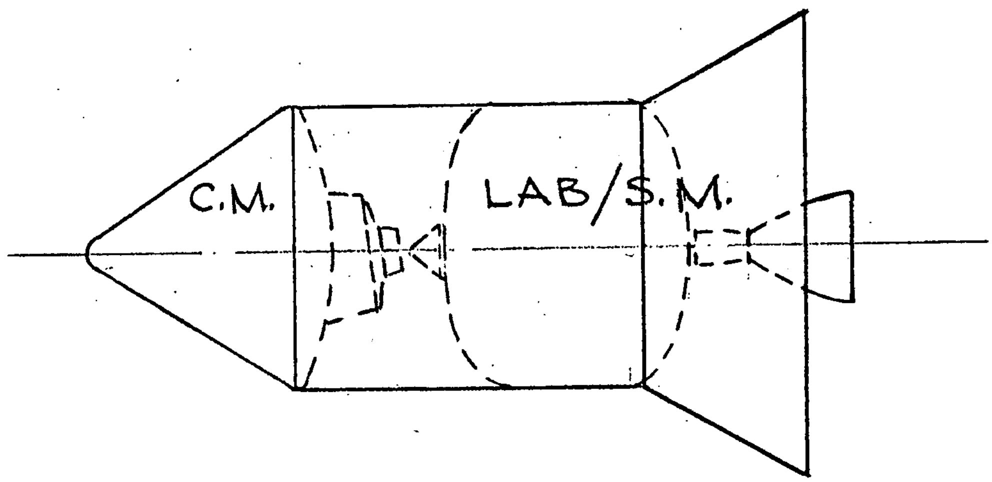

Figure 1-4. Command Module and Combined Laboratory/Service Module Reduced in Volume with the Propulsion System Aft.

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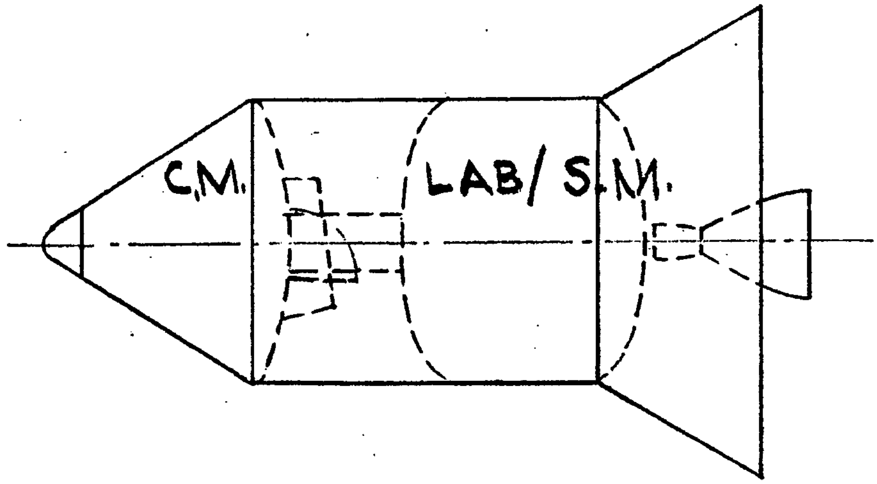

Figure 1-5. Crew Transfer Effected by a Tunnel through Command Module Heat Shield

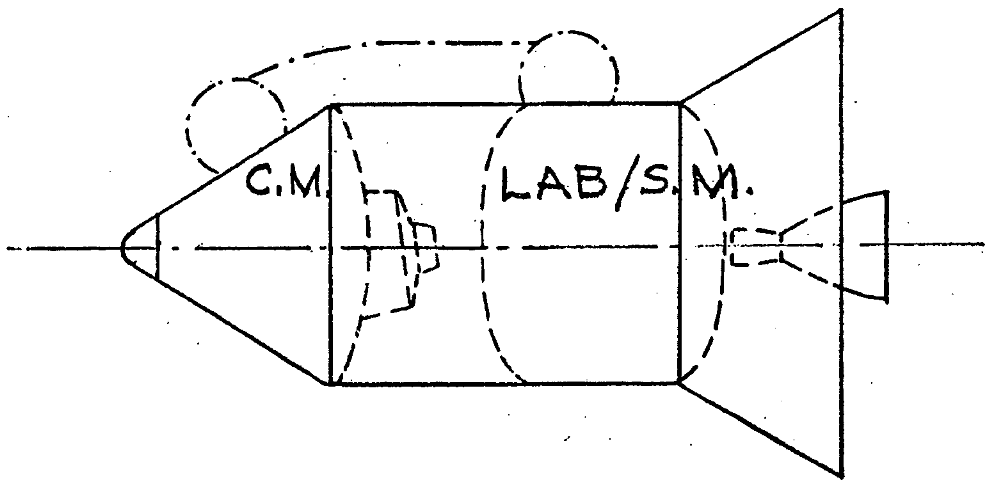

Figure 1-6. Crew Transfer Effected by an Inflatable Pressurized Tunnel from the Command Module to the Laboratory Module

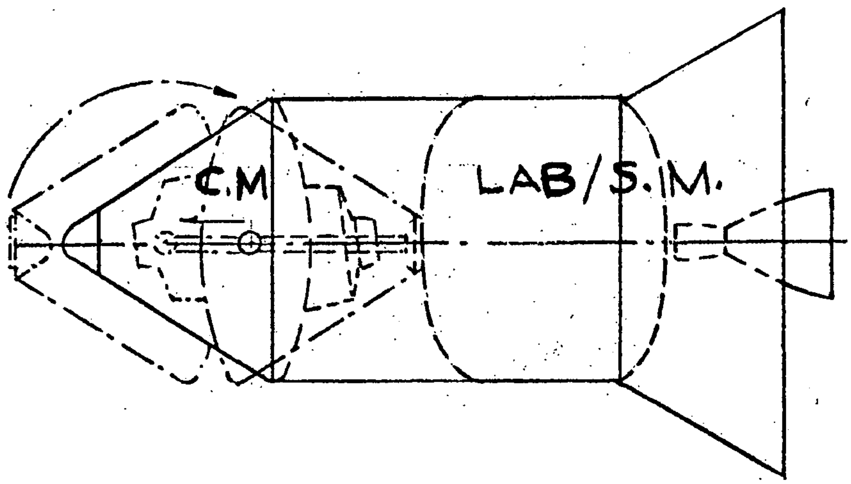

Figure 1-7. Crew Transfer Effected by the Repositioning of the Command Module Using a Mechanical Trunnion

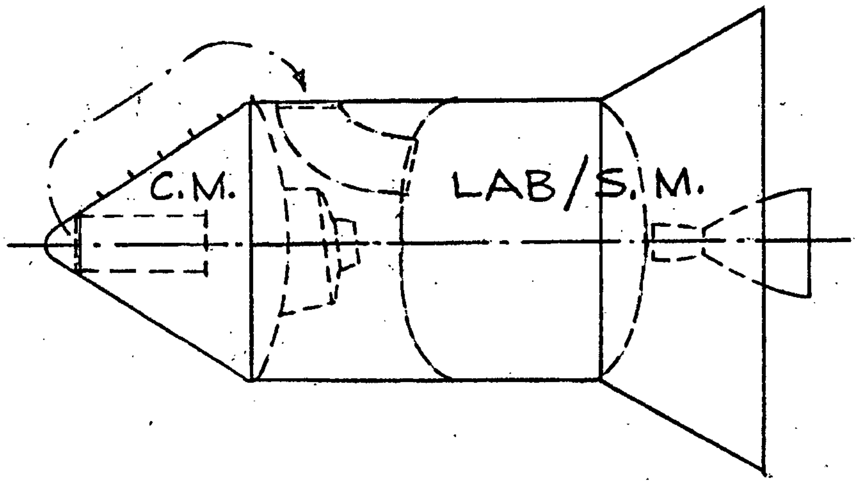

Figure 1-8. Crew Transfer Effected by Extravehicular Excursion of Crew Members through Air Locks

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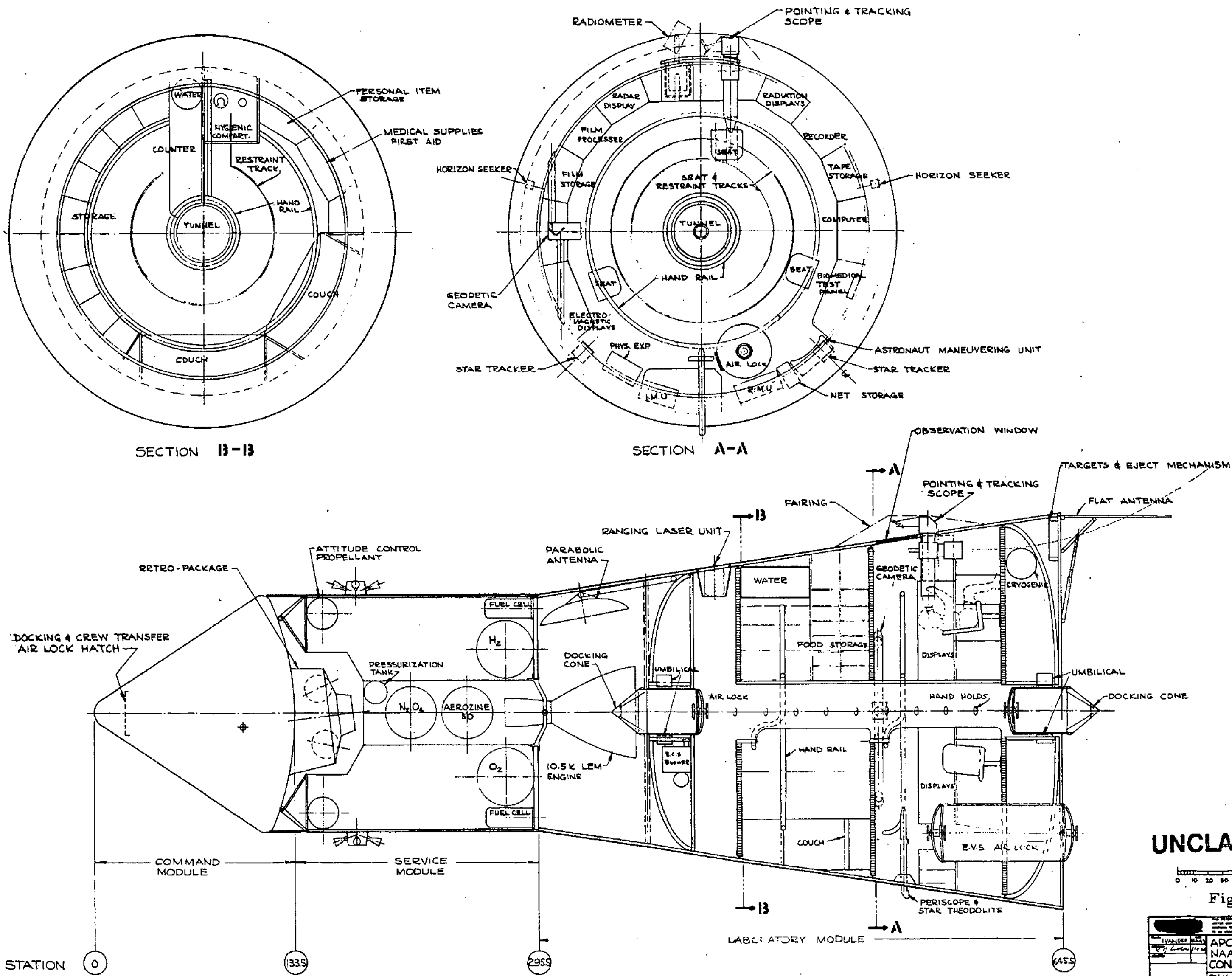

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Figure 1-9

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APOLLO/MOL NAA CONCEPT II CONFIGURATION

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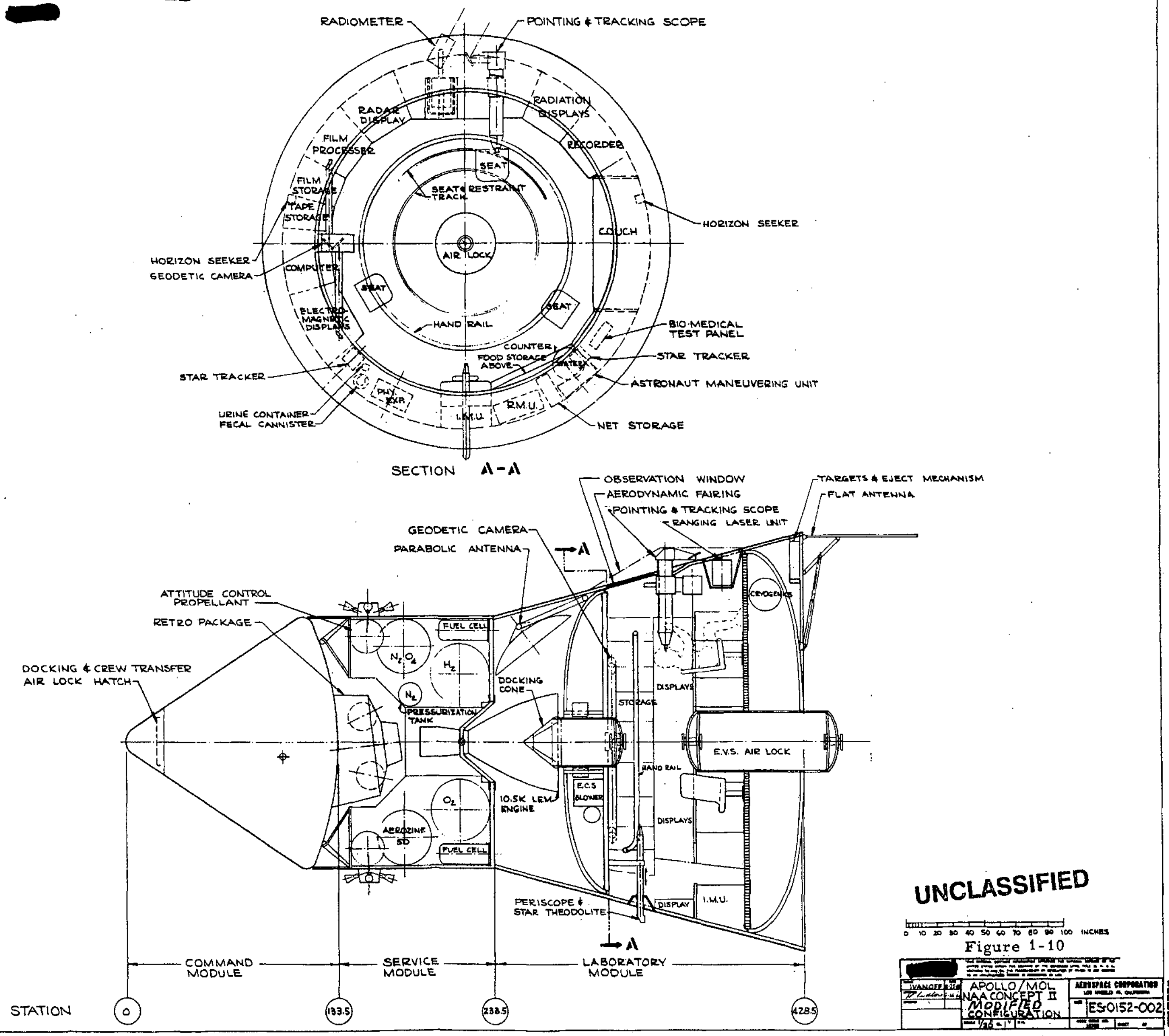

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Figure 1-10

[redacted] IVANOFF	APOLLO/MOL NAA CONCEPT II MODIFIED CONFIGURATION	AEROSPACE CORPORATION ES-0152-002

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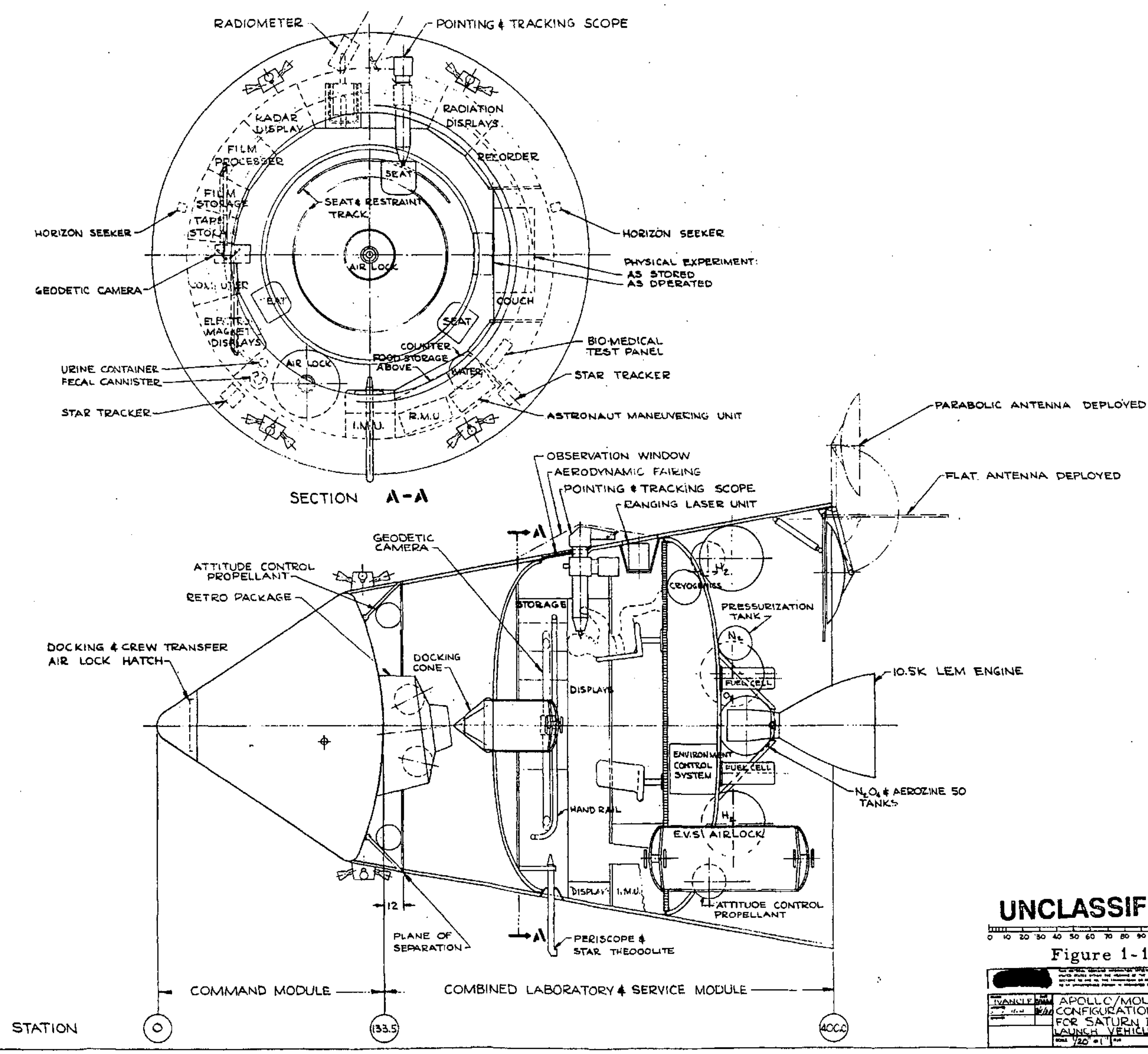

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0 10 20 30 40 50 60 70 80 90 100 INCHES

Figure 1-11

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APOLLO/MOL CONFIGURATION FOR SATURN IB LAUNCH VEHICLE

AEROSPACE CORPORATION

ES-0152-003

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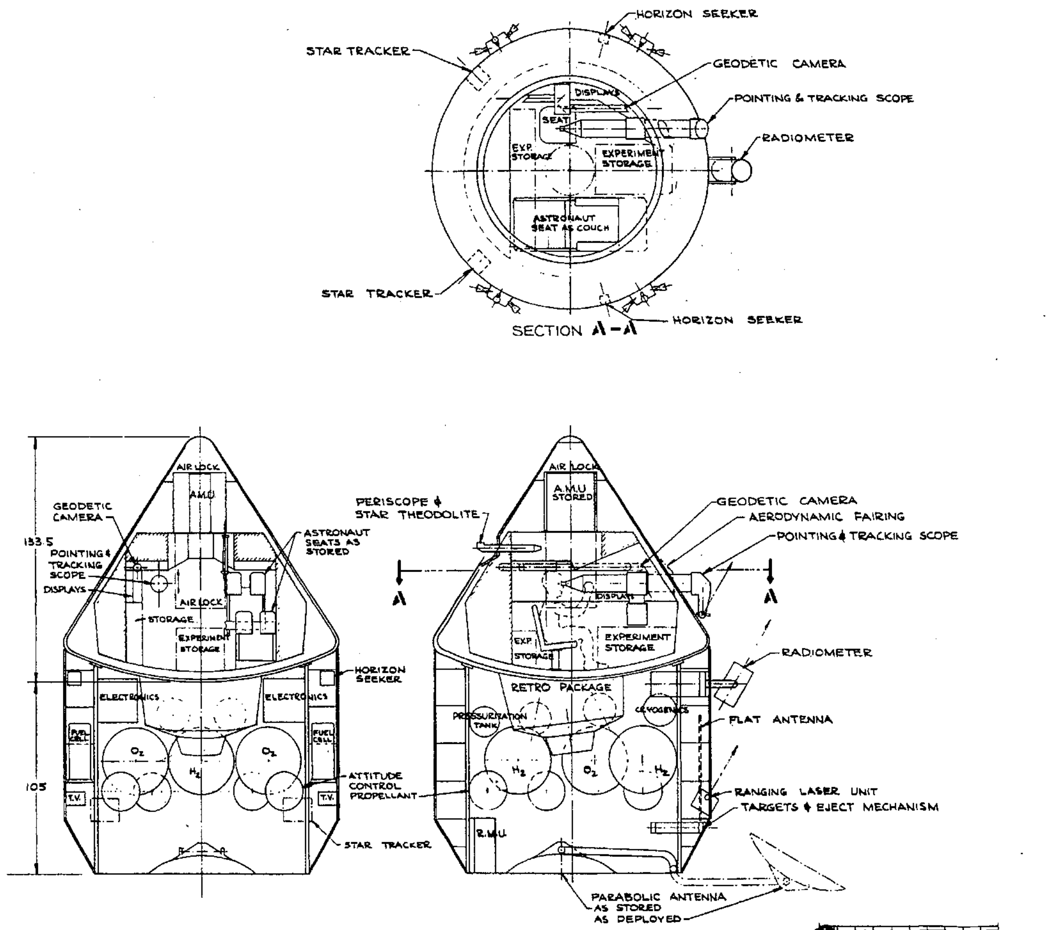

Figure 1-12

[redacted]

APOLLO/MOL CONFIGURATION FOR TITAN IIIc LAUNCH VEHICLE	AEROSPACE CORPORATION	ES-0152-004

UNCLASSIFIED

REFERENCES

1-1 "MOL Preliminary Experiments Data Book," Aerospace Report, TOR-269(4107-35)-1, 30 June 1964 (Secret).

1-2 "Extended Mission Apollo Study," North American Aviation Inc., Report SID63-1370, November 1963 (Confidential).

1-3 NASA Report, "Project Apollo - Spacecraft Description," 1 September 1963 (Confidential).

1-4 "Modified Apollo Logistics Spacecraft," North American Aviation Inc., Final Review, Report No. SID63-1459, 3 December 1963 (Confidential).

SECTION 2

WEIGHTS

C. Stewart

SUMMARY

Preliminary weight estimates are computed for a space station system using an Apollo re-entry vehicle, a service module and an experimental laboratory that would orbit at 160 nautical miles.

These weight estimates enable the weight available for experiments to be determined for various space station configurations and mission durations, and for two launch vehicles: the Titan IIIC and the Saturn IB.

The results show that, for the space station configuration launched by the Titan IIIC, there is an experiment weight capability of approximately 2280 pounds, and for the various space station configurations launched by the Saturn IB this experiment weight capability ranged from approximately 180 pounds to 6255 pounds. All configurations include an additional velocity increment of 200 feet per second that is available for experiments.

CONTENTS

TABLES

2. WEIGHTS

2.1 INTRODUCTION

Preliminary weight estimates are made for Apollo/MOL space station configurations using, as a launch vehicle, the Titan IIIC in one case and the Saturn IB in various other configurations. These weight estimates present the weight available for experiments, in the various space station systems, all of which orbit at 160 nautical miles. The basis for these estimates is data from North American Aviation supplemented by analytical weight estimating techniques.

2.2 PRELIMINARY WEIGHT STUDY FOR APOLLO/MOL USED WITH TITAN IIIC

Preliminary weight estimates were computed to assess the experimental payload capability of a MOL system using a modified Apollo crew module and a service module. The mission postulated for this study was an integral launch, east from ETR into a 160-nautical mile orbit using the Titan IIIC launch vehicle. (See page 7-62, Section 7.)

2.2.1 Vehicle Description

2.2.1.1 Apollo Crew Module

The Apollo re-entry vehicle considered for this study was based on the standard NASA vehicle modified to meet the following requirements:

a. The crew size was reduced from three to two men.
b. The vehicle equipment and structural heat shield were modified for the earth orbit mission.
c. The mission duration was set at 30 days.
d. A de-orbit system was required in the re-entry vehicle.

2.2.1.2 Service Module

This vehicle was a new design and its size was constrained by the booster capability and the weight of the Apollo re-entry vehicle. It was designed to contain the service equipment required for the 30-day mission as well as the volume of the proposed experiments.

2.2.2 Basis for Weight Analysis

The weight of the crew module (Apollo) was based on data contained in Reference 2-1 supplemented by more recent weight data (Reference 2-3) obtained from NASA, Cape Kennedy on the Block 2 vehicle system. This supplementary data listed a weight increase of approximately 1300 pounds over that quoted in the November 1963 monthly weight status, Reference 2-1. This increase was due to the addition of docking requirements, an exit heat shield, and a new maintenance concept. The weight increments for the reduction in heat shield for the low orbit mission and the addition of a de-orbit system were based on data shown in Reference 2-2.

The weight statement includes a 2000-pound growth contingency in order that a payload comparison can be made between this configuration and the Gemini/MOL design.

The weights for the service module section were based upon the dimensional data as depicted in Section 1 of this report and analytical weight estimating methods. The weights for the equipment in the service module were based on data appearing in Reference 2-1, modified to include a fuel cell power supply capable of operating for 30 days.

2.2.3 Results

A weight summary showing the weight of the Apollo/MOL configuration is presented in Table 2-1.

Weight statements for the Apollo crew module and the service module are shown in Tables 2-2 and 2-3.

2.2.4 Conclusions

The study indicated an experimental payload capability of 2280 pounds for the Apollo/MOL/Titan IIIC launch configuration. This weight compared with 3432 pounds of experiments carried on a Gemini B/MOL/Titan IIIC configuration for a 30-day mission. These results suggest that more flights would be required to accomplish the same program covered by the Gemini B/MOL, or that a reduced program of experimentation could be performed with the Apollo/MOL system.

The weights derived for this study were based on preliminary data and should be regarded as approximate.

2.3 PRELIMINARY WEIGHT STUDY FOR APOLLO/MOL USED WITH SATURN IB

A preliminary weight estimate was made of three MOL system configurations designed around the present Apollo command module and boosted by the Saturn IB launch vehicle. The study included perturbations of the three baseline configurations for mission durations of 30, 60, and 120 days using alternate power supply system concepts. A nominal experimental payload weight of 4395 pounds was used throughout the study.

2.3.1 System Description

The three baseline configurations are described in the following brief summaries. These studies were made for mission durations of 30, 60, and 120 days.

2.3.1.1 Minimum Modification Apollo/MOL

The minimum modification configuration is composed of the present Apollo command module modified for a two-man crew, the present service module with a new propulsion system, and a laboratory module. This system is shown in Figure 1-9, Section 1.

2.3.1.2 Modified Apollo/MOL

The modified configuration is similar except the command module heat shield is reduced to earth orbit requirements, the service module is shortened, a smaller laboratory module is utilized, and the electrical power system is changed. Studies were made on the modified version, designed for mission durations of 30 to 120 days, comparing current technology fuel cells and a LiOH carbon dioxide removal system with a system using solar panels and molecular sieves. An outline drawing showing this configuration is contained in Figure 1-10.

2.3.1.3 Alternate Configuration

This configuration is similar to the modified version except that the propulsion system is moved from the service module and mounted on the aft bulkhead of a combined laboratory and service module. This configuration is shown in Figure 1-11.

2.3.2 Experimental Payload Definition

The experimental payload of 4395 pounds was established for this study by the Program Office and is based on estimates made by the Weight Prediction Section for the Gemini/MOL Program. This nominal payload weight was used so that a relative comparison could be made among the various configurations studied.

2.3.3 Mass Property Data

The mass property data shown in Table 2-4 include an approximate estimate of the inertia values made to provide basic data for control system analysis. The longitudinal centers of gravity for vehicle conditions 1 and 2, shown in the table, were measured forward of the space station to S-IVB booster interface. These values for the vehicle in condition 3 were measured aft of the crew module nose leading edge. The reason for this change in reference datum between vehicle conditions 1, 2, and 3 is due to the 180° rotation of the Apollo re-entry vehicle during the docking maneuver. The vertical centers of gravity were measured from the vehicle centerline and in all cases were below this centerline reference datum.

2.3.4 Basis for Weight Analysis

The weights for this study were based on configuration drawings shown in Section 1. The weight used for the Apollo Command Module was based on Reference 2-1, the North American Aviation monthly weight status of November, 1963, supplemented by more recent weight data (Reference 2-3) obtained from NASA on the Block 2 vehicle system. The supplemental weight data included approximately 1300 pounds increase in weight for docking capability, an exit heat shield and a new maintenance concept. In this study the Apollo re-entry heat shield weight was reduced by 600 pounds for the low orbit mission and a de-orbit retro rocket system was added. These data were based on the North American Aviation XMAS study, Reference 2-2. The space station system service and supply section weights were derived for mission periods of 30, 60, and 120 days considering fuel cell and solar array power supplies as alternatives. (A weight allowance of 650 pounds was included for the effective weight of the abort tower.) A weight contingency of 2000 pounds based on that used in the Gemini/MOL weight studies was included to facilitate the comparison to be made between these studies and the Gemini/MOL studies.

2.3.5 Results

The results of this weight study are presented in Table 2-5 as estimated weight statements for the various configurations. Included for comparative purposes were the reference North American Aviation configurations for 3 man 14-day and 3 man 120-day missions. This table includes a fixed experiments weight of 4395 pounds and indicates the total weights of all the configurations for this experimental weight. Table 2-6 is prepared to show the available experimental weight capability for a Saturn IB booster capability of 33,460 pounds. Also included for all configurations is a velocity increment of 200 feet per second, available for the experiments.

2.3.6 Conclusions

Configurations D, G, H, and K are capable of carrying the proposed payload of 4395 pounds.

Configurations C, D, and K may be compared on a mission basis with Gemini/MOL; of these, D and K have 5860 and 5460 pounds available for experiments (after considerable modification to the Apollo system) versus 3432 for the Gemini B/MOL/Titan IIIC; configuration C (relatively unmodified Apollo system) can only carry 180 pounds of experiments.

Table 2-1

Weight Summary
Apollo/MOL Configuration for Titan IIIC Launch Vehicle

Item		Weight (lb)
Crew Module		12,300
Apollo (including 2 men and retro section)	11,070	
Experimental payload	1,230*	
Service Module		6,400
Structure and equipment	5,350	
Experimental payload	1,050*	
Payload Contingency		2,000
Total Booster Payload		20,700

*These values differ very slightly from those presented in Volume I because they are based on later information.

Table 2-2

Weight Statement
Apollo Crew Module
Apollo/MOL Configuration for Titan IIIC

Item	Weight (lb)
Structure	4,000
Crew Systems	210
Communications	302
Instruments	132
Control and Displays	160
Guidance and Navigation	282
Stability and Control	220
Reaction Control	323
Electrical Power	480
Environmental Control	300
Transfer Provisions	210
Earth Landing System	753
Weight Growth since contract	1,300
CREW MODULE - EMPTY WEIGHT	8,672
Crew Systems	705
Reaction Control	270
Environmental Control	153
Experimental Payload	1,230
CREW MODULE - RE-ENTRY WEIGHT	11,030
Retro-Rockets	936
Retro Package Structure	102
External Power Supply	232
CREW MODULE - DE-ORBIT WEIGHT	12,300

Table 2-3

Weight Statement
Service and Equipment Module
Apollo/MOL Configuration for Titan IIIC

Item	Weight (lb)
Structure	1,550
Electronics	177
Reaction Controls, Fixed	400
Electrical Power, Fixed	1,020
Environmental Control, Fixed	150
Experimental Payload	1,050
SERVICE MODULE - EMPTY WEIGHT	4,347
Reaction Control - Useful Load	500
Electrical Power - Useful Load	1,160
Environmental Control	393
SERVICE MODULE - LOADED WEIGHT	6,400

Table 2-4

Moment of Inertia for Minimum Modification Vehicle
Mass Properties

Vehicle Condition	Weight Pounds	Center of Gravity Inches		Moment of Inertia Slug/ft^2		
		Longitudinal	Vertical	Pitch	Yaw	Roll
1. Apollo/MOL - After Docking	35,695	293*	2.4***	216,785	216,285	33,200
2. Apollo/MOL - Less Mission Propellant	34,275	285*	2.5***	204,010	203,650	33,195
3. Apollo - At Docking Maneuver	21,540	150**	4.0***	30,305	29,840	10,200

* Measured forward of space station to S-IVB/booster interface.

** Measured aft of the command module nose leading edge.

*** Measured from vehicle longitudinal centerline.

TABLE 2-5

ESTIMATED WEIGHT STATEMENTS FOR APOLLO/MOL ON SATURN IB
(Based on a fixed experiment capability of 4,395 pounds)

	A	B	C	D	E	F	G	H	I	J	K	L	M
VEHICLE TYPE	Reference Vehicle		Min. Mod	←		Modified N.A.A. Concept II				→	←Alternative Vehicle→		
CONTRACTOR	NAA	NAA	←					Aerospace					→
CREW SIZE	3 men	3 men	2 men	2 men	2 men	2 men	2 men	2 men	2 men	3 men	2 men	2 men	2 men
MISSION DURATION	14 days	120 days	30 days	30 days	60 days	120 days	30 days	60 days	120 days	120 days	30 days	60 days	120 days
POWER SUPPLY	Fuel Cells	Solar Cells	Fuel Cells	Fuel Cells	Fuel Cells	Fuel Cells	Solar Cells	Solar Cells	Solar Cells	Solar Cells	Fuel Cells	Fuel Cells	Fuel Cells
COMPONENT													
COMMAND MODULE	(11,000)	(9443)	(10,135)	(9,535)	(9,715)	(10,075)	(9,465)	(9,465)	(9,465)	(9,815)	(9,535)	(9,715)	(10,075)
Structure	4,561	4,200	4,600	4,000	4,000	4,000	4,000	4,000	4,000	4,000	4,000	4,000	4,000
Crew Systems	298	475	210	210	210	210	210	210	210	300	210	210	210
Communications	392	360	310	310	310	310	310	310	310	310	310	310	310
Instruments	175	193	130	130	130	130	130	130	130	130	130	130	130
Controls and Displays	282	286	280	280	280	280	280	280	280	280	280	280	280
Guidance and Navigation	420	475	280	280	280	280	280	280	280	280	280	280	280
Stabilization and Control	220	251	220	220	220	220	220	220	220	220	220	220	220
Reaction Control	323	328	325	325	325	325	325	325	325	325	325	325	325
Electrical Power	488	470	490	490	490	490	490	490	490	490	490	490	490
Environmental Control	303	399	300	300	300	300	410	410	410	410	300	300	300
Earth Landing	673	598	675	675	675	675	675	675	675	675	675	675	675
Weight Increase (per NASA)	1,270	-	1,270	1,270	1,270	1,270	1,270	1,270	1,270	1,270	1,270	1,270	1,270
Useful Load - Crew Systems	908	1,054	555	555	555	555	555	555	555	815	555	555	555
- Reaction Propellants	270	270	270	270	270	270	270	270	270	270	270	270	270
- Environment	167	76	220	220	400	760	40	40	40	40	220	220	220
- Experiments	250	-	-	-	-	-	-	-	-	-	-	-	-
RETRO SYSTEM	-	(1,030)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)	(1,270)
Rockets	-		936	-	-	-	-	-	-	-	-	-	-
Structure	-	~~1,030~~	102	-	-	-	-	-	-	-	-	-	-
Batteries	-	-	232	-	-	-	-	-	-	-	-	-	-
SERVICE MODULE	(10,200)	(9,359)	(9,710)	(7,220)	(10,010)	(15,155)	(4,845)	(5,830)	(7,750)	(5,230)	(7,820)	(10,610)	(15,705)
Structure	2,210	2,285	2,210	1,400	1,450	1,525	1,375	1,400	1,400	1,450	2,000	2,050	2,075
Electronics	177	177	175	175	175	175	175	175	175	175	175	175	175
Reaction Control	580	1,010	585	585	620	680	600	660	765	765	585	620	680
Electrical Power	1,363	245	2,790	1,610	2,465	3,755	200	200	200	200	1,610	2,465	3,755
Environment Control	92	654	190	190	290	495	190	290	495	635	190	290	495
Propulsion	3,038	800	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000
Weight Increase (per NASA)	510	-	510	-	-	-	-	-	-	-	-	-	-
Useful Load - Reaction Propellant	838	2,855	860	860	1,220	1,940	1,085	1,675	2,855	2,855	860	1,220	1,940
- Environment	208	1,268	220	220	430	860	280	430	860	1,150	220	430	860
- Electrical Power	503	-	1,170	1,180	2,360	4,725	-	-	-	-	1,180	2,360	4,725
- Residuals, Reserve	681	65	-	-	-	-	-	-	-	-	-	-	-
SPACE STATION	-	(10,670)	(8,755)	(6,165)	(6,290)	(6,540)	(7,915)	(8,040)	(8,290)	(8,590)	(5,965)	(6,090)	(6,340)
Structure	-	7,600	7,090	4,500	4,500	4,500	4,500	4,500	4,500	4,500	4,300	4,300	4,300
Electrical	-	1,437	200	200	200	200	1,950	1,950	1,950	1,950	200	200	200
Communications	-	50	50	50	50	50	50	50	50	50	50	50	50
Environmental Control	-	5	175	175	175	175	175	175	175	175	175	175	175
Instruments	-	222	225	225	225	225	225	225	255	225	225	225	225
Accomodations	-	295	550	550	550	550	550	550	550	600	550	550	550
Crew Systems	-	761	165	165	290	540	165	290	540	790	165	290	540
Spares	-	300	300	300	300	300	300	300	300	300	300	300	300

TABLE 2-5

ESTIMATED WEIGHT STATEMENTS FOR APOLLO/MOL ON SATURN IB

(Based on a fixed experiment capability of 4,395 pounds)
(Continued)

	A	B	C	D	E	F	G	H	I	J	K	L	M
EXPERIMENTS	-	(880)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)	(4,395)
PROPELLANT - MISSION	-	(760)	(1,430)	(1,210)	(1,625)	(2,420)	(1,500)	(2,000)	(2,845)	(2,890)	(1,210)	(1,625)	(2,420)
Mission ΔV	-	760	1,430	1,210	1,345	1,590	1,200	1,260	1,370	1,415	1,210	1,345	1,590
Orbit Sustenance	-	-	-	-	280	830	300	740	1,475	1,475	-	280	830
CONTINGENCY	-	-	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)	(2,000)
TOTAL WEIGHT	-	32,142	37,695	31,795	35,305	41,855	31,390	33,000	36,015	37,190	32,195	35,705	42,205

TABLE 2-6

WEIGHT AVAILABLE FOR EXPERIMENTS AND/OR MANEUVER

(Based on Saturn IB booster capability of 33,460 pounds)

	A	B	C	D	E	F	G	H	I	J	K	L	M
TOTAL WEIGHT (from TABLE 2-5)			37,695	31,795	35,305	41,855	31,390	33,000	36,015	37,190	32,195	35,705	42,205
Less:													
Experiments			4,395	4,395	4,395	4,395	4,395	4,395	4,395	4,395	4,395	4,395	4,395
1 Propellant - Mission ΔV (380 fps)			1,430	1,210	1,625	2,420	1,500	2,000	2,854	2,890	1,210	1,345	1,590
WEIGHT LESS EXPERIMENTS AND MISSION ΔV			31,870	26,190	29,565	35,870	25,795	27,345	30,250	31,380	26,590	29,965	36,220
Add:													
Available weight for Experiments			180	5,860	2,485	-1,000	6,255	4,705	1,800	670	5,460	2,085	-1,350
2 Propellant - 200 fps for Experiments			780	780	780	780	780	780	780	780	780	780	780
2 Propellant - 180 fps circularization			630	630	630	630	630	630	630	630	630	630	630
SATURN IB BOOSTER CAPABILITY			33,460	33,460	33,460	33,460	33,460	33,460	33,460	33,460	33,460	33,460	33,460
(Page 7-64, Section 7)													

1 Based on Total Weight shown at top.

2 Based on Total Weight of 33,460 pounds.

REFERENCES

2-1. Monthly Weight and Balance Report for Apollo Spacecraft (C), S. I. D. 62. 99. 21, dated 1 November 1963, North American Aviation, Inc.

2-2. Extended Mission Apollo Study, S. I. D. 63. 1370, dated November 1963, (C), North American Aviation, Inc.

2-3 Informal Weight Data on Apollo - Block 2 Vehicle, NASA, F. C. Strible, dated 3 August 1964

SECTION 3

EXPERIMENTS INTEGRATION

J. J. Fastiggi

SUMMARY

The Apollo/MOL/Saturn 1B concept appears to be roughly competitive with Gemini/MOL. The Apollo/MOL/Titan IIIC is feasible but clearly not competitive with Gemini/MOL. Depending on the configuration utilized, the MOL mission objectives may be met with from three to nine launches of the Saturn 1B. Some of the Apollo/MOL/Saturn 1B configurations are capable of carrying all the Air Force primary experiments and a few of these configurations are capable of carrying additional experiments. Crew duty cycles may pose problems when all experiments are to be performed on a two-man 30-day mission. However, the available data on experiment duty cycles was found to be incomplete at this time, thereby prohibiting a complete evaluation of this problem.

CONTENTS

3. EXPERIMENTS INTEGRATION

3.1 INTRODUCTION

It is the primary purpose of this study to investigate the feasibility of utilizing the Apollo command module, Apollo subsystems, and Apollo hardware as a vehicle system capable of performing the MOL mission objectives. To meet this end, from an experiment integration standpoint, the number of flights of a boost vehicle - MOL combination required to perform the mission objectives has been determined. The boost vehicles considered in this study were the Titan IIIC and Saturn 1B.

Since weight is probably the most significant parameter, experiments were allotted to each flight primarily on a weight basis. In all but a few cases, the laboratory and experiment configurations, which were generated to determine the minimum number of flights required, do not exceed the anticipated booster capability. Because of the limited time available and the preliminary design nature of the equipment, it was not possible to include all design parameters and possible configurations.

In addition to the weight problem, crew duty cycles (defined as the time required daily to perform experiments) must be considered. However, crew duty cycles are mainly supposition at this time and many experiments have yet to be completely defined. Therefore, only a cursory investigation of this problem has been performed.

It should be emphasized that the detailed data presented in Table 3-1 are based on preliminary and incomplete information; and the summaries presented in Table 3-6 are subject to the same limitations.

3.2 DISCUSSION

3.2.1 Configurations

The configurations used for this study are described in Section 1. Included in these are Apollo/MOL/Titan IIIC, Apollo/MOL NAA Concept II (minimum modification), Apollo/MOL NAA Concept II (modified), and Apollo/MOL Aerospace alternate configuration. All the basic configurations listed above utilize LiOH to remove CO_2, and fuel cells as a power source for a 30-day mission. Perturbations were introduced into the Apollo/MOL NAA Concept II (modified) configuration with the substitution of molecular sieves in lieu of the LiOH system, and solar cells in lieu of fuel cells.

3.2.2 Experiments

In order to generate a realistic comparison with the Gemini B/MOL system, only those experiments considered in previous integration studies have been included in this study. These include the following Air Force primary experiments:

P-1	Acquisition and Tracking of Ground Targets
P-2	[redacted]
P-3	Direct Viewing of Ground and Sea Targets
P-4	Electromagnetic Signal Detection
P-5	Eliminated
P-6	Extravehicular Activities
P-7	[redacted]
P-8	Autonomous Navigation
P-9	Deleted
P-10	Multiband Spectral Observation
P-11	General Human Performance in Space
P-12	Biomedical Experiments

No attempt has been made to integrate secondary experiments although some configurations provide weight allowances for additional experiments. Table 3-1 provides a complete component weight breakdown for all experiments. Data in Table 3-1 were compiled from previous Gemini B/MOL studies and Ref. 3-1.

3.2.3 Allocation Constraints and Ground Rules

All experiments and experimental components possess characteristics which influence the allocation method. These characteristics usually impose a certain amount of constraint on where, how, and when the experiment may be allocated. Constraints may be imposed by some or all of the following characteristics:

1. Weight
2. Size
3. Volume
4. Propulsion Requirements
5. Duty Cycle Requirements
6. Power Requirements
7. Environmental Criteria
8. Directional Constraints
9. Aperture Requirements
10. Mounting and Location Requirements
11. Heat Output
12. Stability Limits
13. Maintenance Requirements
14. Spare Part Requirements
15. Pyrotechnic Hazards

In addition to basing the experiment allocation on the above constraints, certain ground rules must be followed. These ground rules are determined by Air Force mission requirements and in some cases by common sense. They are as follows:

1. Each experiment is allocated to at least two flights in order to provide a backup experiment in the event of first flight failure.

2. Experiments P-11 and P-12 are to be allotted to all flights since they are concerned with human performance and as large a population of data points as possible should be examined in order to produce meaningful data.

3. Secondary experiments as such are not considered at this time due to payload restrictions.

4. Experiments utilizing common components are allocated to the same flight when possible.

5. On-orbit propulsion, which is required for Experiments P-2 and P-8, is provided when total payload weight permits.

6. The Apollo power system provides the basic power needs of the experiments.

7. The Apollo environmental control system is utilized to cool the experiments except on Experiment P-10 where a cryogenic system is provided.

8. The Apollo telemetry system is utilized as part of the experiment telemetry system.

9. The Apollo radio links are used by the experiments for oral communication with ground systems and the astronauts performing extravehicular activities.

10. For the Apollo/MOL/Titan IIIC system, all equipment which does not require access or a conditioned environment is located in the unpressurized service module.

11. The first flight in any flight series is reserved for systems checkout.

3.2.4 Experiment Allocation

Using the above constraints and ground rules, an experiment allocation was performed. All the ground rules were adhered to, but it was not considered feasible to attempt to satisfy all the listed constraints at this time. The first six constraints were considered as the most important with major emphasis placed on the first three. Except for the Apollo/Titan IIIC configuration, size and volume presented no problems. Weight is probably the most significant parameter is this study. The experiments are currently in a preliminary design or conceptual stage, thereby prohibiting the acquisition of good, useful data on the remaining constraints. Therefore, this experimental allocation study was primarily based on weight.

In order to perform an experiment allocation on a weight basis, the boost vehicle payload capability and the space station weight less experiments and experimental propulsion must be known. These weights are obtained in Section 2. With the weights known, it is a relatively simple, although tedious procedure to allocate experiments to flights. The experiment allocations for each configuration considered feasible are shown in Tables 3-2 through 3-5. When a particular configuration indicated that only the two biomedical and one other experiment may be flown, the configuration was considered impractical. Table 3-6 presents a complete allocation summary, condensing the information contained in Tables 3-2 through 3-5. The boost vehicle payload in-orbit capability is cross tabulated with the various concepts and the required number of flights indicated in their appropriate positions. This table presents a number of configurations which are impractical at the lower booster capability and a number of configurations which are capable of performing the mission with three flights. Configurations which would allow the inclusion of a third crew member are so noted in Table 3-5.

In the event the mission is planned for the minimum number of flights (three), the crew may be duty-cycle limited. Section 4 of this report indicates the two-man Apollo/MOL crew to have 11 man-hours per day in which to perform experiments. The times required to perform experiments were taken from Reference 3-1 and are tabulated in Table 3-7. Since the experiments are not all completely defined, the times referred to from Reference 3-1 are by no means exact, and the times obtained from Section 4 are also inexact. Therefore, no firm commitment can be stated regarding duty cycles. Based on available information, however, it appears that duty cycles are marginal for an all-experiment flight. No problems are expected on flights which do not carry the full complement of experiments or which provide three crew members.

3.3 CONCLUSIONS

It should be noted that this is a "first cut" at MOL experiments integration for the Apollo/MOL concept and has been performed for the purpose of determining the feasibility of such a system. However, based on the preliminary and conceptual data available, the following conclusions may be made:

1. The Saturn 1B/Apollo/MOL is a feasible system with the present-day anticipated boost vehicle payload capability.

2. For a 30-day duration flight, the MOL mission objectives may be realized with half as many flights utilizing the Saturn 1B/Apollo/MOL concept as with the Gemini B/MOL/Titan IIIC system. Thus, Apollo/MOL appears roughly competitive with Gemini/MOL.

3. The Apollo/MOL/Titan IIIC is limited in both payload weight and volume to perform the MOL experiments. A minimum of nine flights are required to meet the MOL objectives of performing all experiments at least twice.

4. Two- and three-man flights of 120-day durations are not feasible for this program unless molecular sieves are employed in the life support system and solar cells or [redacted] are utilized in the power system.

3.4 RECOMMENDATION

A complete integration study should include the effects of the reliability and the cost implications. It is therefore recommended that the data presented herein be weighted by the appropriate reliability and cost parameters to provide a more meaningful comparison to Gemini B/MOL/Titan IIIC.

Since the data utilized in this study has been based on concepts and preliminary experiment data, an upgraded integration study should be performed when more complete data becomes available. This would allow the remaining constraints to be included in the study.

[redacted]

Table 3-1. MOL Primary Experiments Equipment List.
(Based on Preliminary and Incomplete Data)

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-1 (Acquisition and Tracking of Ground Targets)							[redacted]					[redacted]					
Pointing & Tracking Scope (includes servos & camera)	P		12.0	375	330	X	[redacted]	X				[redacted]	X		X		
Computer	P	(a)	(a)	(a)	100	X	[redacted]	X	X			[redacted]	X				X
Film Processing Unit	P	4 x 12 x 36	1.0	75	80	X	[redacted]	X				[redacted]	X				
Film & Cartridges	P			10		X	[redacted]	X				[redacted]					
Development web	P			15		X	[redacted]	X				[redacted]					
Panels & Controls	P			20		X	[redacted]	X				[redacted]					
Pressure Seals	P, U			15		X	[redacted]	X				[redacted]					
Wiring & Circuitry	P, U			20		X	[redacted]	X				[redacted]					
Mounts & Brackets	P, U			50		X	[redacted]	X				[redacted]					
Radio Transmitter and Receiver	P			(b)		X	[redacted]	X				[redacted]					
Cold Plates, and Miscellaneous	P, U			50 / 630	30	X	[redacted]	X				[redacted]					
[redacted]																	
Experiment P-3 (Direct Viewing of Ground and Sea Targets)							[redacted]					[redacted]					
P-1 Equipment	P	(c)	(c)	(c)	280	X	[redacted]	X				[redacted]					
Recorder	P	(a)	(a)	(a) / 630	50		[redacted]	X			X	[redacted]	X		X	X	X

(a) These items are shared with many other experiments, therefore, their weight and volume are listed under general provisions.

(b) These items are basic MOL units.

(c) These items are shared with experiments which have been charged for their weight.

In the event Experiment P-10 is not flown on the same flight as Experiment P-8, an additional 85 pounds must be assessed to P-10 for star trackers.

SYMBOLS

P - denotes location in pressurized section.

U - denotes location in unpressurized section.

X - denotes equipment utilized on this experiment.

[redacted]

Table 3-1. (Continued).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-4 (Electromagnetic Signal Detection)																	
Computer	P	(a)	(a)	(a)	100	X	[redacted]	X	X			[redacted]	X				X
Receiver A	P	15 x 15 x 15	2	50	100				X								
Receiver B	P	16 x 16 x 16	2.2	60	100				X								
Antenna Array	U	1 x 36 x 53		45					X								
Antenna Array	U			35					X								
Signal Processor	P	18 x 18 x 18	4	125	200				X								
Recorder and Tape Playback	P			100	50				X								
Recorder Tapes and Reels	P			50					X								
Inflatable Antenna and Control Unit	U	60 dia. x 18		50	200				X								
Control Panel and Display	P			75	150				X								
Wiring and Circuitry	P, U			45					X								
Brackets and Mounts	P, U			60					X								
Cold Plates and PCU	P, U			40					X								
				735													
Experiment P-5	Deleted																
Experiment P-6 (Extra-Vehicular Activities)																	
Recorder	P	(a)	(a)	(a)	50			X			X		X		X	X	X
Movie Camera	P	(a)	(a)	(a)	100						X				X	X	
Still Camera	P	(a)	(a)	(a)	100						X				X	X	X
Film, Reels and Storage	P			20							X						
Net Box	P	6 x 6 x 12	.25	5							X						
Tools and Box	P	6 x 12 x 24	1.0	20							X						

[redacted]

[redacted]

Table 3-1. (Continued).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	EXPERIMENTS P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-6 (Cont'd)							[redacted]					[redacted]					
Astronaut Maneuvering Unit	P	12 x 30 x 36	7.5	200							X						
A.M.U. Control	P	3 x 6 x 6	.06	5							X						
A.M.U. Refurbishment	P, U	12 x 30 x 36	7.5	200							X						
Attachment Device Box	P, U	6 x 6 x 12	.25	10							X						
Task Panel	P	6 x 24 x 24	1.00	5							X						
ECS Hose	P, U	3 x 24 x 24	1.00	10							X						
Hardline	P, U	3 x 6 x 6	.06	10							X						
Physiological Measuring Recorder	P	12 x 12 x 24	2.0	50							X						
Tetherline	U	4 x 12 x 12	.33	10							X						
Periscope	P	3 dia. x 42	.17	5							X						
Handholds	U			5							X						
T.V. System	U	12 x 12 x 12	1.0	45							X						
Lights	P, U			5	200						X						
Recording Tapes and Reels	P			10							X						
Mounts and Brackets	P, U			60							X						
Wiring and Circuitry	P, U			15							X						
Cold Plates & Misc.	P, U			30							X						
				720													

[redacted]

[redacted]

UNCLASSIFIED

[redacted]

Table 3-1. (Continued).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	EXPERIMENTS P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
[redacted]							[redacted]					[redacted]					
Experiment P-8 (Autonomous Navigation)							[redacted]					[redacted]					
Horizon Scanner	P	8 x 8 x 8	.3	25	15		[redacted]					[redacted]	X				
Geodetic Camera	P	5 x 8 x 14	.3	20	2		[redacted]					[redacted]	X				
Geodetic Ranging Capacitor	P	8 x 8 x 8	.3	10	5		[redacted]					[redacted]	X				
Geodetic Ranging Laser	P	5 x 5 x 6	.09	5			[redacted]					[redacted]	X				
Inertial Measuring Unit	P	14 x 14 x 14	1.5	110	240		[redacted]					[redacted]	X				
Star Theodolite	P	8 x 8 x 8	.5	10			[redacted]					[redacted]	X				
Electronic Assembly	P, U	3 x 10 x 12		20			[redacted]					[redacted]	X				
Star Trackers (2)	P			85	26		[redacted]					[redacted]	X		X		
Display and Control Panel	P			20	25		[redacted]					[redacted]	X				
Pointing and Tracking Scope	P	(c)	(c)	(c)	250	X	[redacted]	X				[redacted]	X		X		
Computer	P	(a)	(a)	(a)	200	X	[redacted]	X	X			[redacted]	X				X
Data Recording Tapes	P			10			[redacted]					[redacted]	X				
Recorder	P	(a)	(a)	(a)	50		[redacted]	X			X	[redacted]	X		X	X	X

UNCLASSIFIED

[redacted]

Table 3-1. (Continued).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-8 (Cont'd)							[redacted]					[redacted]					
Mounts & Brackets	P, U			30									X				
Wiring and Circuitry	P, U			30									X				
Cold Plates & Misc.	P, U			25									X				
				400													
Experiment P-9 (Deleted																	
Experiment P-10 (Multiband Spectral Observation)																	
Optical Unit	P			65											X		
Pointing and Tracking Scope	P	(c)	(c)	(c)	550	X		X					X		X		
Star Tracker	P			(c)									X		X		
Spectral Radiometer	P			10											X		
Spatial Radiometer	P		12	20											X		
U.V. Radiometer	P			15											X		
Cryogenic System	U	20 Dia.	2.0	100											X		
Recorder	P	(a)	(a)	(a)				X			X		X		X	X	X
Movie Camera	P	(a)	(a)	(a)							X				X	X	
Still Camera	P	(a)	(a)	(a)							X				X	X	X
Display and Control Panel	P			35											X		
Pressure Structure	P, U			50											X		
Tapes and Film	P			10											X		
Mounts and Brackets	P, U			30											X		
Wiring and Circuitry	P, U			20											X		
Cold Plates and Misc.	P, U			30											X		
				385													

Table 3-1. (Continued).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-11 (General Human Performance in Space)							[redacted]					[redacted]					
Recorder	P	(a)	(a)	(a)	50			X			X		X		X	X	X
Movie Camera	P	(a)	(a)	(a)							X				X	X	
Still Camera	P	(a)	(a)	(a)							X				X	X	X
Tapes and Film	P			10												X	
Test Unit and Control Stick	P	3 x 8 x 15 2 x 2 x 5	.2	15	20											X	
Wiring and Mounts	P			10 35												X	
Experiment P-12 (Biomedical Experiment)																	
Computer	P			(a)		X		X	X				X				X
Recorder	P			(a)	50			X			X		X		X	X	X
Still Camera	P			(a)							X				X	X	X
Heart Rate Monitors	P			15													X
Blood Pressure Equip.	P			10													X
Blood O_2 and Cardiac Output	P			5													X
Spirometer and PaO_2, $PaCO_2$ Detector	P			20													X
Gas Meter	P		19 cu.ft. Total	10													X
EEG, EOG and Tape Recorder	P			30													X
Lyophilizer	P			20													X
Biochemical Analysis Unit	P			30	30												X
Body Mass Measuring Device	P			20													X
Exercise Equipment	P			40													X
Urine Collector and Volume Measurement	P			10													X

[redacted]

Table 3-1. (Concluded).

COMPONENT	LOCATION	SIZE INCHES	VOLUME CU.FT.	WEIGHT POUNDS	POWER WATTS	P-1	P-2	P-3	P-4	P-5	P-6	P-7	P-8	P-9	P-10	P-11	P-12
Experiment P-12 (Cont'd)							[redacted]					[redacted]					
Centrifuge	P		19 cu.ft. Total	15	50												X
Mass Spectrometer	P			30													X
Misc. Exp. Equipment	P			30													X
Spare Part and Tool	P			50													X
Tapes and Reels	P			10													X
Mounts and Bracketry	P			35													X
Wiring and Circuitry	P			20													
				400													
General Provisions																	
Computer	P	12 x 12 x 18	1.5	125		X		X	X				X				X
Recorder	P	12 x 12 x 24	2.0	50				X			X		X		X	X	X
Targets (2)	U	6 x 12 x 24	1.0	80													
Ejectors	U	6 x 12 x 4	.17	20													
Movie Cameras	P	3 x 3 x 6	.06	10							X				X	X	
Still Camera	P	3 x 3 x 6	.06	5							X				X	X	X
Telemetry and Data Cond.	P, U			75		X		X	X		X		X		X	X	X
Airlock Repressurization	P, U			70		X		X	X		X		X		X	X	X
Repressurizing Gas Tankage	U			20		X		X	X		X		X		X	X	X
				455													
Experiments Weight (Less Propulsion)				4,395													

[redacted]

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[redacted]

Table 3-2. Apollo/MOL/Titan IIIC Flight Allocation.

BOOST VEHICLE PAYLOAD CAPABILITY	20,700																	
EXPERIMENT CAPABILITY	1,230 Command Module (C.M.) + 1,050 Service Module (S.M.) = 2,280																	
	FLIGHTS																	
	1		2		3		4		5		6		7		8		9	
Experiment	CM	SM	CM	SM	CM	SM	CM	SM	CM	SM	CM	SM	CM	SM	CM	SM	CM	SM
P-1			-	-	630	-	630	-	-	-	-	-	630	-	630	-	-	-
P-2	[redacted]																	
P-3			-	-	(a)	-	(a)	-	-	-	-	-	(a)	-	(a)	-	-	-
P-4			-	-	-	-	-	-	370	365	-	-	-	-	-	-	370	365
P-5			-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
P-6		out	520	200	-	-	-	-	-	-	520	200	-	-	-	-	-	-
P-7	[redacted]																	
P-8			-	-	100	220	-	-	-	-	-	-	100	220	-	-	-	-
P-9		System	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
P-10			-	-	-	-	175	210	-	-	-	-	-	-	175	210	-	-
P-11			35	-	35	-	35	-	35	-	35	-	35	-	35	-	35	-
P-12			400	-	400	-	400	-	400	-	400	-	400	-	400	-	400	-
General Provisions			65	390	65	390	65	390	65	390	65	390	65	390	65	390	65	390
			1020	590	1230	610	1305	600	1105	1150	1020	590	1230	610	1305	600	1105	1150
In Orbit Propulsion						440		375		-				440		375		-
TOTAL			1610		2280		2280		2255		1610		2280		2280		2255	

(a) Equipment weight for these experiments is charged to Experiment P-1.

[redacted]

Table 3-3. Flight Allocation.

CONFIGURATION	APOLLO CONCEPT II	MODIFIED APOLLO LiOH AND FUEL CELLS 2 Men 30 Days			MODIFIED APOLLO LiOH AND FUEL CELLS 2 Men 60 Days						MODIFIED APOLLO LiOH AND FUEL CELLS 2 Men 120 Days
BOOST VEHICLE PAYLOAD CAPABILITY	33,460	33,460			33,460						33,460
EXPERIMENT CAPABILITY	180	5,860			2,485						-1,000
EXPERIMENT PROPULSION	780	780			780						780
TOTAL	960	6,640			3,265						- 220
	FLIGHTS	FLIGHTS			FLIGHTS						FLIGHTS
	1	1	2	3	1	2	3	4	5	6	
Experiment	No flights feasible with this configuration.	System Ch[redacted]t			System Ch[redacted]t						No flights feasible with this configuration
P-1			630	630		630	630	630	630	630	
P-2		[redacted]									
P-3			(a)	(a)		(a)	(a)	(a)	(a)	(a)	
P-4			735	735		-	-	735	-	735	
P-5			-	-		-	-	-	-	-	
P-6			720	720		720	-	-	-	720	
P-7		[redacted]									
P-8			400	400		400	-	400	-	-	
P-9			-	-		-	-	-	-	-	
P-10			385	385		-	385	-	385	-	
P-11			35	35		35	35	35	35	35	
P-12			400	400		400	400	400	400	400	
General Provisions			455	455		455	455	455	455	455	
TOTAL			4,395	4,395		2,640	2,540	2,650	2,540	2,970	
Experiment Propulsion			780	780		625	725	615	725	295	
TOTAL			5,175	5,175		3,265	3,265	3,265	3,265	3,265	

(a) Equipment weight for these experiments is charged to Experiment P-1.

Table 3-4. Flight Allocation.

CONFIGURATION	MODIFIED APOLLO MOLECULAR SIEVES AND SOLAR CELLS (b) 2 Men 30 Days			MODIFIED APOLLO MOLECULAR SIEVES AND SOLAR CELLS (b) 2 Men 60 Days			MODIFIED APOLLO MOLECULAR SIEVES AND SOLAR CELLS (b) 2 Men 120 Days									MODIFIED APOLLO MOLECULAR SIEVES AND SOLAR CELLS (b) 3 Men 120 Days
BOOST VEHICLE PAYLOAD CAPABILITY	33,460			33,460			33,460									33,460
EXPERIMENT CAPABILITY	6,255			4,705			1,000									670
EXPERIMENT PROPULSION	780			780			780									780
TOTAL	7,035			5,485			2,580									1,450
	FLIGHTS			FLIGHTS			FLIGHTS									FLIGHTS
	1	2	3	1	2	3	1	2	3	4	5	6	7	8	9	
Experiment	System [redacted]out			System [redacted]out			System [redacted]out									No flights feasible with this configuration
P-1		630	630		630	630		630	630	630	630	630	630	630	630	
P-2		[redacted]														
P-3		(a)	(a)		(a)	(a)		(a)	(a)	(a)	(a)	(a)	(a)	(a)	(a)	
P-4		735	735		735	735		-	735	-	-	-	735	-	-	
P-5		-	-		-	-		-	-	-	-	-	-	-	-	
P-6		720	720		720	720		720	-	-	-	720	-	-	-	
P-7		[redacted]														
P-8		400	400		400	400		-	-	400	-	-	-	400	-	
P-9		-	-		-	-		-	-	-	-	-	-	-	-	
P-10		385	385		385	385		-	-	385	-	-	-	385	-	
P-11		35	35		35	35		35	35	35	35	35	35	35	35	
P-12		400	400		400	400		400	400	400	400	400	400	400	400	
General Provisions		455	455		455	455		455	455	455	455	455	455	455	455	
TOTAL		4,395	4,385		4,395	4,395		2,240	2,255	2,305	2,155	2,240	2,255	2,305	2,155	
Experiment Propulsion		780	780		780	780		340	325	275	425	340	325	275	425	
TOTAL		5,175	5,175		5,175	5,175		2,580	2,580	2,580	2,580	2,580	2,580	2,580	2,580	

(a) Equipment weight for these experiments is charged to Experiment P-1.

[redacted]

Table 3-5. Flight Allocation.

CONFIGURATION	AEROSPACE ALTERNATE LiOH AND FUEL CELLS 2 Men 30 Days			AEROSPACE ALTERNATE LiOH AND FUEL CELLS 2 Men 60 Days						AEROSPACE ALTERNATE LiOH AND FUEL CELLS 2 Men 120 Days
BOOST VEHICLE PAYLOAD CAPABILITY	33,460			33,460						33,460
EXPERIMENT CAPABILITY	5,460			2,085						-1,350
EXPERIMENT PROPULSION	780			780						780
TOTAL	6,240			2,865						570
	FLIGHTS			FLIGHTS						FLIGHTS
	1	2	3	1	2	3	4	5	6	
Experiment										
P-1		630	630		630	630	630	630	630	
P-2	[redacted]									
P-3		(a)	(a)		(a)	(a)	(a)	(a)	(a)	
P-4	Sys[redacted]Checkout	735	735	Sys[redacted]Checkout	-	-	735	-	735	
P-5		-	-		-	-	-	-	-	No flights feasible with this configuration
P-6		720	720		-	720	-	-	720	
P-7	[redacted]									
P-8		400	400		400	-	400	-	-	
P-9		-	-		-	-	-	-	-	
P-10		385	385		385	-	-	385	-	
P-11		35	35		35	35	35	35	35	
P-12		400	400		400	400	400	400	400	
General Provisions		455	455		455	455	455	455	455	
TOTAL		4,395	4,395		2,305	2,875	2,655	2,540	2,975	
Experiment Propulsion		780	780		560	-	210	325	-	
TOTAL		5,175	5,175		2,865	2,875	2,865	2,865	2,975	

(a) Equipment weight for these experiments is charged to Experiment P-1.

[redacted]

Table 3-6. Flight Allocation Summary. (Based on Preliminary and Incomplete Information)

GEMINI MOL	APOLLO MOL	APOLLO MOL	MODIFIED APOLLO							AEROSPACE ALTERNATE		
T-IIIC	T-IIIC	CONCEPT II	LiOH and Fuel Cells			Molecular Sieve & Solar Cells				LiOH and Fuel Cells		
			2 Men 30 Days	2 Men 60 Days	2 Men 120 Days	2 Men 30 Days	2 Men 60 Days	2 Men 120 Days	3 Men 120 Days	2 Men 30 Days	2 Men 60 Days	2 Men 120 Days
Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights	Flights
6	9	(a)	3(c)(d)	6	(a)	3(c)(d)	3(b)(c)	9	(a)	3(c)(d)	6	(a)

(a) Gross payload in orbit requirements prohibit experiments on these flights.
(b) Experiments may be duty cycle limited on these flights.
(c) Additional experiments could be carried on these flights.
(d) An additional crew member could be carried on these flights.

[redacted]

TABLE 3.7

PROPOSED DAILY TIME ALLOCATION FOR EXPERIMENTS

(Based on an approximate 11 hour duty cycle)

Experiment	Time Req'd. to Conduct Experiment	Frequency of Experiments for 30 Day Mission	Day Allocated for Performance of Experiments During a 30 Day Mission				
			1-20	21-23	24&25	26-28	29&30
P-1	0:50	20 Times	0:50	-	-	-	-
P-2	[redacted]						
P-3	0:30	20 Times	0:30	-	-	-	-
P-4	0:30	50 Times	1:00 (a)	-	-	1:00 (a)	1:00 (a)
P-5	-	-	-	-	-	-	-
P-6	3:00	6 Times	-	3:00	-	3:00	-
P-7	[redacted]						
P-8	1:35	Daily	1:35	1:35	1:35	1:35	1:35
P-9	-	-	-	-	-	-	-
P-10	2:00	Daily	2:00	2:00	2:00	2:00	2:00
P-11	1:00	Daily	1:00	1:00	1:00	1:00	1:00
P-12	3:10	Daily	3:10	3:10	3:10	3:10	3:10
		TOTAL	10:05	11:35	10:50	11:45	11:00

All times are in hours and minutes

(a) This experiment is performed twice daily.

REFERENCES

3-1. "MOL Primary Experiments Data Book," TOR-269(4107-35)-1, Aerospace Corporation, El Segundo, California, 30 June 1964.

SECTION 4

CREW ACTIVITY TIME ALLOCATIONS

E. F. Schmidt
F. E. Cook

SUMMARY

The time available for the performance of experiments is presented for two-, three-, and four-man crews. These estimates are based largely on a comparative evaluation of the results of several previous industry and Aerospace studies of crew time allocations for various manned systems. It is found that the time available for experiments is 11 hours for both the two-man Gemini/MOL or Apollo/MOL.

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TABLES

4. CREW ACTIVITY TIME ALLOCATIONS

4.1 INTRODUCTION

A brief investigation was made to determine reasonable time allocations for the various duties of two-, three, and four-man crews for Apollo/MOL. The primary objective was to determine the time available for the performance of experiments. The approach taken was to review the results of the various studies already performed in this area (References 4-1 and 4-2 and previous Aerospace studies) and to attempt to resolve the differences in the time allocated to various crew duties, where differences existed. Allocations were then established for Apollo/MOL, primarily on the basis of the existing information that was judged to be the best supported.

4.2 DISCUSSION

Table 4-1 is a tabulation of the information extracted from the various references. An attempt was made to break down the time allocations to a common level of detail, but this was not possible in all cases. The XMAS allocations for exercise and recreation were combined, but the total is in good agreement with all of the others excepting the MORL figure. Although the latter is comparatively large, a degree of confidence is inspired by the fact that Douglas appears to have performed more detailed study and experiments in this area than have the others. However, confidence in any figure can only be expressed relatively until actual flight data is obtained. The allocation for exercise and recreation for Apollo/MOL of six hours might be considered to represent a weighted average of the others, with more confidence having been attached to a single high allocation of MORL than to the several low values for the other systems.

Large discrepancies appear in the values assigned by the various studies to station-keeping and maintenance. However, the functions included in these categories differ from one system to another. For example, the MOL figures for station-keeping include monitoring time, whereas for MORL monitoring is considered a part of maintenance. The differences between the total times for the two categories are not large except for the XMAS values. The latter appear to be unreasonably low, and the allocations assigned to Apollo/MOL reflect confidence in the higher time estimates. A more complete breakdown of the latter, given in Table 4-2, supports the use of the larger number. The time available for experiments for the various systems is quite consistent, except for the XMAS values. This discrepancy may be attributed primarily to the low figure used for station-keeping and maintenance, as discussed above.

The column headed "Gemini B/MOL" represents the values used by the MOL Program Office in a recent briefing.

Since both the MORL and XMAS concepts are oriented primarily toward biomedical and behavioral experiments, an attempt was made to allocate the Apollo/MOL time for these experiments (P-11 and P-12) separately. The allocations given are based on recent discussions with the MOL Program Office. The time allocated to P-11 of one man-hour per day is an average, and represents the performance of two one-half hour experiments every other day by each astronaut. The P-12 allocation is also an average value based on conversations with Air Force Aeromedical Division personnel. Each of the tests included in P-12 is not performed every day, nor are both astronauts sampled each day. The daily time required will, therefore, vary.

Table 4-2 tabulates the time allocations for two-, three-, and four-man crews. The task breakdown is detailed to the level required to determine the allocations which increase with crew size. The estimated time requirements for biomedical and behavioral experiments for the three- and four-man crews were based on discussions with the MOL Program Office. An increase in time available for other experiments of nine man-hours for each additional crew member over the initial two is realized.

Table 4-3 summarizes the Apollo/MOL two- and three-man work loads, and compares the allocations for various categories with those given by the Gemini/MOL Program Office.

Table 4-1. Two-Man Crew Work Load.*

Task	Minimol	Apollo (NAA-XMAS) Concept 2 SP-1**	Apollo (NAA-XMAS) Concept 1 SP-2**	MORL	Acc. MOL	Gemini B/ MOL	Apollo/MOL
Sleep	16:00	16:00	16:00	16:00	16:00	16:00	16:00
Eat	2:10	2:00	2:00	3:36	2:20	2:00	2:00
Personnel Hygiene	2:30	2:00	2:00	1:48	2:30	2:00	2:00
Exercise	2:30			5:00	2:30	2:00	3:00
Recreation	2:00	4:00	4:00	3:00	2:00	3:00	3:00
Station Keeping	9:40	3:42	3:42	5:00	8:40	4:00	5:30
Maintenance	1:20	1:00	1:00	4:24	4:00	4:00	5:30
Experiments							
Biomed & Behavioral	{11:50	7:24	13:12	7:12	{10:00	{11:00	4:10
Other		11:54	6:06	2:00			6:50
Miscellaneous	--	--	--	--	--	4:00	--
TOTAL	48:00	48:00	48:00	48:00	48:00	48:00	48:00

*Time is expressed in hours and minutes.

**Safety package (Biomedical Experiments)

[redacted]

Table 4-2. Apollo/MOL Crew Work Load.*

Task	2 Astronauts	3 Astronauts	4 Astronauts
Sleep	16:00	24:00	32:00
Eat	2:00	3:00	4:00
Personnel Hygiene	2:00	3:00	4:00
Exercise	3:00	4:30	6:00
Recreation	3:00	4:30	6:00
Station Keeping	5:30	6:00	6:30
Data Management	(2:40)	(3:00)	(3:20)
Station Management	(1:50)	(2:00)	(2:10)
Orbit Keeping & Navigation	(1:00)	(1:00)	(1:00)
Maintenance	5:30	5:30	5:30
Preventive & Repair	(1:30)	(1:30)	(1:30)
Checkout & Monitoring	(4:00)	(4:00)	(4:00)
Experiments	11:00	21:30	32:00
Biomedical	(4:10)	(5:40)	(7:10)
P-11	(1:00)	(1:30)	(2:00)
P-12	(3:10)	(4:10)	(5:10)
Other	(6:50)	(15:50)	(24:50)
TOTAL	48:00	72:00	96:00

*All time is expressed in hours and minutes.

[redacted]

Table 4-3. Crew Work Load.*

Task	Gemini/MOL 2 Man	Apollo/MOL 2 Man	Apollo/MOL 3 Man
Sleep	16:00	16:00	24:00
Eat	2:00	2:00	3:00
Personnel Hygiene	2:00	2:00	3:00
Exercise	2:00	3:00	4:30
Recreation	3:00	3:00	4:30
Station Keeping	4:00	5:30	6:00
Maintenance	4:00	5:30	5:30
Experiments			
Biomedical/Behavioral	} 11:00	4:10	5:40
Other		6:50	15:50
Miscellaneous	4:00	--	--
TOTAL	48:00	48:00	72:00

*All time is expressed in hours and minutes.

4.3 CONCLUSIONS

The time available for experiments in an Apollo/MOL mission with a two-man crew appears to be approximately the same as that for Gemini/MOL. If a third man is added, the available time appears to be almost doubled.

[redacted]

REFERENCES

4-1 "Report on a System Comparison and Selection Study of a Manned Orbital Research Laboratory," Volume I, Appendix 3, Douglas Missile and Space Systems Division Document SM-44599

4-2 North American Aviation SID 63-1370-1, "Extended-Mission Apollo Study," Final Report, Volume 1, 24 November 1963

[redacted]

SECTION 5.

POWER SYSTEMS

H. J. Killian

SUMMARY

A study of electrical power systems for an Apollo/MOL spacecraft has been performed and is reported herein. Objectives of the study were to (1) define an electrical power system for Apollo/MOL making maximum use of the present Apollo fuel cell hardware, (2) select and define the best electrical subsystem for Apollo/MOL on the basis of comparative evaluations of candidate power system concepts, (3) evaluate the growth potential of an Apollo/MOL electrical power subsystem for extended mission durations, and (4) compare the best Apollo/MOL electrical power subsystem with that recommended for (Gemini) MOL in previous studies. Apollo fuel cells, current technology fuel cells, solar cells and [redacted] were evaluated as candidate power systems on the basis of a set of power system requirements formulated as a part of this study. It was concluded that the current technology fuel cells were preferable for the early (two-year subsystem availability) missions even though their mission duration capabilities were limited to 30 to 40 days. Follow-on missions could be extended in duration to 120 days by incorporation of [redacted] Solar cells appeared to be generally unsatisfactory for the Apollo/MOL mission. Power system characteristics for the five Apollo/MOL configurations of interest were defined assuming use of the recommended power system(s).

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ILLUSTRATIONS

5.1 INTRODUCTION

5.1.1 Candidate Power Systems

Much of the prior work done at Aerospace in connection with Gemini/MOL was directly applicable to the analysis of power systems for Apollo/MOL. In particular, the same basic power system concepts under consideration for Gemini/MOL were competitive for application to Apollo/MOL; viz., fuel cells, solar cells, and [redacted] power systems. Solar dynamic power systems were not evaluated since they are not current state-of-the-art and are not expected to be operational in this decade. Each of the three power system concepts evaluated has special advantages and shortcomings. A fuel cell system has a low fixed weight, but consumes stored reactants and therefore becomes quite heavy for longer mission durations. Solar cell systems have a high fixed weight, but this weight remains substantially constant for longer duration high altitude missions. [redacted] This left fuel cells and solar cells as the primary candidates which received major emphasis for the Apollo/MOL power system study.

5.1.2 Scope of Comparison Effort

It was evident from the onset of the power subsystem study that the Apollo fuel cell power subsystem would require substantial reconfiguration to adapt it to Apollo/MOL needs and that solar cells were very competitive with fuel cells for this mission. Consequently, it was realized that technical justification for a selection of one in preference to the other would involve extensive analysis of both power systems. For unmanned spacecraft above a couple of hundred nautical miles altitude, the fuel cell, because of its continuous reactant consumption, is not weight competitive with solar cells beyond about two weeks. However, for manned spacecraft, a fuel cell system may integrate well with the life support system (thereby saving weight), whereas no integration is possible with a solar cell system. Also, for orbit altitudes below a couple hundred nautical miles, aerodynamic drag on solar cell panels creates propulsion weight penalties in favor of the fuel cell system which does not interface with propulsion. The net effect is that the MOL mission is in an area of overlapping applicability of both fuel cells and solar cells and analyses to select between the two must consider the sometimes subtle involvement of many other subsystem areas.

5.1.3 Objectives

Specific objectives of the electrical power subsystem study were as follows:

1. Define an electrical power system for Apollo/MOL making maximum use of the present Apollo fuel cell hardware.

2. Select and define the best electrical power subsystem(s) for Apollo/MOL on the basis of comparative evaluations of candidate power system concepts.

3. Evaluate the growth potential of an Apollo/MOL electrical power subsystem for extended mission durations.

4. Compare the best Apollo/MOL electrical power subsystem with that recommended for Gemini/MOL in previous studies.

Design requirements which formed a base for the power system analyses are presented below followed by the definition of a minimum modification Apollo fuel cell system for Apollo/MOL. A subsequent comparison of all promising power system concepts for Apollo/MOL leads to recommended power systems for the several Apollo/MOL system configurations of interest. These results are compared to characteristics and capabilities of the (previously recommended) Gemini/MOL electrical power subsystem, and over-all study conclusions are drawn.

5.2 DESIGN REQUIREMENTS

5.2.1 Electrical Power Requirements

Estimated Apollo/MOL electrical power requirements are shown in Table 5-1. These estimates are based on the continuous utilization of Apollo and all of its subsystems which would be retained for the Apollo/MOL mission. In addition, the continuous maintenance of a "shirt sleeve" environment in an attached laboratory module for the housing and conduct of experiments was assumed. Experiment power is based on estimates made in connection with Gemini/MOL of the average power required for all primary experiments. These power estimates typify expected requirements with either a two- or three-man crew.

Table 5-1 was prepared before the final results from other subsystem studies were available. Consequently, the subsystem power requirements indicated may differ slightly from those finally determined in the subsystem studies. The total average power requirements shown, however, are consistent with similar data from numerous Aerospace Corporation and industry studies of comparable missions. They will therefore result in valid power system analyses pertaining to Apollo/MOL itself and, it is believed, in valid power system comparisons with (Gemini) MOL or other mission concepts.

Table 5-1. Apollo/MOL Estimated Power Requirements.

	Watts Average	Watts Peak
Apollo:		
Stabilization and control	200	210
Environmental control and life support	380	420
Guidance and navigation	50	500
Control and displays	170	270
Illumination	80	100
Commun. and instru. (including T/M)	190	490
Subtotal	1070	1990

	Watts Average	Watts Peak
Laboratory Module:		
Environmental control and life support	300	500
Illumination	100	300
Experiments	300	1000
Subtotal	700	1800
Total	1770	3790

5.2.2 Mission Duration

Mission duration was a key design parameter covering the range 30 to 120 days.

5.2.3 Orbit

Circular orbit altitudes between 100 and 200 nautical miles were considered. An easterly launch from AMR (32.5-degree orbit inclination) was assumed where necessary for power system definition or optimization. It resulted, however, that the power systems defined in the study were adequate for any orbit inclination.

5.2.4 Orientation

An earth-stabilized spacecraft (roll axis aligned with velocity vector and yaw axis aligned withlocal vertical) was assumed to exist throughout the mission.

5.2.5 Life Support Integration

A fuel cell system would supply more than enough water for crew needs as a natural by-product of its power generation. Complete water recovery was assumed in connection with solar cell and [redacted] power systems. However, the weight and power penalties which would be associated with a recoverable water supply were not assessed against the power systems. Therefore, in the power supply comparisons presented in this report, fuel cells do not receive credit for the advantage they offer in connection with water supply. Past studies have shown that this credit would amount to an equivalent weight of 100 to 150 pounds. On the other hand, no penalties were assessed against fuel cells for the much greater thermal load they present to the spacecraft cooling system. Radiator, plumbing and control hardware weights associated with fuel cell cooling would exceed 50 pounds. These two counteracting omissions are both small enough so as to not have significant quantitative impact on the power system comparisons presented herein.

5.2.6 Reliability

A 30-day reliability goal of 0.9995 was assumed. This number was based on a preliminary subsystem reliability allocation made early in the (Gemini)MOL program. It was used to establish levels of redundancy for various power system elements so as to better definitize power system weight and volume characteristics and to provide a common reliability base for comparison of the candidate power system concepts.

5.2.7 Auxiliary Power

Power for Apollo during the launch and pre-station activation period would be supplied from the MOL. Also, an emergency power capability corresponding to full power (without experiments, 1.5 kilowatts) for 1.5 hours or reduced power for a longer period was assumed as a requirement for the over-all Apollo/MOL spacecraft. In addition to this emergency power, an emergency capability would be provided to obtain power in the Apollo command module only for an extended period after abandonment of the MOL and during re-entry. An emergency is defined as a total loss of power from the regular power source.

5.2.8 Availability

Man-rated and flight-ready hardware within two years from go-ahead was assumed as a requirement for the initial development program consistent with (Gemini) MOL planning. This limits design technology to essentially current state-of-the-art for at least the 30-day (initial) MOL.

[redacted]

5.3 APOLLO FUEL CELL POWER SYSTEM

The Apollo fuel cell qualified for special consideration as a power source in Apollo/MOL because of the study objective to make maximum use of existing hardware. It was decided that qualification under this objective limited such special consideration to "essential modification" versions of the Apollo fuel cell. Assumptions of repackaging, life extension or other desirable but not essential developments in connection with the Apollo fuel cell would render it no more extant than other fuel cell concepts which are important competitors. No major performance extensions, then, beyond what is presently specified for the Apollo program, were permitted the Apollo fuel cell in this "existing hardware" category.

5.3.1 Basic Fuel Cell Module Configuration

An Apollo fuel cell module can continuously deliver power at any level within a large range: approximately 560 watts to almost 2000 watts. In view of the 1.77 kilowatts of average power required by Apollo/MOL, it must be determined whether one, two, or three modules should be operated simultaneously to deliver this much power.

A major factor that influences modules requirements is the 15-day life of Apollo fuel cells as compared to the 30-day and longer Apollo/MOL missions being considered. Obviously, module wearout and replacement with spare modules must form the basis for module requirements.

Selection of a module configuration was based on consideration of the following constraints:

1. Delivery of 1.77 kilowatts average power while keeping the primary DC bus voltage within a four- to five-volt range.
2. Minimize over-all weight.
3. Achieve a 30-day reliability of approximately 0.9999 for the fuel cells.

[redacted]

A secondary goal was to maximize peak power capability although this aspect of power delivery can be conveniently handled by secondary (rechargeable) batteries fed from the fuel cell(s).

Figure 5-1 shows how the Apollo fuel cell module operating temperatures are constrained (in Apollo) by voltage tolerances and other constraints. Clearly, the generation of higher power levels requires higher fuel cell (electrode-electrolyte) temperatures to stay with the voltage limits. However, module life is almost solely dependent on temperature with the result that it varies with module power output (in Apollo) as shown in Figure 5-2. Assuming that the Apollo fuel cell meets qualification test requirements, a relative life of 1.0 corresponds to 400 hours (~ 15 days) and higher average power outputs would degrade this capability substantially. Although a variety of factors such as relaxation of voltage requirements, restriction of minimum power to higher levels and addition of another cell (32 total instead of 31), different temperature modulation, etc., could ameliorate this power output-life relationship, it is still highly questionable whether a single module could, with "minimum modification", deliver an average power of 1.77 kilowatts for as long as 15 days.

Aside from intrinsic module capability, simple operational considerations indicate the desirability of at least two modules operating at all times. If one module fails, then the other can handle the full load until a spare module can be switched onto the line (approximately one hour warm-up time required).

Two modules operating simultaneously can handle twice the peak load that a single module could. Two modules could deliver 2840 watts without falling below 27 volts and could supply 4400 watts at 22 volts.

Figure 5-3 shows the weight of cryogenic reactants and their tankage for 30 days per kilowatt of net module output (net output = gross output - parasitic load) versus average module power output. This weight begins to increase below a

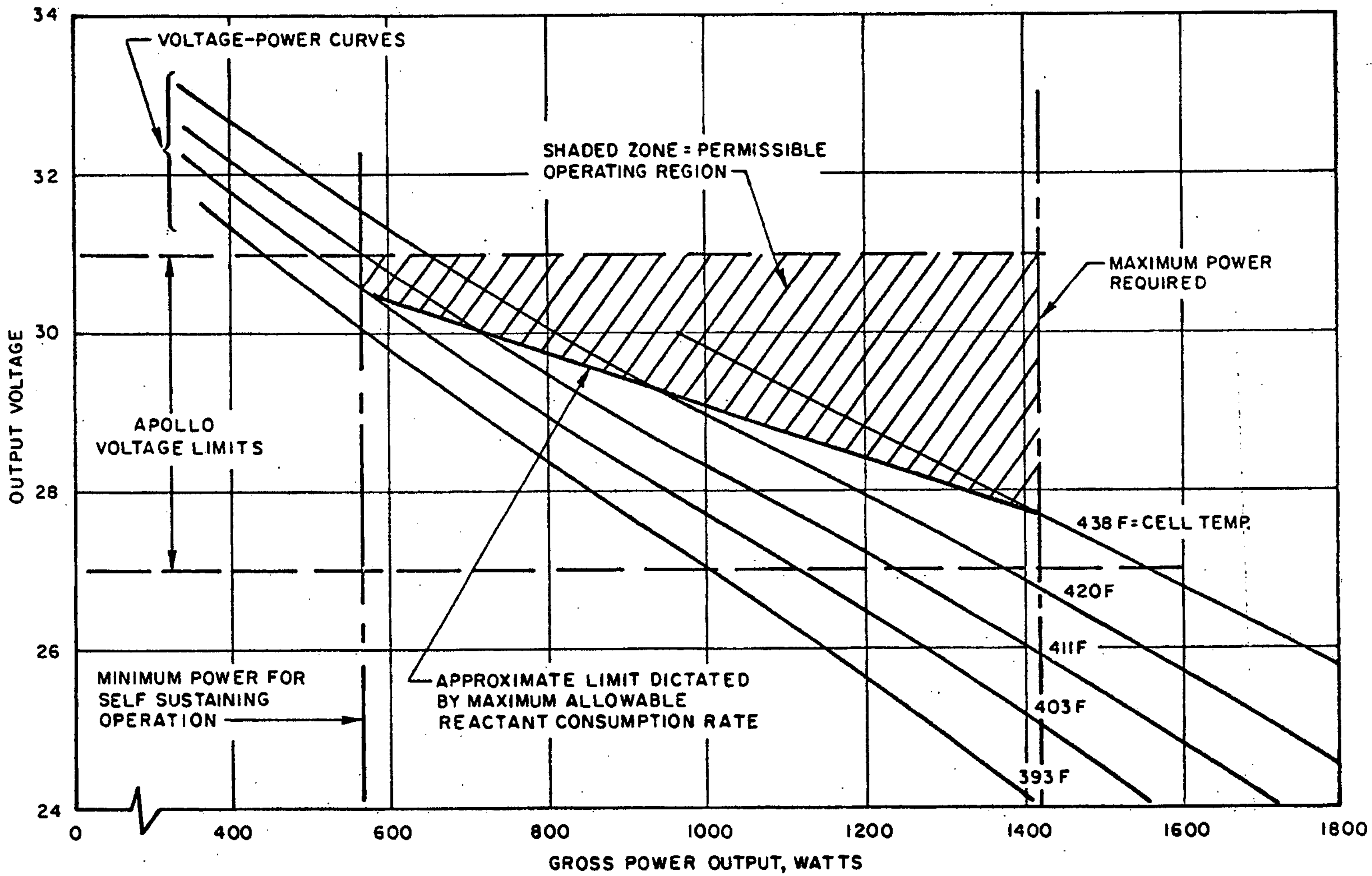


Figure 5-1. Apollo Fuel Cell Module Performance Constraints (Pratt and Whitney Data).

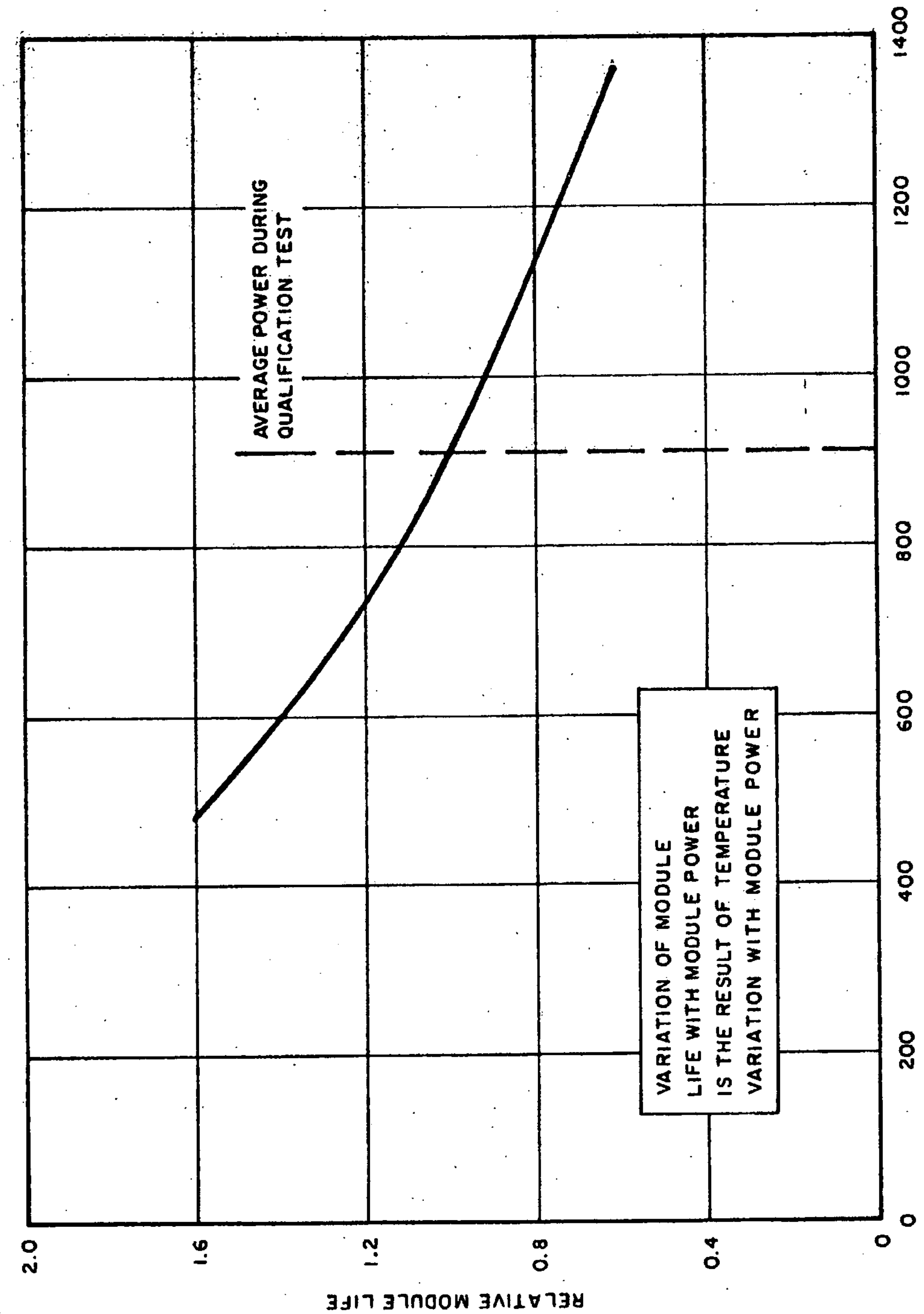


Figure 5-2. Estimated Effect of Apollo Fuel Cell Module Power on Module Life (Pratt and Whitney Data)

[redacted]

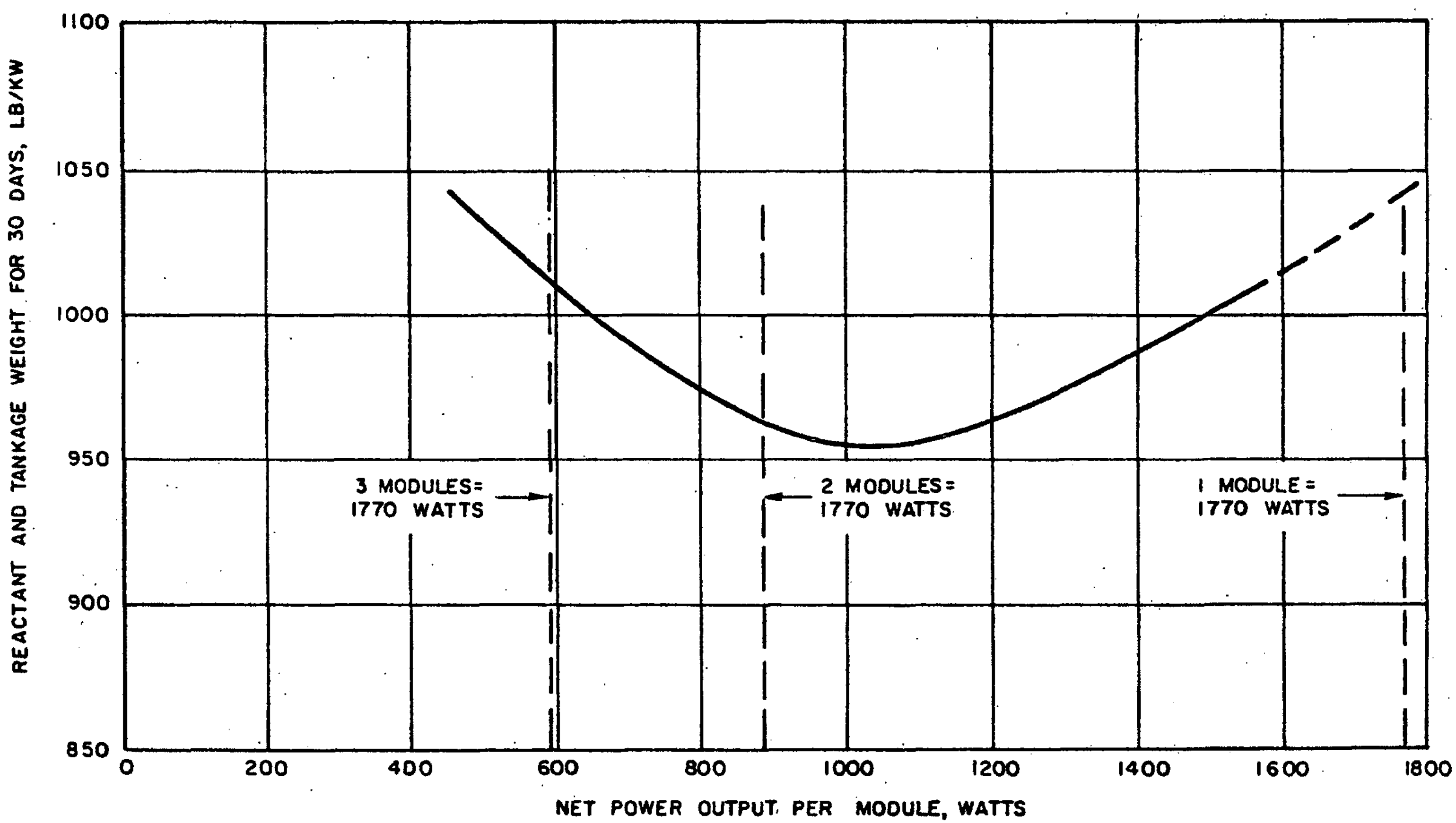


Figure 5-3. Reactant and Tankage Specific Weight for the Apollo Fuel Cell.

[redacted]

certain power level despite improving fuel cell efficiency because of the increasing importance of the fixed parasitic power losses (150 watts). It is clear from Figure 5-3 that no benefits are obtainable in this area by operating three units simultaneously.

It was concluded that two fuel cell modules operating at all times was the optimum module arrangement. This means that two modules would be started, would operate for 15 days (until wearout) and would then be replaced by two new modules. Standby replacement modules would then be added to the basic four modules to obtain the necessary reliability. Figure 5-4 shows 30-day reliability versus total number of modules. On the basis of this graph, a total of eight modules was selected as the minimum necessary for 30 days of operation.

5.3.2 Weight

Table 5-2 gives a tabular summary of total Apollo/MOL power system weight when using Apollo fuel cells. Brief comments are made below for each weight item.

Modules: Each module weighs 210 pounds. The number of modules was determined as cited previously.

Startup Batteries: Each module (except the initial two started prior to launch) requires 3500 watt-hours of energy for warmup. This would require about 43 pounds of silver-zinc battery to be able to start up each module on-orbit.

Water Recovery: Apollo experience

Oxygen: Approximately 0.82 pound per kilowatt-hour including a two percent purge allowance.

Oxygen Storage and Supply: Tankage, insulation, valving, etc., weights varied from 0.24 to 0.28 times oxygen weight depending on total oxygen weight.

Hydrogen: Based on 0.103 pound per kilowatt-hour which includes a two percent purge allowance.

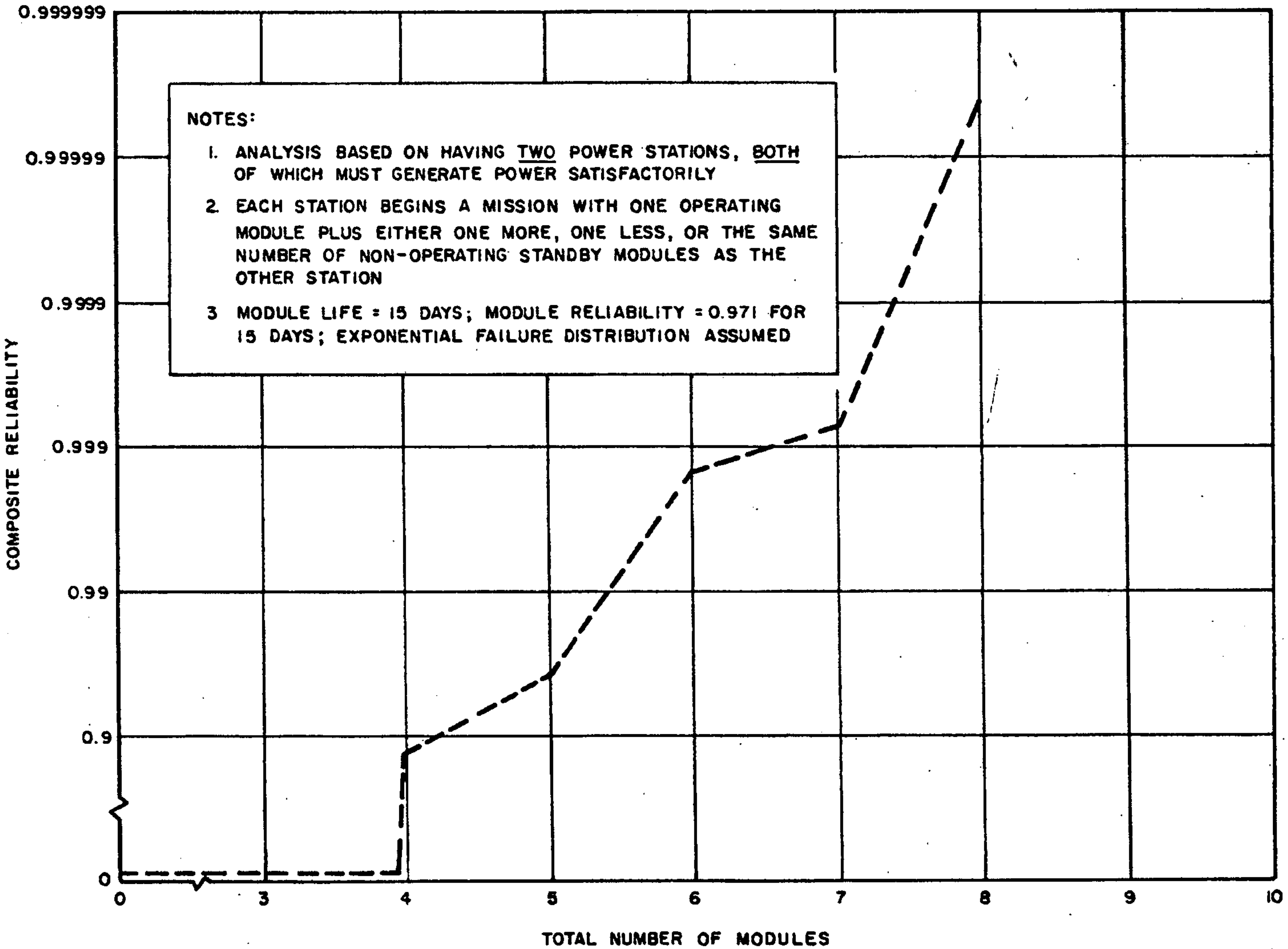


Figure 5-4. Reliability of Apollo Fuel Cell Modules for 30-Day Mission.

Table 5-2. Power System Weight Summary Using Apollo Fuel Cells (pounds).

Item	30 Days	60 Days	90 Days	120 Days
Modules (No. of modules)	1680 (8)	3260 (16)	5640 (24)	6520 (32)
Start-up batteries	260	610	955	1300
Water recovery	7	7	8	8
Oxygen	1040	2080	3120	4160
Oxygen storage and supply	280	514	760	990
Hydrogen	131	262	393	524
Hydrogen storage and supply	230	375	535	698
Power conditioning, control and distribution	495	495	495	495
Support structure	269	519	836	1031
Auxiliary battery	50	50	50	50
Total	4442 lb	8172 lb	12,892 lb	15,776 lb
(consumables)	(1171 lb)	(2342 lb)	(3513 lb)	(4684 lb)

Hydrogen Storage and Supply: Tankage, insulation, valving, etc., weights varied from 1.3 to 2.2 times hydrogen weight depending on total hydrogen weight.

Power Conditioning, Control and Distribution: A standard set of power conditioning, control and distribution equipment was defined for use with all types of power sources considered in this study. The "set" was comprised of:

A 1.25-kilowatt rated regulator plus two spares	73 lb
A 1.25-kilowatt rated inverter plus two spares	111 lb
Fault protection and automatic switching Instrumentation and control Distribution, installation provisions, misc.	306 lb

Justification of the spares provisioning is given in the section on reliability.

Support Structure: Weight was computed at eight percent of the combined weight of the fuel cells, reactants and tankage.

Auxiliary Battery: A nominal 50-pound allowance was made for a battery plus a charge regulator to accommodate extreme emergency (total loss of complete power system) and re-entry power requirements.

5.3.3 Reliability

Table 5-3 shows the results of a gross reliability analysis estimating the power system reliability for successful completion of a 30-day mission. The purpose of the analysis was to allow power system weight estimates to reflect reasonable levels of redundancy where needed and to establish a common reliability base for comparison among different power system concepts.

5.3.4 Volume

Results of volume calculations are shown in Table 5-4. Actual volumes occupied by the equipment items were multiplied by a factor estimated to account for unusable volume associated with the installation of items in addition to the volume of the items themselves.

Table 5-3. Power System Gross Reliability Using Apollo Fuel Cells.

Item	Basic 30-Day Reliability	Redundancy	30-Day Reliability With Redundancy
Fuel cell modules (4) (see Figure 5-4)	0.889	100% per module	~ 0.9999 (est)
Fuel cell start-up	~ 1.0 (assumed)	------	~ 1.0
Reactant tankage and supply (Tanks segmented, three per reactant)	0.99994 (before segmenting)		0.99982 (after segmenting)
Power conditioning			
Regulators	0.961	200%	0.99996
Inverters	0.912	200%	0.9997
Auxiliary batteries	~ 1.0 (assumed)	------	~ 1.0
Net	~ 0.76		~ 0.9994

Table 5-4. Power System Volume Using Apollo Fuel Cells.

Item	Volume (ft^3) 30 Days	Volume (ft^3) 120 Days	Factor	Installed Volume (ft^3) 30 Days	Installed Volume (ft^3) 120 Days
Modules (No. of modules) (24 in. dia. x 40.5 in. long)	85 (8)	340 (32)	1.5	128	510
Start-up batteries	2	10	2	4	20
Oxygen and assoc. hardware (3 tanks)	23	85	1.2	28	102
Hydrogen and assoc. hardware (3 tanks)	47	170	1.2	56	204
Power conditioning, control and distribution	8	8	2	16	16
Auxiliary battery	0.4	0.4	2	~ 1	~ 1
Total				233 ft^3	853 ft^3

5.3.5 Costs

Cost estimates (see Table 5-5) were divided into development costs and "unit" costs where "unit" refers to a full set of flight hardware. Fuel cell costs were based on informal data from Pratt and Whitney and cryogenic system costs were based on informal data from AiResearch. Unit costs proved to be relatively high because of the large number of fuel cell modules involved. The 200,000-dollar cost per module value used is believed to be conservative (low). Cost estimates for the categories entitled "electrical system engineering" and "other development and test" (including hardware) are compatible if not identical to similar cost categories used with other power system cost estimates (in this report). Consequently, cost comparisons with other power systems should be reasonably valid.

5.3.6 Availability

Earliest possible availability for an Apollo/MOL power system using Apollo fuel cells was estimated to be 18 months under the reasonable assumption that the fuel cells themselves would be the pacing items. The basis for the 18-month estimate was the following milestone schedule constructed from discussions with Pratt and Whitney.

Define and engineer necessary design changes	3 months
Procure material and parts	9
Qualification testing	3
Production and delivery	3
	18 months

5.4 COMPARISON OF CANDIDATE POWER SYSTEM CONCEPTS

Current technology fuel cells,* solar cells and [redacted] are alternative power sources to the Apollo fuel cell that were considered in detail in this study.

*"Current technology" is a term that has been adopted to refer to fuel cells incorporating existing technology which is quite advanced over the Apollo fuel cell concept. Refer to Appendix A for more discussion of this point.

Table 5-5. Power System Costs Using Apollo Fuel Cells (M = Million).

Development	
Fuel cell development and test	$ 5.0 M
Hardware (20 modules)	4.0 M
Cryogenic system development and test	3.5 M
Hardware (72 tanks)	2.2 M
Other development and test	3.0 M
Hardware (6 systems)	3.0 M
AGE development plus 4 sets of equipment	6.0 M
Electrical system engineering	3.4 M
Total	~ $ 30.0 M
Unit (30-Day System)	
Fuel cells ($ 200,000 per module x 8 modules)	$ 1.6 M
Cryogenics ($ 30,000 per tank x 6 tanks)	~ 0.2 M
Other	0.5 M
Total	~ $ 2.3 M

Appendices A, B and C summarize the results of the work on Apollo/MOL power systems incorporating these sources. They are defined in these appendices in approximately the same detail as the Apollo fuel cell power system was defined.

As indicated previously, [redacted] system availability would be such that it could not be considered for the initial Apollo/MOL design. It could, however, be made available for follow-on Apollo/MOL missions. Because of this and because of its very attractive characteristics, it was compared in a direct manner with the more readily available fuel cells and solar cells. A constraint on its selection, however, was that it could only be recommended for follow-on (generally thought of as longer duration) applications.

5.4.1 Quantitative Comparison

Table 5-6 compares total power system weights (plus aerodynamic-drag-associated weight penalties in the case of the solar cell system), volumes and costs. As indicated, these data are based on a common 30-day reliability of about 0.9995 for each system. Figure 5-5 illustrates the weight comparison data.

Fully oriented (two-degree of freedom) solar arrays were used to define the solar cell system on the basis of their lighter weight for almost all altitude-mission duration regimes of interest. An earth-oriented spacecraft was assumed at all times. Substantial aerodynamic drag would be incurred by solar arrays so that a propulsion penalty was associated with the solar cell system. The amount of this penalty was, of course, a function of the altitude that would be maintained by the Apollo/MOL spacecraft. Consequently, solar cell power system weight is shown as a function of orbit altitude. Weights of both the fuel cell and [redacted] systems would be altitude independent since they do not include drag inducing parts.

Table 5-6. Quantitative Comparison of Candidate Power Systems.

	Mission Duration (Days)			
	30	60	90	120
Weight (lb)				
Apollo fuel cell	4,442	8,172	12,892	15,776
Current technology fuel cell	2,772	4,867	6,986	9,091
Solar cell				
100-nautical mile altitude	3,787	5,458	7,218	8,969
125-nautical mile altitude	2,582	3,123	3,676	4,219
200-nautical mile altitude	2,022	2,076	2,132	2,185
[redacted]				
Reliability				
All systems	~ 0.9995			
Volume (ft^3)				
Apollo fuel cell	233			853
Current technology fuel cell	111			363
Solar cell				
125-nautical mile altitude	35.1			59.4
200-nautical mile altitude	27.8			30.2
[redacted]				

Costs	(Development)	(Unit)
Apollo fuel cell	$ 30 M	$ 2.3 M
Current technology fuel cell	32 M	1.0 M
Solar cell	25 M	1.8 M
[redacted]		

[redacted]

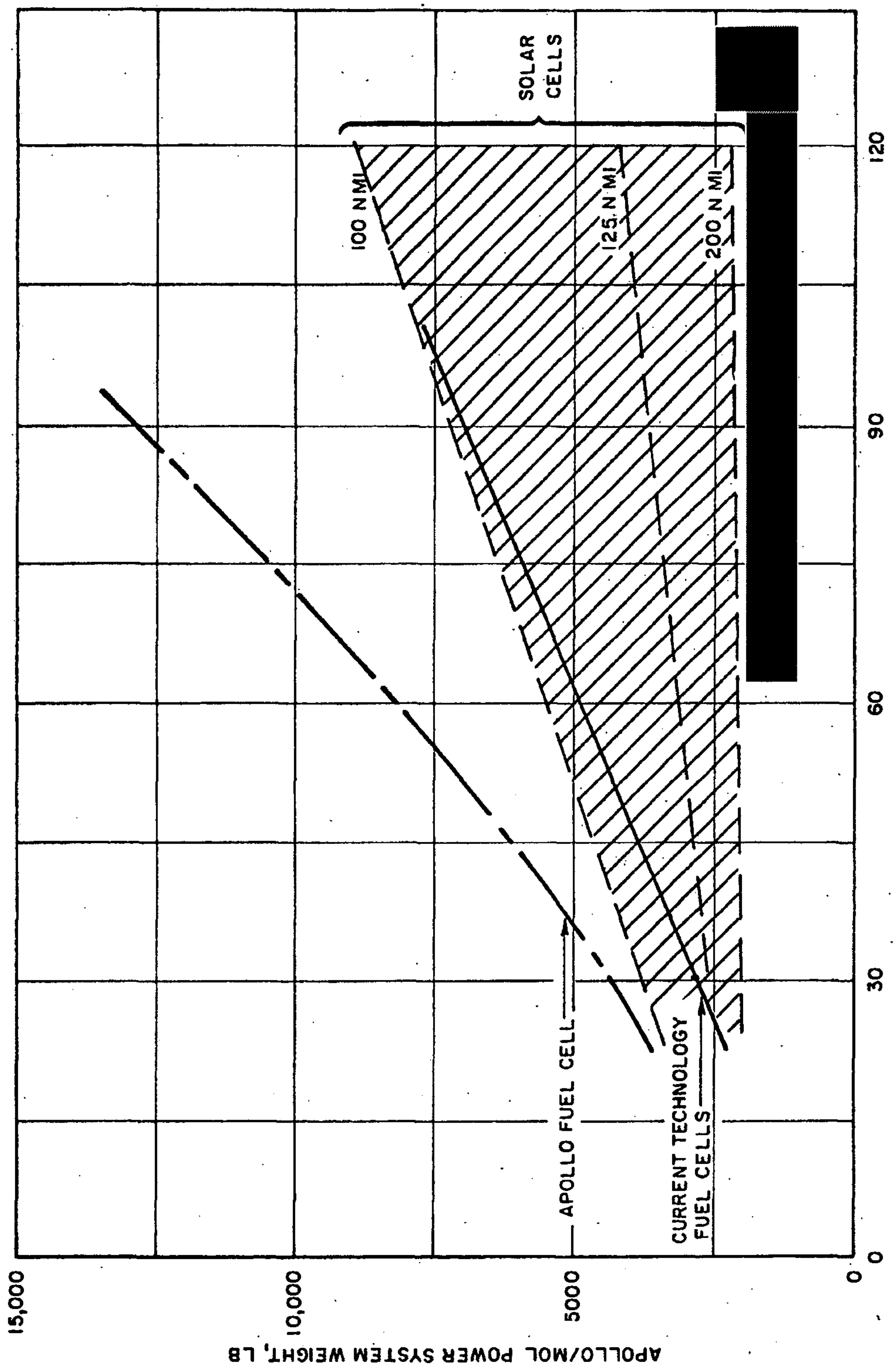


Figure 5-5. Apollo/MOL Power System Weights

[redacted]

[redacted]

Assessing power system weights on the basis of a relatively high reliability proved quite a bit more penalizing to some systems than to others. Reliability improvement with the solar cell and current technology fuel cells up to the 0.9995 level resulted in modest weight increases for these systems: roughly 100 and 300 pounds, respectively. This same change cost almost 400 pounds with the [redacted] system and about 950 pounds with the Apollo fuel cell system. Although no two of these power system types would receive the same reliability allocation, even in identical applications, there was no satisfactory alternative in this study but to compare the different systems on the basis of equal reliabilities for all, keeping in mind what this reliability cost in each case.

Volume data for the fuel cell systems reflect the relative compactness of the current technology fuel cells. Total volume of reactants was essentially identical for both fuel cell systems. The 200-nautical mile, 30-day solar cell system volume is indicative of the volume of the solar cell system itself. Larger values for other mission conditions include appreciable amounts of propellant and associated hardware volume.

Cost estimates in Table 5-6 are rough at best and probably are all optimistic. In essence, they indicate that all of the power systems would cost about the same to develop. "Unit cost" refers to the cost of a full set of flight hardware. Differences in unit costs among the systems are more pronounced than development cost differences and realistically so. An important point to remember when considering the cost of developing a new power source for follow-on missions is that, within the guidelines of this study, all parts of the power system other than the source would not need redeveloping. Therefore, only a portion of development cost indicated for each system in Table 5-6 would be incurred if that system were to replace another system for follow-on missions. [redacted] development (13 million dollars) plus the initial set of flight hardware (1.5 million dollars).

It is reasonably clear from the quantitative comparison data presented in Table 5-6 and Figure 5-5 that the Apollo fuel cell system is not very competitive for Apollo/MOL application. Also, fuel cells of either type studied are not competitive for longer duration missions out to 120 days. Factoring in the availability constraint on the [redacted] system leaves the following possibilities which remain (at this point) attractive.

1. Two-year availability (emphasis on 30-day mission capability)
   - current technology fuel cells
   - solar cells
2. Three- to four-year availability (emphasis on growth to 120-day mission capability)
   - solar cells

[redacted]

5.4.2 Qualitative Comparison

A great many factors are important to power system selection other than the few just discussed in quantitative terms. A large number of factors associated with development, mission and operational flexibility could be importantly influential in selection decisions. Specifically, prelaunch operations, launch interfaces, laboratory activation, on-orbit operations interfaces, experiments interference, emergency power, crew safety, crew demands, complexity, growth, and development risk are all categories that could have important bearing on such decisions. All of these factors are discussed in a qualitative manner in Appendix D as they would be affected by selection of a solar cell power system and by selection of a fuel cell power system and comparisons are made. These discussions were prepared originally for the

(Gemini)/MOL program office and are directly applicable to this study effort. A tabular summary of the discussion and comparison results is given in Table 5-7.

Table 5-7 shows the over-all preponderance of practical advantages that would accrue to Apollo/MOL from the selection of a fuel cell power system in preference to a solar cell power system. A similar comparison between [redacted] and solar cell systems was not prepared. Study of Appendix D will show, however, that all bad features of solar cells would still exist in a comparison with [redacted] systems and many of the good features of fuel cells apply to the [redacted] system. Also, it is clear from Appendix D that, if fuel cells were selected to meet the two-year availability requirement (as this study shows they should be), redesigning the power system for a longer mission capability would be immensely more complicated if solar cells were selected for the growth system instead of [redacted]

5.5 RECOMMENDED POWER SYSTEMS

As a result of the power system comparisons in the previous section, the recommended Apollo/MOL electrical power systems are as follows:

1. Two-year availability; select current technology fuel cells.
2. [redacted]

These recommendations are summarized in Table 5-8 which characterizes the power systems selected for each of the five Apollo/MOL configurations specified by the program office. Also shown in Table 5-8 are the power system characteristics in Apollo and in (Gemini)/MOL.

An important over-all program recommendation rendered from the power system study is that early (two-year availability) missions should be designed around a 30-day capability. If a long duration (e.g., 120-day) capability had to be designed within two years, the power system would have to be built around solar cells (as indicated in Table 5-8). Since this study indicated that a solar cell power system would be basically undesirable for the Apollo/MOL mission, it follows that the deferment of a 120-day design until a [redacted] system could be obtained would be preferable.

Table 5-7. Summary of Qualitative Comparison Results Between Fuel Cells and Solar Cells for Apollo/MOL.

	The Comparison is				
	Strongly in Favor of Solar Cells	In Favor of Solar Cells	Neutral or Uncertain	In Favor of Fuel Cells	Strongly in Favor of Fuel Cells
1. Prelaunch Operations				x	
2. Launch Interfaces				x	
3. Laboratory Activation					x
4. On-Orbit Operations Interfaces					
a. Environmental Control System		x			
b. Life Support				x	
c. Maneuvering				x	
d. Attitude Control					x
e. Orbit Keeping Propulsion				x	
f. Communications				x	
g. Rendezvous and Docking				x	
5. Experiment Interference					x
6. Emergency Power			x		
7. Crew Safety			x		
8. Crew Demands			x		
9. Complexity			x		
10. Growth					
a. Different Altitudes					x
b. Longer Mission Duration	x				
c. Different Power Levels					x
d. Polar (Operational) Missions			x		
e. Vulnerability (Operat'l Missions)					x
11. Development Risk					
a. Power System Equipment					x
b. MOL Power Requirements					x

Table 5-8. Recommended Power Systems for Apollo/MOL.

		Apollo	Minimum Adaptation Apollo	Apollo/MOL Optimized Configurations		Apollo/MOL Recommended Configuration	Gemini/MOL
Duration		14 days	30 days	30 days	[redacted]		30 days
Crew size		3 men	2 men	2 men			2 men
Average power requirements (kw)		1.65	1.77	1.77			1.6
Selected power source		P and W fuel cells	P and W fuel cells	Current technology fuel cells			Fuel cells
Reasons for power source selection		--------	Makes use of Apollo hardware	Simplicity Weight			Simplicity
Power system weight (lb)		2423	4442	2777			2324
Reliability		0.994	~ 0.9995	~ 0.9995			0.96 (est)
Volume (ft^3)			233	111			~ 100
Cost	Develop.	$ 80 M	$ 30 M	$ 32 M			$ 32 M
	One flight system	$ 1.7 M	$ 2.3 M	$ 1.0 M			$ 1.0 M
Key development tasks		--------	Cryogenic tankage	Reliability Cryogenic tankage			Life and reliability Cryogenic tankage

5.6 COMPARISON WITH GEMINI/MOL POWER SYSTEM

Power system requirements with Apollo/MOL differ in no essential way from those in Gemini/MOL. Probably the only difference of importance is that "maximum use of existing hardware" in Gemini/MOL causes special consideration to be given to the Gemini fuel cell instead of to the Apollo fuel cell.

Studies performed for Gemini/MOL to select a power system have been very similar in approach to that for Apollo/MOL described in this report. In particular, the results have been the same as regards recommendation of fuel cells over solar cells as the primary power source. Much more detailed attention has been given to the question of which fuel cell, however. As yet no decision has been made as to which fuel cell concept(s) will be developed for (Gemini) MOL. In fact, several Air Force funded studies by industry are in progress with the objective of providing a proper basis for making such a decision.

Regardless of which power source might be expected for use with Gemini/MOL, it is logical to suppose that the best power system for Apollo/MOL is also best for Gemini/MOL and vice versa. This logic stems from the close similarity in power system design requirements in the two cases. Viewed in this way, there should be no significant differences between an Apollo/MOL power system and a Gemini/MOL power system.

5.7 CONCLUSIONS

1. Power source candidates for Apollo/MOL are limited to solar cells, fuel cells and [redacted] Of these the fuel cell should be considered as a limited duration (30 to 40 days) source, whereas the solar cell and [redacted] systems qualify readily as 120-day power sources.

2. [redacted] although they appear to offer the best all around power system by a wide margin. Consequently, only solar cells and fuel cells are contenders for early missions.

3. Current technology fuel cells are probably superior to the present Apollo fuel cells. Comparisons between current technology fuel cell power systems and a solar cell power system indicated that many practical advantages would accrue to Apollo/MOL if the fuel cell system were selected. Consequently, the current technology fuel cell system is judged preferable for the initial Apollo/MOL missions.

4. If a long duration (120-day) Apollo/MOL had to be available with a two-year lead time, the power system would have to be built around solar cells since [redacted] Since this study indicated that a solar cell power system would be basically undesirable for Apollo/MOL, it was concluded that the 120-day design should be deferred [redacted]

APPENDIX A

APOLLO/MOL POWER SYSTEM USING CURRENT TECHNOLOGY FUEL CELLS

An important reason for the selection of Pratt and Whitney's "Bacon" fuel cell for the Apollo mission was its high heat rejection temperature. At the time of selection, it was planned that the Apollo spacecraft would land on the lunar surface where heat rejection would be required in spite of lunar surface temperatures as high as 250 F. Since that selection, which occurred when fuel cells were little more than laboratory curiosities, considerable insight has been gained into which electrochemical concepts offer the best fuel cell performance and life characteristics. In particular, the medium temperature (185 - 200 F) catalyzed electrode approach is currently demonstrating major advances over the Bacon (375 - 425 F) and ion exchange membrane (75 - 100 F) fuel cells being developed for the Apollo and Gemini missions, respectively. Better known examples of medium temperature fuel cells are the Allis-Chalmers fuel cell, the Union Carbide fuel cell, and the Pratt and Whitney "compact" fuel cell.

It is entirely possible that the Apollo and Gemini fuel cell concepts may be at higher states of development than the medium temperature approach. Because of this possibility, they may be preferable for use with Apollo/MOL despite their indicated performance inferiority. At any rate, the medium temperature fuel cell, hereafter referred to as the "current technology" fuel cell, is an important contender for application to the Apollo/MOL mission. Data used in the following definition of a power system using "current technology" fuel cells are based on the Allis-Chalmers fuel cell for which the most abundant supply of information was available.

A. Availability

Fully qualified flight hardware could be made available in 18 to 24 months. This includes all power system hardware other than the fuel cells since it should not take any longer to develop this other hardware for current technology fuel cells than the 18 months maximum estimated in connection with the Apollo fuel cell power system.

B. Basic Module Configuration

A single module such as that designed for the Saturn IVB can deliver more than enough average power (two kilowatts). Also, this fuel cell has a life capability equal to or greater than the 720-hour basic mission duration of interest. As with the Apollo fuel cell, however, it is desirable to have a pair of modules always functioning so that if one fails, the other can handle the entire load without any power interruption.

Unlike the Apollo fuel cell, a second operating fuel cell is not necessary to satifactory power system performance over the entire mission. Thus, a second operating current technology fuel cell is in active redundancy and contributes markedly to reliability. Assuming each module had the same failure rate (82×10^{-6} failures/module-hour) as the Apollo fuel cell gives a 30-day reliability for two operating modules of 0.9967. Adding a third module as a non-operating spare gives a 30-day reliability of 0.99992. It was concluded that a total of three modules would be required for a 30-day mission, two operating and one held in passive standby, in order to meet the power system reliability objective established for this study.

C. Weight

A summary of the weight versus mission duration of a current technology fuel cell system is given in Table A-1. Note that the current technology fuel cell is self-starting; that is, it does not require an energy input to warm up and begin delivering power. Hydrogen and oxygen reactant consumption rates proved to be almost identical to those of the Apollo fuel cell. All other weight items (except, of course, the fuel cell modules themselves) are the same or equivalent to the Apollo fuel cell power system.

D. Reliability

Table A-2 shows the power system success probability determination for a 30-day mission. It is important to note that the module reliabilities are based on a failure rate assumed equal to that of the Apollo fuel cell. In view of the great amount of development money that has been spent on the Apollo fuel cell, this may be an optimistic expectation.

E. Volume

Volume values for the current technology fuel cell power system are the same as for the Apollo fuel cell except for the former's smaller and fewer modules and no start-up batteries. Table A-3 gives volume estimates for the current technology fuel cell system.

F. Costs

Power system costs are estimated in Table A-4 using current technology fuel cells. Fuel cell module costs are based on estimates from Allis-Chalmers. All other costs are the same as for the Apollo fuel cell system.

Table A-1. Power System Weight Using Current Technology Fuel Cells.

Item	30 Days	60 Days	90 Days	120 Days
Modules (No. of modules)	375 (3)	750 (6)	1125 (9)	1500 (12)
Oxygen	1050	2100	3150	4200
Oxygen storage and supply	280	514	760	990
Hydrogen	131	262	393	524
Hydrogen storage and supply	230	375	535	698
Power conditioning, control and distribution	495	495	495	495
Support structure	166	321	478	634
Auxiliary battery	50	50	50	50
Total	2777 lb	4867 lb	6986 lb	9091 lb

Table A-2. Power System Reliability Using Current Technology Fuel Cells.

Item	Basic 30-Day Reliability	Redundancy	30-Day Reliability With Redundancy
Modules	0.94284 (assumed)	100% active plus 100% passive	0.99992
Cryogenic tankage (each reactant segmented into 3 tanks)	0.99994 (before segmenting)	----------	0.99982 (after segmenting)
Power Conditioning			
Regulators	0.961	200%	0.99996
Inverters	0.912	200%	0.9997
Auxiliary battery	~ 1.0 (est.)	----------	~ 1.0
Net	~ 0.82		~ 0.9994

Table A-3. Power System Volume Using Current Technology Fuel Cells.

Item	Volume (ft^3)		Factor	Installed Volume (ft^3)	
	30 Days	120 Days		30 Days	120 Days
Modules (No. of modules)	6 (3)	24 (12)	1.7*	10	40
Oxygen and assoc. hardware (3 tanks)	23	85	1.2	28	102
Hydrogen and assoc. hardware (3 tanks)	47	170	1.2	56	204
Power conditioning, control and dist.	8	8	2.0	16	16
Auxiliary battery	0.4	0.4	2.0	~ 1	~ 1
Total				~ 111 ft^3	~ 363 ft^3

*A slightly greater factor was used here than with the Apollo fuel cell to account for less design definition with the current technology fuel cell.

Table A-4. Power System Costs Using Current Technology Fuel Cells (M = Million)

Development	
Fuel cell development and test	$ 8.6 M
Hardware (20 modules)	2.0 M
Cryogenic system development and test	3.5 M
Hardware (72 tanks)	2.2 M
Other development and test	3.0 M
Hardware	3.0 M
AGE development plus hardware (4 sets)	6.0 M
Electrical system engineering	3.4 M
Total	~ $ 32.0 M
Unit (30-Day System)	
Fuel cells ($ 100,000/module x 3 modules)	$ 0.3 M
Cryogenic ($ 30,000 / tank x 6 tanks)	~ 0.2 M
Other	0.5 M
Total	~ $ 1.0 M

APPENDIX B

APOLLO/MOL POWER SYSTEM USING SOLAR CELLS

A. Array Orientation

One of the most fundamental decisions to be made in configuring a solar cell power system is how the solar arrays will be oriented. A study ground rule was that the spacecraft would be earth oriented (roll axis coincident with velocity vector and yaw axis aligned with local vertical). A variety of associated array orientation control modes has been studied in the past (in connection with MOL) in which the arrays had zero, one, and two degrees of freedom relative to the spacecraft. Only two cases proved to be of interest from a weight standpoint; two degrees of freedom (full array orientation) and one degree of freedom in which the large array surfaces were always parallel to the velocity vector (roll control).

Figure B-1 shows the altitude - mission duration regimes in which each of these array types would be lighter weight. This weight comparison includes consideration of the arrays themselves, batteries, altitude sustenance propulsion due to aerodynamic drag on the arrays, attitude control propellant required because of array drag, and array stowage, deployment, support and orientation weights. It can be seen that roll control arrays would be lighter weight only with long duration, low altitude missions which are of doubtful over-all spacecraft capability and, therefore, of doubtful interest. Since the fully oriented arrays would be most competitive with other power system types, they were selected for use in defining the characteristics of a solar cell system. Total area of fully oriented arrays was 605 ft^2.

B. Weight

A weight summary for an Apollo/MOL solar cell power system is presented in Table B-1. Important facts and assumptions underlying the weight determinations are discussed below for each item.

Solar arrays: Solar arrays weights are based on Program 461 experience corrected for differing charged particle radiation environments. Program 461 solar arrays are the largest arrays which have been flown and closest of any designed to meeting the Apollo/MOL requirements from the standpoint of stowage, deployment and orientation. Therefore, they are the best available indicator of what could be achieved with Apollo/MOL. Despite this, there is little doubt that an analysis of the dynamic response requirements of the arrays, as dictated by attitude control needs, would show the array weights in Table B-1 to be grossly underestimated. Such analyses have been conducted in connection with (Gemini) MOL and have shown that obtaining the required array rigidity would increase array weights over those shown by something in excess of several hundred to several thousand pounds. An alternative method

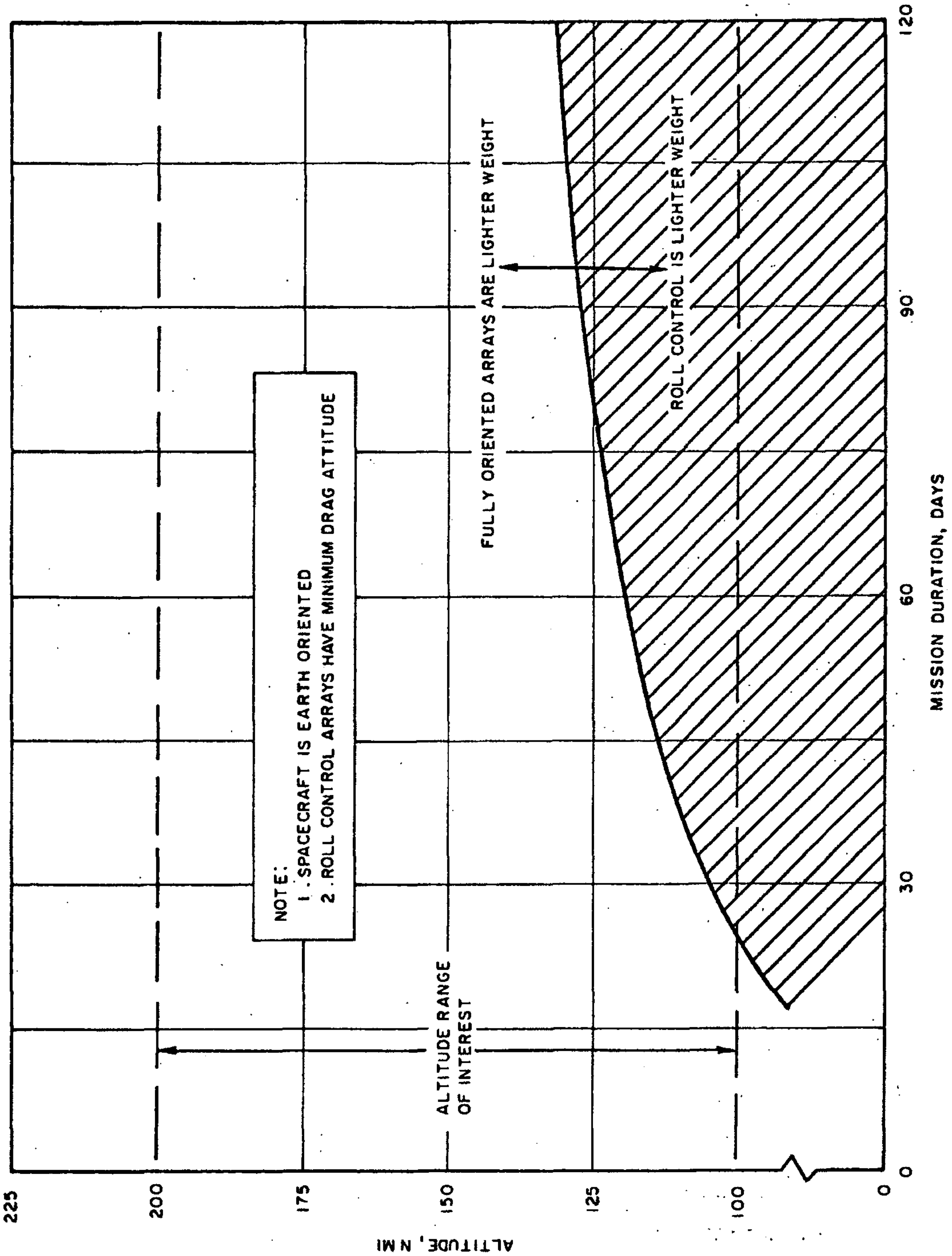


Figure B-1. Solar Cell Array Weight Comparisons.

Table B-1. Power System Weight Using Solar Cells (lb).

	30-day redundancy	100-naut mi. alt.* 30 days	60	90	120	125-naut. mi. alt.* 30 days	60	90	120	200-naut. mi. alt.* 30 days	60	90	120
Solar arrays	0			768				768				768	
Array stowage, deployment and support	0			140				140				140	
Array orientation	200 %			60				60				60	
Secondary battery	75 %			249				249				249	
Battery charge regulator	100 %			48				48				48	
Reserve and pyrotechnic batteries	0			202				202				202	
Power conditioning, control and distribution**	(200 %)			495				495				495	
Associated penalties													
Altitude sustenance propulsion	0	1800	3430	5150	6860	622	1150	1690	2220	73	125	179	230
Attitude control propellant	0	40	81	121	162	13	26	39	52	2	4	6	8
TOTAL		3787	5458	7218	8969	2582	3123	3676	4219	2022	2076	2132	2185
(Consumables)		(1470)	(2941)	(4411)	(5882)	(457)	(914)	(1371)	(1828)	(35)	(70)	(105)	(140)

*Numbers are in pounds
**Regulators and inverters

of obtaining the required dynamic decoupling would be to increase the flexibility of the arrays with a design akin to a window shade arrangement. Such a design would be highly experimental and non-conforming to the "current state-of-the-art" design requirement.

Array Stowage, Deployment and Support: These weights were estimated on the basis of folded arrays stowed under external fairings during launch and subsequently deployed by lazy tong devices.

Array Orientation: The weight estimated includes drive members, gear trains and a drive motor with two spares for reliability purposes. Two sets of orientation gear were assumed consistent with having two arrays, one on each side of the spacecraft.

Secondary Battery: Silver-cadmium batteries at 15 watt-hours per pound were sized on the basis of a 50 percent depth of discharge. Reliability calculations subsequently showed the need for a 75 percent redundancy level in connection with the batteries which was then added in.

Battery Charge Regulators: Four secondary batteries each with its own charge regulator was assumed as the basic battery arrangement. A redundancy of 100 percent on the regulators gave a total of eight of these devices.

Reserve and Pyrotechnic Batteries: Reserve batteries would be required to power Apollo/MOL during the period between launch and power system activation on orbit. A total of 15,000 watt-hours was allocated for this purpose. This same battery (or a portion thereof) could be used to supply power to the re-entry vehicle after separation from the laboratory. Consequently, no additional weight was included for this function. A special (high current) pyrotechnic battery at 15 pounds was included to handle squib firings, motor startings and the like.

Power Conditioning, Control and Distribution: The same standard "set" of this type of equipment that was used in connection with the Apollo fuel cell system was applied to the solar cell system.

Associated Penalties: Aerodynamic drag on the solar arrays could significantly influence the functions of altitude sustenance and spacecraft attitude control performed by the propulsion and reaction control subsystems, respectively. These affects were evaluated for various circular orbit altitudes with varying mission durations. Propellant requirements for altitude sustenance (attributable solely to the solar arrays) were calculated from Reference 1 data and from drag coefficient data used for Gemini/MOL. These propellant weights were then multiplied by correction factors to account for tankage and other propulsion hardware associated with propellant utilization. Alternatively, it would be possible to launch to a sufficiently high initial altitude such that continuous decay of the orbit would not cause re-entry over the duration of the mission. However, the presence of solar arrays would cause the drag and decay rate to increase such that a higher initial altitude would be required with a solar cell system than with a fuel cell or [redacted] system. This higher initial altitude requirement creates a payload loss which would have to be taken as a weight penalty for the solar cell system. Analysis showed that the weight penalty so obtained was equivalent to the solar array propulsion weight penalty obtained by maintaining a constant orbit altitude of about 170 nautical miles. To obtain attitude control penalties, maximum disturbing torques of solar array drag origin were assumed to be continuously balanced by thrusting from the reaction control subsystem. The propellant requirements resulting from this ultra-conservative analysis were small enough so that refinement of the analysis was not warranted.

C. Reliability

An estimation of the over-all reliability of a solar cell power system is shown in Table B-2. Redundancy was added in a manner that increased weight the least until the goal of 0.9995 for 30 days was attained. Array reliability stemmed from having sufficient excess operating solar cells so that array degradation could occur without power output "failure". Array deployment and orientation reliabilities were based solely on probabilities of drive motor failures. Power conditioning equipment reliabilities were taken from Westinghouse data while the other basic reliability data were from various industry sources. Extension of the 30-day estimate to obtain a 120-day estimate was done using the simple exponential model $R = e^{-\lambda t}$.

D. Volume

Power system and associated equipment volumes are tabulated in Table B-3. Note that the internal volume consumed by the basic power system is relatively small but that the contribution to total volume from "power system associated" propulsion can be quite large for longer duration missions.

E. Costs

Estimated solar cell power system costs are shown in Table B-4. Costs in the "other" and "AGE" categories are essentially the same as those estimated for other power system types in these or equivalent categories.

Table B-2. Power System Reliability Using Solar Cells.

Item	Basic 30-Day Reliability	Redundancy	30-Day Reliability With Redundancy
Arrays	0.99998	0	0.99998
Array Deployment	0.99998	0	0.99998
Array Orientation	0.96890	200 %	0.99997
Secondary Batteries	0.995	75 %	0.99990
Battery Charge Regulator(s)	0.99569	100 %	0.99998
Reserve and Pyrotechnic Batteries	~ 1.0 (assumed)	------	~ 1.0
Power Conditioning			
Regulators	0.9611	200 %	0.99996
Inverters	0.912	200 %	0.9997
Net	~ 0.83		~ 0.9995

[redacted]

Table B-3. Power System Volume Using Solar Cells.

Item		Volume (ft^3) 30 Days	Volume (ft^3) 120 Days	Factor	Installed Volume (ft^3) 30 Days	Installed Volume (ft^3) 120 Days
Solar arrays		0	0	-------	0	0
Array stowage, deployment and support		0	0	-------	0	0
Array orientation		1.5	1.5	1	1.5	1.5
Secondary batteries (130 #/ft^3)		1.92	1.92	2	3.8	3.8
Battery charge regulators (50 #/ft^3)		0.96	0.96	2	1.9	1.9
Reserve and pyrotechnic bat. (130 #/ft^3)		1.55	1.55	2	3.1	3.1
Power conditioning, control and distribution		8	8	2	16	16
Associated penalties						
Altitude sus. + attitude cont (75 #/ft^3)	100 n.m.	24.6	94	1.1	27.1	103.6
	125 n.m.	7.9	30		8.7	33.0
	200 n.m.	1.3	3.5		1.4	3.8
Total	100 n.m.				53.5 ft^3	130.0 ft^3
	125 n.m.				35.1 ft^3	59.4 ft^3
	200 n.m.				27.8 ft^3	30.2 ft^3

[redacted]

Table B-4. Power System Costs Using Solar Cells (M = Million).

	Solar Cell Unique	Other	Total
Development			
Engineering	$ 4.6 M	$ 4.5 M	$ 9.1 M
Hardware	3.0 M	1.5 M	4.5 M
Test	2.5 M	3.4 M	5.9 M
AGE (including 3 sets of hardware)			5.0 M
TOTAL			$ 25.0 M
Unit			
Solar cell arrays			$ 1.3 M
Other			0.5 M
TOTAL			$ 1.8 M

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

APPENDIX D

QUALITATIVE COMPARISON OF FUEL CELLS AND SOLAR CELLS FOR APOLLO/MOL

Elements of the Apollo/MOL mission which could be affected by power system selection through factors other than weight, cost, volume, and reliability have been categorized as follows:

1. Prelaunch Operations
2. Launch Interfaces
3. Laboratory Activation
4. On-Orbit Operations Interfaces
5. Experiments Interference
6. Emergency Power
7. Crew Safety
8. Crew Demands
9. Complexity
10. Growth
11. Development Risk

Each of these categories is discussed below in terms of the use of fuel cells or solar cells for Apollo/MOL, followed by a tabular summary of the qualitative discussion results (presented in the main body of the report).

1. Prelaunch Operations

No unusual burdens from either fuel cell or solar cell systems are foreseen for prelaunch (or postlaunch) ground support activities. A singularly important reliability advantage accrues to the fuel cell in this category, however, because of its amenability to thorough preflight checkout. In fact, operation of a fuel cell system would be started some number of hours before launch and would continue up to, through, and after launch. A solar cell system, on the other hand, cannot be checked out on the pad prior to flight. The necessarily flimsy panels cannot support their own weight under one-g, and solar simulation and illumination of even large panel segments would require prohibitively elaborate equipment provisions. Because of the great risk to damage involved with even superficial prelaunch check-out of solar cell panels, current practice in one major Air Force program has resulted in no prelaunch check-out, except for simple electrical continuity checks, once the panels have left the fabricating contractor's facility.

[redacted]

2. Launch Interfaces

Fuel cell systems would be entirely contained within Apollo/MOL and would have no significant launch-peculiar interfaces. The dynamic environment during a launch apparently poses no threat to fuel cell operation. Both Gemini and Apollo fuel cells have successfully undergone "double severity" vibrations tests without a trace of trouble. Solar cell panels, it is assumed, would be folded and housed externally to Apollo/MOL beneath protective fairings. Two important interfaces are created with this approach; viz., aerodynamics of the launch vehicle/payload with side protuberances, and structural dynamics of such a configuration. Present rough estimates indicate that the problems created by these interfaces may not be important. However, they would definitely constitute an engineering and design burden of significant magnitude to ensure that they would not be a source of mission degradation or failure.

3. Laboratory Activation

As was noted earlier, a fuel cell system would be operating throughout launch and ascent. Consequently, laboratory (and Apollo) power will exist continually and will not be a problem for the crew during laboratory activation. General Dynamics recently reported on a sequence-time study of solar cell power system activation for (Gemini) MOL which showed that 222 minutes of elapsed time were required before the power system could be declared fully operational. A substantial portion of the 222 minutes (for at least one crewman) was devoted exclusively to the solar cell power system.

4. On-Orbit Operations Interfaces

a. Environmental Control System - A fuel cell would have an important interface with the ECS since about 30 percent of all heat rejected by the ECS would come from cooling the fuel cell. This burden is offset to some extent by storing excess water from the fuel cell for use in the ECS water boiler during periods of high heat load. Also, some alleviation is possible by utilizing the heat capacity of the cryogenic reactants consumed by the fuel cell. The net effect of using a fuel cell power system, however, is a substantial thermal burden on the ECS. A solar cell system would necessitate battery and battery charge control unit temperature control which would be a minor interface with ECS.

b. Life Support - A beneficial interface of fundamental significance between fuel cells and life support is water supply. A fuel cell system would provide more than enough water for all crew drinking water and wash water needs. The interface would simplify water management in Apollo/MOL by eliminating water purification provisions. With a fuel cell, the only provisions required would be for collecting the water as it comes from the fuel cell. Another interface which could be obtained if it proved desirable would be to integrate life support and fuel cell oxygen storage. A solar cell power system would not interact with the life support system.

[redacted]

c. Maneuvering - Very limited, if any, spacecraft maneuvering would be possible with fully deployed solar panels because of their frail construction. Methods for retraction of panels to permit maneuvering are conceivable but would be complex and time-consuming during operation. A fuel cell power system would not affect maneuvering.

d. Attitude Control - A solar cell system would have a large impact on attitude control and vice versa. The Apollo/MOL spacecraft roll moment of inertia would be increased markedly by solar cell panels, thereby increasing propellant consumption for stabilization about the roll axis. Exhaust impingement on the solar panels from the reaction control thrustors would constrain the design and possibly the relative positioning of these equipments. Aerodynamic drag on the paddles would disturb attitude control about the pitch and yaw axis. Whether this disturbance would be beneficial or detrimental as regards spacecraft stabilization is not known at this time. In the other direction, precision attitude control requirements would dictate sufficient solar panel stiffness (substantially more than is otherwise required) so that panel flutter induced by reaction control pulses would damp out rapidly. A fuel cell system would not interact significantly with the attitude control system.

e. Orbit Keeping Propulsion - Aerodynamic drag on solar cell panels would create a need for larger amounts of propellant and more frequent thrusting if a nominal orbit altitude were to be maintained. Beyond this, acceleration forces from thrusting to regain lost altitude would be a constraint on the design of solar cell panel structure and possibly panel orientation control mechanisms. No propulsion interfaces would be involved with the use of a fuel cell power system.

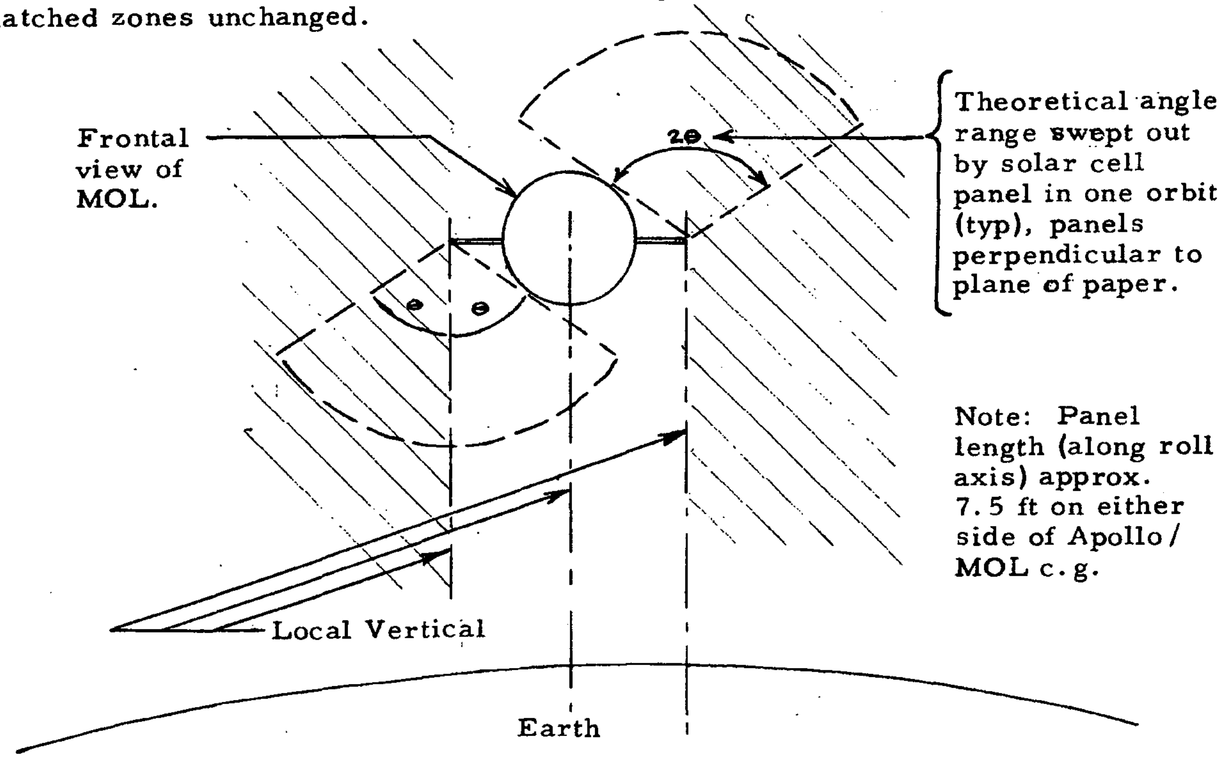
f. Communication - The requirement for broad beam antenna patterns from Apollo/MOL to ensure adequate ground coverage guarantees communications interference from solar cell panels. The degree of interference would vary with panel orientation and could result in periodic waxing and waning of signal strength. How severe this interface problem would actually be cannot be assessed at this time. A fuel cell system would not interface with communications.

g. Rendezvous and Docking -Close-in maneuvers by a ferry vehicle to achieve rendezvous and docking with an Apollo/MOL spacecraft would almost certainly be influenced by the presence of large fragile solar panels extending 15 to 20 feet from either side of Apollo/MOL. It is not possible at this time to estimate reliably how rendezvous and docking might be constrained by such influence or the importance of this interface. However, no such interface would exist in this area for fuel cells.

[redacted]

5. Experiments Interference

Probably the most conclusive factors against the use of solar cell panels for Apollo/MOL arise from considerations of experiment interference. A sketch is shown below of the Apollo/MOL spacecraft with solar cell panels and the range of panel positions which would occur as the panels are oriented for maximum insolation. Panel size and spacecraft size are in proper proportion in the sketch for about 1.5 kilowatts average power and fully oriented panels (roll-only orientation of panels would necessitate panels almost twice as large as shown). Note that the theoretical angle range (2θ) swept out by each solar cell panel is symmetric about local vertical (neglecting solar eclipse). For purposes of illustration in the sketch, it is convenient to assume that the panels are parallel to the flight direction (roll axis). However, the same basic panel motion will result from full panel orientation in which such parallelism is not maintained. Neglecting eclipse affects would give a maximum value for θ of 90 degrees. Considering that panel orientation is not required during solar eclipse periods limits the maximum value of θ in the non-cross-hatched zones to 50 to 60 degrees while leaving θ in the cross-hatched zones unchanged.



[redacted]

Experiment interference for each of the experiments currently in planning for Apollo/MOL is considered below in context with the above sketch.

Experiments P1, P2, and P3 - [redacted]

Experiment P4 - This experiment may involve a five- to eight-foot diameter parobolic receiving antenna deployed immediately beneath the Apollo/MOL spacecraft. Both physical interference and signal interference from a solar panel would be a problem with this antenna.

Experiment P5 - Deleted

Experiment P6 - Extravehicular activity is the subject of this experiment and would involve an astronaut leaving via an escape hatch in the top of the spacecraft. The presence of an overhead solar panel could interfere with such an operation.

Experiment P7 - [redacted]

Experiment P8 - Autonomous navigation by the Apollo/MOL crew is attempted in this experiment. At least one facet of this experiment which would be confronted with interference by solar cell panels is manual star "shooting" through a view port in the top of the spacecraft.

Experiment P9 - Maintenance - Undefined.

Experiment P10 - A manually directed radiometer would be pointed towards earth in this experiment both at preplanned targets and at targets of opportunity. Obscuration of the field of view of the radiometer would be a problem with solar cell panels for this experiment.

Experiment P11 - General (astronaut) performance in a military space system - no interference.

Experiment P12 - Biomedical - no interference

A fuel cell power system poses no experiment interference problems for Apollo/MOL.

[redacted]

6. Emergency Power

Power system sizing was accomplished on the basis of essentially equivalent emergency power provisions for both the fuel cell and solar cell system. No advantage is believed to accrue to either power system in this area.

7. Crew Safety

Neither power system type possesses features that would significantly affect crew safety.

8. Crew Demands

It was pointed out in Section 4 that a substantial amount of crew time and effort would be involved in activating a solar cell power system. This is the only unique demand on crew time that can be foreseen. All other demands such as periodic power system check-out should be about the same for both systems. This assumes that solar paddle orientation is fully automated throughout the mission and that the fuel cells purge themselves automatically.

9. Complexity

Types of failures and failure modes are difficult to predict even in the late stages of mission development. Failures which can result from inadequate quality control, improper handling, misuse or reasons unknown are not accounted for in most system failure models. Consequently, preliminary reliability analyses, which are necessarily based on failure models, are generally insufficient to establish relative probabilities of failure between two systems. An additional measure of failure probability is the quantity of possible failure sources, i.e., complexity. A solar cell system would contain, for example, many gear trains and mechanical linkages whose predictable probabilities of failure could be negligible. Nonetheless, they are potential failure sources whose satisfactory performances are reasonably sensitive to quality control, handling, etc. Similar comments are applicable to the valving and complex structuring in fuel cells. It is estimated that fuel cells and solar cells are about equivalent over-all relative to such complexity considerations and that arguments favoring either power system in this regard would be uncertain at best.

10. Growth

a. Different Altitudes - Fuel cell system characteristics are independent of altitude. A solar cell system penalizes Apollo/MOL from an altitude sustenance propulsion standpoint as explained earlier. Consequently, lower altitude Apollo/MOL missions would be substantially compromised by a solar cell power system in terms of smaller payload, shorter duration or in some equivalent manner due to increased propulsion requirements. On the other hand, of course, a solar cell powered Apollo/MOL could decrease its altitude sustenance propulsion for higher altitude orbits. Higher altitudes are not so attractive, however, because of the personnel shielding requirements that arise, and because of reduced resolution for observations of earth.

b. Longer Mission Duration - A solar cell system has a life capability well beyond 30 days and could perform for many months without significant design or electrical performance changes. A fuel cell power system is much more limited in this regard. Since a fuel cell system can, within bounds, trade power level for mission duration, modest (e.g., 10 to 15 days) extensions of mission duration capability could be obtained by reducing average Apollo/MOL power consumption to some appropriate level. Beyond this, reactant resupply from a ferry vehicle would be required to render a basic 30- to 40-day fuel cell design adequate for long mission extensions.

c. Different Power Levels - A fuel cell consumes reactants only as needed and at a rate roughly linear with power output. Also, they can operate over a wide range of power densities. Fuel cells readily adapt, therefore, to either lower or higher average power levels and to either lower or higher peak power levels. In every case, the volume and weight cost per unit of energy delivered is (within bounds) roughly the same. A solar cell system must be redesigned with larger solar panels to produce a higher average power output. Higher sustained peak power levels also require a redesign involving larger batteries and possibly larger solar panels. Lower power levels can be accommodated without solar cell system redesign with, of course, no reduction in power system weight.

d. Polar (Operational) Missions - Fuel cell performance is independent of orbit parameters. Similarly, a solar cell power system would produce its design power at any orbit inclination. Therefore, either type of power system is amenable to Apollo/MOL adaption to operational missions.

e. Vulnerability (Operational Missions) - An important advantage of fuel cells in an operational mission environment is their much superior vulnerability characteristics. Solar cell panels would increase the radar cross-section of Apollo/MOL by a factor of at least two or more, thereby facilitating missile tracking. Also, the power generating properties of solar cells can be readily destroyed by distant nuclear detonations. Fuel cells would have no influence on the basic Apollo/MOL spacecraft vulnerability.

11. <u>Development Risk</u>

a. Power System Equipment - Both fuel cell and solar cell technologies are sufficiently advanced to warrant their selection for an accelerated Apollo/MOL development program. It can be seen from the previous discussions, however, that a solar cell power system involves much greater complexity in operating procedures and interfaces with other subsystems.

Consequently, the number of opportunities for development troubles (i.e., development pitfalls) is estimated to be much greater for a solar cell system than for a fuel cell system. For example, large appendages housing stowed panels on the sides of Apollo/MOL may prove dynamically unsound. Also, current experience in the deployment and orientation of solar panels has proven so troublesome that a major Air Force program has abandoned oriented panels in favor of fixed panels, and is starting, for the second time, with development of conceptual deployment schemes. On the other hand, there is limited zero-g experience with both fuel cells and their associated cryogenic tankage. While there is not any serious question of the ability of either of these equipments to function in a zero-g environment, the degree to which their in-orbit performance (e.g., specific reactant consumption, reactant boiloff rate) will meet expectations will not be known for some time. Thus, in comparing solar cell risks to the fuel cell risks, the possibility of obtaining, at the worst, a few days less mission duration capability from fuel cells is preferable to the large number of uncertain design problems and interfaces, any one or all of which could be much more harmful to the Apollo/MOL mission.

b. Apollo/MOL Power Requirements - Power requirements for an electrically complex system such as Apollo/MOL become accurately defined late (if ever) in the design of the spacecraft. At this conceptual-planning stage for Apollo/MOL, it is a certainty that the eventual power requirements for Apollo/MOL will be importantly different from current power profile projections, with a probable trend to higher power levels. Solar cell power systems have a fixed power generating capacity once their design is frozen. Higher (average) power from them requires redesign with resultant reverberations through all of its interfaces. To avoid this, either a highly conservative solar cell system design would have to be employed initially, the design would somehow have to be kept flexible until late in development, or power requirements would have to be frozen early in development. None of these undesirable design constraints is necessary with a fuel cell system which simply consumes reactants faster if more power is drawn from it. A conservative cooling design in the fuel cell units plus a trade-off of mission duration for higher average power requirements on a one-for-one basis causes power profile variations to be of only modest design significance for fuel cell powered spacecraft.

TABULAR SUMMARY

Table 5-7 in the main body of the report summarizes the results of the foregoing comparisons. No attempt was made at assigning relative levels of importance to the various comparison factors; some should undoubtedly be considered more important than others in selecting the Apollo/MOL power system.

CONCLUSIONS

It is concluded from the foregoing discussions and the revealing summary of Table 5-7 that a fuel cell power system for Apollo/MOL has an overwhelming number of advantages in comparison to a solar cell system. Taken one at a time, each solar cell system disadvantage probably could be conceptually circumvented or judged tolerable. Viewed en masse, as in this appendix, such arguments become rather academic to the over-all preponderance of practical advantages that would accrue to Apollo/MOL from the selection of a fuel cell power system.

[redacted]

REFERENCES

5-1. Buglia, J. J. and J. F. Newcomb, "A Method for Determining the Propellant Required to Maintain an Earth Orbital Vehicle in a Useful Orbit Between Two Specified Orbits," NASA TN D 1817, August 1963.

[redacted]

[redacted]

SECTION 6

LIFE SUPPORT

C. C. Wright

SUMMARY

A study was conducted of the life support subsystems for various Apollo/MOL vehicle configurations. The weights, volumes and power requirements of two candidate systems were estimated and compared with those of the Gemini B/MOL. In addition, the life support system characteristics, location of equipment in the vehicle, compartment volumes and major features for each vehicle configuration were identified and compared.

The two candidate systems selected for study were (1) lithium hydroxide method of CO_2 removal with water supply from the fuel cells and (2) molecular sieve method of CO_2 removal system with water reclamation and a solar cell power supply system. The cross-over point between these two systems for two men at an orbit of 160 nautical miles (with altitude sustenance and one repressurization) is about 35 days. The lithium hydroxide CO_2 removal system has the lowest weight for mission durations less than 35 days whereas the molecular sieve CO_2 removal system is lighter for mission durations greater than 35 days. There are no significant differences for the life support and environmental control systems between the Gemini/MOL and Apollo/MOL.

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TABLES

ILLUSTRATIONS

6.1 INTRODUCTION

A study was conducted of the life support subsystem characteristics of two candidate systems for various possible Apollo/MOL configurations. Weights, volumes, and power requirements were estimated and compared with those of the Gemini B/MOL. Life support subsystem characteristics, location of equipment, compartment volumes and major features for each configuration were also identified and compared.

In accordance with the Apollo/MOL study ground rules, the various vehicle configurations and crew requirements that were studied are as follows:

1. Basic Apollo, three men for 14 days
2. NAA Concept II, two men for 30 days
3. Semi-optimum NAA Concept II, two men for 30 days
4. Semi-optimum NAA Concept II, two men for 120 days
5. Semi-optimum NAA Concept II, three men for 120 days
6. Aerospace Recommended Configuration, two men for 30 days
7. Gemini B/MOL, two men for 30 days
8. Gemini B/MOL, two men for 90 days

The vehicle configuration for the North American Aviation Concept II consists of the Apollo command module, the Apollo service module and a large pressurized laboratory module in the LEM adapter. The atmospheric control system is located on the command module. This configuration is based on the NAA extended mission Apollo Study (Reference 6-1) and is described in Section 1. The semi-optimum configurations (Section 1) have shorter service modules and smaller laboratory modules. In the Aerospace recommended configuration (Section 1) the service and laboratory modules are combined, the cryogenic storage tanks and propulsion system are relocated in the aft section, and the laboratory contains its own environmental control and life support subsystem similar to the Gemini B/MOL. All Apollo/MOL configurations use the Saturn IB launch vehicle, whereas the Gemini B/MOL uses the Titan-IIIC launch vehicle.

The two candidate EC/LS subsystems selected for analysis are (1) lithium hydroxide type of CO_2 removal system with water supply from the fuel cells and (2) molecular sieve type of CO_2 removal system with water reclammation and a solar cell power supply system.

In the first system, the CO_2 expired into the air by the astronauts is removed by a chemical reaction in LiOH beds. No regeneration is possible, and the beds must be replaced with a fresh charge at regular intervals. In the second system, the CO_2 is removed from the air, which is first dried with silica gel, by absorption on the surface of zeolite granules called molecular sieves. Here, regeneration (desorption) of the beds is possible by periodic exposure to the vacuum of space. Two silica gel-molecular sieve units operating cyclically in parallel are required so that while one unit is drying the air and absorbing CO_2, the other unit is being desorbed by CO_2 and water. Although the CO_2 is lost to space, the water, which was absorbed by the silica gel bed ahead of the molecular sieve, is returned to the purified air stream.

6.2 BASIC WEIGHT AND BULK DENSITY DATA

The weights and volumes of the various fixed and variable items of the Apollo/MOL life support subsystems may be estimated from the basic weight and bulk density data shown in Table 6-1, which was obtained or adapted from various sources, principally References 6-1 and 6-2 and the Gemini/MOL program.

6.3 POWER REQUIREMENTS AND WEIGHT PENALTY

The estimated power requirements for the various Apollo/MOL study configurations are summarized in Table 6-2. These numbers are crude estimates, but they should be adequate for preliminary comparison purposes.

According to Section 5, the total weight power penalty for present improved version fuel cells in the 1.5- to 2.0-kilowatt range is

W_{fc} = sum of the fixed power system weight, replacement weight, and variable weight, lb

$$W_{fc} = 500 + 200\ PN + 34.8\ PD \text{ for } 1.5<2.0 \text{ kilowatts} \quad (1)$$

Table 6-1. Basic Weight and Bulk Density Data
Apollo/MOL Life Support Subsystems.

Item	Weight	Bulk Density
LiOH CO_2 removal granules, charcoal filters and canisters	3.0 lb/man-day	25 lb
Metabolic oxygen	2.0 lb/man-day	-----
Cabin leakage (assumed)	2.0 lb/day at 5 psia 2.8 lb/day at 7 psia	-----
Repressurization(1)	144 PV/RT lb	-----
Cryogenic oxygen tankage (supercritical at 1500 psia)	0.48 lb/lb fluid	0.0175 ft^3/lb O_2 (inc. 5% ullage)
Food (freeze dried, including storage containers)	1.5 lb/man-day	45 lb/ft^3
Sanitation and clothing	0.58 lb/man-day	40 lb/ft^3
Fixed Hardware (2 - 3 men): LiOH CO_2 Removal System	390 lb(2)	35 lb
Fixed Hardware (2 - 3 men): Molecular Sieve CO_2 Removal System	500 lb(2)	
Fixed Hardware (2 - 3 men): Water Reclamation System	50 lb	

Notes:

1. P = cabin pressure, psia
   V = total volume to be pressurized, ft^3
   R = gas constant (for O_2 = 48.3 ft#/lb-R)
   T = cabin temperature (nominal value = 530 R)
2. In all Concept II versions of Apollo/MOL an additional 225 lb of thermal control equipment should be added to the laboratory module. Atmospheric control is accomplished with the CO_2 removal equipment in the command module plus booster fans and ducts.

Table 6-2. Estimated EC/LS System Power Requirements, Apollo/MOL Study.

EC/LS System	LiOH CO_2 Removal, Fuel Cell Power			Molecular Sieve CO_2 Removal, Solar Cell Power		
Vehicle System	Apollo (3 men)	Apollo/MOL(1) (2 men)	Gemini B/MOL (2 men)	Apollo/MOL (2 men)	Apollo/MOL (3 men)	Gemini B/MOL (2 men)
Air circulation	60	70 (50)	50	70	95	50
Temperature control	40	10	10	10	10	10
Transport fluid pump	60	80 (60)	60	80	80	60
Lighting	80	200 (160)	120	200	200	120
Instrumentation	30	50	50	50	50	50
Suit compressor	105	40	40	40	40	40
Thermal pressurization of cryogenic tanks	15	10	10	10	15	10
Waste management	--	30	30	30	45	30
Water management	--	20	20	20	30	20
Urine reclamation	--	--	--	50	75	50
Molecular sieve system	--	--	--	150	225	150
Contaminant removal	--	--	--	20	30	20
Total Power, watts	390	510 (430)	390	730	895	610

(1) Figures in parentheses refer to Aerospace recommended version.

(2) All figures are in watts.

where

P = average power level, kw

$\bar{N}$ = number of 30-day periods

D = mission duration, days

Assuming a total average power level of 1.77 kilowatts (which is typical for the MOL mission), the weight penalty chargeable to the EC/LS system for a fuel cell power supply is

$$
\begin{aligned}
W_{fc} &= (P_{ec/ls}/P_{1.77})\ (500 + 200\ PN + 34.8\ PD) \\
&= P_{ec/ls}\ (283 + 200\ N + 34.8\ D)
\end{aligned}
\tag{2}
$$

where

$P_{ec/ls}$ = the power required for the EC/LS system

D = mission duration, days

In the case of solar cell power, the weight power penalties for an average power level of 1.77 kilowatts (Section 5) are:

1. 125 nautical miles with altitude sustenance:

   W_{SC} = sun fixed solar cell system weight plus variable weight

   $$W_{SC} = 200 + 18.2\ D \quad \text{for}\quad P = 1.77\ \text{kw} \tag{3}$$

2. 200 nautical miles with altitude sustenance:

   $$W_{SC} = 1968 + 1.8\ D \quad \text{for}\quad P = 1.77\ \text{kw} \tag{4}$$

The weight penalty chargeable to the life support system is $P_{ec/ls}/1.77$ times the above values.

6.4 WEIGHT VERSUS MISSION DURATION

The total weight of the life support subsystem may be computed from the data in Tables 6-1 and 6-2 together with the appropriate power penalty equation. Assuming a 10 per cent allowance for expendable reserves and an additional

10 per cent allowance for cryogenic residuals, the weight versus mission duration of the two candidate life support subsystems may be computed as follows:

6.4.1 LiOH CO_2 Removal System, 5.0 Psia Oxygen Atmosphere, Fuel Cell Power and Water Supply

$$W = W_{fixed} + 1.10\ (W_{LiOH} + W_{food} + W_{sanitation\ and\ clothing}) + 1.20\ (W_{O_2} + W_{tankage}) + W_{power\ penalty} \quad (5)$$

Assuming one complete repressurization (2800 ft^3), the EC/LS weights for the NAA Concept II version of the Apollo/MOL are:

W_{fixed} = referring to Table 6-1, the fixed hardware weight in command and laboratory modules is = 390 + 225 = 615 lb

W_{LiOH} = 3.0 MD, lb

W_{food} = 1.5 MD, lb

$W_{sanitation\ and\ clothing}$ = 0.58 MD, lb

W_{O_2} = combined weight of metabolic oxygen, leakage and repressurization gas for 2800 ft^3 at 5.0 psia

$2.0\ MD + 2.0\ D + 144 \times 5 \times 2800/48.3 \times 530$

$W_{O_2} = 2.0\ MD + 2.0\ D + 78.7$ lb

$W_{tankage}$ = tankage chargeable to the EC/LS system

$= 0.48 \times W_{O_2} = 0.96\ MD + 0.96\ D + 37.8$ lb

$W_{power\ penalty} = 0.510\ (283 + 200\ N + 34.8\ D) = 144 + 102\ N + 17.75\ D$

Substituting these weights into Equation (5), the total weight is:

$$W = 615 + 1.10 \times 5.08 \times MD + 1.20\ (116.5 + 2.96\ MD + 2.96\ D) + 144 + 102\ N + 17.75\ D$$

Combining similar terms, the total weight of the LiOH CO_2 removal EC/LS subsystem, including power penalty, is

$$W = 899 + 102\ N + 9.14\ MD + 21.3\ D\ lb \qquad (6)$$

where

D = mission duration, days

M = number of men

N = number of 30-day periods (maximum useful life of fuel cells because of reliability consideration is 30 days)

For 0 to 30 days, N = 1; for 30 to 60 days, N = 2, etc.

A plot of Equation (6) for a two-man system is shown in Figure 6-1.

6.4.2 Molecular Sieve CO_2 Removal System, 7 Psia Oxygen/Nitrogen Atmosphere, Solar Cell Power With Water Reclamation

In this case, the various itemized weights are:

W_{fixed} = fixed hardware weight in command and laboratory modules 500 + 225 = 775 lb

W_{LiOH} = emergency weight for two days = 3 x 2 x M lb

W_{food} = 1.5 MD lb

$W_{\text{sanitation and clothing}}$ = 0.58 MD lb

$$W_{O_2} = 2.0\ MD + 2.8\ D + (144 \times 7 \times 2800)/(51.5 \times 530)$$

$$W_{O_2} = 2.0\ MD + 2.8\ D + 103.5\ lb$$

$$W_{tankage} = 0.48\ W_{O_2} = 0.96\ MD + 0.96\ D + 49.7\ lb$$

Referring to Equation (3) and Table 6-2, the power penalty for a 125-nautical mile orbit with altitude sustenance is:

$$W_{\substack{\text{power penalty}\\ \text{125 nautical miles}}} = (0.73/1.77)\ (2004 + 18.2\ D) = 828 + 7.51\ D$$

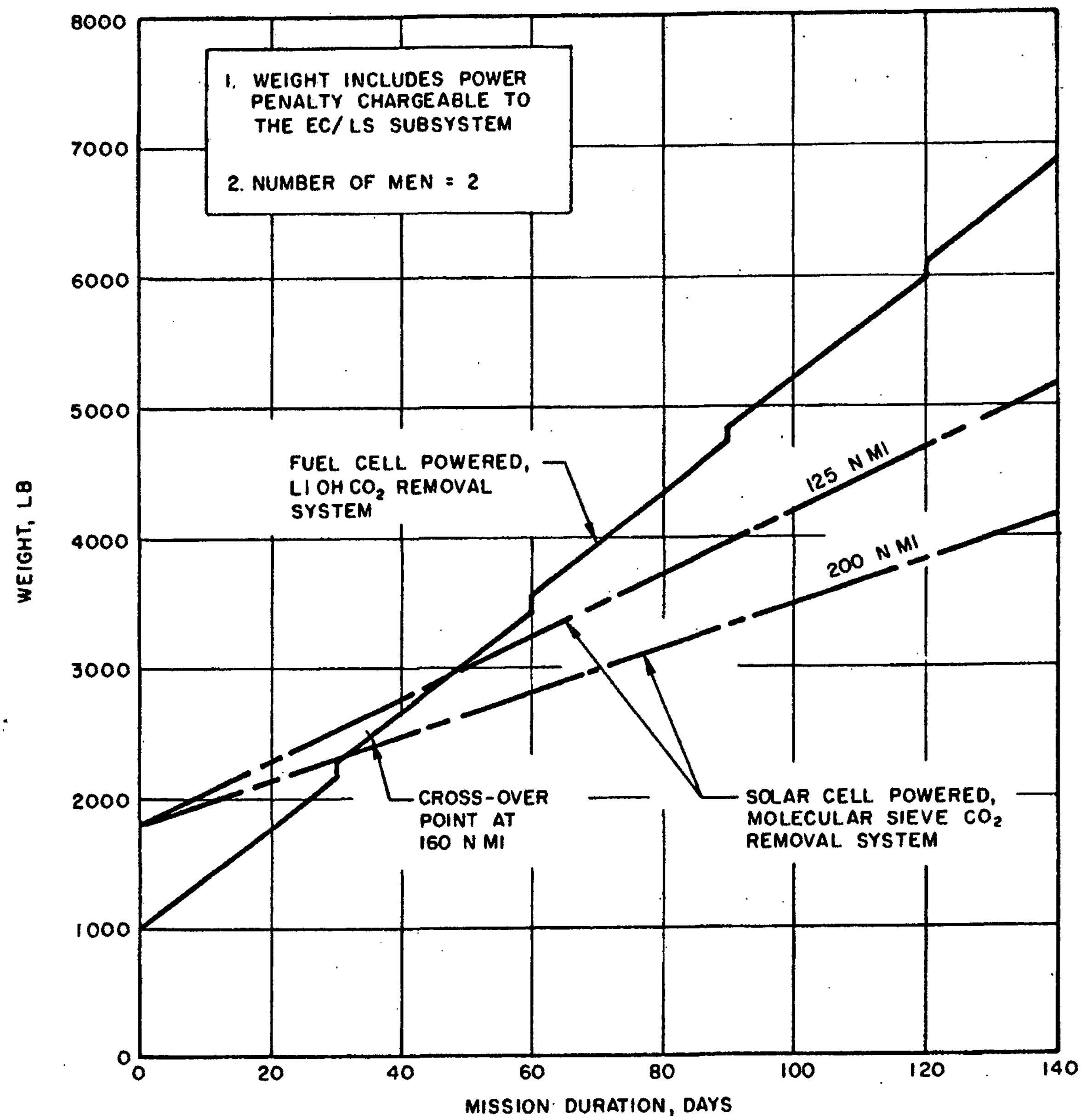

Figure 6-1. Life Support Subsystem Weight Versus Mission Duration.

Referring to Equation (4), the power penalty for a 200-nautical mile orbit with altitude sustenance is

$$W_{\substack{\text{power penalty}\\ \text{200 nautical miles}}} = (0.73/1.77)\ (1968 + 1.80) = 812 + 0.74\ D$$

Combining similar terms in accordance with Equation (5), the total weight of the molecular sieve EC/LS systems, including power penalty, is

$$W_{125\ \text{naut mi}} = 1787 + 6\ M + 5.84\ MD + 12.48\ D\ \text{lb} \tag{7}$$

$$W_{200\ \text{naut mi}} = 1771 + 6\ M + 5.84\ MD + 5.71\ D\ \text{lb} \tag{8}$$

where

D = mission duration, days

M = number of men

A plot of Equations (7) and (8) for a two-man system is shown on Figure 6-1. Inspection of this figure reveals that the cross-over point between a fuel cell powered, lithium hydroxide CO_2 removal system with water supply from the fuel cells and a solar cell powered, molecular sieve CO_2 removal system with water reclamation is between 32 and 46 days for 200-nautical mile and 125-nautical mile orbits, respectively. The cross-over point at an orbit of 160 nautical miles with solar cell altitude sustenance is about 35 days.

6.5 COMPARISON OF LIFE SUPPORT SUBSYSTEM DATA

A comparison of the weights, volumes and power requirements of the life support subsystem for each study configuration is shown in Table 6-3. In addition, the life support system characteristics, location of equipment in the vehicle, compartment volumes, and major features are identified and compared. In accordance with Figure 6-1, which shows the cross-over point between the two candidate life support systems at about 32 to 46 days, all configurations having a 30-day mission were assumed to be supplied with a fuel cell powered, LiOH CO_2 removal system whereas all configurations having a 90- or 120-day mission were assumed to be supplied with a solar cell powered, molecular sieve CO_2 removal life support system.

Inspection of Table 6-3 reveals the over-all characteristics, similarities and dissimilarities of the life support systems of each configuration. Note that the Aerospace recommended version of the 30-day, two-man Apollo/MOL is very similar to the 30-day, two-man Gemini B/MOL, except for cabin volume.

The differences in the life support subsystem data between the various Apollo/MOL configurations in Table 6-3 may be explained as follows:

6. 5. 1 Vehicle Configuration, Mission Duration, and Number of Men

Specified as a study ground rule.

6. 5. 2 Power Supply

See Section 5. A fuel cell power supply was selected for the 30-day missions, whereas a solar cell or [redacted] power supply was selected for the 120-day missions.

6. 5. 3 Atmosphere

Selecting an atmosphere involves trade-off considerations between the man subsystem, the environmental subsystem, and the power system. The constraints are (1) physiological considerations, (2) mission safety, (3) reliability, (4) hardware availability, (5) thermal management consideration, and (6) vehicle penalty comparisons.

In order to satisfy man's physiological requirements, the alveolar oxygen partial pressure must be greater than about 100 mm Hg in order to prevent hypoxia. The corresponding inspired oxygen partial pressure should be about 185 mm Hg. This number represents present space suit conditions and is a good basis for the transition from a one- to two-gas atmosphere. Also required is a water partial pressure of about 3 to 28 mm Hg and a carbon dioxide partial pressure of about 1 to 8 mm Hg. A 260 mm Hg (5 psia) oxygen atmosphere is probably satisfactory for the 30-day MOL mission, but longer missions may be detrimental because of oxygen toxicity.

An inert gas, such as nitrogen, helium or neon, may also be beneficial. It is also thought that man may, in some unknown way, require nitrogen for body chemical reactions. However, there may be difficulties with the use of

Table 6-3. Comparison of Life Support Subsystem Data
Apollo/MOL Study.

Program	Apollo	Apollo/MOL					Gemini B/MOL	
Vehicle Configuration	Basic Apollo	NAA Concept II	Semi-Optimum Versions of NAA Concept II			(A) Recommended	Basic	Variant
Mission Duration and Number of Men	14 days, 3 men	30 days, 2 men	30 days, 2 men	120 days, 2 men	120 days, 3 men	30 days, 2 men	30 days, 2 men	90 days, 2 men
Power Supply	Fuel cell	Fuel cell	Fuel cell	[redacted]		Fuel cell	Fuel cell	Solar cell
Atmosphere — Composition	1 gas ~ 100% O_2	1 gas ~ 100% O_2	1 gas ~ 100% O_2	2 gas ~ 50% O_2 50% N_2	2 gas ~ 50% O_2 50% N_2	1 gas ~ 100% O_2	1 gas ~ 100% O_2	2 gas ~ 46% O_2 54% N_2
Atmosphere — Total Pressure, psia	5	5	5	7	7	5	5	7
Compartment Volumes, ft^3 total and (free) — Pressurized — Command Module	366 (215)	366 (300)	366 (300)	366 (300)	366 (300)	366 (300)	(188 inc. tunnel)	(188 inc. tunnel)
Pressurized — Laboratory Module	--------	5620 (5000)	2900 (2500)	2900 (2500)	2900 (2500)	2600 (2200)	1200 (1000)	1200 (1000)
Unpressurized — Service Module	1670	1670	400	400	400	----------	TM 400	TM 400
Unpressurized — Laboratory Module	6240 (Lem ad'r)	620	680	680	680	2960	580	580
Location of Equipment — Thermal Control Hardware	CM	CM and LM	CM and LM	CM and LM	CM and LM	CM and LM	Gemini B and LM	Gemini B and LM
CO_2 Removal Equipment	CM (26 canstrs under seats)	CM	CM	CM	CM	CM and LM	Gemini B and LM	Gemini B and LM
Cryogenic Tankage	SM	SM	SM	SM	SM	LM (APC)	LM (APC)	LM (APC)
Radiator	Skin of SM	Skin of SM and LM	Skin of SM and LM	Skin of SM and LM	Skin of SM and LM	Skin of LM	Skin of TM and LM	Skin of TM and LM
Method of CO_2 Removal	LiOH	LiOH	LiOH	Molecular sieves	Molecular sieves	LiOH	LiOH	Molecular sieves
Water Supply	Fuel cell	Fuel cell	Fuel cell	Reclaimed	Reclaimed	Fuel cell	Fuel cell	Reclaimed
Contaminant Control	Charcoal filter	Charcoal filter	Charcoal filter	Char. filter, Cat. burner	Char. filter, Cat. burner	Charcoal filter	Charcoal filter	Char. filter Cat. burner
Leakage Rate, Pound per Day	2	2	2	2.8	2.8	2	2	2.8
System Weights, lb (Includes 10% Reserves and 10% Residuals) — LiOH, Filters and Canisters	139	198	198	13	20	198	198	13
Metabolic oxygen	101	144	144	576	864	144	144	432
Leakage	34	72	72	403	403	72	72	302
Repressurization Gas	7	168	89	111	111	78	141	198
Cryogenic Tankage (1)	72	184	147	523	661	141	171	448
Food (Freeze Dried)	69	99	99	396	593	99	99	297
Sanitation and Clothing	27	38	38	153	230	38	38	115
Fixed Hardware	390	615	615	775	775	690	690	850
Total System Weight (2)	739	1518	1402	2950	3657	1460	1553	2655
System Volumes, ft^3 — LiOH, Filters and Canisters	5.6	7.9	7.9	0.5	0.8	7.9	7.9	0.5
Cryogenic Tankage (1)	2.6	6.7	5.3	19.1	24.1	5.1	6.2	16.3
Food	1.5	2.2	2.2	8.8	13.2	2.2	2.2	6.6
Clothing and Sanitation	0.7	1.0	1.0	3.8	5.8	1.0	1.0	2.9
Fixed Hardware	11.1	17.6	17.6	22.1	22.1	19.7	19.7	24.3
Total System Volume (2)	21.5	35.4	34.0	54.3	66.0	35.9	37.0	50.6
Power Requirements, Watts	390	510	510	730	895	430	390	610

Notes: (1) EC/LS and fuel cell oxygen are integrated into a common storage system. The listed tankage weights and volumes are those chargeable to the EC/LS system.

(2) Includes weight or volume of EC/LS in both command and laboratory modules.

(3) Number of repressurizations during mission: Apollo/MOL, one; Gemini/MOL, one + 6 partials.

nitrogen as a diluent because of the possibility of the "bends" occurring when changing from a higher pressure two-gas system to a 185 mm Hg "pure oxygen" suit system. Helium may be physiologically more attractive than nitrogen.

In regard to mission safety, both explosion and fire hazards must be considered. The possible hazard caused by explosions is reduced (the minimum spark ignition energy for explosive combustion of gas mixtures is increased) and the rate of consumption of materials in the case of fire is decreased by (1) lowering the oxygen partial pressure while maintaining the total pressure constant, or (2) by raising the total pressure while maintaining the oxygen partial pressure constant. Combustible mixtures can also be reduced or eliminated by using a catalytic burner contaminant control.

Low fire hazard materials have a high thermal conductivity, a high ignition temperature, and a low heat of reaction with oxygen. Experiments conducted by F. A. Parker of General Electric (Reference 6-3) showed that stainless steel can burn in an atmosphere of 85 per cent O_2 and 13 per cent N_2 at a total pressure of 750 mm Hg. Aluminum is a better material because of its higher thermal conductivity (heat is conducted away from reaction zone, thus reducing the heat required to sustain the reaction) and its tough oxide coating. It should be noted, however, that materials that burn well in a free convection environment (under the influence of gravity) may not be capable of a self-sustaining reaction in a zero gravity environment because the products of combustion tend to dilute the reaction constituents. The result is the material temperature at the reacting surface may drop below the self-sustaining level. In general, a mixed gas atmosphere (with inert mixing gases) is safer than a pure oxygen atmosphere, a lower oxygen partial pressure atmosphere is safer than a higher partial pressure atmosphere, and the possibility of a fire, especially an uncontrollable fire, can be minimized by utilizing low fire hazard materials.

Reliability considerations indicate that, on the basis of hardware requirements, a mixed gas atmosphere tends to have a slightly lower reliability than a pure oxygen atmosphere. Nevertheless, a two-gas atmosphere can be made just as reliable if sufficient effort is expended. In fact, the over-all reliability of a combined man-equipment system may be higher with a properly selected mixing gas atmosphere than with a pure oxygen atmosphere. Also, during an emergency mode of operation both mixed gas and pure oxygen atmospheres have the same reliability.

From the point of view of available hardware, there appears to be no compelling reason for selecting one type of atmosphere over another. Even helium-oxygen atmospheres are within present state-of-the-art capability. Thermal management considerations indicate that the order of preference is O_2/He, O_2/N_2 and pure O_2 atmospheres. Finally, over-all vehicle considerations indicate that the desired pressure level range for minimum vehicle weight is about 5 to 9 psia with the minimum occurring at about 7 psia. Below 5 psia the pumping power penalty becomes excessive and above 9 psia the structural weight is too large.

Although the type of atmosphere for the Gemini/MOL vehicle has not been established, the present industry EC/LS study contracts are based on a 5 psia one-gas atmosphere (pure oxygen) for 30-day missions and a 7 psia two-gas atmosphere (46 per cent O_2 and 54 per cent N_2) for 90-day missions. Other atmospheres are also possible. For example, a recent preliminary qualitative study indicates that the optimum atmosphere for the MOL mission is a 5 psia, O_2He mixture having a composition of 165 mm Hg of O_2, 75 mm Hg of He, 15 mm Hg of H_20 and 5 mm Hg of CO_2.

For purposes of the Apollo/MOL study the atmosphere for 30 days or less was assumed to be 5.0 psia "pure oxygen", whereas for longer durations the atmosphere was assumed to be a 7 psia, two-gas atmosphere consisting of 50 per cent O_2 and 50 per cent N_2. Except for leakage and repressurization requirements, variations from the above assumptions will not significantly effect the weight and volume estimations presented in Table 6-3.

6.5.4 Compartment Volumes

Compartment volumes were estimated from the sketches of Reference 6-1 and Section 1. The difference between the free volumes of the command module in the basic Apollo and Apollo/MOL configurations is due to the electronic equipment and electronic cold plates associated with the Apollo mission. Much of this hardware is not necessary for the Apollo/MOL mission.

6.5.5 Location of Equipment

The equipment location becomes fairly obvious by referring to Reference 6-1 and Section 1. It should be noted, however, that the CO_2 removal equipment in the semi-optimum versions of Apollo/MOL is located in the command module. In the Aerospace recommended version, CO_2 removal equipment is also located in the laboratory module. This eliminates the need for ducts and blowers (plus interface connecting fittings at the junction between the command and laboratory modules) for circulating the atmosphere between the two compartments.

6.5.6 Method of CO_2 Removal and Water Supply

Inspection of Figure 6-1 reveals that the weight cross-over point between the two candidate combined life support-power systems is about 35 days at an altitude of about 160 nautical miles. The use of [redacted] power would give about the same cross-over point as fuel cells. Using this as a criteria for selection, all 30-day or less missions would use lithium hydroxide CO_2 removal equipment with a fuel cell water supply. On the other hand, all missions greater than 30 days would use a molecular sieve CO_2 removal system with water reclamation. Water reclamation is required in the latter case because water is not produced as a by-product from the power supply as in the case with fuel cells.

6.5.7 Contaminant Control

For mission durations greater than 30 days, it appears as though catalytic burners will be required to prevent excessive accumulation of contaminants.

6.5.8 Leakage Rate

The figure of two pounds per day for a 5 psia atmosphere was assumed to be the same as for the Gemini B/MOL EC/LS study. At higher pressure levels the leakage rate is directly proportional to the pressure.

6.5.9 System Weights, Volumes and Power Requirements

These items are adequately explained in the sections of this memorandum on weight, bulk density and power requirements. All of the data was either computed or estimated from the reference reports. Note that all variable weights (food, oxygen and tankage, sanitation and clothing) are directly proportional to the number of men and mission durations. Also, the power requirements include lighting requirements.

6.6 CONCLUSIONS

1. The cross-over point between a fuel cell powered, LiOH type of carbon dioxide removal system with fuel cell supplied water and a solar cell powered, molecular sieve type of carbon dioxide removal system with water reclamation is about 35 days at a 160-nautical mile circular orbit. Missions less than 35 days in length should use the LiOH type of CO_2 removal system whereas missions of longer length should use the molecular sieve type of CO_2 removal system.

2. The life support subsystem in the Aerospace Corporation recommended version of the 30-day, two-man Apollo/MOL is very similar to the 30-day, two-man Gemini B/MOL.

3. There are no significant differences for the life support and environmental control systems between the Gemini/MOL and the Apollo/MOL.

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SECTION 7.

PERFORMANCE

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SUMMARY

Abort boundaries for the Apollo re-entry vehicle have been computed for use in ascent flight performance calculations. The boundaries were computed for re-entry vehicle weights of 9000 and 12,000 pounds with lift-to-drag ratios of 0.4 and 0.5 and a maximum total load factor of 13 g's. The four boundaries which result are presented in graphical form.

A parametric study of re-entry trajectories for the Apollo re-entry vehicle is presented. Deboost velocity increments of 500 and 1500 feet per second, re-entry vehicle weights of 9000 and 12,000 pounds, and lift-to-drag ratios of 0, 0.25, and 0.5 are used. The results are plots of velocity, altitude, and density versus time.

The ascent performance of an Apollo/MOL system, using the Titan IIIC and Saturn IB launch vehicles, has been evaluated. The Apollo abort recovery ceiling and re-entry performance have also been evaluated using the parametric studies of Sections 7.1 and 7.2. The results are presented in graphical form.

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7. PERFORMANCE

7.1 PRELIMINARY APOLLO/MOL RE-ENTRY VEHICLE ABORT BOUNDARIES

7.1.1 Introduction

Abort boundaries corresponding to a maximum total load factor of 13 g's for the Apollo re-entry vehicle are presented in this section. Re-entry vehicle weights of 9000 and 12,000 pounds were used with lift-to-drag ratios of 0.4 and 0.5. The computed boundaries are shown in the accompanying plots.

7.1.2 Analysis

The two re-entry vehicle weights used here are the expected upper and lower limits of the weight for the Apollo/MOL configuration. The two lift-to-drag ratios are representative of the aerodynamic capabilities of the configuration.

The following data and assumptions were used in computing these boundaries:

1. Spherical, rotating earth; $R_E = 20.9029 \times 10^6$ ft, $g_o = 32.2284$ ft/sec^2
2. ARDC 1959 standard atmosphere
3. Initial latitude = 32 degrees, Initial azimuth = 90 degrees
4. Constant mass (i.e., no inert weight loss due to ablation)
5. Neither total heat nor heating rate were considered.
6. Aerodynamic data (taken for Mach number 4.65)

   $L/D = 0.4$, $C_L = 0.46$ = constant
   $C_D = 1.15$ = constant

   $L/D = 0.5$, $C_L = 0.505$ = constant
   $C_D = 1.01$ = constant

   Reference area = 133 ft^2

7.1.3 Results

Figure 7-1 illustrates the abort boundaries for both 9000 and 12,000 pounds with a L/D of 0.5. The boundaries corresponding to the L/D of 0.4 with the same weights are shown in Figure 7-2.

These apogee boundaries represent upper limits which must not be exceeded by the ascent flight instantaneous apogees to ensure that the re-entry vehicle will not exceed the stated load factor during re-entry. The instantaneous apogee during the ascent results from the current booster velocity plus the velocity increment due to the abort rocket if it is assumed that the re-entry vehicle trajectory is not affected by the atmosphere between separation from the booster and attainment of the maximum altitude. Note that the ascent flight instantaneous apogee could be lowered by executing a pullover maneuver prior to exit from the atmosphere. The lower bound on the plots is merely the constant dynamic pressure line corresponding to the stated limit load factor.

7.2. PRELIMINARY APOLLO/MOL RE-ENTRY TRAJECTORIES

7.2.1 Introduction

Re-entry trajectories of the Apollo re-entry vehicle from a 160-nautical mile circular orbit are presented in this report. The trajectories were obtained for deboost velocity increments of 500 and 1500 ft/sec using re-entry vehicle weights of 9000 and 12,000 pounds. Lift-to-drag ratios of 0, 0.25, and 0.5 were used. The time histories of velocity, altitude, and density are presented in graphical form for use in conducting heating analyses on the twelve trajectories.

7.2.2 Analysis

The two re-entry vehicle weights used here are the expected upper and lower limits of the weight for the Apollo/MOL configuration. The three values of lift-to-drag ratios are taken to represent the vertical components of L/D where the vertical lift is modulated by banking the vehicle. In each of the cases, the drag corresponds to a total L/D of 0.5. All of the trajectories were initiated by the application of a straight-back retro-thrust to deboost from the 160-nautical mile circular orbit. Velocity increments due to retro-thrusts of 500 and 1500 ft/sec were used here.

The following data and assumptions were used in computing these boundaries.

1. Spherical, rotating earth; $R_E = 20.9029 \times 10^6$ ft

2. ARDC 1959 standard atmosphere; no atmospheric effects above an altitude of 400,000 feet.

3. Trajectory computations were initiated at the 400,000-foot altitude and terminated at 70,000 feet.

4. Initial latitude = 32 degrees, Initial azimuth = 90 degrees

5. Constant mass (i.e., no inert weight loss due to ablation)

6. Neither total heat nor heating rate were considered.

7. Aerodynamic data (taken for Mach number 4.65)

   $C_D = 1.01$ = constant
   $C_L = 0$, 0.2525, and 0.505 for L/D's of 0, 0.25, and 0.5, respectively
   Reference area = 133 ft^2

7.2.3 Results

The velocity, altitude, and density time histories are presented in the accompanying plots. A summary of the input parameters is given in the following table.

Figure No.	Deboost ΔV	Weight	L/D
7-3a, b, c	500	9000	0.5
7-4a, b, c	500	9000	0.25
7-5a, b, c	500	9000	0
7-6a, b, c	500	12000	0.5
7-7a, b, c	500	12000	0.25

Figure No.	Deboost ΔV	Weight	L/D
7-8a, b, c	500	12000	0
7-9a, b, c	1500	9000	0.5
7-10a, b, c	1500	9000	0.25
7-11a, b, c	1500	9000	0
7-12a, b, c	1500	12000	0.5
7-13a, b, c	1500	12000	0.25
7-14a, b, c	1500	12000	0

For each figure number, the velocity, altitude, and density are plots a, b, and c, respectively.

7.3 PRELIMINARY APOLLO/MOL ASCENT AND RE-ENTRY PERFORMANCE

7.3.1 Introduction

Ascent and re-entry performance for an Apollo/MOL system has been evaluated and is reported in this section. The section is divided into four parts: the Titan IIIC ascent performance with an Apollo payload, the Saturn IB ascent performance, the Apollo abort recovery ceiling, and the Apollo re-entry performance.

7.3.2 Titan IIIC Ascent Performance

7.3.2.1 Method of Analysis

Titan IIIC ascent performance is based on the Revision VI definition of the launch vehicle, Reference 7-2, with suitable corrections to account for the Apollo spacecraft payload. A sequence of events, a sequential weight statement, and a propulsion summary for this launch vehicle are presented in Tables 7-1, 7-2 and 7-3. Note that the 7050-pound Apollo launch escape system is included in the weight statement and is jettisoned with the solid motors. The axial force coefficient data for the Titan IIIC/Apollo was obtained from the Fluid Mechanics Department (see Section 10.3) and is presented in Figure 7-15.

The ascent trajectory consisted of a 10-second vertical rise followed by a kick (an instantaneous rotation of the missile attitude and velocity vector) into a gravity turn trajectory. After the solid motors are jettisoned, a constant inertial pitch rate is established for the remainder of the Core Stage I flight. A different constant inertial pitch rate is introduced at Core Stage II ignition and maintained to Core Stage III (transtage) initial burnout.

Values for the kick angle and the two constant pitch rates were automatically determined by the computer so that the burnout flight path angle was horizontal at a specified burnout altitude (60 or 100 nautical miles) and the aerodynamic heating index, $\int q\, Vdt$, was 0.95×10^8 - the Titan IIIC limit. Reference 7-3 indicates that the latter constraint, $\int q\, Vdt = 0.95 \times 10^8$, maximizes the ascent performance without violating the launch vehicle aerodynamic heating or dynamic pressure limits.

Actual burnout of the transtage occurred when the velocity required to ascend to a desired orbit altitude was achieved. The payload weight was adjusted so that sufficient propellant remained in the transtage at the initial burnout in order to perform the final injection maneuver at the desired orbit altitude. In addition, a propellant margin was included at the final burnout into orbit to account for -3σ dispersions in the launch vehicle performance. This margin was based on the root-sum-square of three percent of the ideal velocity of the individual stages.

7.3.2.2 Results

Maximum payload capability as a function of final orbit altitude is presented in Figure 7-16 and is based on a 106-degree azimuth launch from ETR with an initial transtage burnout into a 60- or a 100-nautical mile perigee. At apogee, the transtage is restarted and injects the payload into the final circular orbit.

The payload data presented in Figure 7-16 is based on optimum transtage propellant loading. The transtage has a maximum capacity for 22,841 pounds of usable propellant; however, off-loading between 4000 to 7500 pounds of transtage propellant may result in as much as a 2.3 percent improvement in payload capability. The optimum amount of propellant off-loading and resulting performance improvement vary with burnout altitude and final orbit altitude.

The payload estimates for the Apollo/MOL configuration are approximately 700 pounds lower than those for Gemini/MOL. It is estimated that half of the difference is due to the inclusion of the 7050-pound launch escape system for the Apollo spacecraft. The remainder may be attributed to the increased drag resulting from the 13-foot diameter Apollo spacecraft.

7.3.3 Saturn IB Ascent Performance

7.3.3.1 Method of Analysis

A sequence of events, a sequential weight statement, a propulsion summary, and the aerodynamic drag characteristics for the Saturn IB launch vehicle are presented in Tables 7-4, 7-5, 7-6 and in Figure 7-17, respectively, and are based on the data obtained from Reference 7-4. This configuration is characterized by a 32,500-pound minimum (-3σ) payload capability for a 105-nautical mile circular orbit (based on a 72-degree azimuth launch from ETR).

The ascent trajectory consisted of a 25-second vertical rise followed by a kick (an instantaneous rotation of the missile attitude and velocity vector) into a gravity turn trajectory. Following ignition of the S-IVB stage, a constant inertial pitch rate is introduced and maintained for 260 seconds of the second stage flight. A different constant pitch rate is introduced at this time and maintained to burnout of the S-IVB stage.

Values of the kick angle and the two constant pitch rates were automatically determined by the computer so that the burnout velocity was maximized (for a fixed payload weight) for a specified burnout altitude and a zero-degree flight path angle. Burnout of the S-IVB stage occurred with sufficient propellants remaining on board to account for -3σ dispersions in the launch vehicle performance. This propellant margin was based on the root-sum-square of 2.333 percent of the ideal velocity of the individual stages.

Maximum burnout velocity was determined as a function of payload weight for burnout altitudes of 60, 80 and 100 nautical miles. From this, apogee altitude could be calculated as a function of payload weight. Note that the Saturn burnout was into a perigee condition. At apogee, the propulsion system included in the Apollo/MOL payload is used for injection into the final circular orbit. The LEM descent engine was assumed for this maneuver and is characterized by 10,500-pound thrust level and a 305-second vacuum specific impulse.

7.3.3.2 Results

Payload propellant expended to achieve the final orbit and payload weight in this final orbit are presented in Figure 7-18 as a function of the final orbit altitude. This data is based on a 106-degree azimuth launch from ETR and considers burnout altitudes of 60, 80 and 100 nautical miles. Note that the sum of the propellant weight expended and the payload weight in orbit constitute the Saturn boosted payload weight.

The performance data presented in Figure 7-18 is based on a Saturn IB burnout into the perigee of an elliptic transfer orbit. At apogee, the payload propulsion system injects the payload into the final circular orbit. Another ascent mode requiring two injection maneuvers by the payload propulsion system was also considered. In this mode, the Saturn launch vehicle burnout is into a circular parking orbit. The payload propulsion is used to inject into the elliptic Hohmann transfer orbit and subsequently for injection into the final circular orbit.

This type of ascent mode increases the payload weight in the final orbit by only 300 pounds for a 300-nautical mile final orbit altitude, and by only 100 pounds for a 150-nautical mile orbit while approximately doubling the payload propellant requirement. While this ascent mode does offer a small payload gain, it has two disadvantages: (1) it requires an additional propulsive maneuver, and (2) the parking orbit ascent mode is undesirable at altitudes below 100 nautical miles. For these reasons, this ascent mode was dropped from consideration.

[redacted]

There are several factors in addition to maximum payload capability that must be considered in selecting a Saturn burnout altitude. The burnout altitude must be high enough so that the spacecraft will have sufficient life in orbit (in this case, the elliptic tranfer orbit) to effect a satisfactory recovery in the event of an abort. It is estimated that at least two or three orbits are required in order to accurately predict the spacecraft impact point. Secondly, the launch vehicle aerodynamic heating and load limits may determine a minimum burnout altitude, below which these constraints are violated. Neither of the above considerations has been examined in this study.

7.3.4 Apollo Abort Recovery Ceilings

Manned payloads generally limit the launch vehicle ascent trajectory by the requirement that the spacecraft and crew be recovered from any abort situation which could occur during the ascent to orbit. The abort recovery constraint is based on spacecraft design and crew tolerance limits and is usually presented in the form of an abort ceiling - a limiting curve of apogee altitude versus apogee velocity. Any combination of velocity, flight path angle, and altitude which has an apogee condition in excess of this ceiling will result in a spacecraft abort trajectory profile that exceeds the spacecraft design or crew acceleration tolerance limits associated with the abort ceiling.

Abort ceilings for the Apollo spacecraft and the Gemini B spacecraft are compared in Figure 7-19. These curves are based on a 13-g maximum re-entry load factor limit and, in the case of Gemini, an afterbody temperature limit. In the region of interest, no afterbody temperature limit exists for the Apollo spacecraft. Apollo abort ceiling data was determined in Section 7.1 for aerodynamic lift-to-drag ratios of 0.4 and 0.5 with re-entry vehicle weights of 9000 and 12,000 pounds. It was found that these aerodynamic and weight variations resulted in only minor perturbations to the abort ceiling.. The more conservative ceiling based on the L/D of 0.4 and a re-entry vehicle weight of 12,000 pounds was used in Figure 7-19.

[redacted]

The ascent trajectory apogee traces for the Saturn IB and Titan IIIC launch vehicles are compared to the Apollo abort ceiling in Figure 7-20. Burnout altitudes of both 60 and 100 nautical miles are considered in this figure. The apogee trace is merely the locus of apogee altitude and apogee velocity points calculated each instant along the ascent trajectory. Note that the apogee trace for the Saturn IB ascent trajectory to a 100-nautical mile burnout altitude exceeds the abort ceiling. This violation disappears when an abort ceiling based on a 10,500-pound re-entry vehicle weight is used.

7.3.5 Apollo Re-entry Performance

The Apollo re-entry from orbit is initiated by firing the retro-rockets. According to Reference 7-5, the nominal Apollo re-entry trajectory would have an effective bank angle of 45 degrees. If no lateral range is required, the vehicle is alternately rolled from left to right in order to maintain the trajectory in the orbital plane. A trajectory time history for this nominal re-entry is presented in Figure 7-21 and was taken from the above reference.

Lateral maneuvering is achieved by biasing the bank angle to one side. Maximum re-entry range is achieved by a zero-degree bank angle while minimum range is achieved by a 90-degree bank angle. The landing footprint for the spacecraft, also obtained from Reference 7-5, is presented in Figure 7-22 for aerodynamic lift-to-drag ratios of 0.4 and 0.5.

The Apollo spacecraft for the lunar mission has a trim lift-to-drag ratio of 0.5; however, removing unnecessary ablation material from the heat shield for orbital missions causes the spacecraft center of gravity to move aft and results in a trim lift-to-drag ratio of 0.4.

Finally, the re-entry trajectories used for Figures 7-21 and 7-22 were based on a -2 degree re-entry flight path angle. Re-entry trajectories for other re-entry angles, spacecraft aerodynamics, and spacecraft wing loadings are presented in Section 7.2.

7.4 CONCLUSIONS

1. Within the conditions of the study, it appears that near-optimum ascent trajectories may be used with either Titan IIIC or Saturn IB as the launch vehicle.
2. Recovery conditions for the Apollo/MOL appear to be within acceptable bounds.

Table 7-1. Titan IIIC/Apollo Sequence of Events.

Event	Time, sec
Liftoff	0
End Vertical Rise, Start Pitch Over	10.00
Solid Motor Web Burnout	104.20
Start Staging Sequence (SS) at a Sensed Axial Load Factor of 2.34 g's, Jettison Boattail	SS
Core Stage I Ignition	SS + 1.00
Solid Motor Burnout	114.80
Jettison Solid Motors and Launch Escape Tower	SS + 10.00
Core Stage I Burnout	SS + 152.96
Jettison Core Stage I and Core Stage II Ignition	SS + 153.96
Core Stage II Cutoff Signal, Start of Tailoff	SS + 359.94
Core Stage II Burnout and Jettison	SS + 369.94
Core Stage III Ignition	SS + 371.94
Core Stage III Burnout	SS + 806.77 (1)

Note: (1) Core Stage III burnout time based on complete depletion of usable propellants.

Table 7-2. Titan IIIC/Apollo Sequential Weights.

	Component Weight (lb)	Total Weight (lb)
Weight at Liftoff		1,400,933
Solid Motor Propellant	838,000	
Solid Motor Expended Inerts (1)	6,752	
Expended TVC Injectant (1)	21,594	
Core Stage I and II Engine Bleed (1)	36	
Core Stage I Boattail (2)	1,058	
Core Stage I Start Charge (3)	9	
Core Stage I Propellant (4)	14,921	
Weight at Solid Motor Jettison		518,563
Solid Motor Burnout Weight	157,716	
Apollo Launch Escape Tower	7,050	
Weight After Solid Motor Jettison		353,797
Core Stage I Propellant (4)	237,018	
Core Stage I and II Engine Bleed (5)	22	
Weight at Core Stage I Burnout		116,757
Core Stage I Burnout Weight	15,647	
Core Stage II Start Charge	3	
Weight at Core Stage II Ignition		101,107
Core Stage II Propellant	66,627	
Core Stage II Expended Inerts and Engine Bleed (6)	67	

Table 7-2 Titan IIIC/Apollo Sequential Weights (Continued).

	Component Weight (lb)	Total Weight (lb)
Weight at Core Stage II Cutoff		34,413
Core Stage II Shutdown Propellant	186	
Weight at Core Stage II Burnout		34,227
Core Stage II Burnout Weight	6,455	
Control Module Settling Propellant	8	
Weight at Core Stage III Ignition		27,764
Core Stage III Propellant	22,841	
Core Stage III Expended Inerts (7)	30	
Weight at Core Stage III Burnout		4,893
Core Stage III Burnout Weight	2,531	
Control Module Weight at Burnout	2,362	
Payload Weight		0

Notes: (1) Expended over 104.20 seconds of Solid Motor Web Action Time.

(2) Jettisoned at a sensed axial load factor of 2.34 g's.

(3) Expended 1 second after boattail jettison.

(4) Core Stage I propellant expended both before and after solid motor jettison.

(5) Expended at a constant rate over Core Stage I burning.

(6) Expended at a constant rate over Core Stage II burning.

(7) Expended at a constant rate over Core Stage III burning.

Table 7-3. Titan IIIC/Apollo Propulsion Data(1).

Solid Motors (All Data for Two Motors)

Total Vacuum Impulse, lb/sec	219.59527×10^6
Average Vacuum Specific Impulse, sec	262.04686
Impulse Propellants, lb	838,000
Web Action Time, sec	104.20
Total Action Time, sec	114.80
Total Nozzle Exit Area, $in.^2$	17,840
Nozzle Cant Angle, deg	6.0

Core Stage I, II and III

Core Stage	I	II	III
Vacuum Thrust, lb	474,000	100,890	16,000
Sea Level Thrust, lb	430,000	-	-
Nozzle Cant Angle, deg	2.0	0	0
Propellant Vacuum I_{sp}, sec	285.9	311.9	305.0
Propellant Flow Rate, lb/sec	1657.797	323.469	52.459
Total Flow Rate, lb/sec(2)	1657.942	323.794	52.528

Notes: (1) All data along nozzle centerline, thrust levels should be corrected for nozzle cant angle.

(2) Includes engine inert and engine bleed flow rates.

[redacted]

Table 7-4. Saturn IB Sequence of Events.

Event	Time, sec
Liftoff	0
End Vertical Rise, Start Pitch Over	25.000
Shutdown S-IB Stage 4 Inboard Engines	147.328
S-IB Stage Burnout	153.328
Jettison S-IB Stage and S-IVB Stage Ignition	158.828
Jettison Launch Escape System and Ullage Rocket Cases	168.828
S-IVB Stage Burnout	627.153(1)

Note: (1) Complete depletion of S-IVB stage usable propellants.

[redacted]

Table 7-5. Saturn IB Sequential Weights.

	Component Weights (lb)	Total Weight (lb)
Weight at Liftoff		1,245,718
Propellant Expended	883,178	
Frost, Fuel Additive, and Lube Oil(1)	1,640	
Weight at S-IB Stage Burnout		360,900
S-IB Stage Burnout Weight	102,935	
S-IB/S-IVB Interstage	5,600	
S-IVB Ullage Rocket Propellant	182	
Weight at S-IVB Stage Ignition		252,183
Ullage Rocket Cases(2)	213	
Launch Escape System(2)	6,600	
Propellant Expended(3)	219,871	
Weight at S-IVB Stage Burnout(3)		25,499
S-IVB Stage Burnout Weight	22,939	
Vehicle Instrument Unit	2,660	
Payload Weight		0

Notes: (1) Expended at a constant rate over 153.328 seconds of S-IB stage burning.

(2) Jettisoned 10 seconds after S-IVB stage ignition.

(3) S-IVB stage burnout weight should be adjusted so that sufficient propellant remains at burnout to provide a velocity margin against -3σ performance. This velocity margin should be equal to the root-sum-square of 2.333 percent of the ideal velocity of each stage.

Table 7-6. Saturn IB Propulsion Data.

	S-IB Stage Inboard Engines (1)	S-IB Stage Outboard Engines (1)	S-IVB Stage
Vacuum Thrust, lb (2)	853,050	853,050	200,000
Sea Level Thrust, lb (2)	752,000	752.000	-
Nozzle Cant Angle, deg	3.0	6.0	0
Flow Rate, lb/sec	2937.500	2948.196 (3)	469.484

Notes: (1) The S-IB stage has four H-1 inboard engines and four H-1 outboard engines.

(2) Thrust data is along nozzle centerline and should be corrected by the cosine of the nozzle cant angle.

(3) Includes 10.696 lb/sec of frost, fuel additive, and lube oil expended during S-IB stage burning.

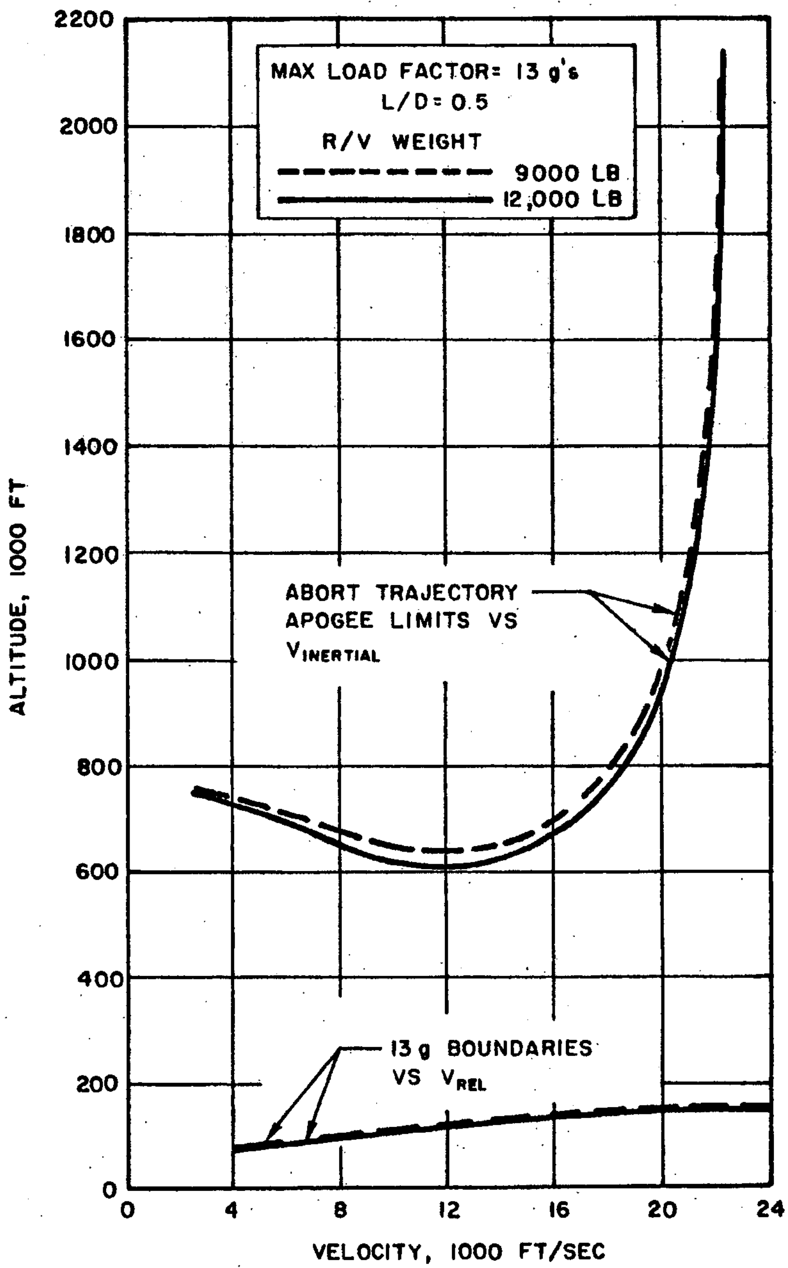

Figure 7-1. 32° Inclination Apollo Abort Boundaries L/D = 0.5

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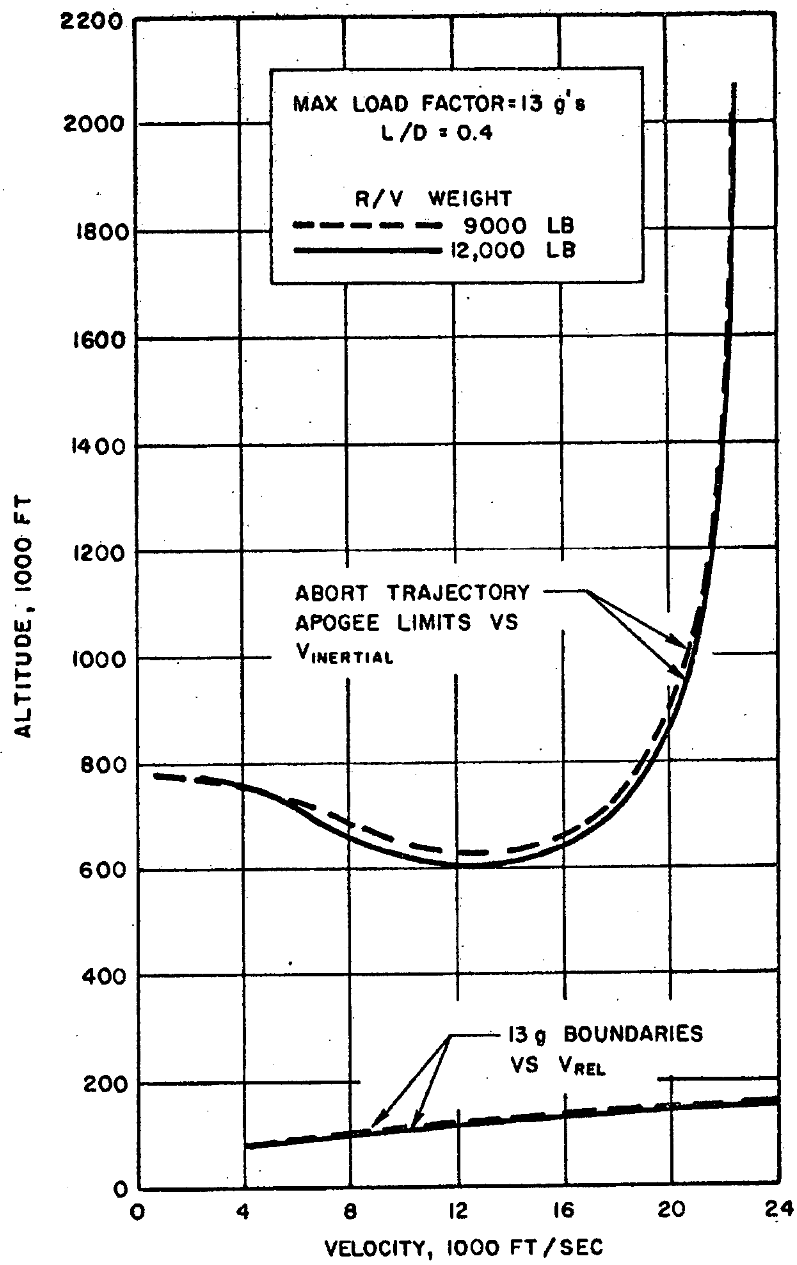

Figure 7-2. 32° Inclination Apollo Abort Boundaries L/D = 0.4

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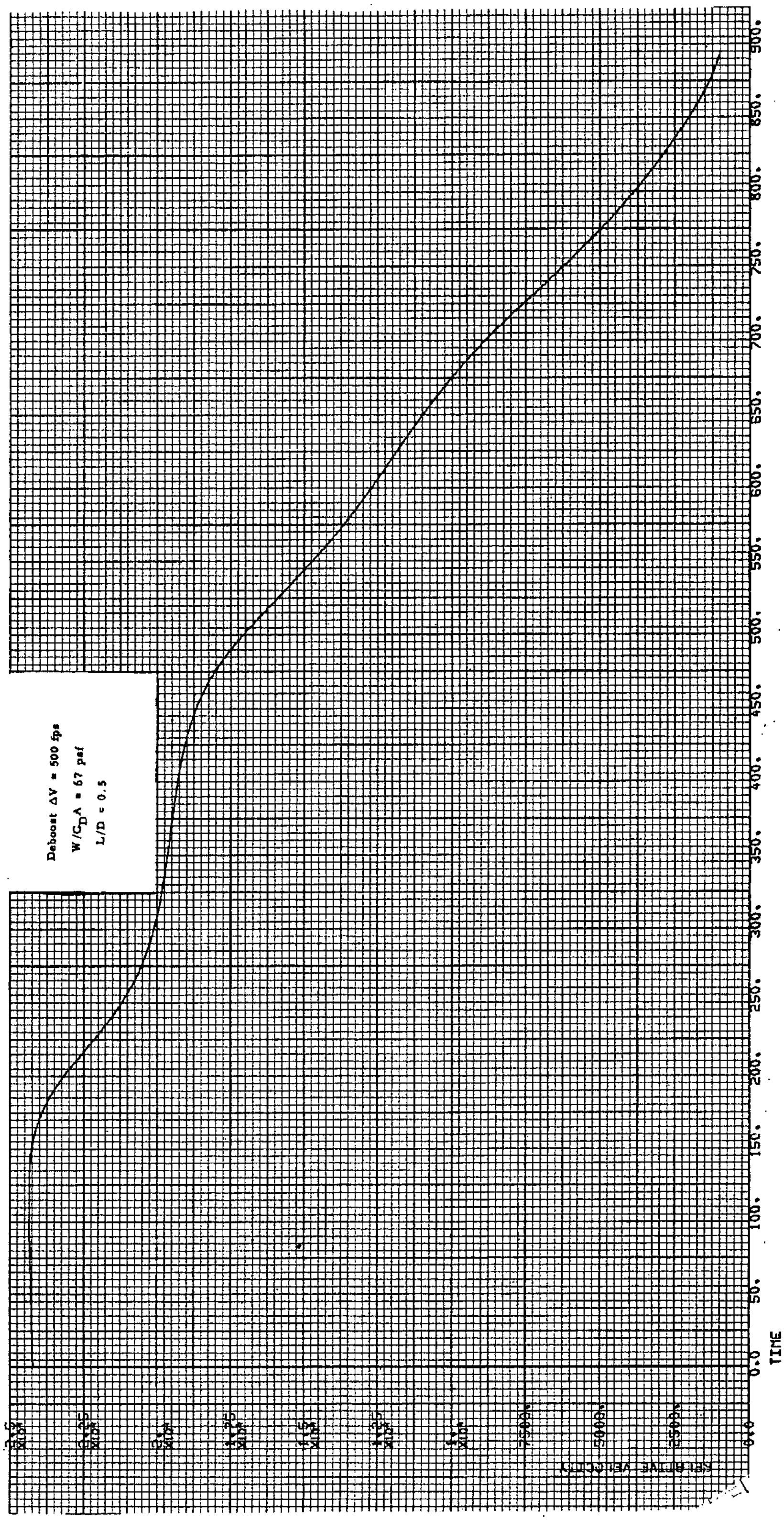

Figure 7-3a. Re-entry Time History.

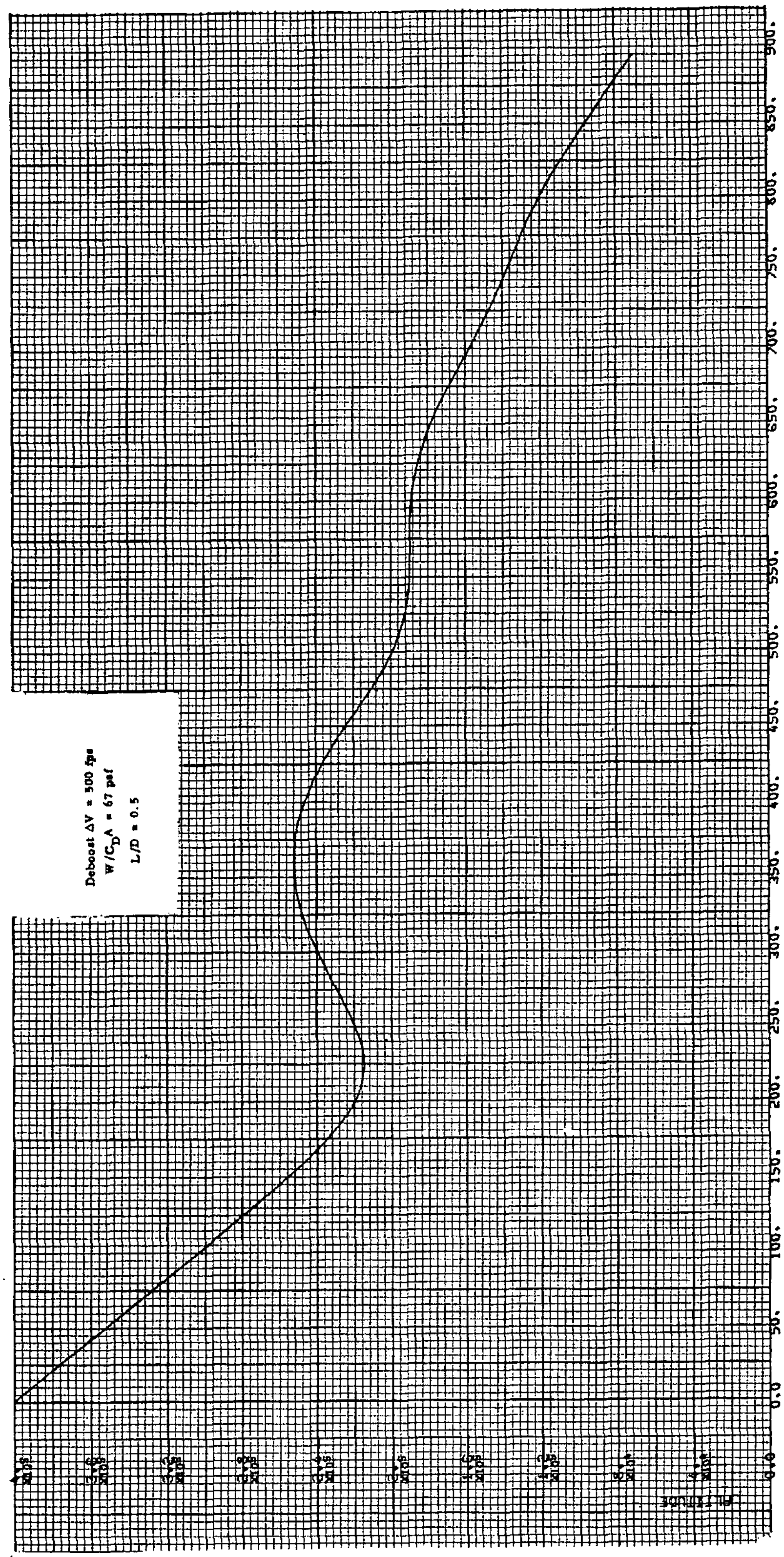

Figure 7-3b. Re-entry Time History.

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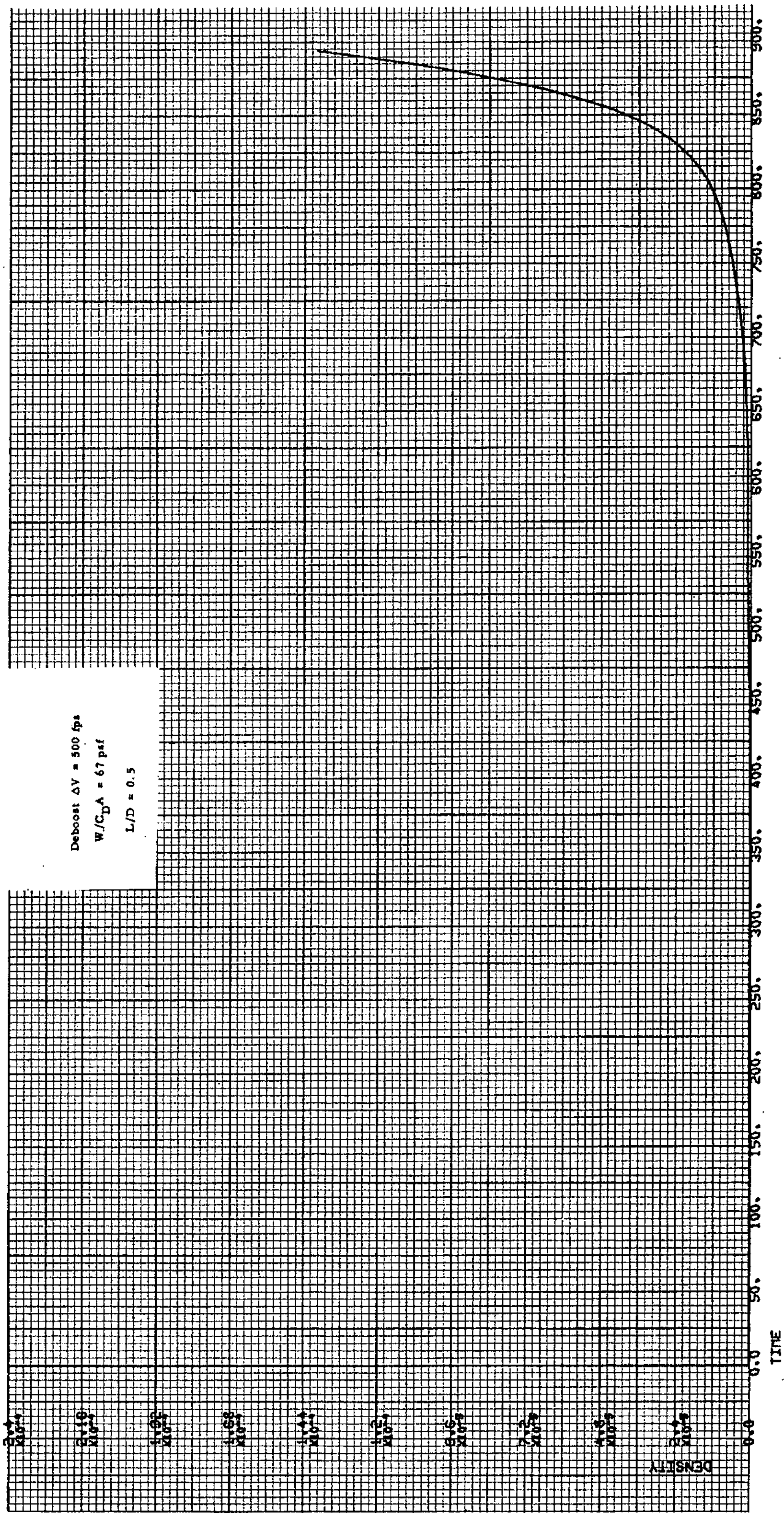

Figure 7-3c. Re-entry Time History.

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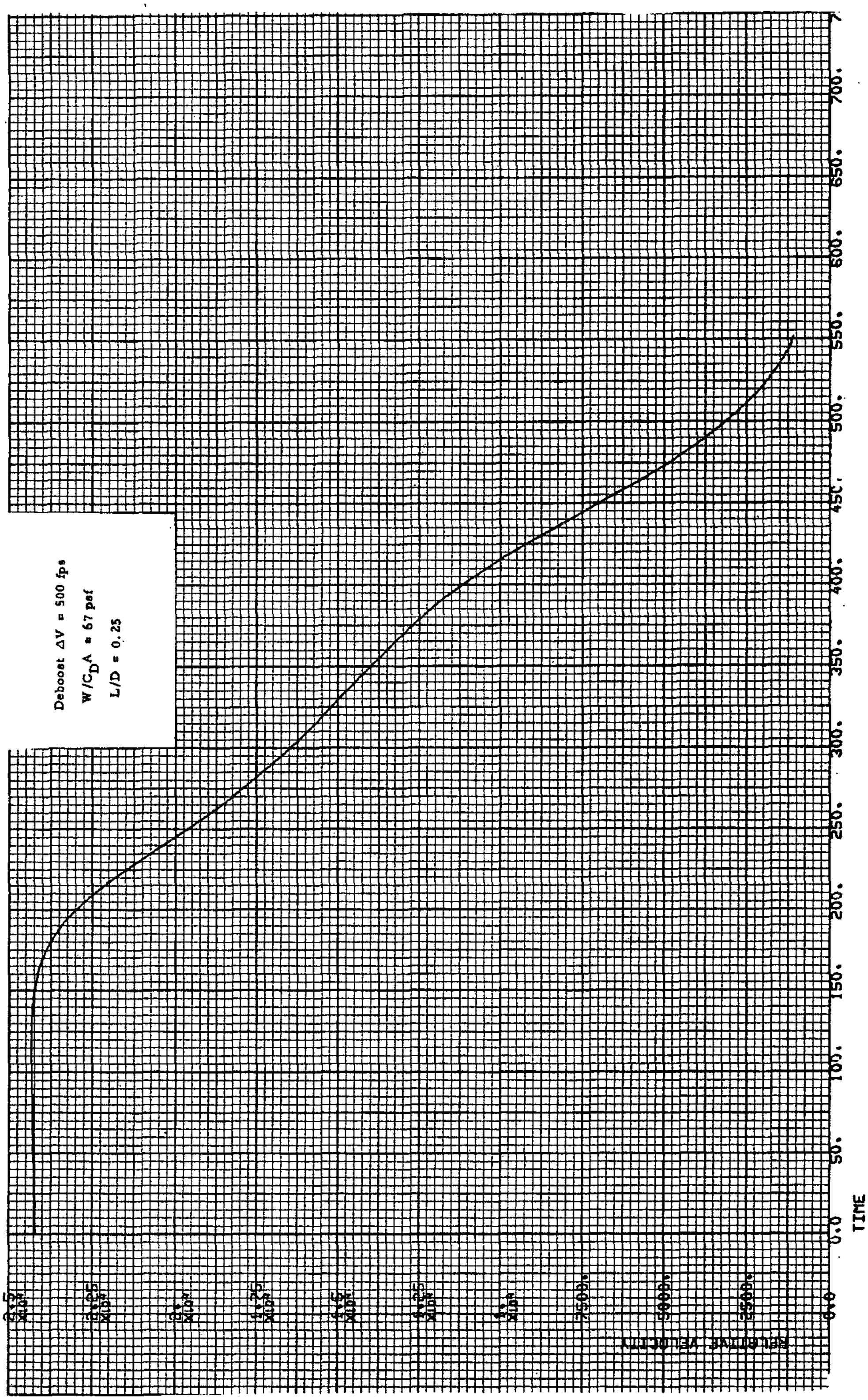

Figure 7-4a. Re-entry Time History.

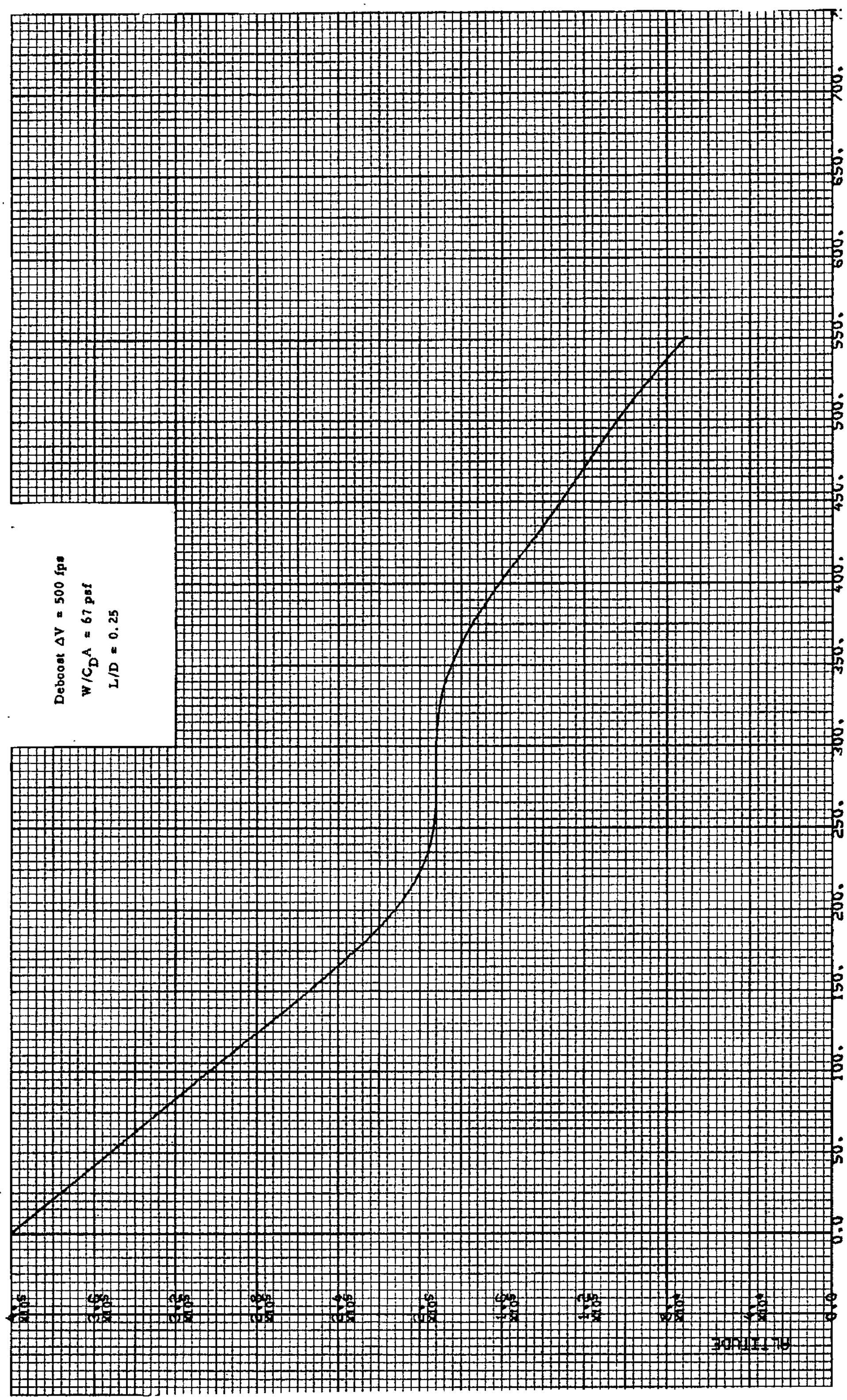

Figure 7-4b. Re-entry Time History.

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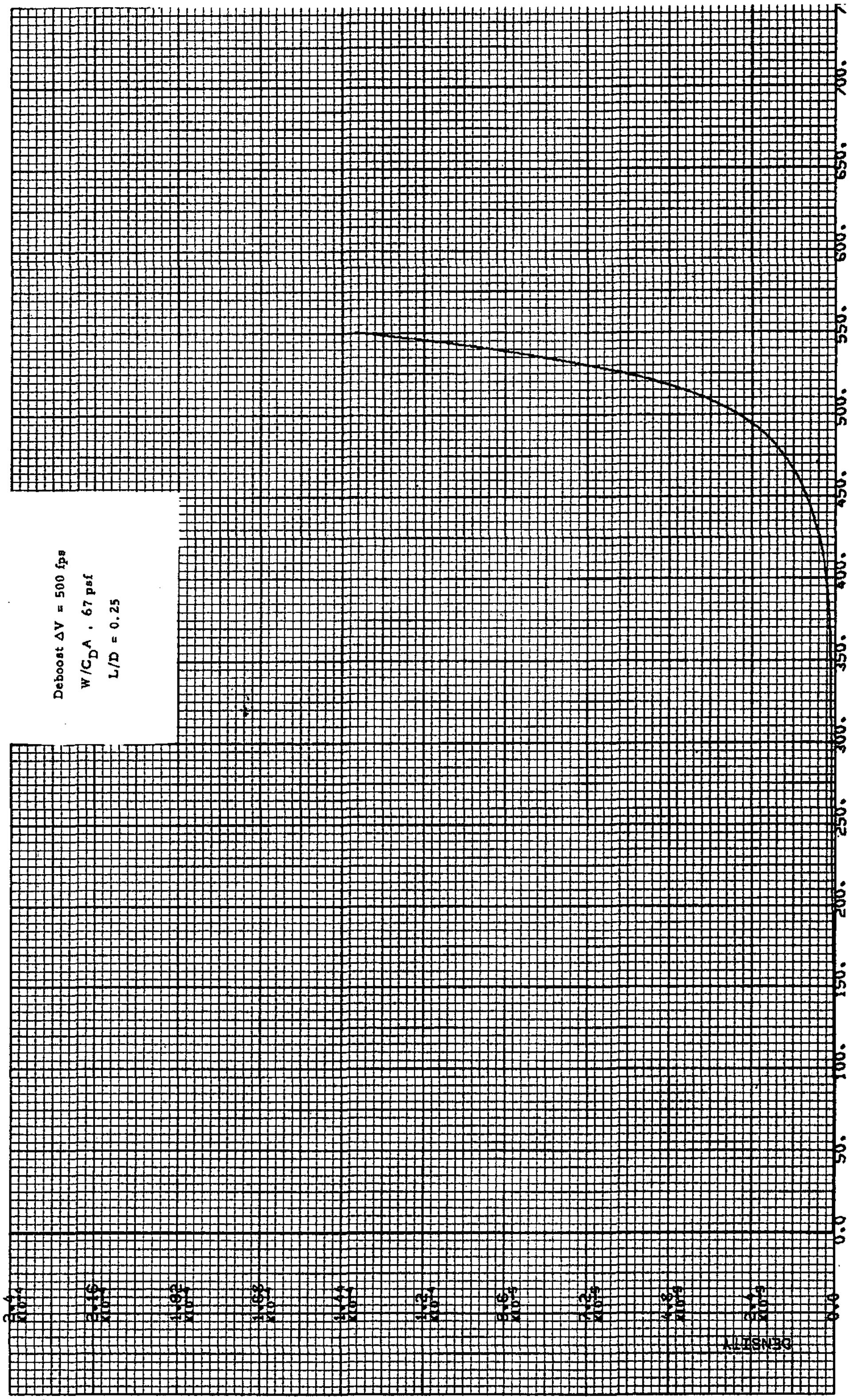

Figure 7-4c. Re-entry Time History.

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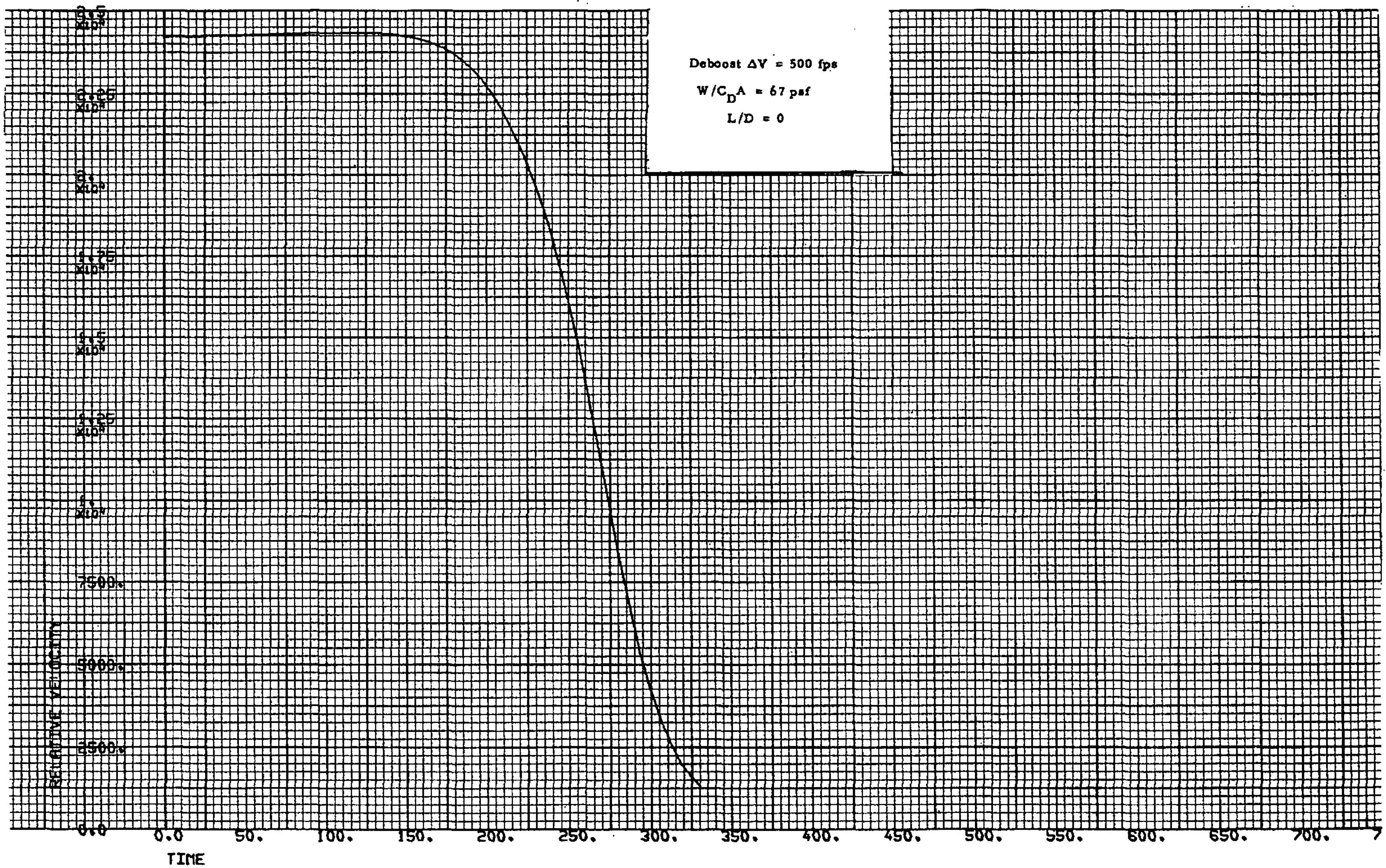

Figure 7-5a. Re-entry Time History.

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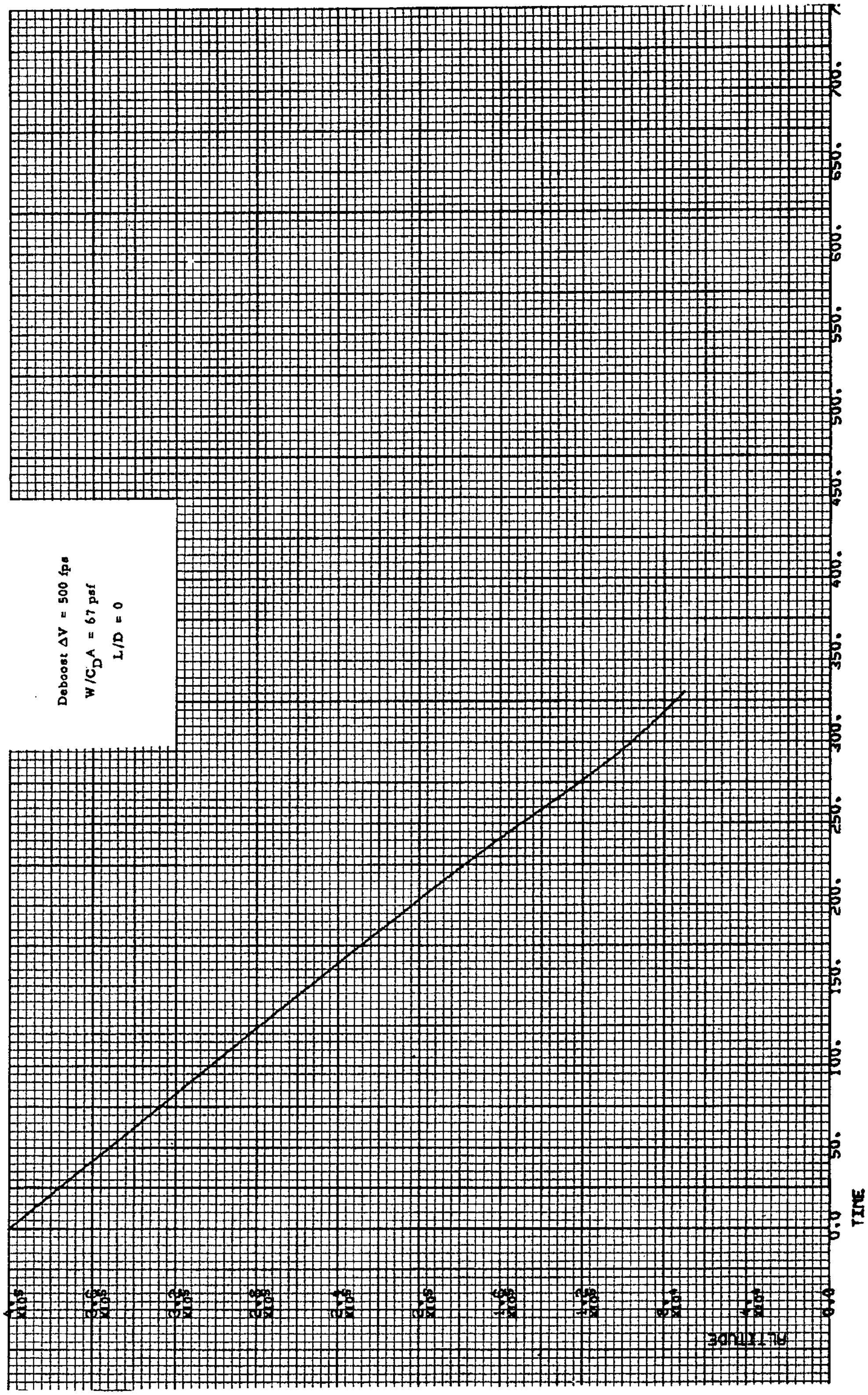

Figure 7-5b. Re-entry Time History.

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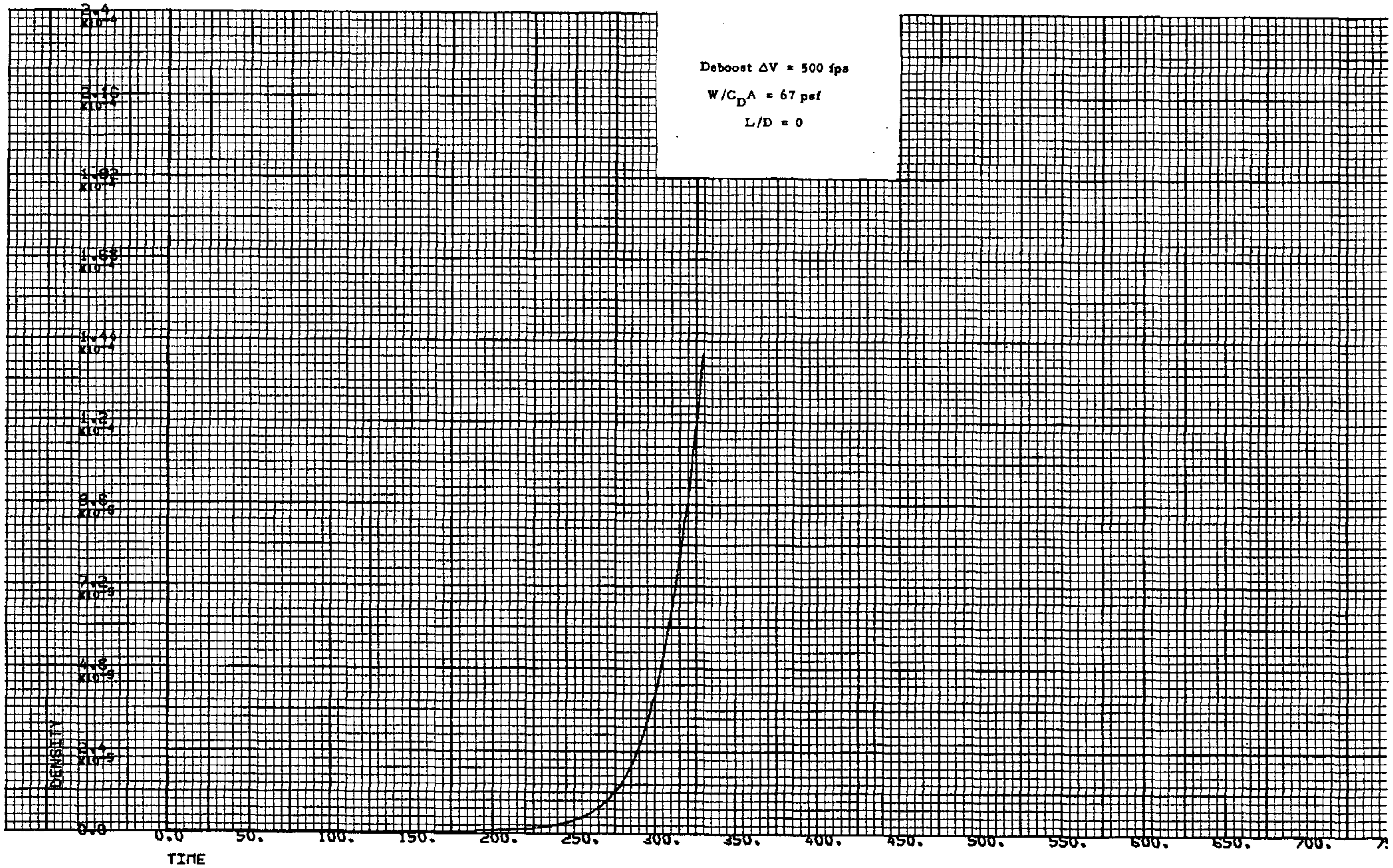
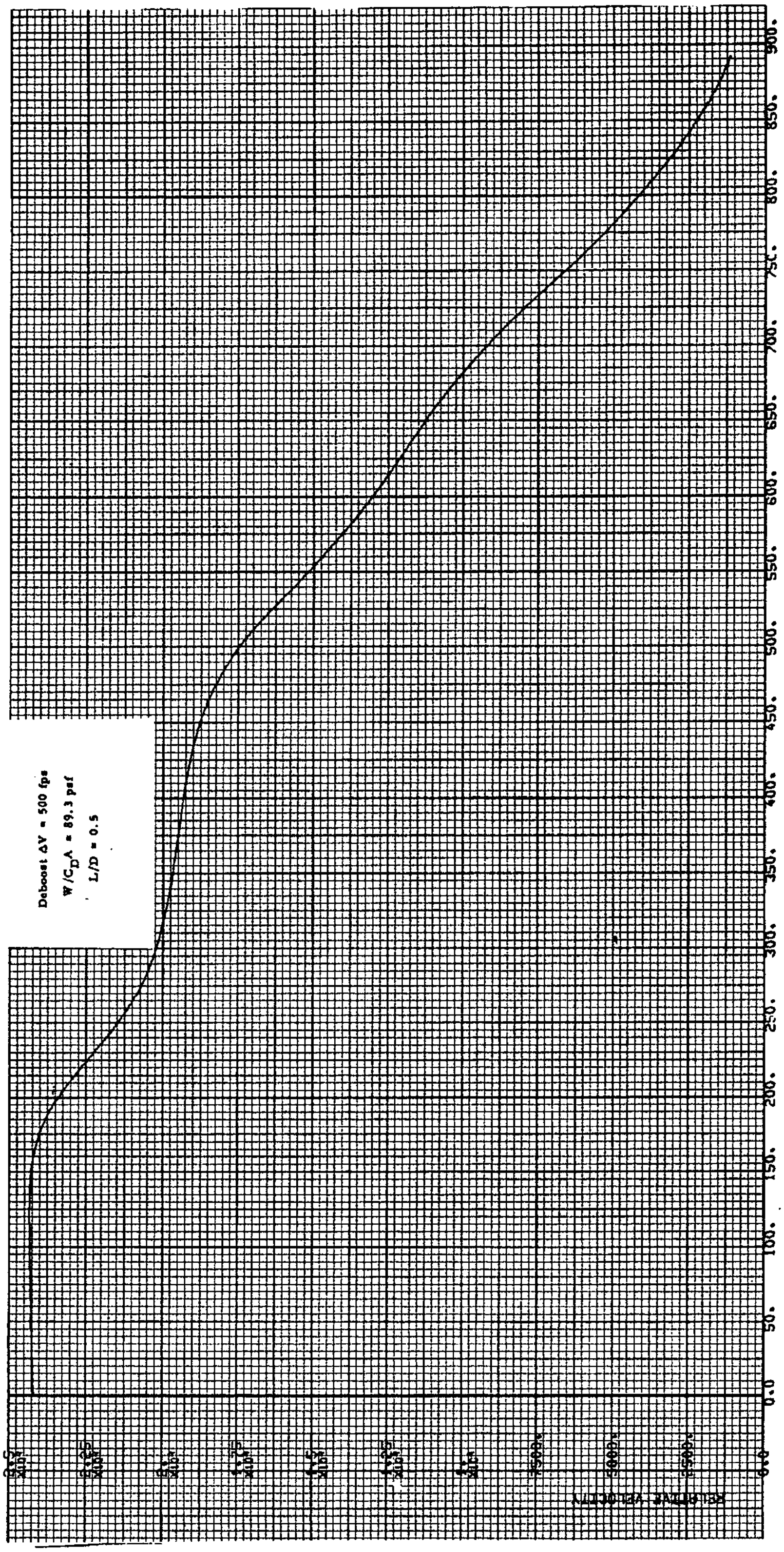


Figure 7-5c. Re-entry Time History.



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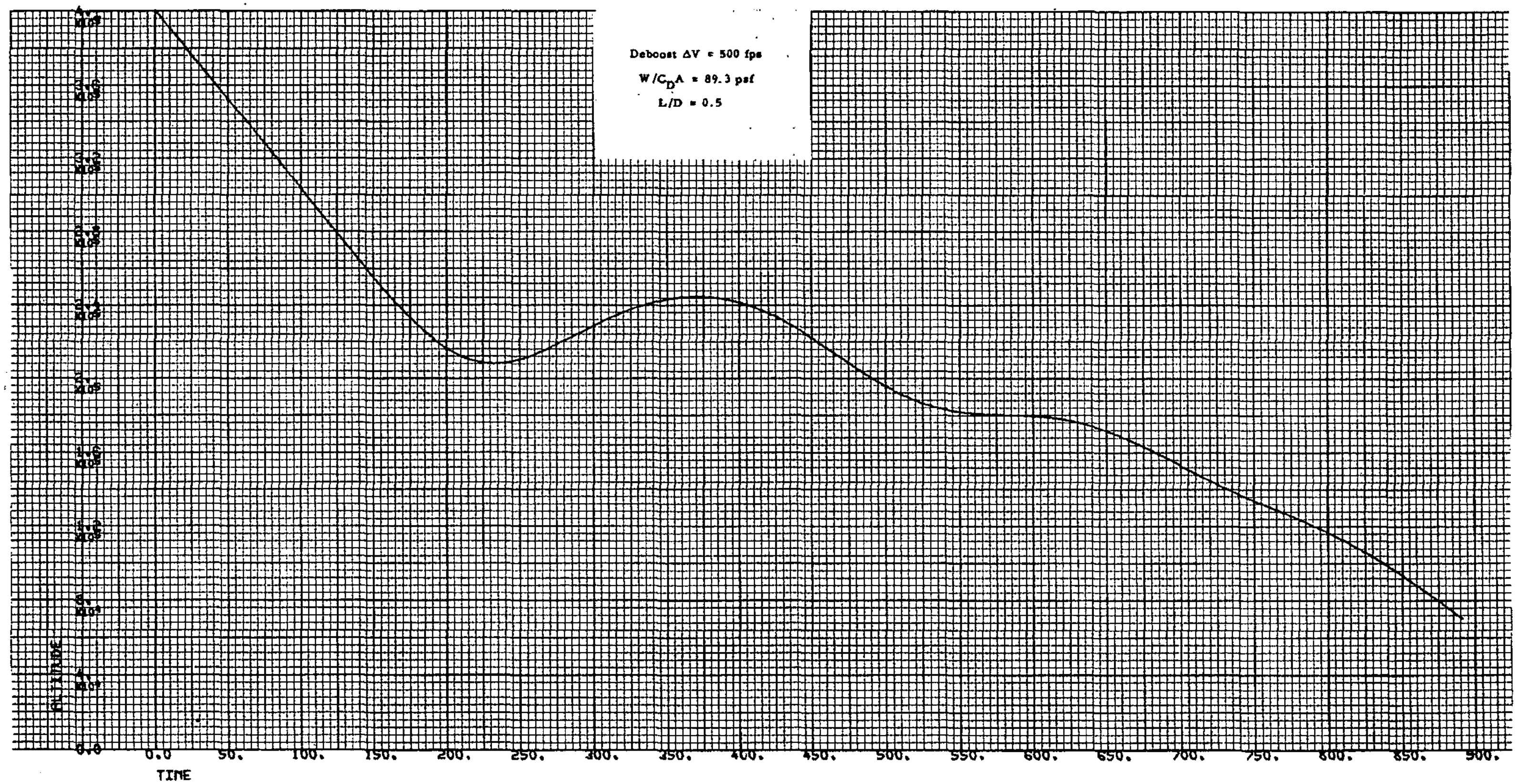


Figure 7-6b. Re-entry Time History.

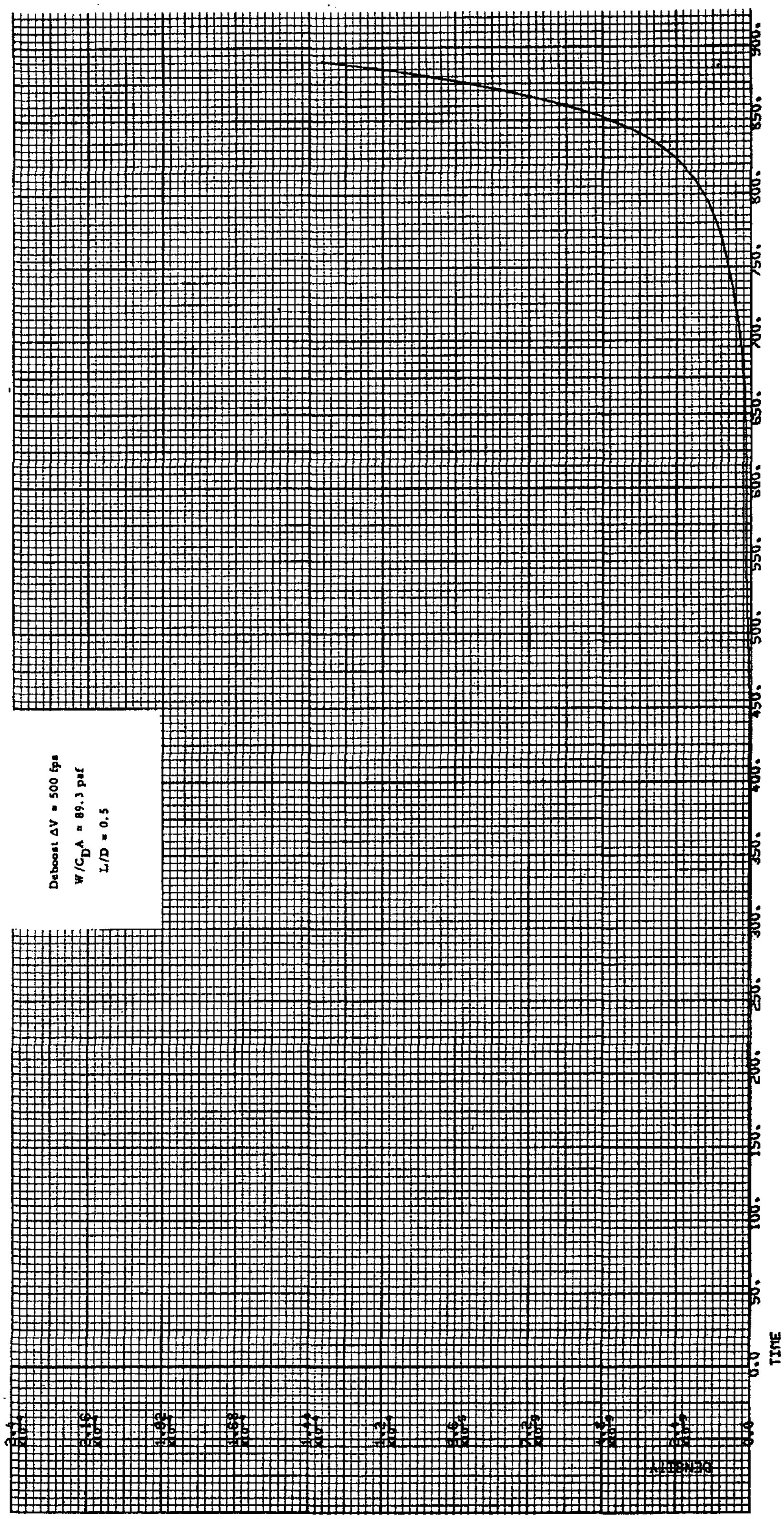


Figure 7-6c. Re-entry Time History.

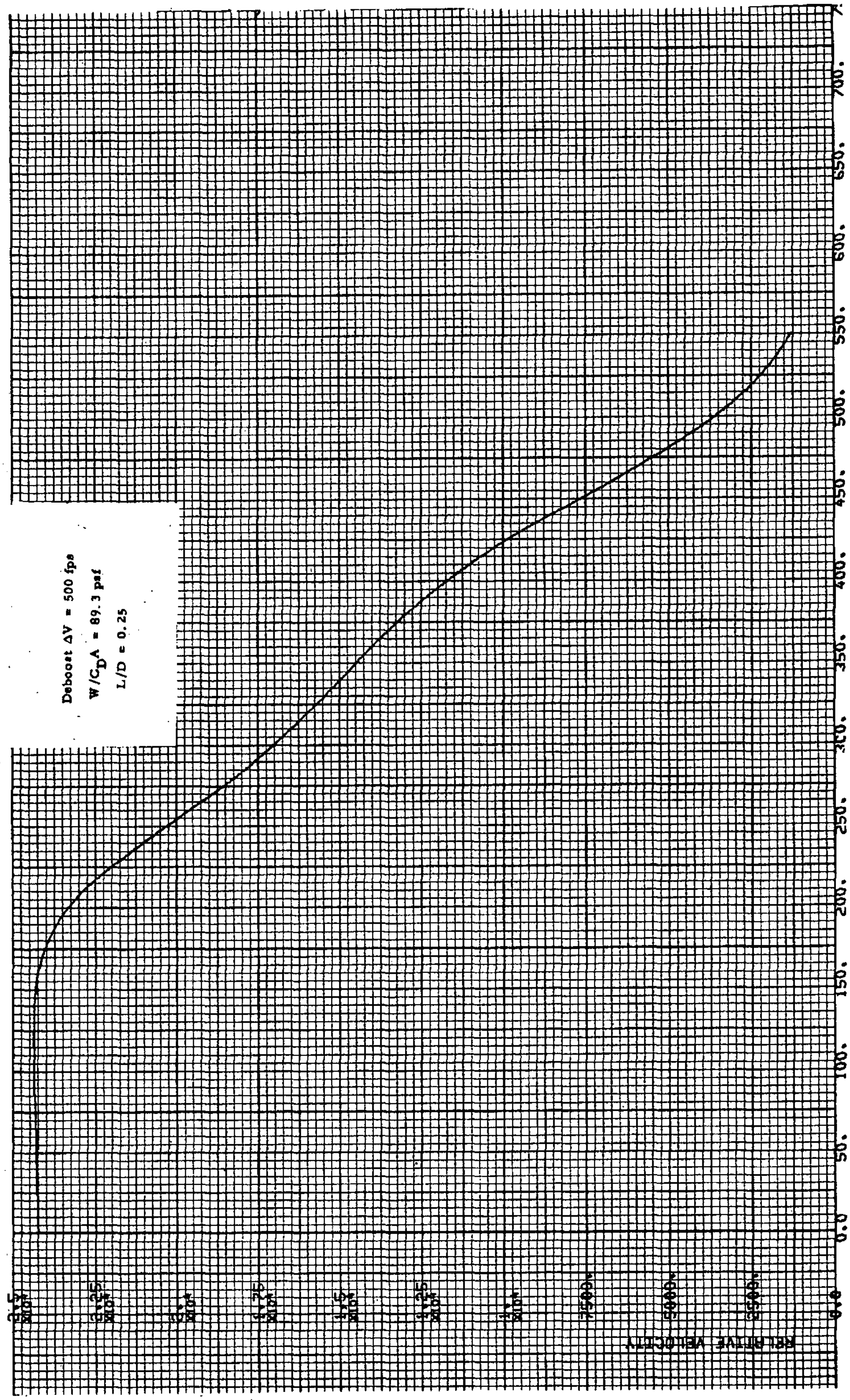


Figure 7-7a. Re-entry Time History.

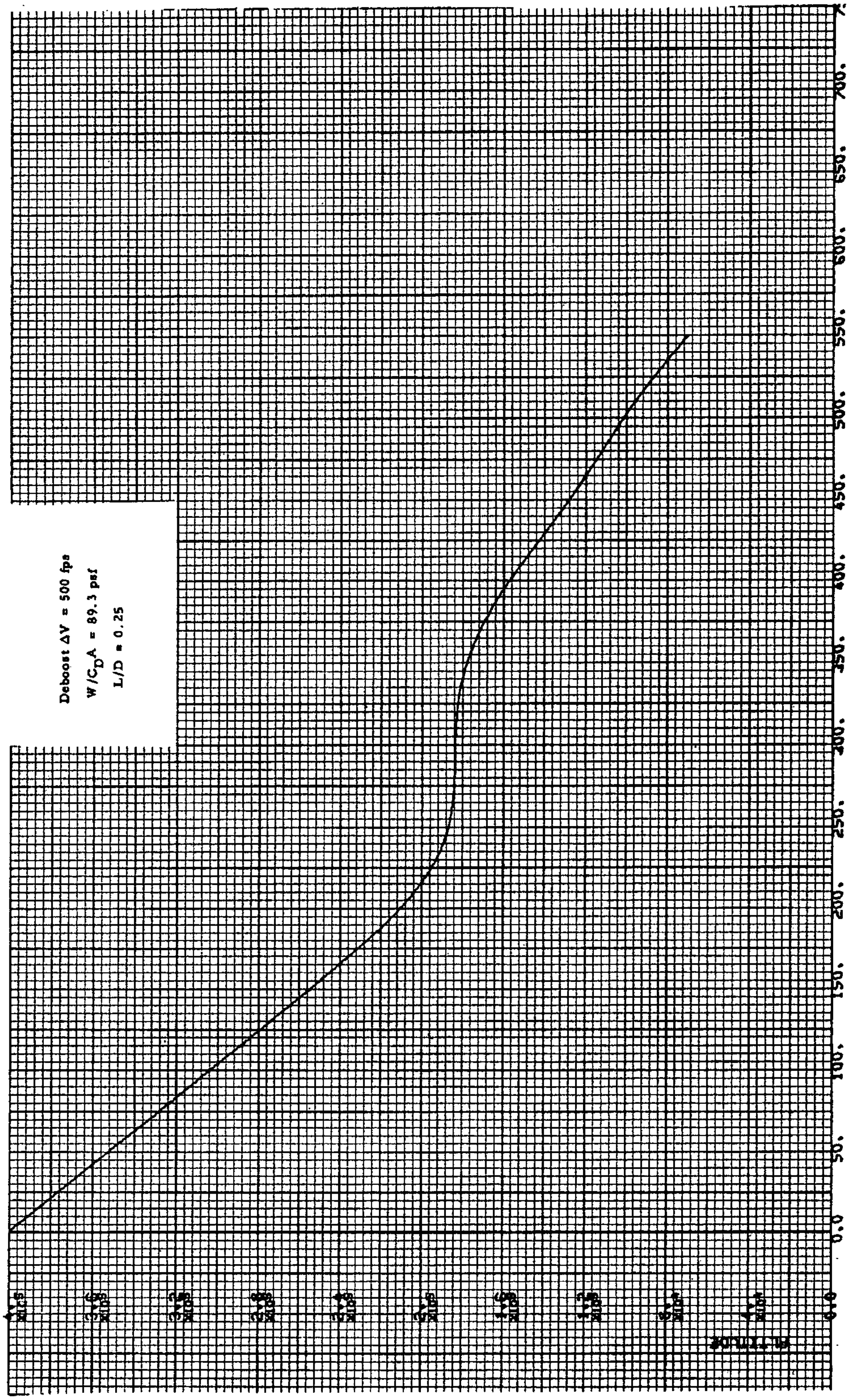


Figure 7-7b. Re-entry Time History.

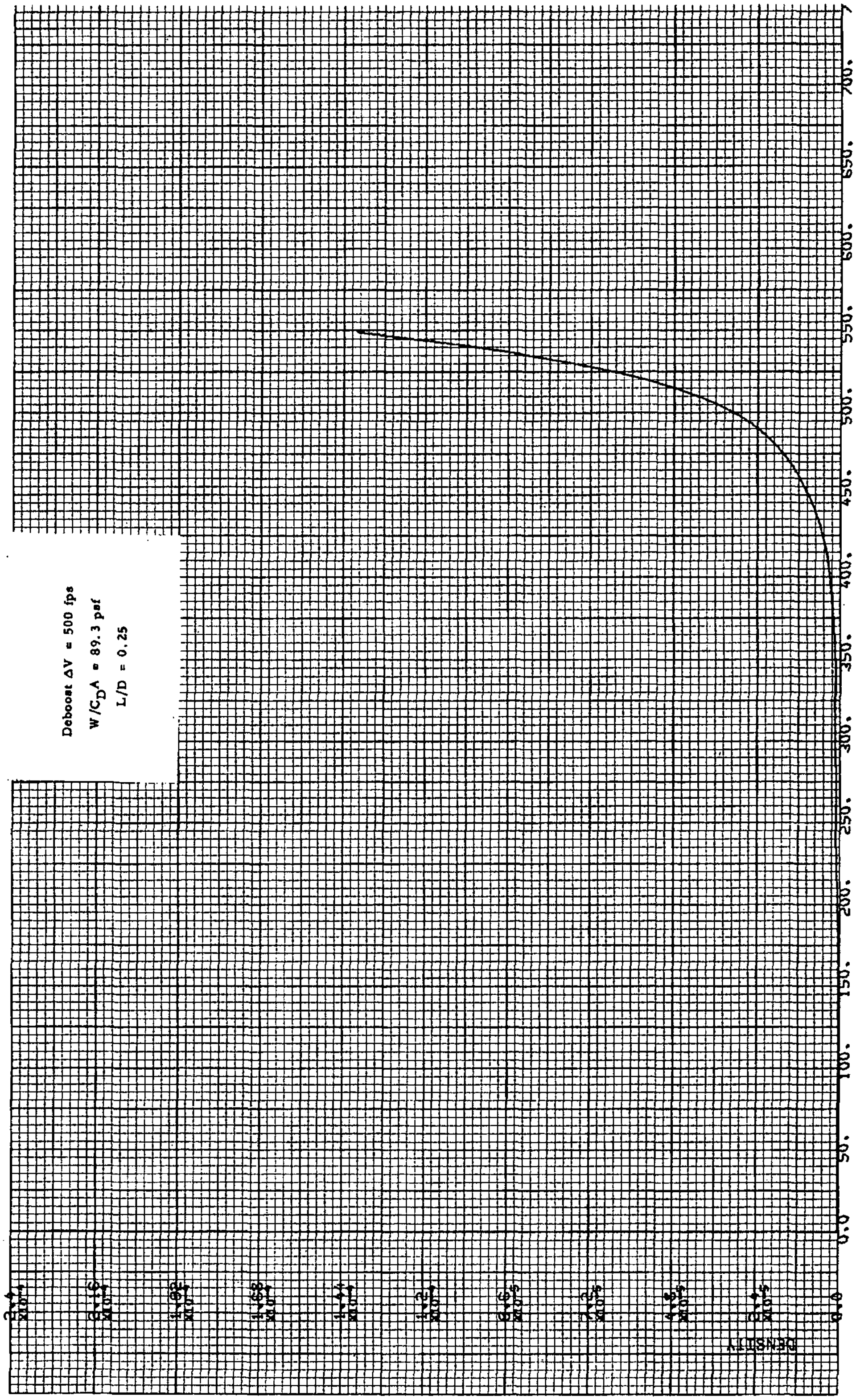


Figure 7-7c. Re-entry Time History.

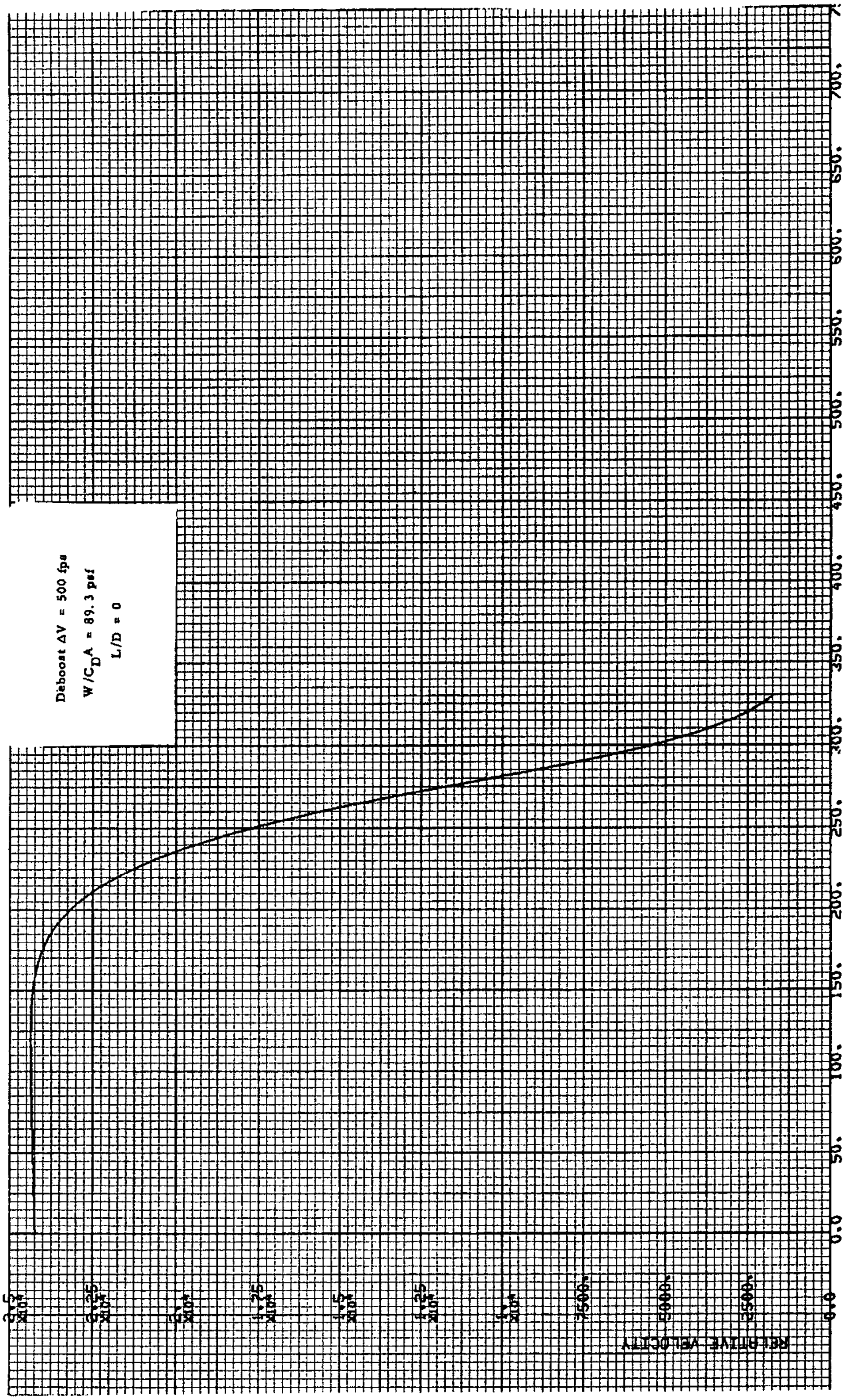


Figure 7-8a. Re-entry Time History.

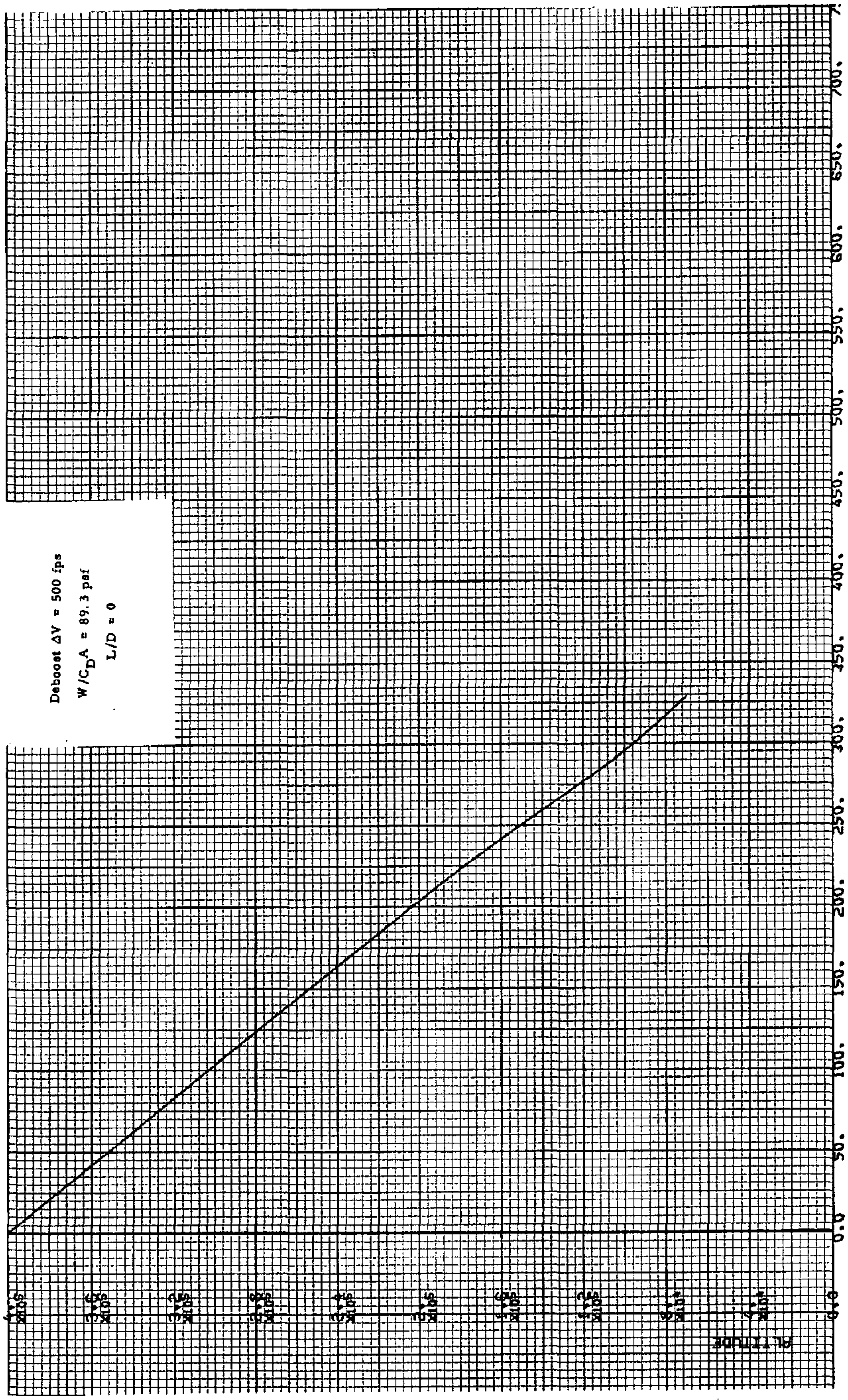
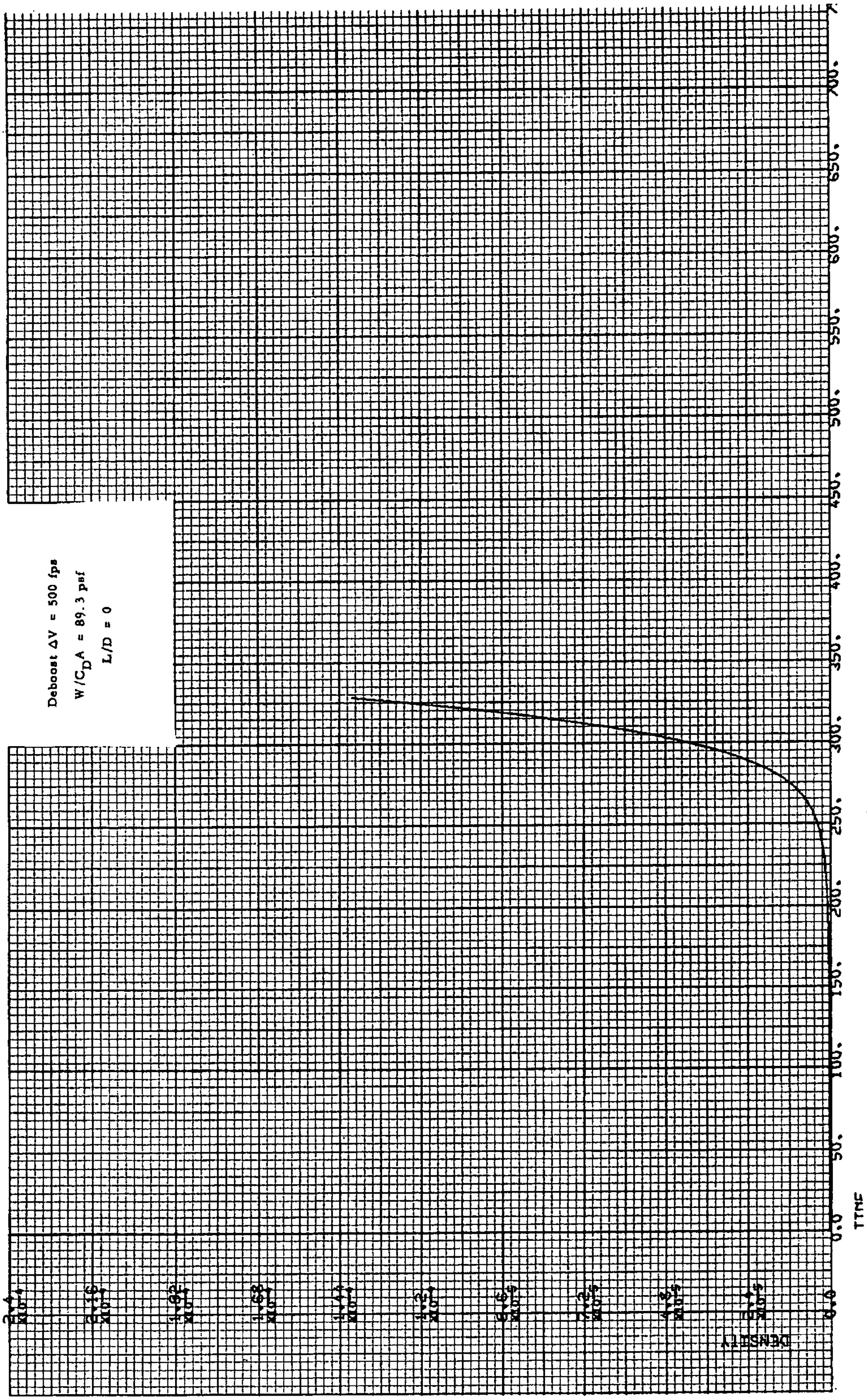


Figure 7-8b. Re-entry Time History.



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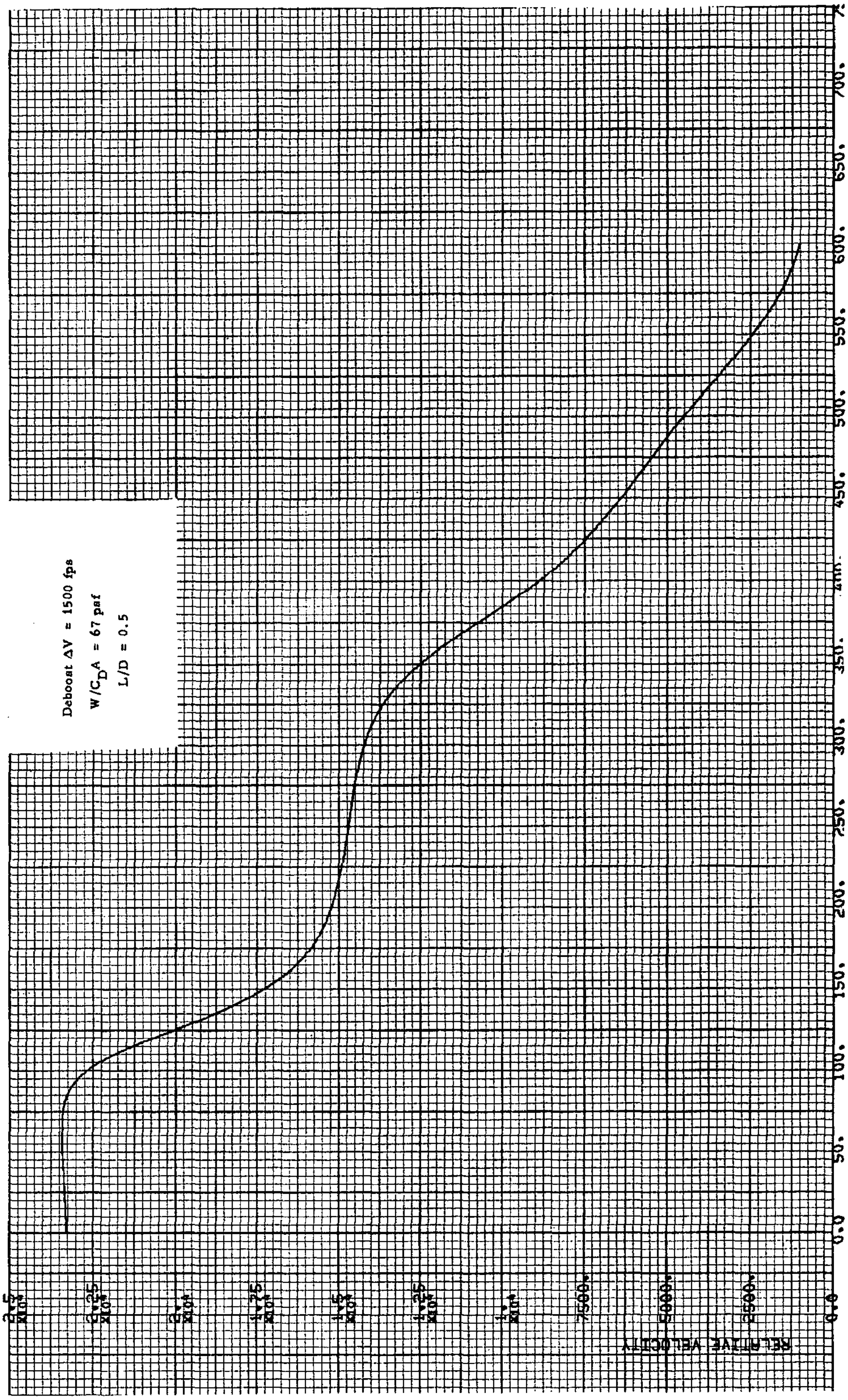


Figure 7-9a. Re-entry Time History.

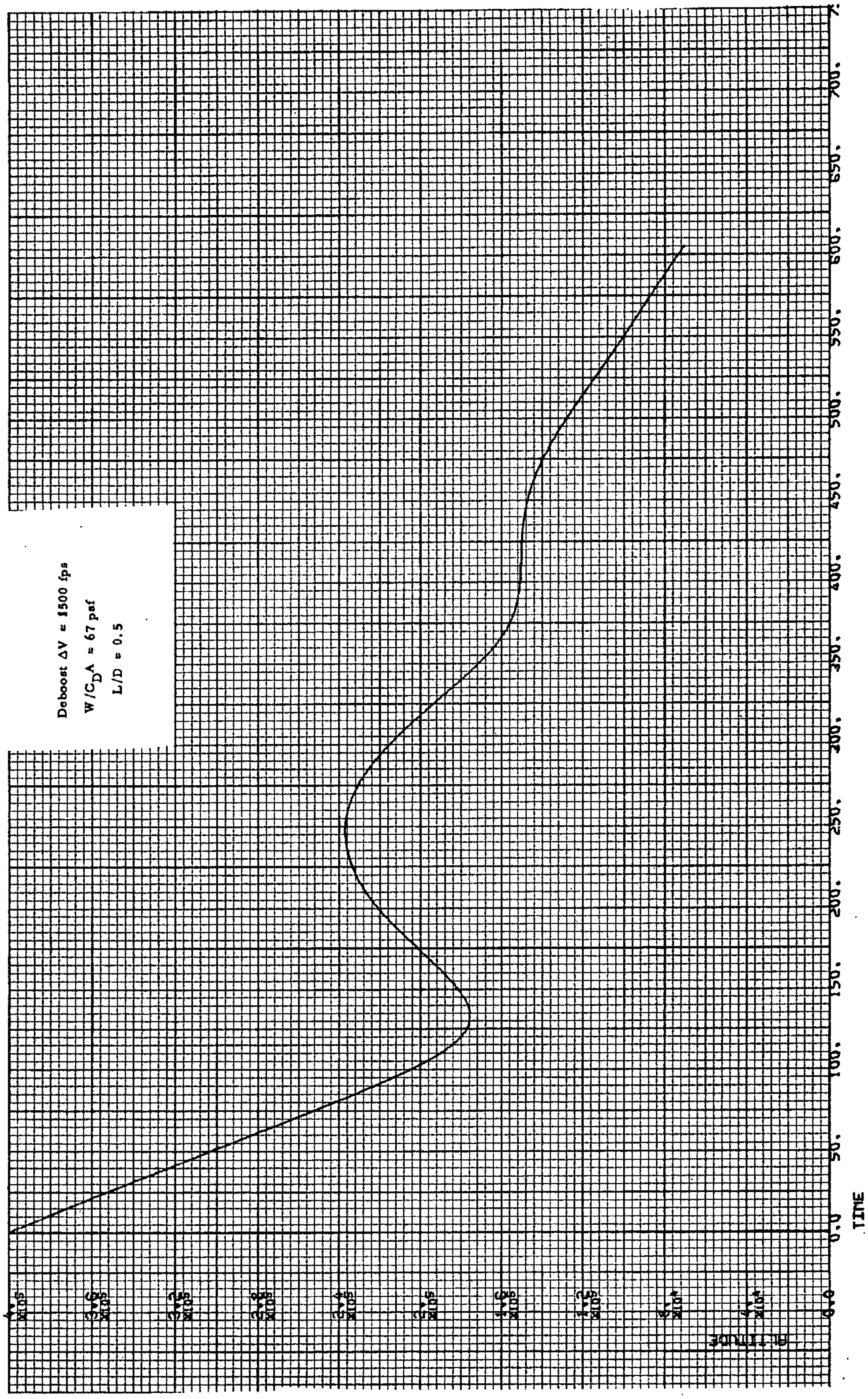
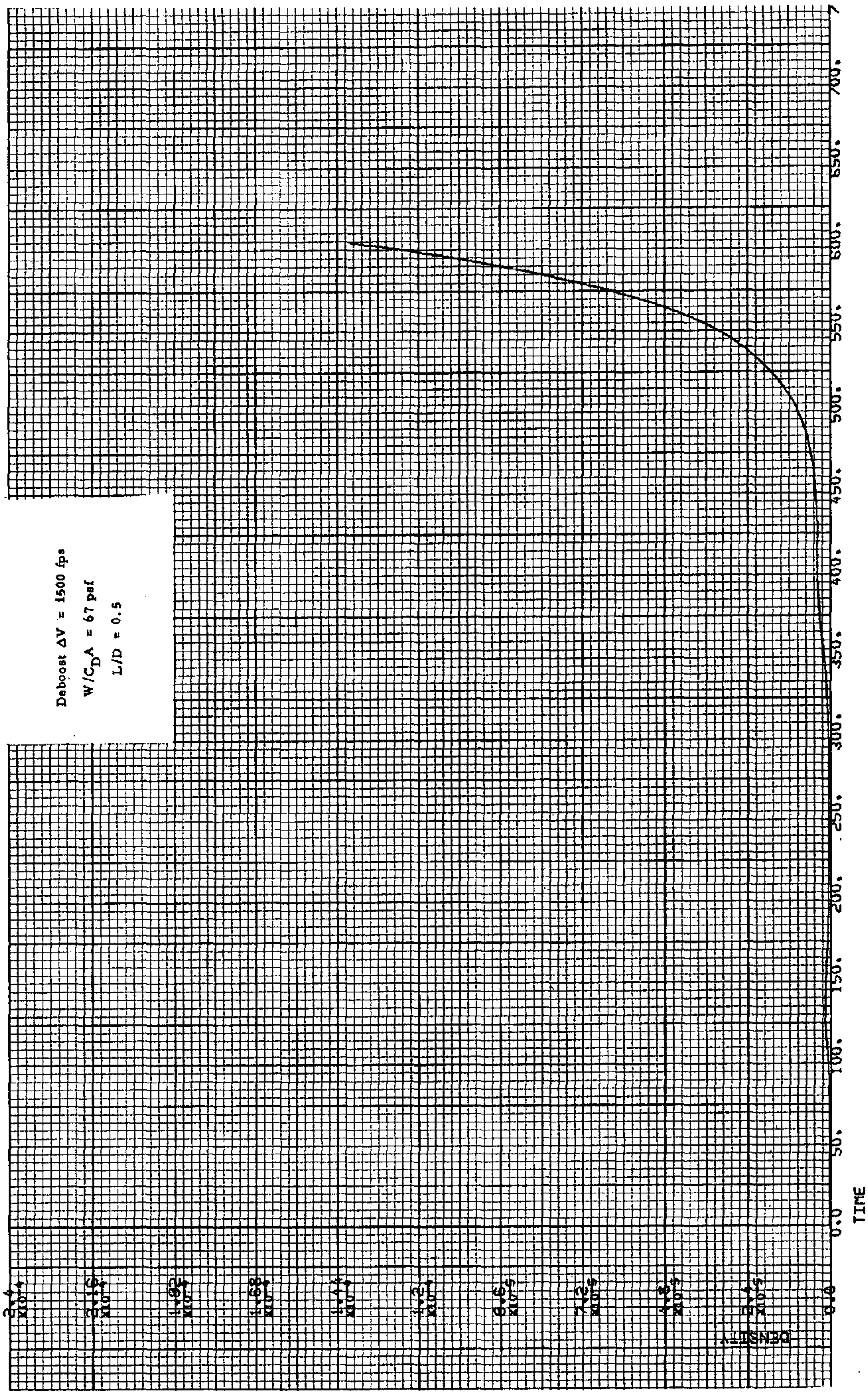


Figure 7-9b. Re-entry Time History.



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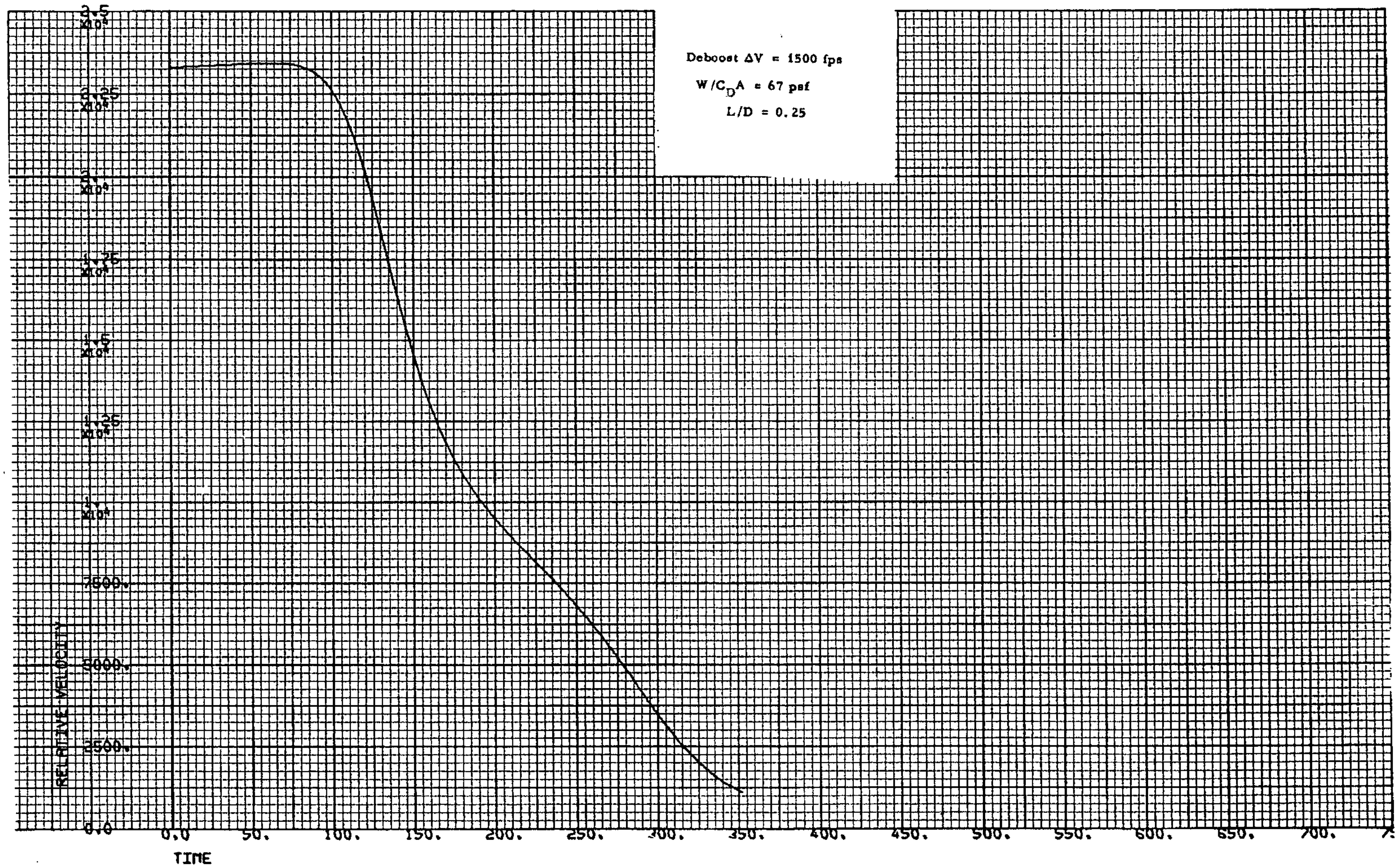

Figure 7-10a. Re-entry Time History.

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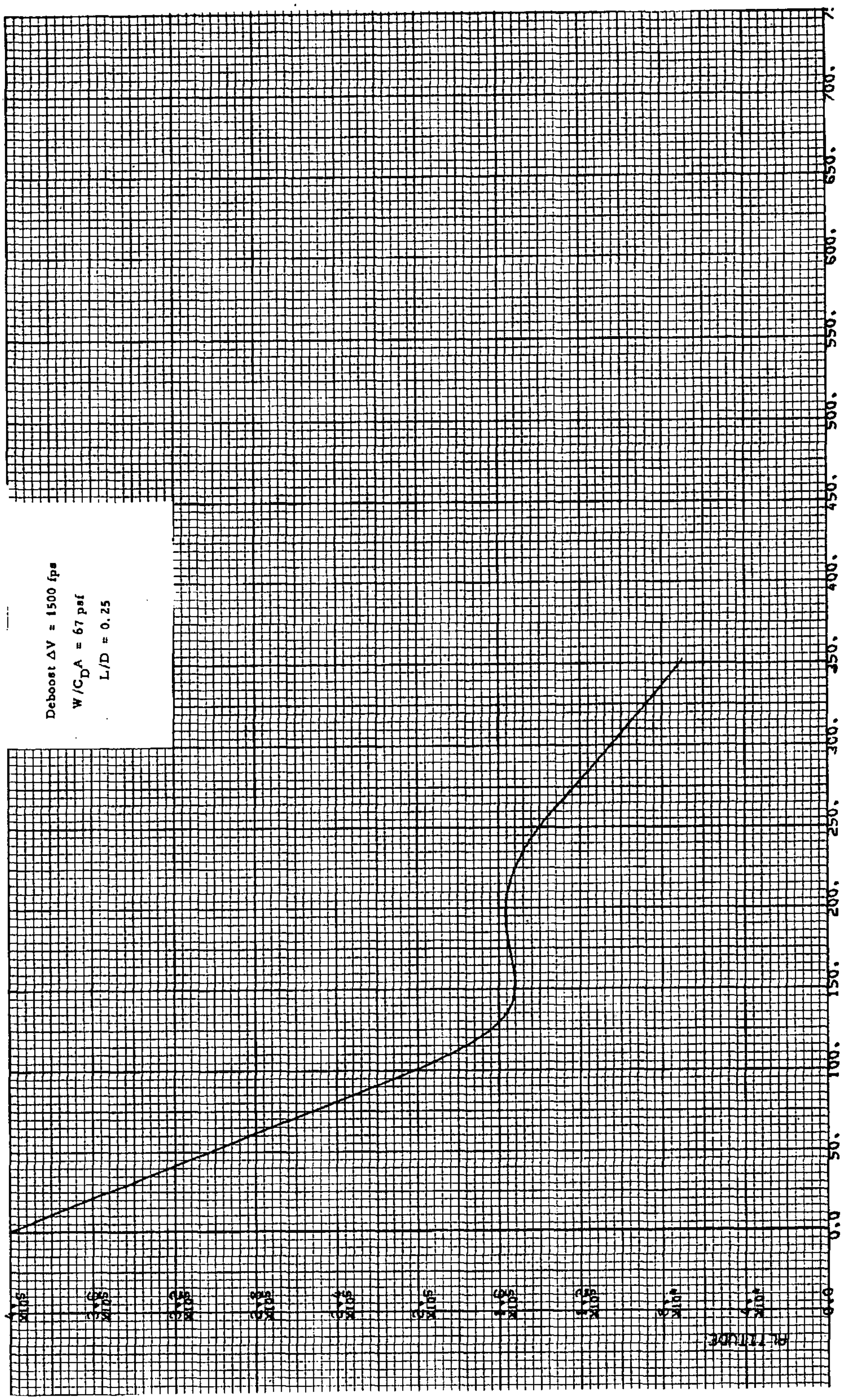

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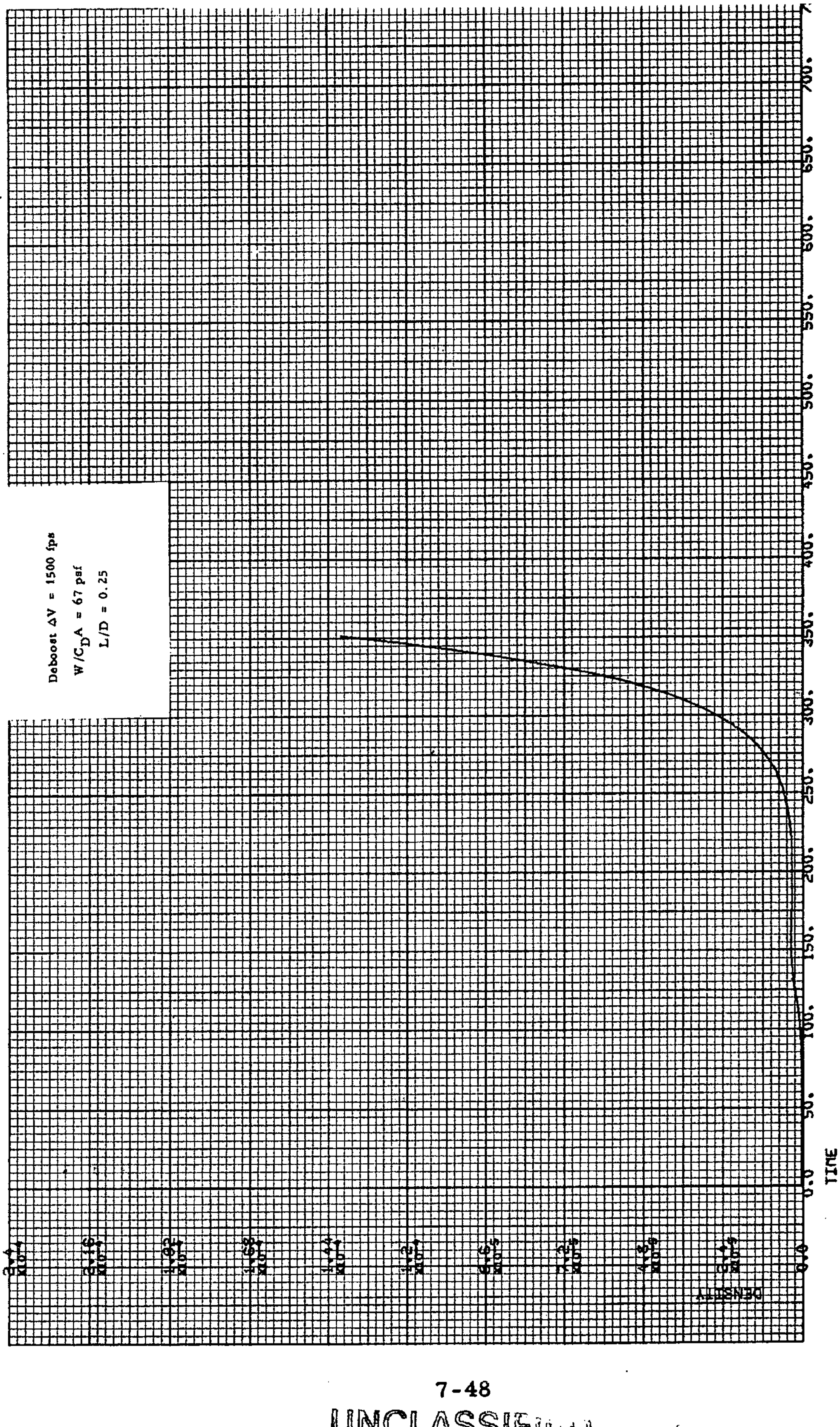

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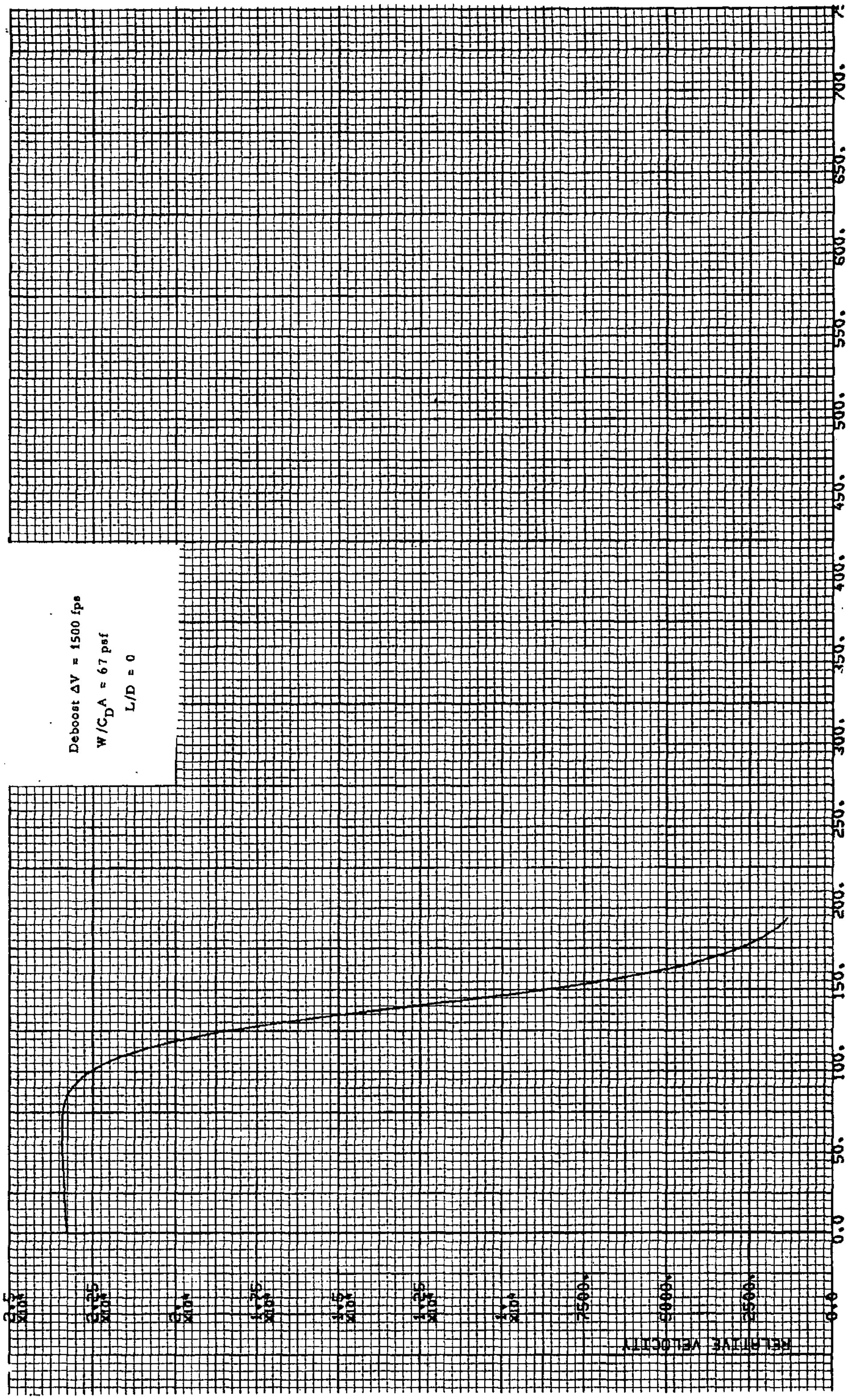

Figure 7-11a. Re-entry Time History.

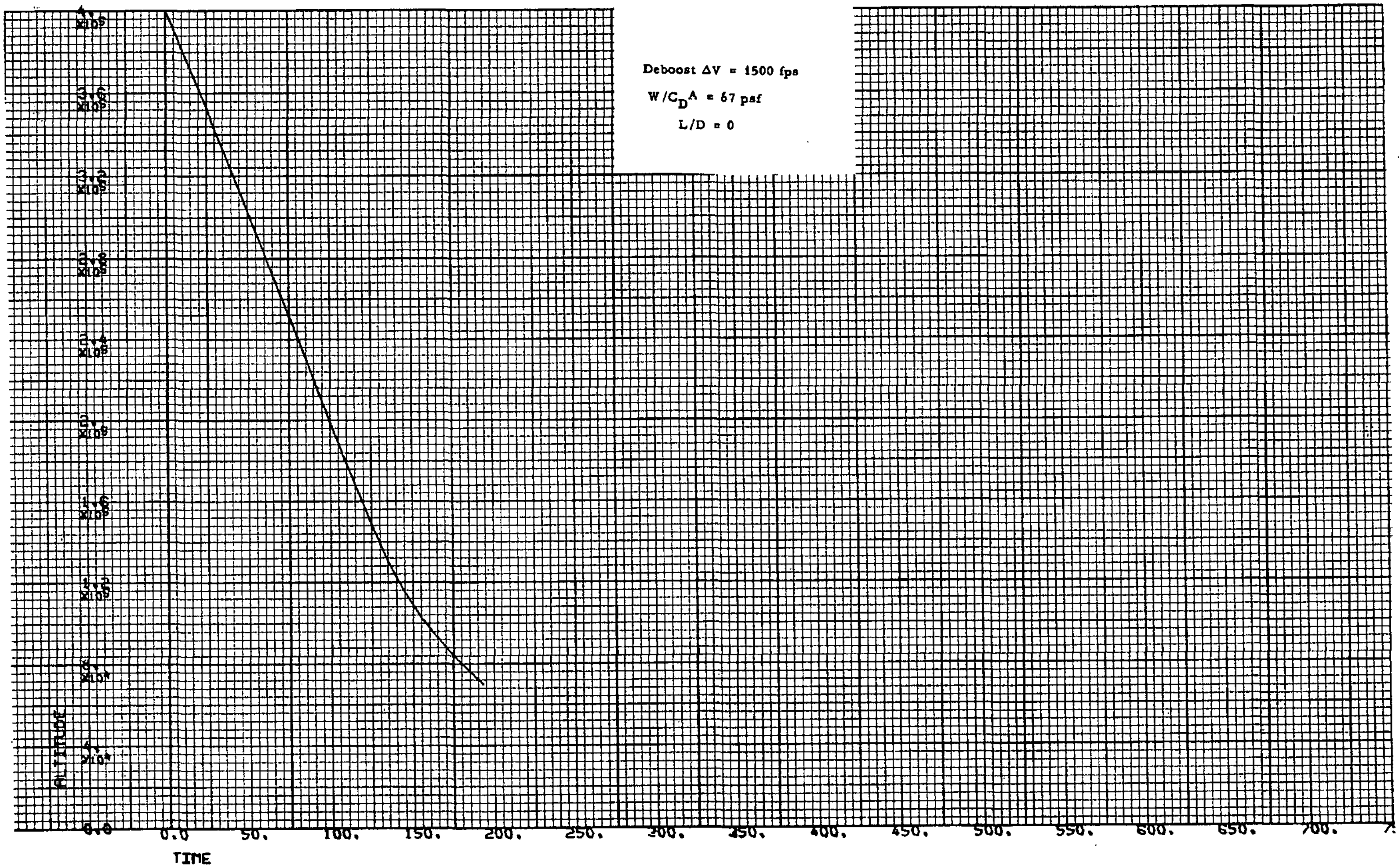

Figure 7-11b. Re-entry Time History.

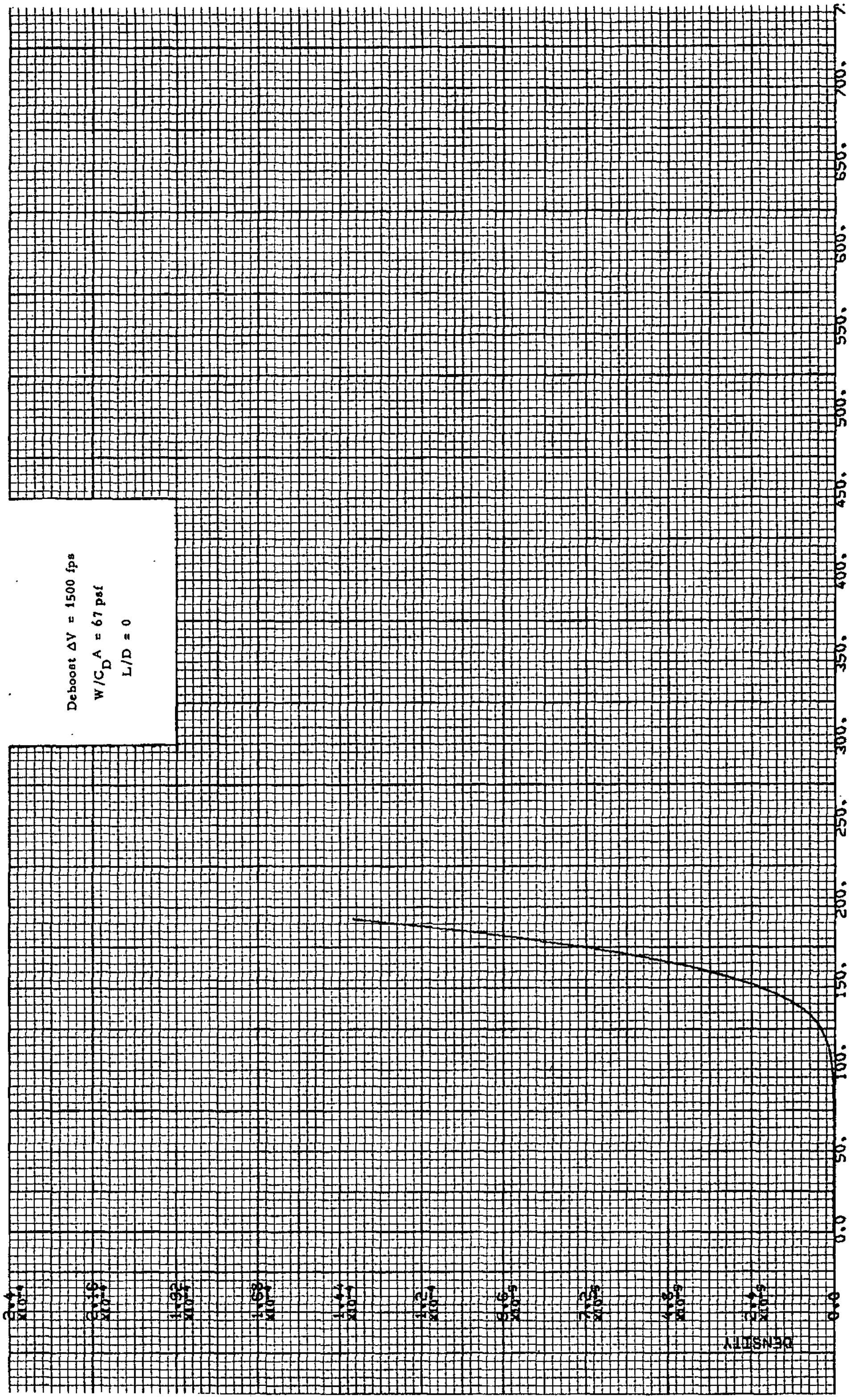

Figure 7-11c. Re-entry Time History.

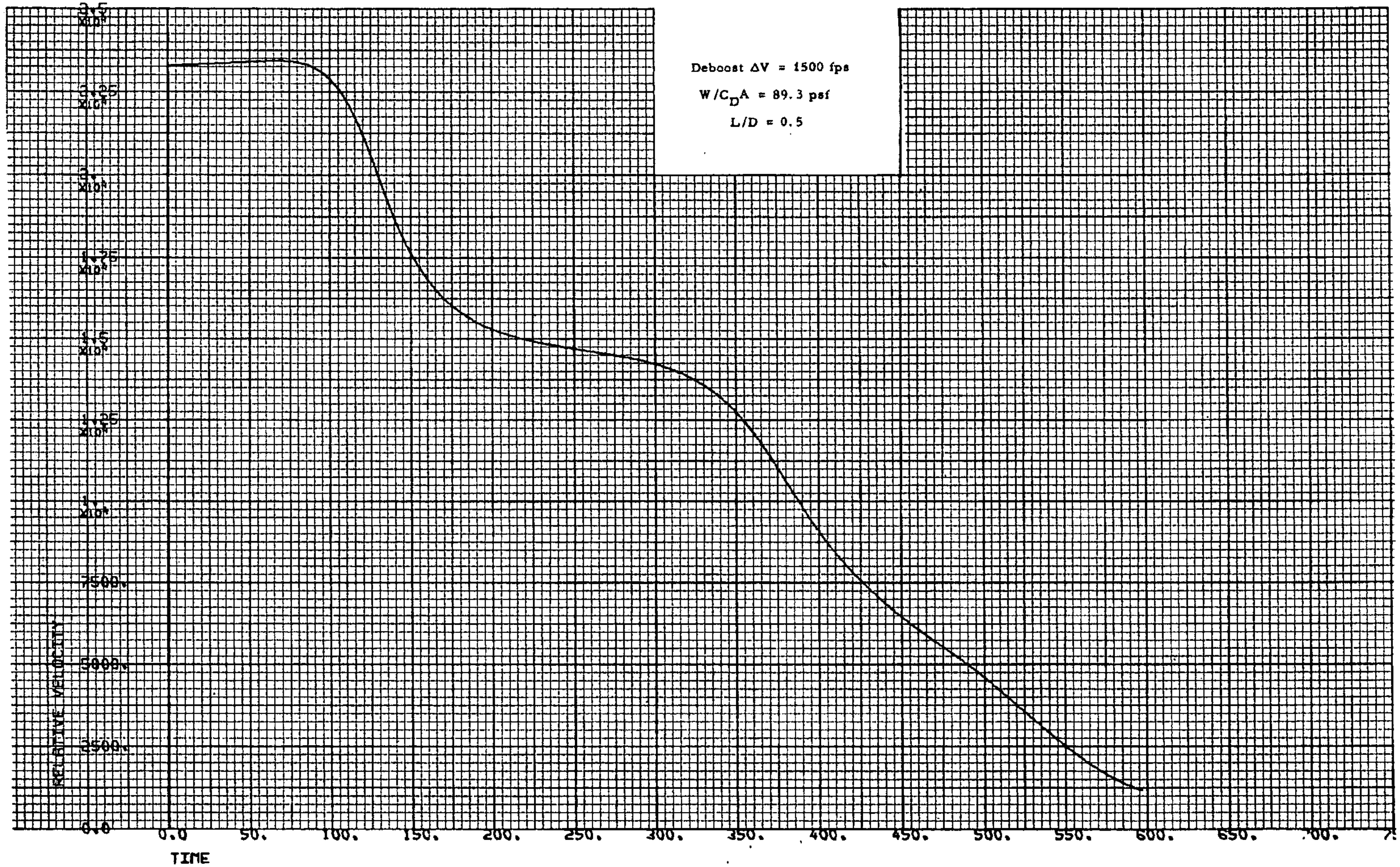

Figure 7-12a. Re-entry Time History.

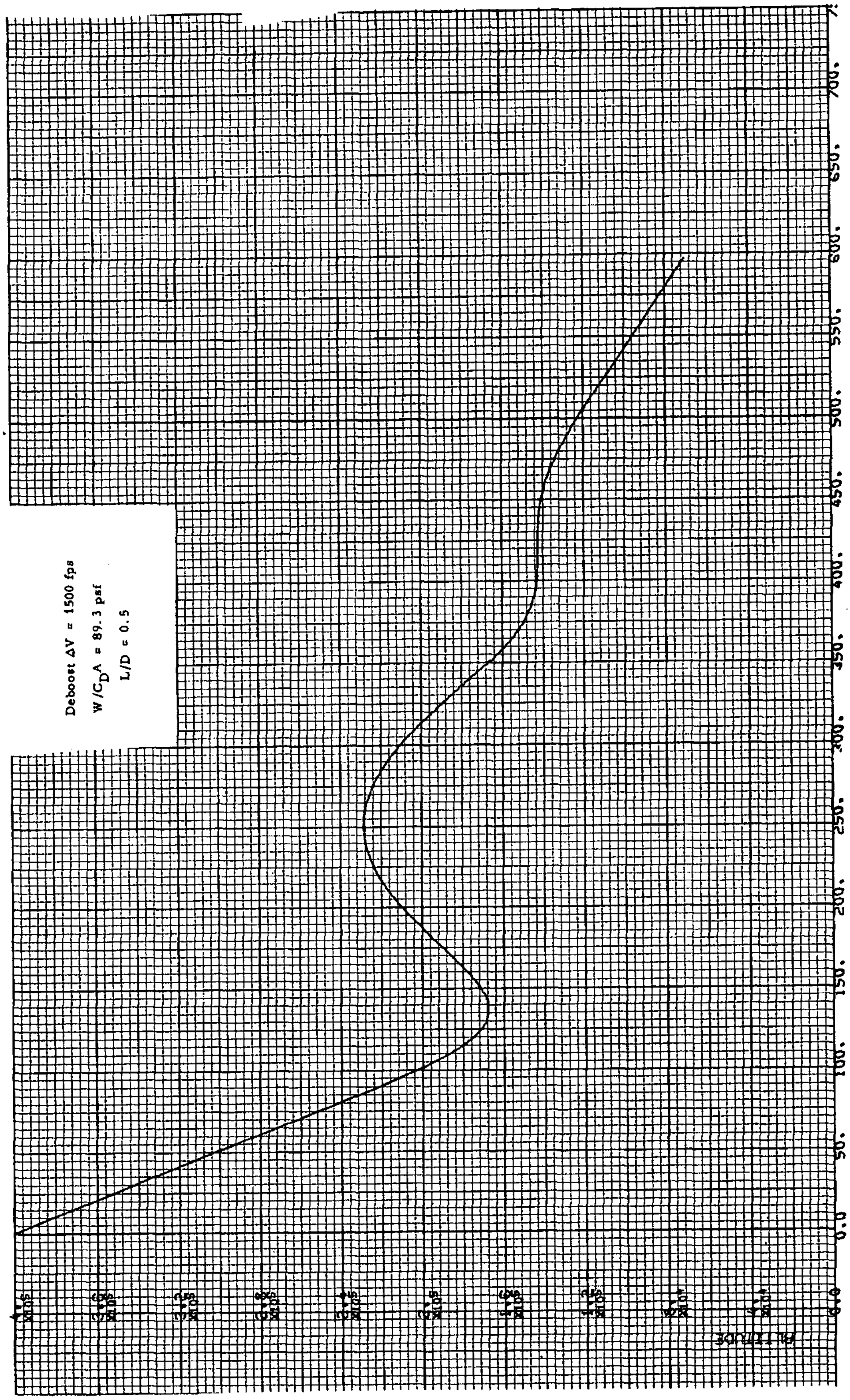

Figure 7-12b. Re-entry Time History.

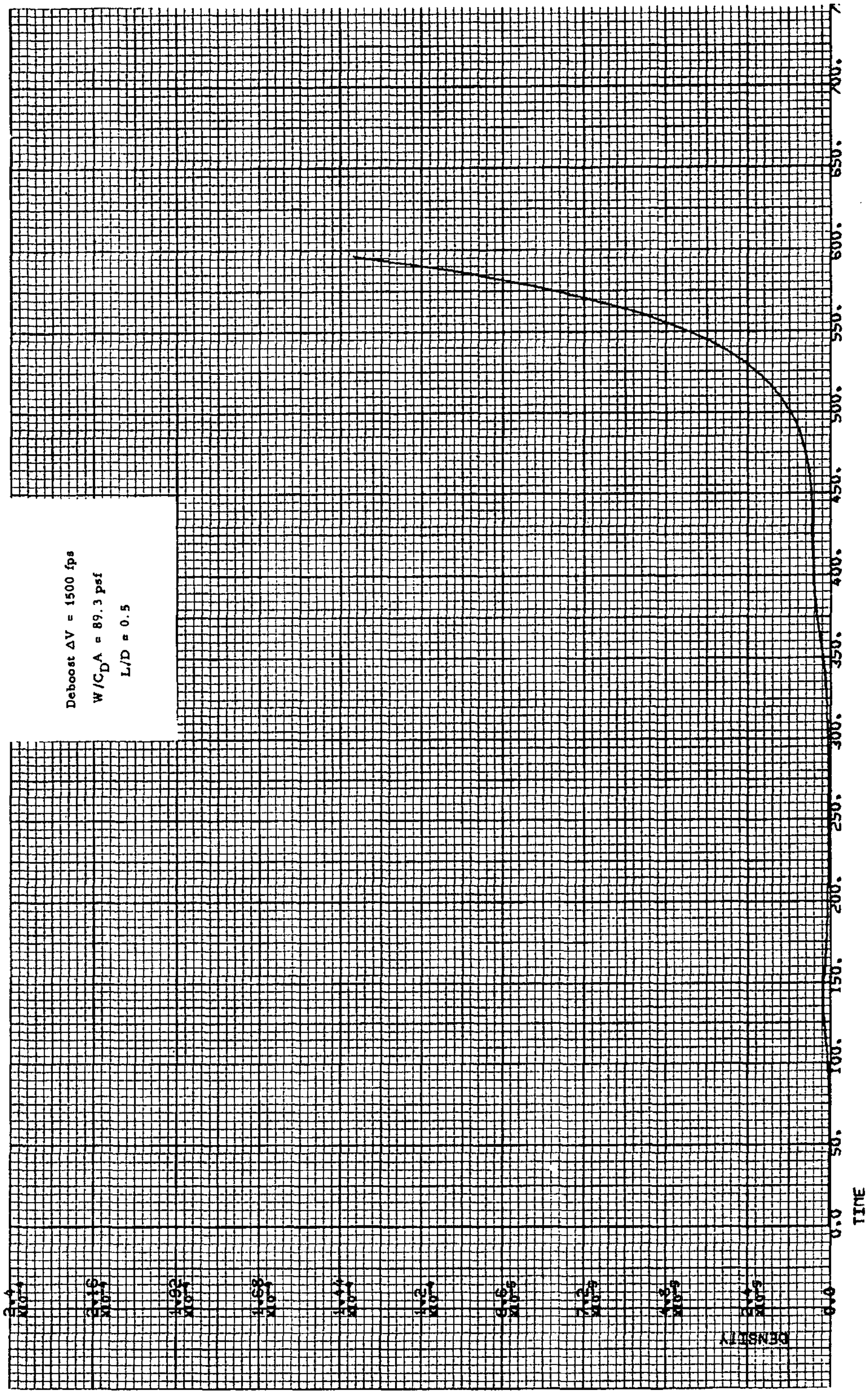

Figure 7-12c. Re-entry Time History.

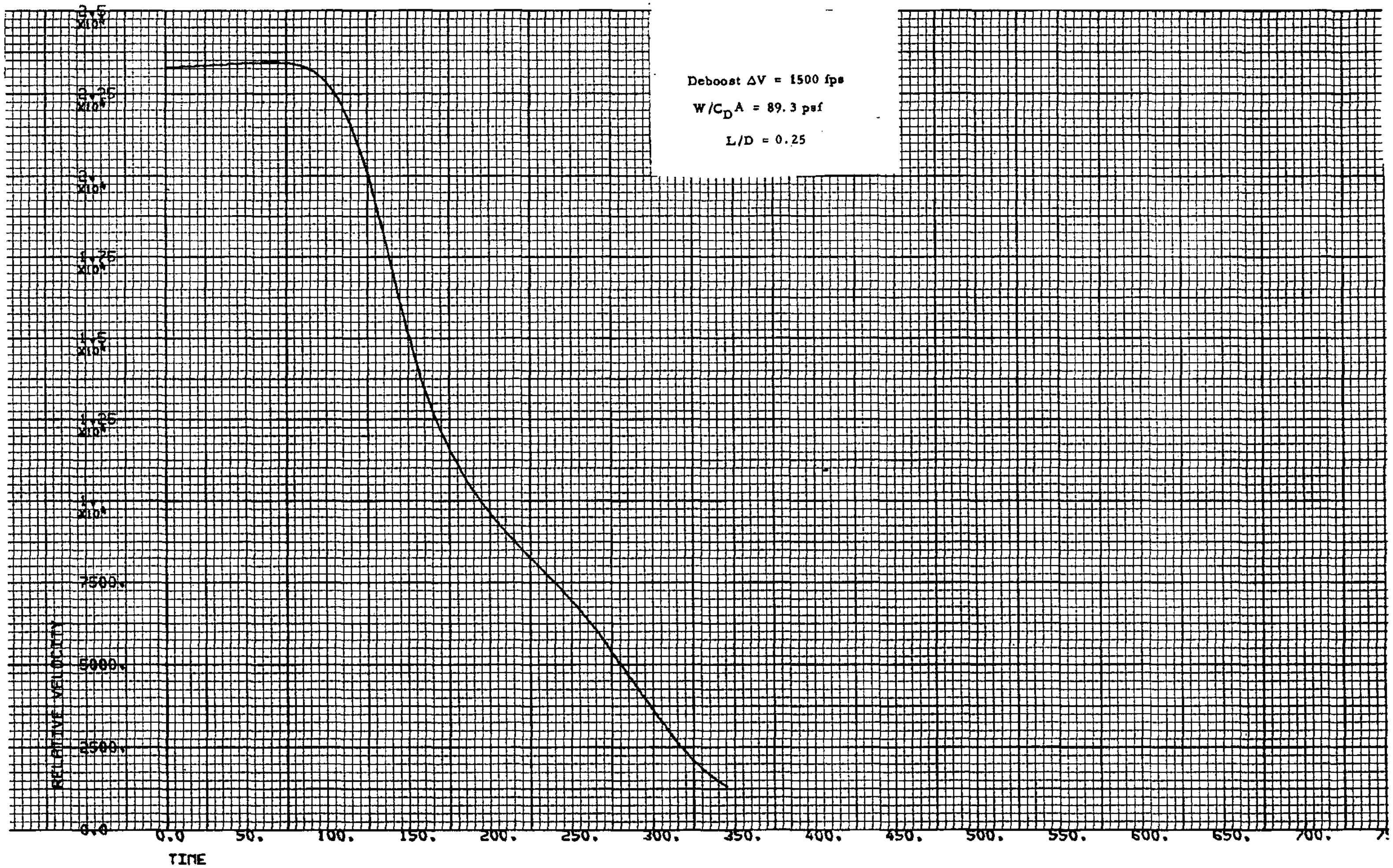

Figure 7-13a. Re-entry Time History.

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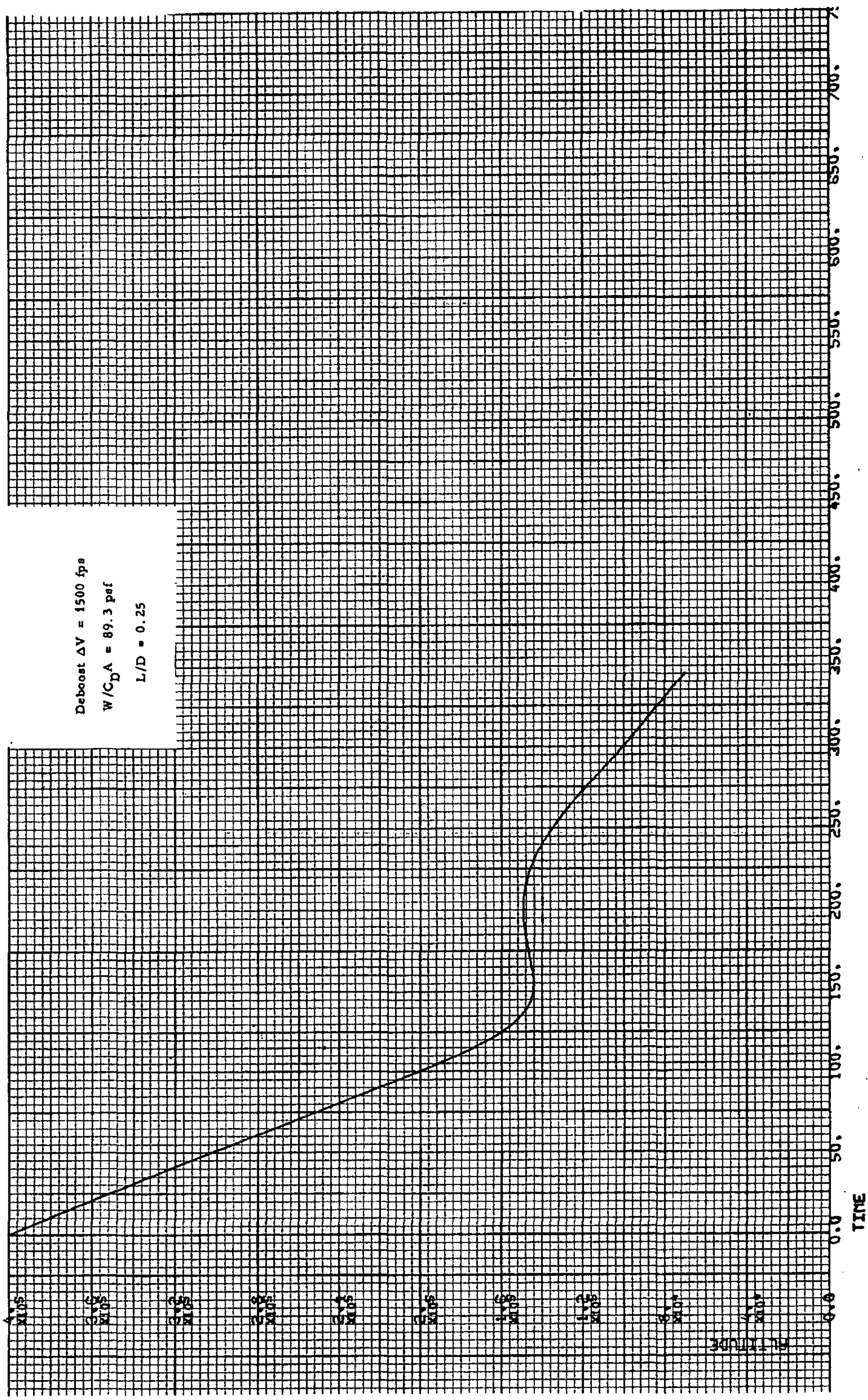

Figure 7-13b. Re-entry Time History.

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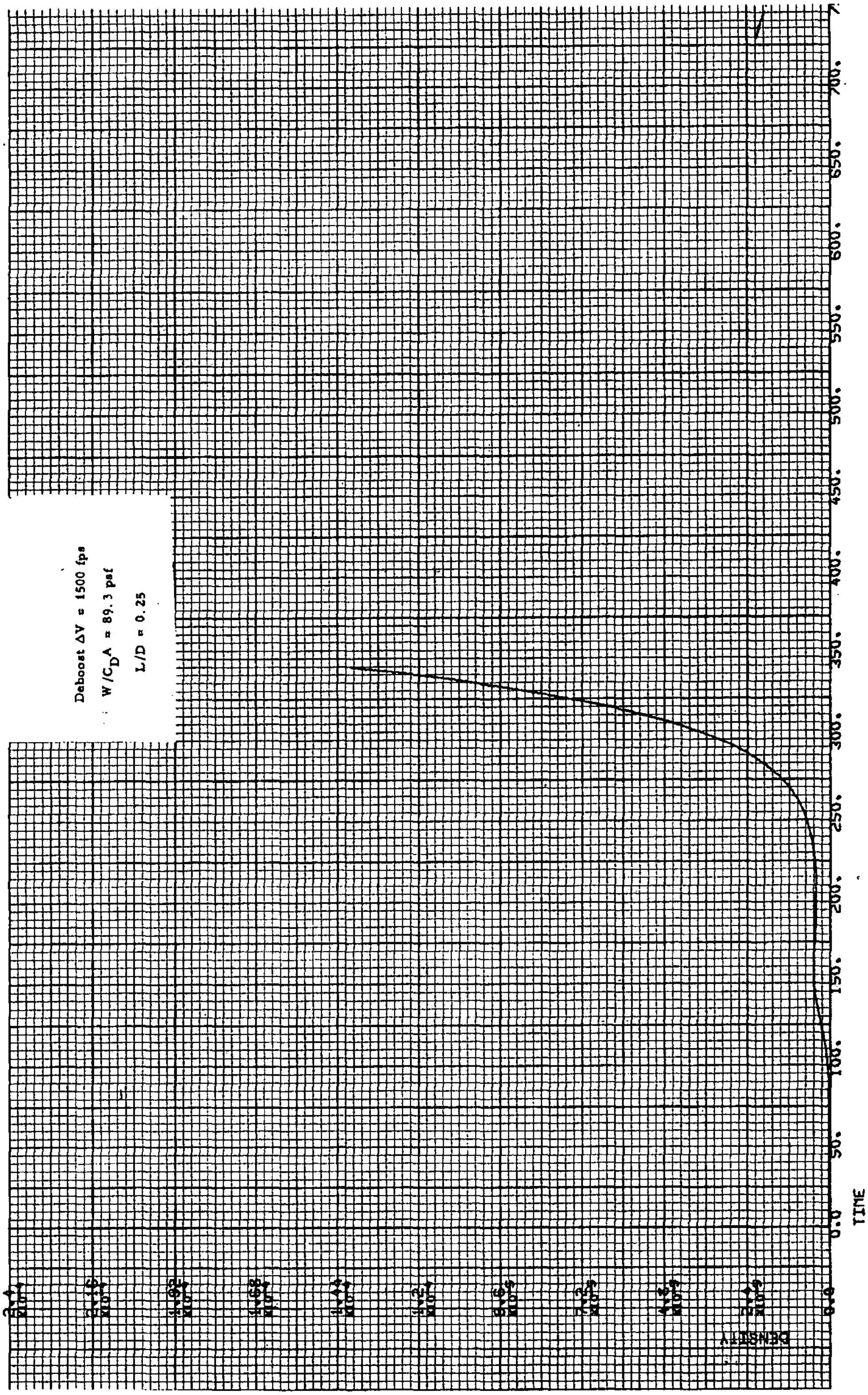

Figure 7-13c. Re-entry Time History.

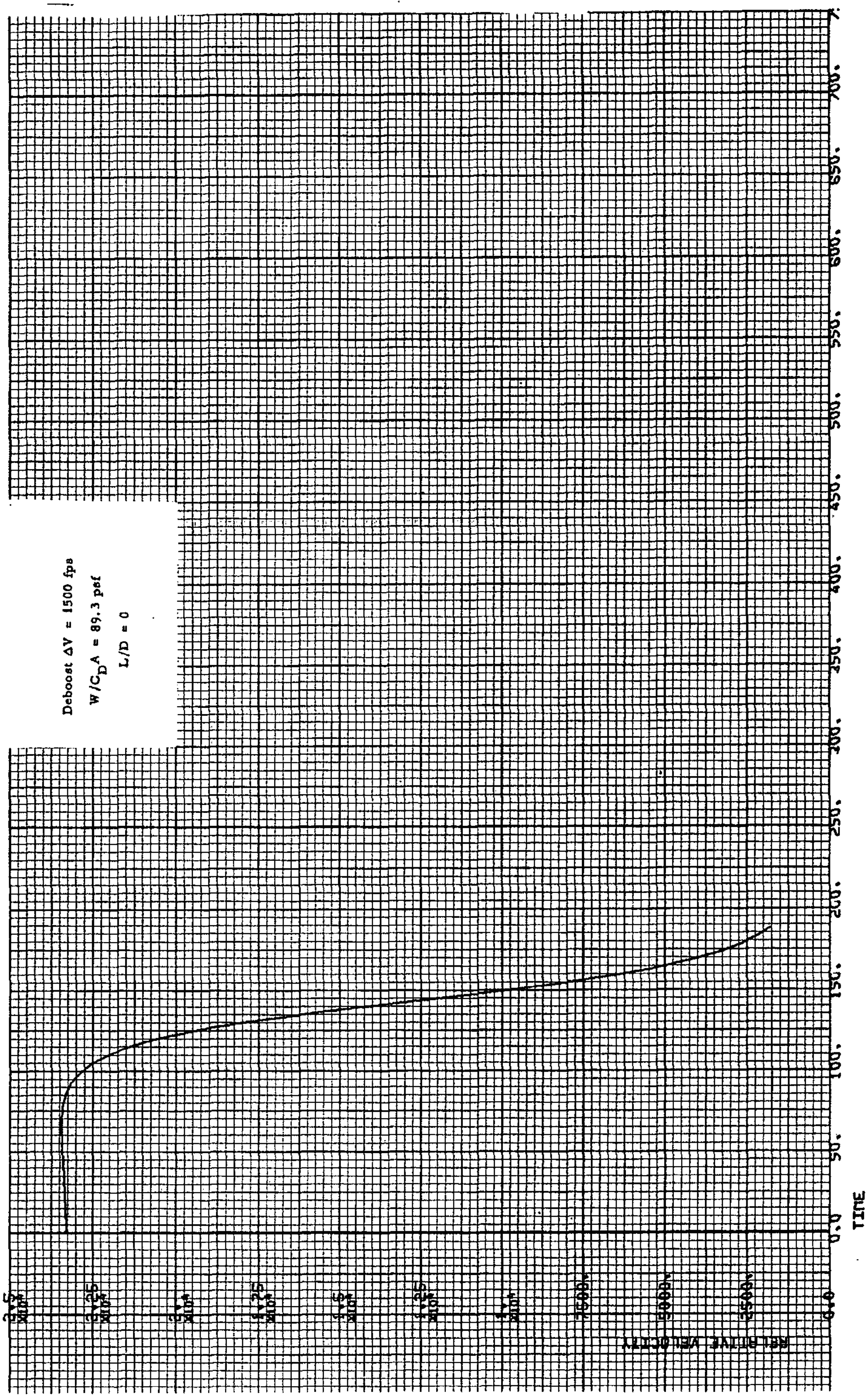

Figure 7-14a. Re-entry Time History.

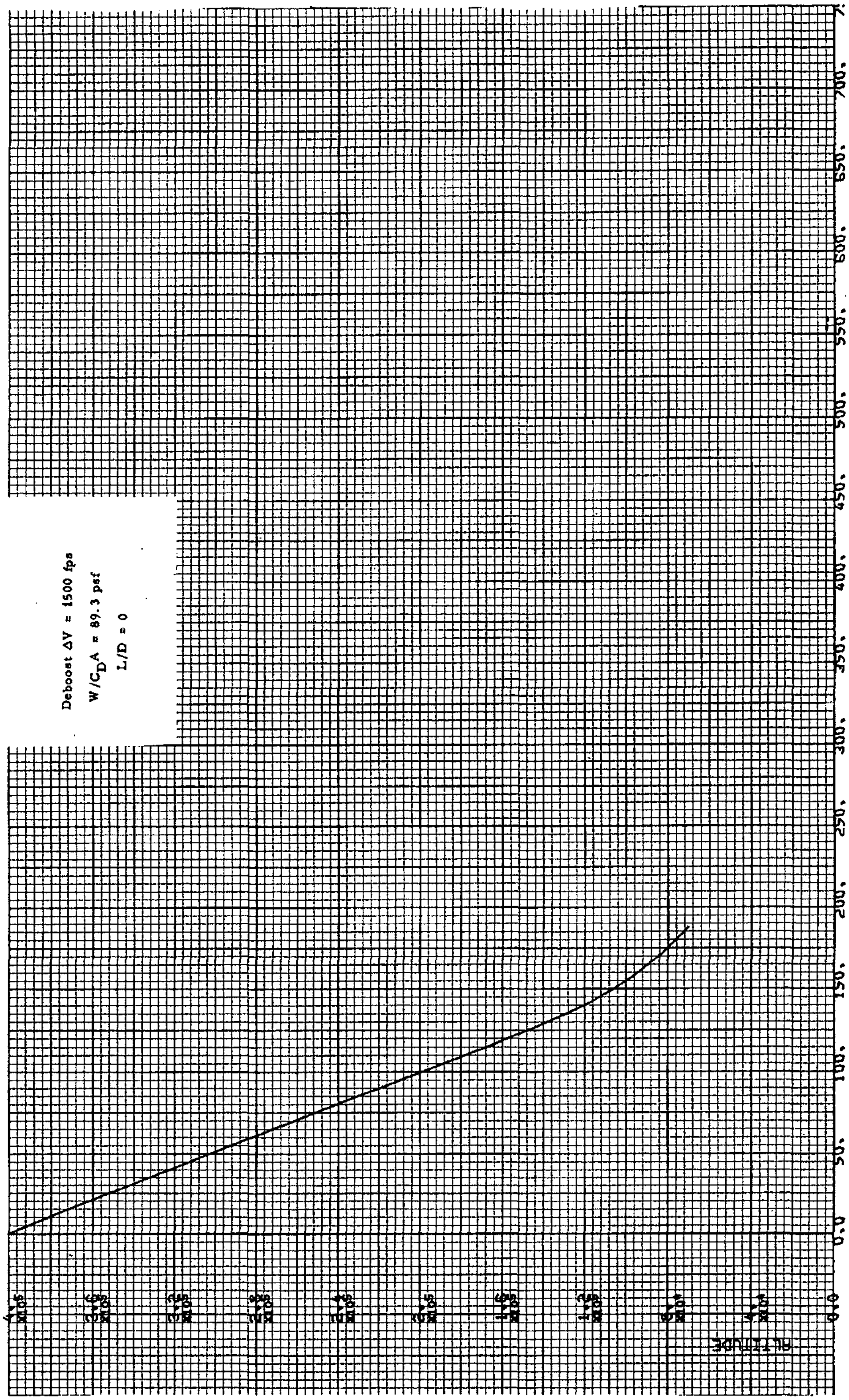

Figure 7-14b. Re-entry Time History.

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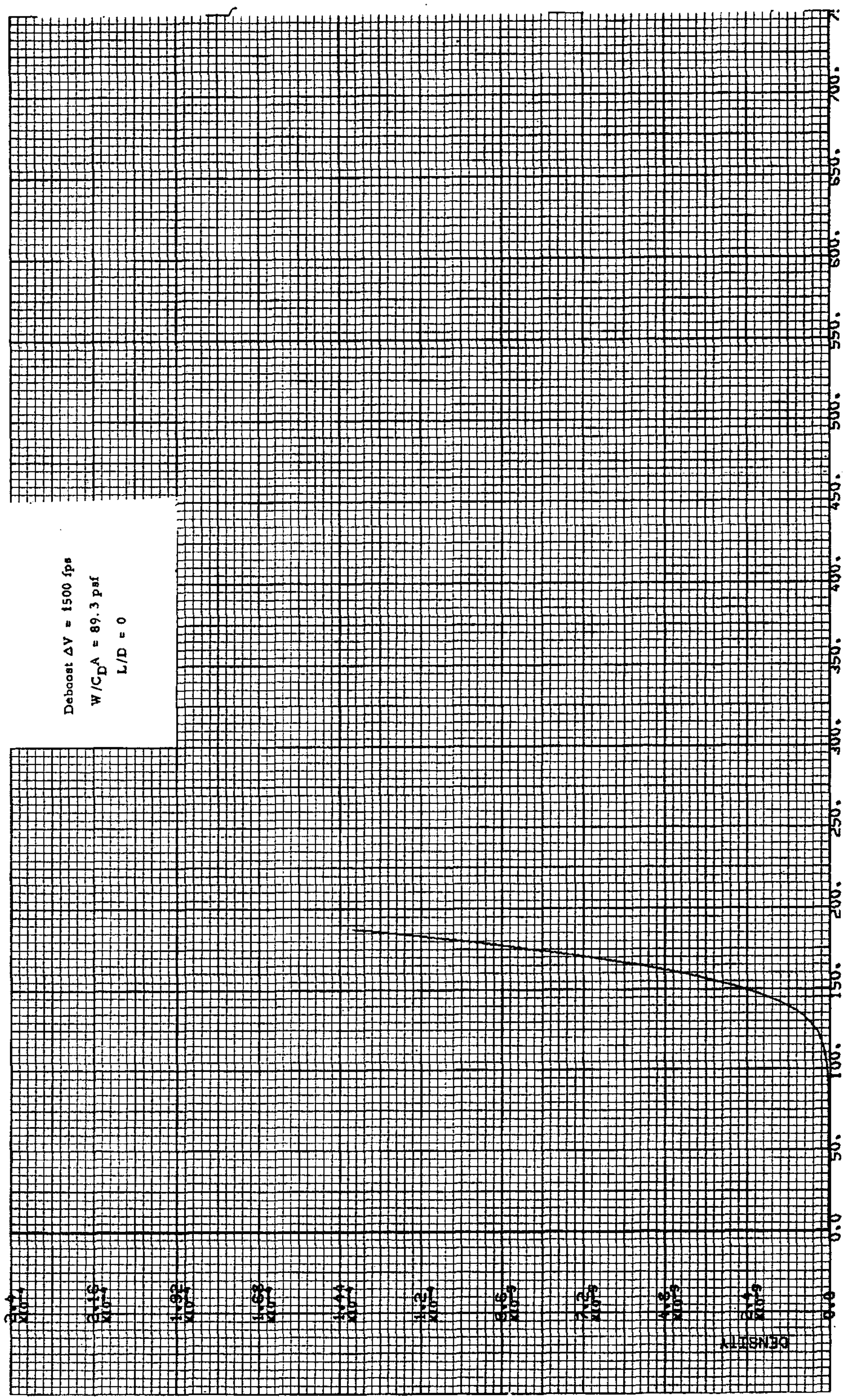

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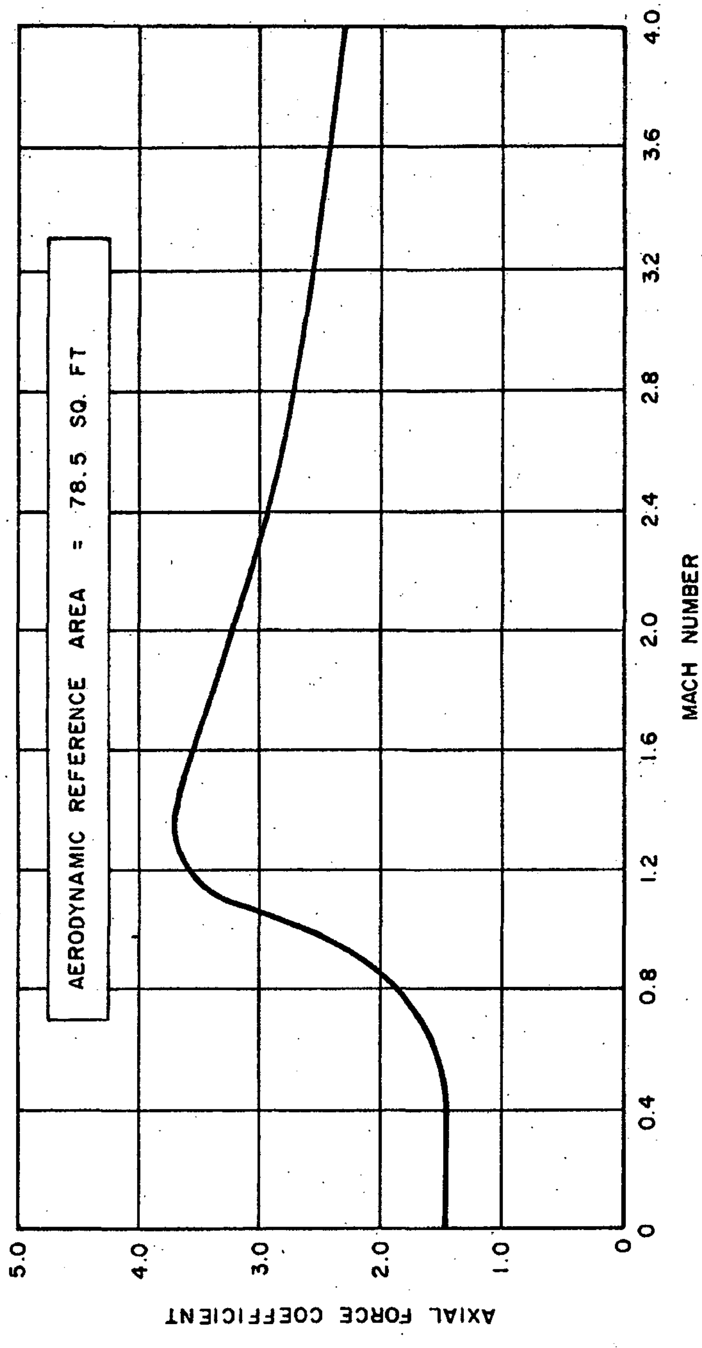

Figure 7-15. Titan IIIC/Apollo Aerodynamic Characteristics

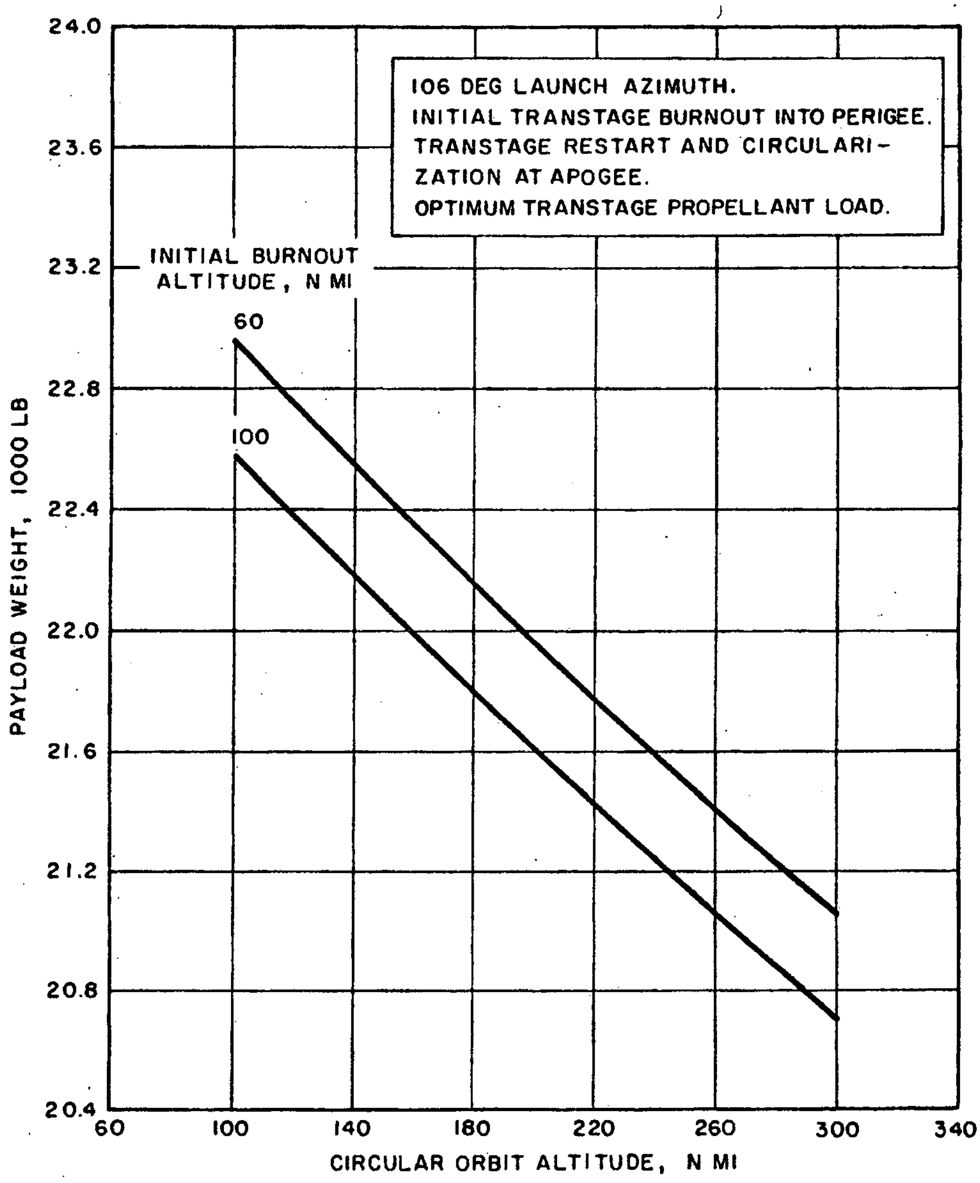

Figure 7-16. Titan IIIC Payload Capability vs Circular Orbit Altitude

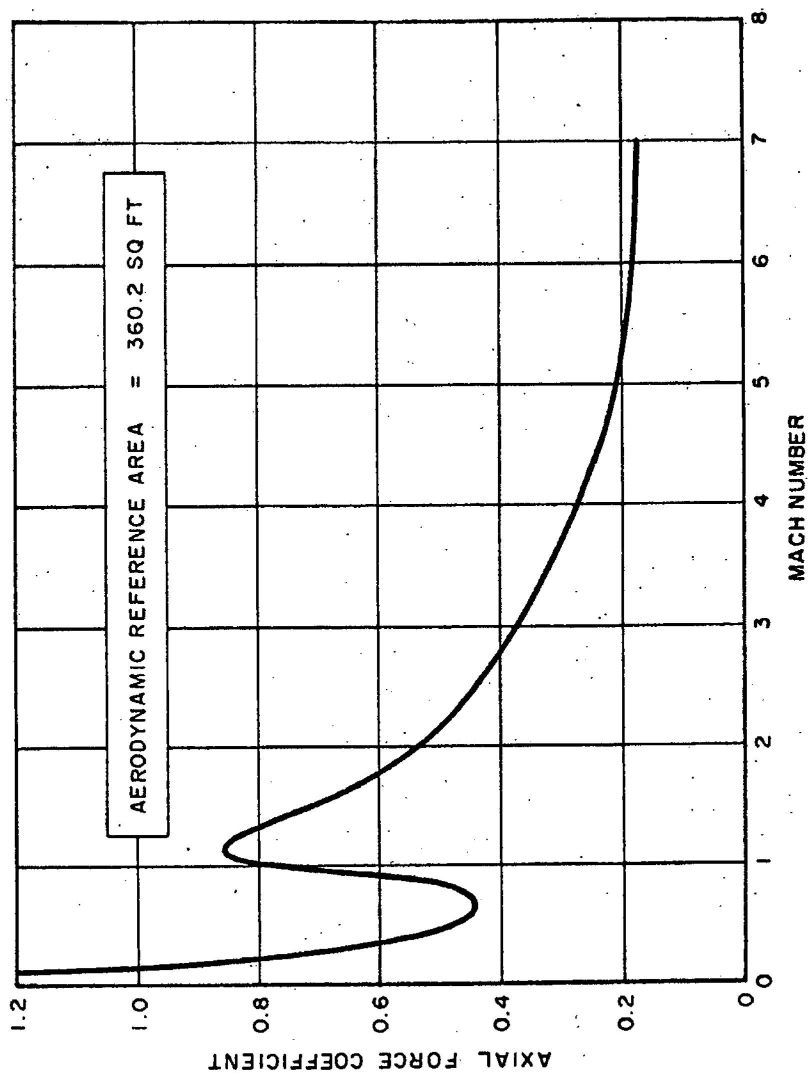

Figure 7-17. Saturn IB/Apollo Aerodynamic Characteristics

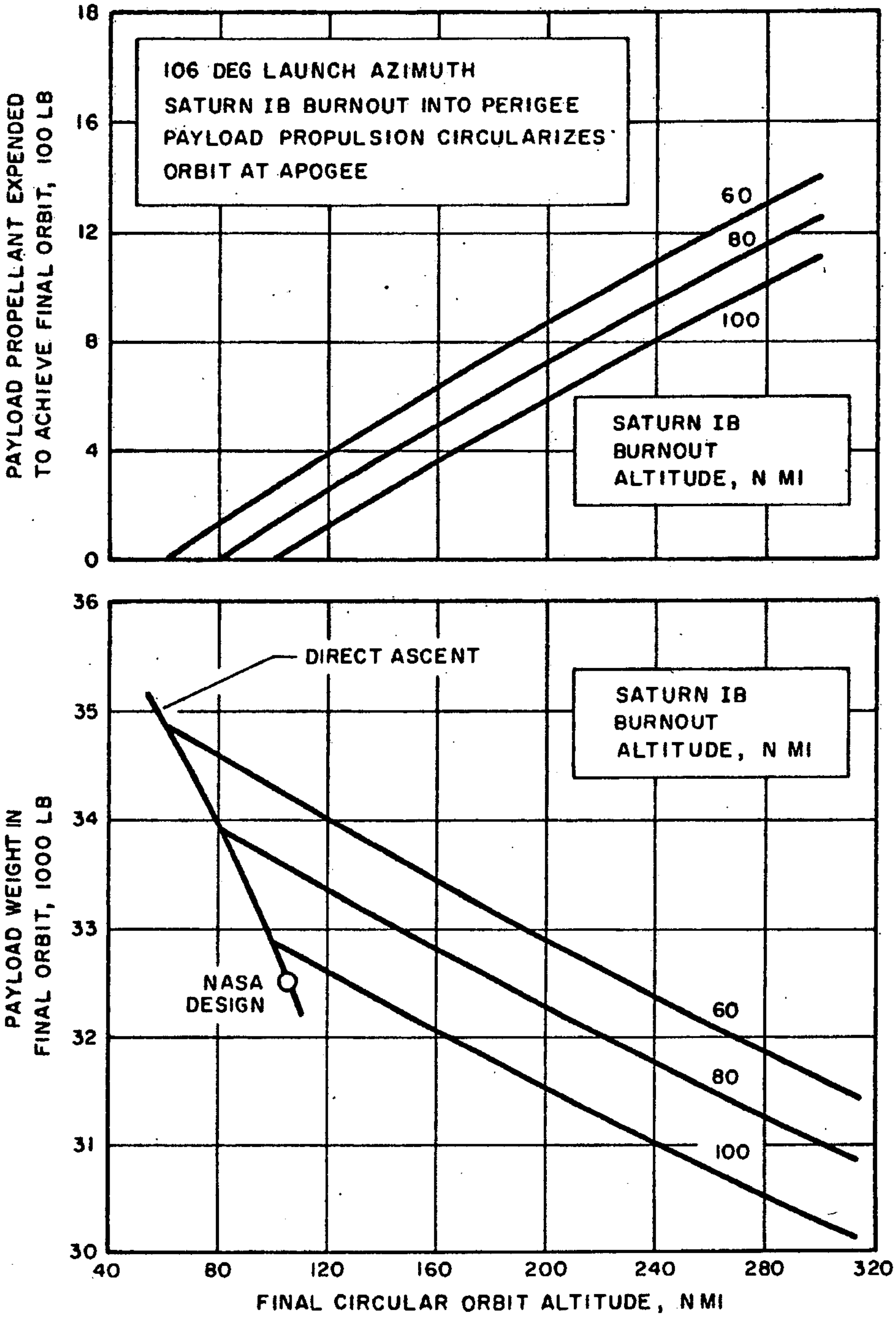

Figure 7-18. Saturn IB Payload Capability

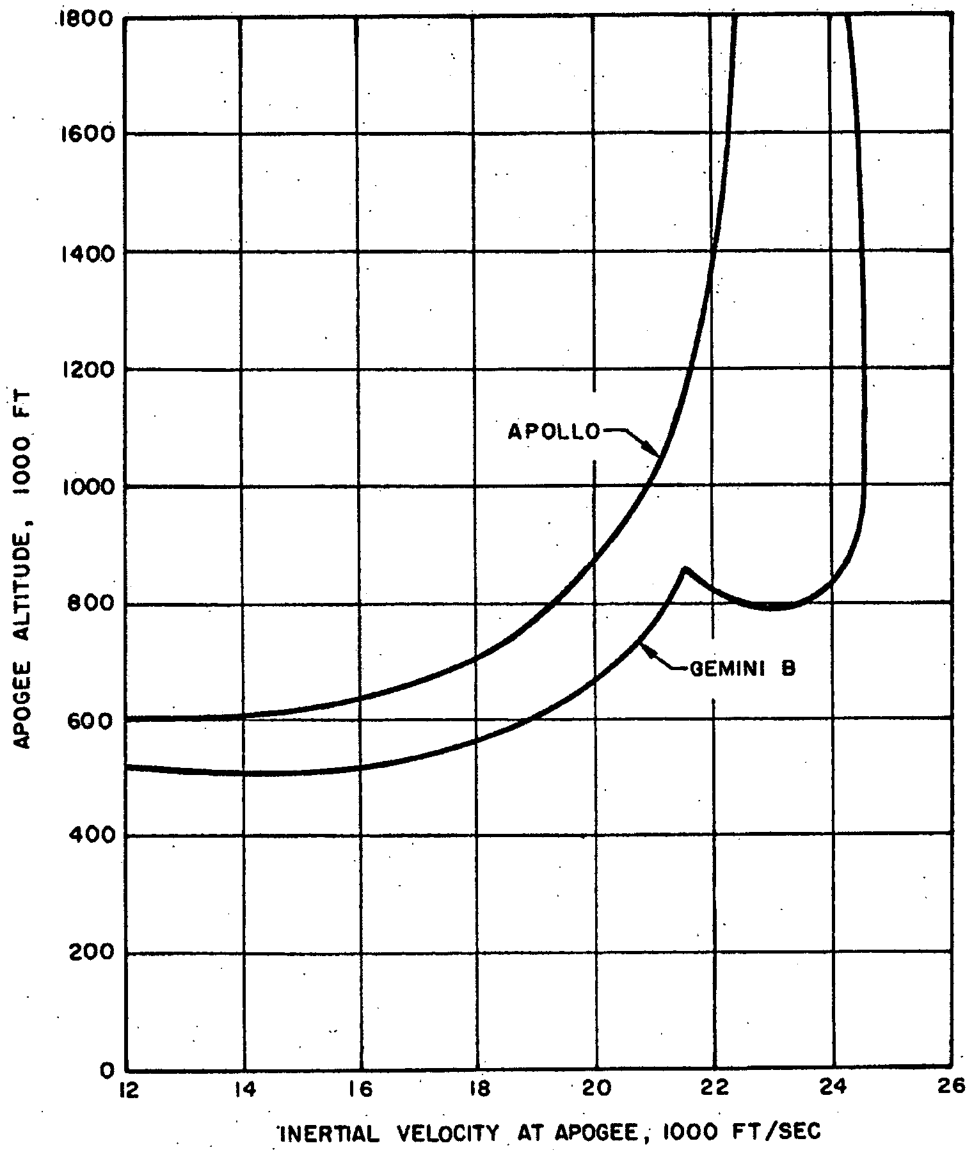

Figure 7-19. Comparison of Apollo and Gemini B Abort Ceilings

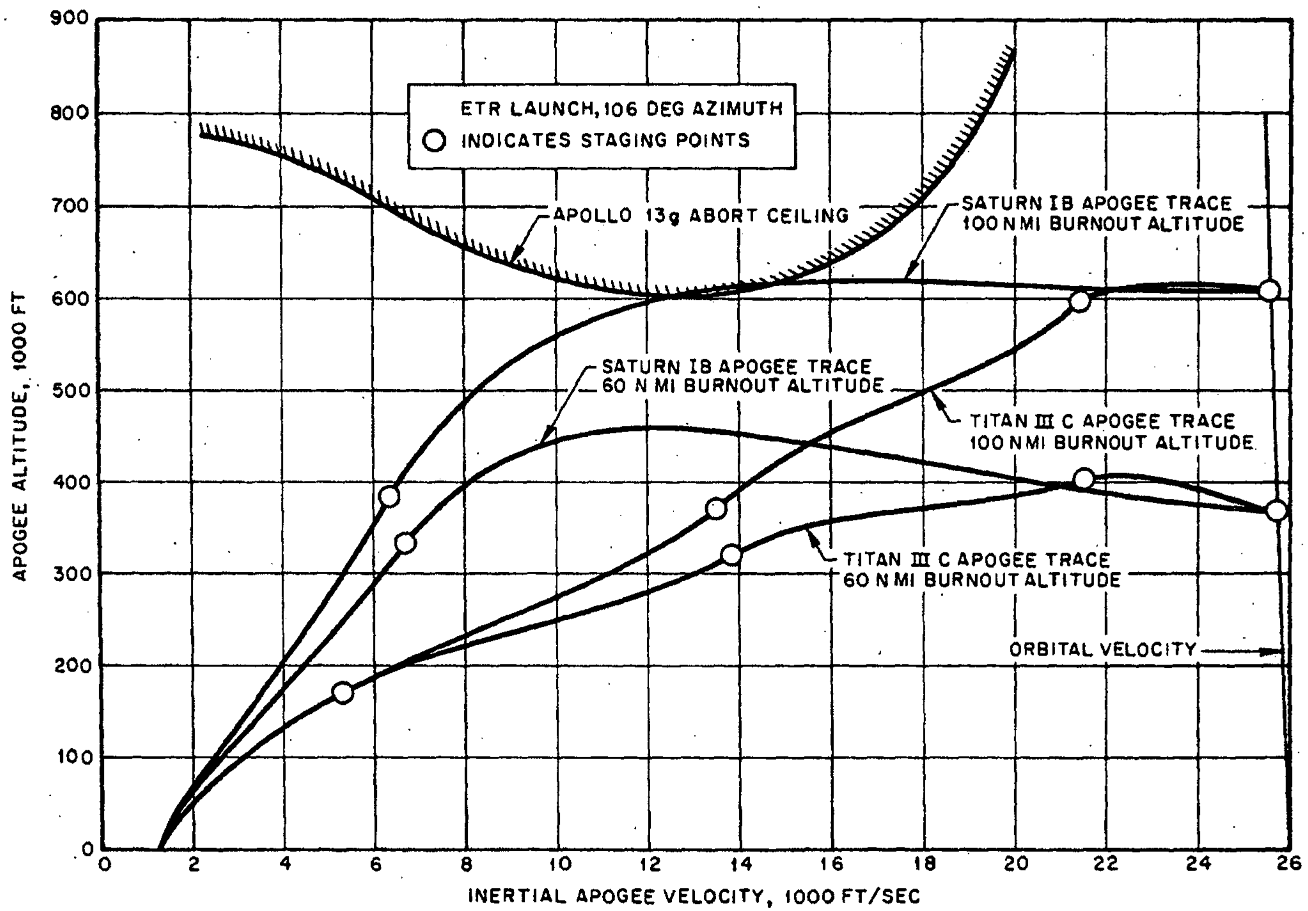

Figure 7-20. Launch Vehicle Apogee Traces and Apollo Abort Ceiling

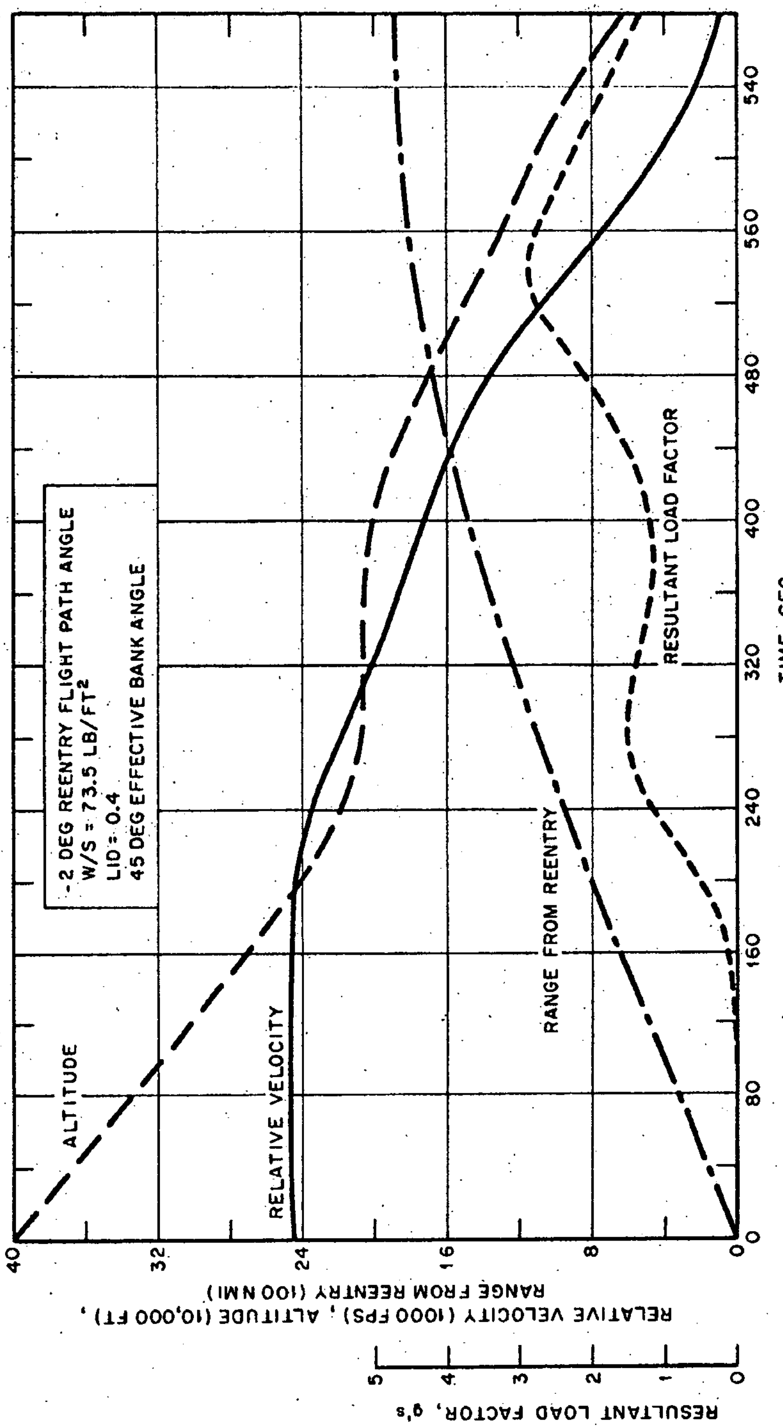

Figure 7-21. Typical Apollo Reentry Trajectory

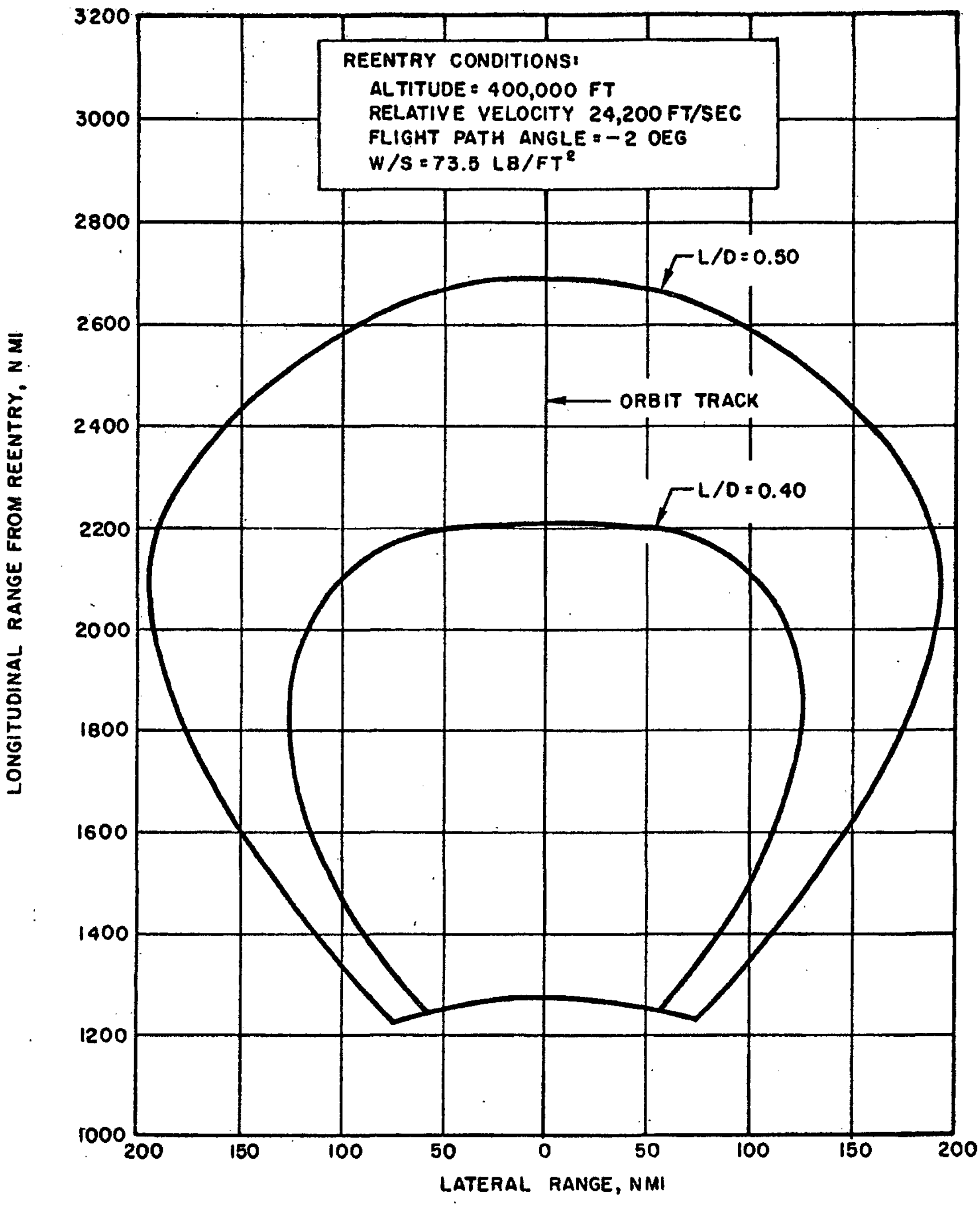

Figure 7-22. Apollo Landing Footprint

REFERENCES

7-1 NASA TM X-603, "Static Longitudinal Aerodynamic Characteristics of a 0.07 Scale Model of a Proposed Apollo Spacecraft at Mach Numbers of 1.57 to 4.65," by James R. Morgan and Roger H. Fournier, September 1961 (Confidential).

7-2 "Program 624A Standard Data Book for Vehicle Performance Calculations, Revision VI," TOR-269(4116-22)-4, 15 June 1964 (Unclassified).

7-3 "Effect of Gemini B Abort Ceilings on Titan IIIC Ascent Performance," TOR-269(4535-15)-2, 29 May 1964, J. M. Dreyfus (Unclassified).

7-4 "Saturn IB Launch Vehicle Design Data, Revision VII," IN-P and VE-V-63-1, 23 March 1964 (Confidential).

7-5 "Extended Mission Apollo Study, Volume III," SID63-1370-3, 24 November 1963 (Confidential).

SECTION 8

TEST OPERATIONS

L. Cooper

SUMMARY

This section contains a comparison of various significant test operations aspects of the Gemini/MOL and Apollo/MOL programs, as currently envisaged, covering the period from final countdown to re-entry. It considers the characteristics of the ground support network required for Apollo/MOL and possible conflicts with other programs in its use. These conflicts can occur throughout the Apollo/MOL programs, but only in the early part of the Gemini/MOL program. Also, the unmanned development flight requirements are examined, and it is estimated that one such flight is required, as compared with four for Gemini/MOL.

CONTENTS

8. TEST OPERATIONS

8.1 INTRODUCTION

This section is concerned with the test operations aspects of the Apollo/MOL, how they differ from the Gemini/MOL and a comparison of these program considerations.

The test operations area is considered to encompass flight operations, covering the period from final countdown to re-entry.

8.2 ASSUMPTIONS AND CONSTRAINTS

In a preliminary study of this nature, any planning or definition must be based on a series of initial assumptions, since no portion of the system has been contractaully defined at this point. For the purposes of this report, the following assumptions have been made:

1. Baseline configuration will be the Saturn IB launch vehicle and the modified Apollo vehicle described in Section 1.3.4, with minimum modifications.

2. Payload capacity will be sufficient to carry all experiments on a single flight.

3. Planning will be oriented to the earliest possible achievement of manned flight.

4. Each experiment will be flown at least twice.

8.3 TEST PLANNING CONSIDERATIONS

The following items delineate areas of primary consideration in flight test planning and comparisons of these areas for the Apollo/MOL and Gemini/MOL programs. Summaries of these comparisons are presented in Table 8-1, Areas of Similarity, and Table 8-2, Areas of Dissimilarity.

8.3.1 Mission Control Center

There are two possible choices for the location of the control center - the NASA MSFC at Houston or the Mission Control Center at CKAFS. The Gemini/MOL Program Office is currently studying these areas, and indications are that the

MCC at Cape Kennedy will be selected. It should be noted that the considerations determining control center selection, such as interference with NASA operations and avilability of secure communications, are the same for both the Apollo/MOL and the Gemini/MOL programs. Therefore, this area does not affect the program comparison.

8.3.2 Trajectory

The trajectory considerations for both the Apollo/MOL and the Gemini/MOL programs are similar. Launch azimuths currently under consideration are 73 and 106 degress, resulting in an orbital inclination of 32.5 degrees. Final selection of the launch azimuth will be based primarily on considerations of experiment, abort recovery, and range safety requirements. Injection into the final circular orbit, at an altitude of 160 nautical miles, will take place in the vicinity of Carnarvon, Australia.

Since the trajectories are similar for both programs, this area does not affect the program comparison.

8.3.3 Ground Support Network

8.3.3.1 Stations

Information supplied by the Gemini/MOL Program Office indicates that current planning calls for a combined NASA/DOD ground network comprised of the following stations:

CKAFS
Antigua
Bermuda (NASA)
Grand Canary (NASA)
Carnarvon (NASA)
Okinawa (to be implemented)
Hawaii
Vandenberg Tracking Station
San Antonio

Current planning for Apollo/MOL calls for utilization of the Unified S-Band communication system. Information, informally obtained, indicates that the NASA Unified S-Band equipment is not compatible with the DOD Integrated S-Band system. Therefore, it must be assumed that the Apollo/MOL program will utilize the NASA ground station network which will contain the following stations:

Guam
Hawaii
Guaymas
West Texas
CKAFS
Antigua
Bermuda
Ascension
Carnarvon

8.3.3.2 Visibility Considerations

An investigation of the Gemini/MOL network station visibility for each orbit is in progress, but has not been completed at this time. Preliminary results indicate that, while all stations do not see the vehicle on every orbit, the laboratory will be visible to at least one ground station on every orbit, for either the DOD/NASA network of the Gemini/MOL program, or the NASA Unified S-Band network of the Apollo/MOL program.

8.3.3.3 Security Considerations

Since the MOL program is a military program, it would seem desirable, especially in view of the requirement for growth capability, to have secure communications available for the ground station network. The Gemini/MOL DOD/NASA network has this capability at the DOD stations, but not at the NASA stations. The Apollo/MOL network stations have no provisions for communication security.

8.3.3.4 Schedule Interference

The NASA Gemini flight test program is scheduled for completion in early 1967. Current Gemini/MOL schedules call for the first manned flight in mid-1967. If all schedules are met, there will be no ground station operational interference between the NASA Gemini and Gemini/MOL programs. Even if the NASA program slips, there will only be interference during the early Gemini/MOL flights.

The NASA Apollo program will be in operation during the same time span (1968-1970) as the Apollo/MOL program. Since the same ground stations will be used for both programs, the probability of program interference must be considered. The schedules for Apollo and Apollo/MOL flight operations are based on the official NASA Apollo schedule, which contemplates four flights per year through 1967 and six flights per year starting in 1968, and on a more probable Apollo schedule, determined from conversations with NASA officials, which contemplates a flight every four months. Based on the official schedule, and assuming 30-day flight durations for the Apollo/MOL and 14-day flight durations for the NASA Apollo earth-orbit missions, there does not appear to be significant interference in ground station operations for the two programs. This, however, does not allow for slippage in either program. The probable actual schedule indicates at least three Apollo/MOL flights on which both MOL and NASA Apollo vehicles will be in orbit at the same time supported by the same NASA ground stations. Schedule slippage could further aggravate this condition. Conversations with NASA Goddard SFC and NASA Hq. personnel indicate that turn-around time for a specific station (the minimum time required from loss-of-sight of one orbital vehicle until a second vehicle can be acquired) could vary from a few minutes to a maximum of approximately half an hour. Therefore, it seems possible to encounter orbits, when coverage is available from only one or two stations due to visibility limitations, where ground station interference could result in complete loss of coverage for one orbit. This possibility could probably be reduced by careful scheduling, however.

8.3.4 Recovery Sites

The minimum number and location of planned landing sites for a mission depends on such factors as the loiter capability, the orbital altitude and inclination, and the L/D ratio of the re-entry vehicle. For the purposes of this report, the orbital altitude and inclination can be considered the same for both programs, and eliminated from the comparison. The L/D for the Apollo/MOL re-entry vehicle is approximately 50 percent greater than for the Gemini/MOL vehicle. For a loiter capability of one orbit, the Gemini/MOL vehicle requires five planned landing sites, while the Apollo/MOL vehicle would probably require only four planned sites. If the loiter capability is increased to one half-day, the effect of the different L/D is reduced, and the required number of landing sites is two for both programs.

8.3.5 Development (Unmanned) Flights

Current Gemini/MOL planning, as stated by the Gemini/MOL Program Office, calls for four unmanned flights associated with the program. The first two flights will determine the effects of the addition of the laboratory on the Titan IIIC/Gemini configuration and will be instrumented to measure the flight environment. They are not considered part of the MOL program; they will utilize Titan III research and development launches which do not have assigned payloads. Although this provides a substantial vehicle cost saving, it also allows possible conflict between Titan III research and development and MOL flight test objectives. The first two flights of the MOL program will also be unmanned and will verify the laboratory structure and the Gemini B capabilities (30-day life, crew safety and life support, etc.).

It is expected that an external configuration similar to the Apollo/MOL vehicle will be flown by the NASA Apollo program (Vehicle 206) in 1966 or 1967, prior to the first flight of the Apollo/MOL program. Since this flight, and subsequent flights, will provide structural, flight control, and environmental data which is considered applicable to the Apollo/MOL vehicle, the two development flights which precede the Gemini/MOL program can be eliminated for the Apollo/MOL. In addition, since the laboratory structure will be based on the previously tested LEM adaptor structure, it seems likely that sufficient ground testing will reduce the second two development flights of the Gemini/MOL to one flight for the Apollo/MOL program. On the other hand, this dependence on the NASA Apollo program for development data could result in program slippage if the NASA program slips significantly. In addition, present indications are that Saturn IB research and development flights will not be available to carry Apollo/MOL development payloads. Therefore, one or more additional Saturn vehicles will have to be purchased.

8.3.6 Earliest Manned Flight

According to the MOL Program Office, the first manned Gemini/MOL flight is scheduled for mid-1967. The earliest time that a manned Apollo/MOL could be achieved is late 1967 or 1968, depending on the NASA Apollo launch schedule. It should be noted that the Titan III research and development program is scheduled for completion approximately one year before the Saturn IB research and development program, and one year before the first scheduled Gemini/MOL manned flight. The Saturn IB research and development program is scheduled for completion less than one year before the first Apollo/MOL manned flight. Therefore, schedule slippage in the Saturn IB program could have a greater effect on the Apollo/MOL program than a corresponding slip in the Titan III research and development program would have on the Gemini/MOL program.

8.3.7 Launch and Orbital Operations

While the vehicles and, possibly, the crew size for the two programs under comparison vary considerably, the over-all launch operations and orbital operations (including range operations and housekeeping, maintenance, and experiment performance) considerations are, in general, quite similar. Therefore, this area does not affect the program comparison.

8.4 CONCLUSIONS

1. Communication security may be a problem with Apollo/MOL because the NASA communication network will be used.

2. It appears that Gemini/MOL might experience some ground station interference during early flights, while Apollo/MOL could experience some interference throughout the program. In both cases, the interference could probably be reduced, if not eliminated, by careful scheduling. The effects of interference could be reduced or eliminated if the necessity for contact with the orbiting vehicle at least once per orbit is modified.

3. For short-time loiter capability, the Gemini/MOL configuration might require a greater number of planned landing sites than the Apollo/MOL. For longer loiter capability, the number of sites required is the same for both programs.

4. The Gemini/MOL program requires two developmental flights (plus two preliminary "free" flights), while a total of one developmental flight is likely for the Apollo/MOL program. The Gemini/MOL might encounter interference problems with Titan III research and development objectives, and the Apollo/MOL program might be forced to slip if the NASA Apollo schedule slips.

Table 8-1. Gemini/MOL Program and Apollo/MOL Program Comparison Test Operations Areas of Similarity.

Mission Control Center
Trajectory
Launch Operations
Orbital Operations
Ground Station Visibility

Table 8-2. Gemini/MOL and Apollo/MOL Comparison Test Operations Areas of Dissimilarity.

	Apollo/MOL	Gemini/MOL
Ground Station Network	All NASA stations	Combination of NASA and DOD stations
Ground Station Secure Communications	No communication security	Secure communications only at DOD stations
Ground Station Schedule Interference	Possible interference on all flights	Possible interference on early flights
Recovery Sites	Four planned sites for one-orbit loiter capability. Two sites for half-day loiter capability	Five planned sites for one-orbit loiter capability. Two sites for half-day loiter capability
Development (Unmanned) Flights	1 or 2	4
Development Flight Vehicles	Saturn IB research and development vehicles not available	Titan III research and development vehicles can be used
Manned Flights	Minimum of 2	Minimum of 3
Crew Size	2 or 3	2
Earliest Manned Flight	Late 1967 or 1968	Mid-1967

SECTION 9

RELIABILITY

F. P. Klein

SUMMARY

Estimates of the reliability of the Apollo/MOL orbiting and re-entry systems are made, using the standard exponential reliability model. They are combined with the Saturn IB reliability, estimated in previous studies, to obtain an estimate of the overall reliability of the Apollo/MOL mission. Similarly, the reliability of the Gemini B and MOL laboratory are estimated and combined with that of the Titan IIIC to obtain the over-all Gemini/MOL reliability. The reliability estimates of the two systems are compared and found to be equal to within the accuracy of the estimates.

CONTENTS

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TABLES

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9. RELIABILITY

9.1 INTRODUCTION

The purpose of this study is to evaluate the capability of the proposed Apollo/MOL configurations to achieve mission success and to compare the mission success reliability capabilities of the Apollo/MOL and the Gemini B/MOL.

This section presents the prediction of system reliability for a 30-day MOL mission. The system as defined consists of launch vehicle (all booster stages), spacecraft, and orbiting laboratory. The MOL mission as defined consists of powered flight, orbital insertion, orbital laboratory operation, de-orbit, re-entry, and landing.

The study was limited to a reliability analysis of the launch vehicle, re-entry (command) module, and orbiting laboratory vehicle exclusive of experiments. The subsystems included in the laboratory vehicle analysis were limited to those necessary for normal vehicle operation (housekeeping) and service to the experiments. No maintainability capabilities or requirements were considered because the spares payload and available maintenance time have not been defined.

Maximum use was made of previous reliability studies of the systems under consideration. The results of these studies were evaluated for applicability to the MOL mission configuration. In cases where subsystem configurations were identical, the reliability analysis was applied directly to the MOL evaluation. If the subsystem configurations were modified for the MOL mission, a new reliability model was constructed which incorporated all reliability data available from the previous studies.

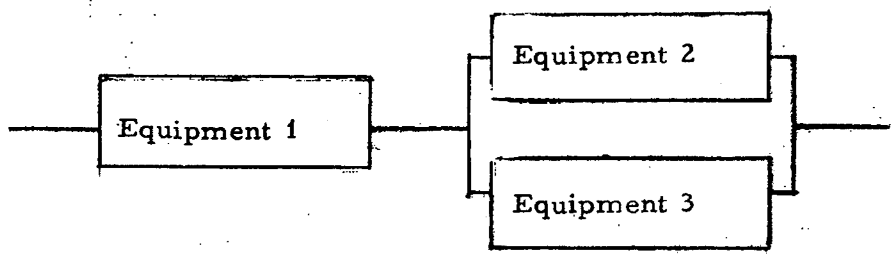
9.2 MOL MISSION ANALYSIS

9.2.1 Reliability Models

The reliability of an item may be represented by the exponential model $R = e^{-\lambda t}$, where λ is the failure rate and t is the operating time, if the failure rate is assumed to be constant during the operating time. This model is a useful approximation of complex equipment reliability if the item being modeled does not have redundant elements. The reliability of a series of such items is given by the product $R_{system} = R_1 . R_2 . . . R_n$, if each R_i is independent. The product of subsystem reliabilities is used to determine the system reliability for the MOL mission.

Where redundant equipment provides alternate paths to accomplish subsystem mission objectives, the reliability model must allow for this. In this case, we are concerned with the probability that either one or the other or both equipments will function and may be represented by the model $R = 1 - (1-e^{-\lambda_1 t})(1-e^{-\lambda_2 t})$ providing that exponentiality is assumed for each equipment.

For a configuration such as:



Where 2 and 3 provide a redundant path, the system reliability may be represented by the model,

$$R = e^{-\lambda_1 t}[1-(1-e^{-\lambda_2 t})(1-e^{-\lambda_3 t})].$$

If equipments 2 and 3 are identical and one is turned off (in standby) when the other is operating, then the model will be

$$R = e^{-\lambda_1 t} . e^{-\lambda_2 t}(1 + \lambda_2 t), \text{ where } \lambda_2 = \lambda_3.$$

This assumes that the equipment has zero failure rate when inoperative. This assumption is made for all reliability analyses performed during this study.

This type of modeling is an integral part of a reliability analysis, and sometimes is the most difficult to accomplish due to incomplete system description. The equipment must be defined so that all redundant paths are considered, the operating times of the individual equipment must be determined, and failure rate information must be available which is appropriate to each equipment in the operational environment.

9.2.2 Mission Definition and Time-Line Analysis

The basic MOL mission is defined to operate for 30 days at an orbital altitude of 150 to 250 nautical miles. Using these ground rules, a simplified mission profile was developed and is shown in Table 9-1. The sources of information for the mission profile are Reference 9-1, 9-2, and 9-3. The time required for each phase is an optimum figure which does not include holds for unscheduled events. Only the major events are considered for a simplified mission analysis.

Each event is examined to determine the subsystems which are required for successful operation. An analysis of the subsystem operating requirements is used to provide information to determine the operating time of each subsystem during the entire mission. Some subsystems are assumed to operate continuously throughout an entire mission phase, while other subsystems are assumed to operate in a regularly scheduled intermittent fashion throughout an entire phase. In order to simplify the reliability calculations for time-dependent subsystems, simplified operating times were derived. Subsystems which were assumed to operate continuously throughout the entire mission were required to operate for 725 hours.

Table 9-1. Mission Profile

Phase	Events	Time in Phase (Hours)	Time from L/O to End of Phase (Hours)
1. Ascent & Injection	Lift off, powered flight, Coast Orbital Injection	10	1.0
2. Pre-Laboratory Operational Orbit	Establish orbit, active lab housekeeping equipment, enter lab.	25	3.5
3. Laboratory Operational Orbits	Commence lab. operation, deactivate unused S/C equipment, perform experiments	720.0	723.5
4. Pre-Separation Orbit	Shut down lab, activate all S/C equipments	2.0	725.5
5. Separation and Re-entry	Separate from S/C, attain proper attitude, fire retro-rockets, maintain re-entry attitude, deploy recovery equipment, land	1.7	727.2

9.2.3 Mission Success Definition

Reliability is defined as the probability that a device will give satisfactory performance without failure for a given period of time when used under specified conditions. This study is concerned with a total mission time of approximately 725 hours, and satisfactory performance is assumed to be successful launch, 30-day orbital operation, and safe landing of the re-entry vehicle. This is an extremely simplified definition of mission-success because certain failures may occur during the mission and be corrected by proper maintenance procedures. A failure which is corrected does not constitute system failure. However, this simplified definition of mission success does consider any failures as a deviation from satisfactory performance. Because of the obvious paradox, the definition for mission success used is only a first approximation of the over-all effectiveness of a MOL mission. However, it is extremely useful for a preliminary reliability analysis and when relative values of reliability are more important than absolute. The reliability figures for each subsystem provide a comparison of the complexities of the various subsystems and indicate possible problem areas. The same comparison can be made between the two systems under consideration. These preliminary reliability predictions which are presented here may be used in reliability/maintenance trade-off studies to determine the spare parts requirements and the subsystems which are expected to require the most maintenance. From these trade-off studies, the effectiveness of the competing designs may be compared to determine which system has the higher probability of completing the mission, regardless of the state of reliability of the system.

9.3 LAUNCH VEHICLES

Studies (Reference 9-4 and A Study By The Aerospace Reliability Department) have been performed which compare the predicted reliability growth of the Saturn IB and the Titan IIIC launch vehicles. Reference 9-4 predicts a reliability of 0.82 for the Saturn IB and 0.77 for the Titan IIIC by 1967, a difference of approximately five percent. The earlier Aerospace study predicts a difference of approximately five percent, with the Titan IIIC as the higher. Reference 9-5 presents a NASA estimated reliability growth curve for the Saturn IB which shows a 0.90 reliability by 1968. Another Aerospace study predicts a reliability of at least 0.84 for theTitan IIIC by the start of 1967.

9.4 GEMINI B/MOL RELIABILITY ANALYSIS

The reliability analysis of the Gemini B/MOL was performed using information from three studies: the MAC 706 Study; a report from McDonnell Aircraft Co. describing the use of the Gemini B spacecraft for a MOL mission; and a typical MOL Laboratory configuration described in a report from the MOL Program Office. Each of the reports was analyzed, and sufficient information was available to make a reliability prediction for each of the subsystems which were expected to comprise the Gemini B/MOL System. The subsystem reliability predictions were divided into separate predictions for the laboratory, described by the McDonnell reports. The subsystem and equipment failure rates which were used in the reliability models of the various subsystems were provided by McDonnell. The MOL laboratory configuration described by the Aerospace MOL Program Office, utilized typical McDonnell subsystems and failure rates. The failure rates assigned by McDonnell are representative of the state-of-the-art equipment and provide a reasonable estimate of the reliability of the particular equipments and subsystems. The reliability block diagrams and assumptions which were used by McDonnell, and the Aerospace MOL Program Office, are consistent with good reliability evaluation techniques and present a conservative prediction of system reliability. In some cases, the mission time used for the reliability calculations provided by Aerospace was longer than the assumed 30-day mission. When this occurred, it was necessary to recompute the reliability for a 30-day mission assuming an exponential reliability model. The result of this reliability analysis is shown in Table 9-2; the over-all Gemini B/MOL spacecraft reliability is approximately 0.88 for a 30-day mission.

Table 9-2. Gemini B/MOL Reliability Analysis

		Reliability
Gemini B S/C	Communications	0.99996
	Telemetry	0.9989
	Power	0.999314
	Environmental Controls	0.9999
	Re-entry Control	0.999
	Earth Landing	0.9963
	Sequentials	0.9991
	Guidance and Control	0.999
	Sub Total:	(0.9915)
Laboratory	Power	0.9844
	Environmental Control	0.9675
	Instrumentation and Telemetry	0.9975
	Reaction Control	0.991
	Attitude Control Electronics	0.985
	Communications	0.960
	Sub Total:	(0.890)

Gemini B/MOL Space Vehicle Reliability = 0.882.

9.5 APOLLO/MOL RELIABILITY ANALYSIS

The reliability analysis of the Apollo/MOL was performed using information from North American studies supplied to Aerospace by NASA. The information, consisting of three separate Apollo configurations, was from the MODAP study (a logistics vehicle), the XMAS study (an extended orbital mission), and the standard Apollo Lunar configuration. The baseline Apollo/MOL configuration chosen for the Aerospace study is described as Configuration II in the North American XMAS study. This configuration consists of the standard Apollo command module, the service module, and an orbital laboratory utilizing the lunar excursion module housing.

In Configuration II, the orbital laboratory utilizes several of the Apollo subsystems to provide housekeeping equipment. These subsystems include communications, telemetry and data, environmental control, service module reaction control, stabilization and control, and power. Because this configuration shares subsystems between the laboratory and the command module, a single analysis was performed which includes both the orbiting laboratory and the command module re-entry vehicle. The subsystem and equipment failure rates which were used in the reliability models of the various subsystems were obtained from the North American reports.

Some of the failure rates and failure probabilities given in the North American reports appear to be more optimistic than comparable failure rates provided by McDonnell. However, they represent comparable state-of-the-art equipment. The subsystems were not always described completely by reliability block diagrams, and in some cases it was necessary to assume a subsystem reliability configuration and model. Because of this lack of complete documentation, the reliability calculations which went into the Apollo/MOL reliability analysis lack the degree of confidence which may be placed in the Gemini B/MOL prediction. In some cases, the mission times used for the reliability calculations given by North American were 14 days for the Apollo lunar mission and 90 days for the XMAS mission. When this occurred, it was necessary to recompute the reliability for a 30-day mission assuming an exponential reliability model. The results of this reliability analysis is shown in Table 9-3 and provide an over-all Apollo/MOL spacecraft reliability of approximately 0.87 for a 30-day mission.

Table 9-3. Apollo/MOL Reliability Analysis

Subsystem	Reliability
Command Module Reaction Control	0.99915
Service Module Reaction Control	0.991
Stabilization and Control	0.94463
Environmental Control	0.974
Guidance and Navigation	0.98
Power (Assumes 2 Spare Fuel Cells)	0.99925
Cryogenic Storage	0.9999
Earth Landing (Assumed)	0.999
Data and Telemetry	0.9852
Communications	0.99977
Ascent Survival (Assumed)	0.995
Apollo/MOL Space Vehicle Reliability =	0.87333

9.6 CONCLUSIONS

A comparison of the reliabilities of the competing systems is shown in Table 9-4, which includes the probability of successful booster operation. Each system has a probability of completing a 30-day mission of approximately 0.7 with less than 1 percent difference in the two estimates. It is concluded from the results of this analysis that neither system provides an advantage when only the reliability of the systems are compared. This is true when the probability of failure-free operation is the only consideration. However, as was previously explained, it is realistic to consider the effects of maintenance upon the over-all probability of continued system operation. The reliability figures presented here can be assumed to be the first approximation of the relative reliabilities of the individual subsystems and may be used as a basis for further reliability/maintainability trade-off studies.

The reliability analysis of the Gemini B/MOL is considered firmer because of the greater credibility of the reliability information provided by McDonnell. This does not mean that the information provided by North American is in error; but is not so complete as the information provided by McDonnell, and consequently we have less confidence in the North American figures. It is believed that the failure rate information used as an input to the Gemini B analysis is more conservative, and it is concluded that there is a greater expectation of achieving the Gemini B/MOL reliability prediction.

Table 9-4. System Reliability

Gemini B/MOL		Apollo/MOL	
Titan IIIC	- 0.80	Saturn IB	- 0.80
Lab	- 0.89	Apollo S/C & Lab (Concept II)	- 0.87
Gemini B S/C	- 0.99		
Mission Rel.:	0.705	Mission Rel.:	0.696

Assumptions

1. Mission success requirements are for 30-day orbital life and safe landing of the re-entry vehicle.

2. Booster reliability is assumed to be at least 0.80 for both launch vehicles by 1967.

REFERENCES

9-1. Informal Report on Gemini B Applications to MOL Mission, compiled from McDonnell Aircraft Corporation Internal Technical Memorandums issued during First Six Months of 1964 (Confidential).

9-2. "Final Report Program 706, Phase 0, Volume III: Analysis of Performance and Requirements," McDonnell Aircraft Corporation, 15 November 1963 (Secret).

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9-4. "Report on a System Comparison and Selection Study of a Manned Orbital Research Laboratory, Volume I: Technical Summary, Appendix 19, Safety, Reliability and Maintainability," Douglas Missile and Space Systems Division, September 1963 (Secret).

9-5. "Modified Apollo Logistic Spacecraft Study (MODAP), Final Report, Volume VII: Operational Implementation," North American Aviation, 30 December 1963 (Confidential).

SECTION 10
FLUID MECHANICS

N. R. O'Brien
M. J. Adams

SUMMARY

The effect of the retropack on re-entry stability and trim characteristics is computed to allow for the case of retropack hang-up.

Considering the possibility of utilizing the Apollo command module for other missions than the moon flights, the following questions were raised:

1. What are the capabilities and limitations of the Apollo earth landing system?
2. What are the possible ways of increasing the allowable landing weights?

From available information, an increase of the allowable landing weight of the command module is possible if (1) a new earth landing system design is allowed, and (2) increased space and weight is provided for the new earth landing system. The new earth landing system would require some combination of new drogue chutes, main parachutes, and/or impact attenuation system. The degree of change would depend on the extent of the weight increase and the capabilities of the final design for the Apollo earth landing system.

A new impact attenuation system would most likely be the first new item. Adequate impact attenuation could be provided by a system composed of retro rockets (velocity attenuation) and shock struts (possibly with an extended heat shield); however, other techniques and combinations are possible.

Aerodynamic characteristics of Titan III C, with an appropriate bulbous payload, and of the Apollo re-entry vehicle are presented for use in performance calculations.

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ILLUSTRATIONS

10. FLUID MECHANICS

10.1 EFFECT OF RETROPACK ON APOLLO HYPERSONIC TRIM CHARACTERISTICS

This section presents the pitching moment coefficient versus angle-of-attack in response to a request for the effect of a possible hang-up of the Apollo retro-pack at re-entry.

The solid curve shows the experimental pitching moment coefficient of the basic Apollo re-entry configuration (without retropack) as obtained from Reference 10-1. The dashed curve represents the pitching moment coefficient curve for the Apollo with the retropack, and was obtained by adding a theoretical (Newtonian) increment for the retropack (Reference 10-2) to the experimental data (solid curve) for the basic configuration. The results shown were referenced to the offset center of gravity indicated in Figure 10-1.

The change in trim angle of attack (i.e., at $C_{M_{CG}} = 0$) due to the presence of the retropack is of the order of two degrees for the indicated configuration. The retropack tends to reduce the required trim angle of attack. Without the retropack, $(L/D)_{trim} = 0.45$ at $\Delta_{trim} = 29^{o}$ and with the retropack, $(L/D)_{trim} = 0.43$ at $\Delta_{trim} = 27^{o}$.

The net change in pitching moment and trim is due to the two components:

(a) The retropack cylinder with front face excluded, and
(b) The part of the Apollo spherical surface shielded by the retropack.

The contribution of the retropack cylinder excluding the front face reduces the pitching moment (i.e., nose up), but this loss is compensated by the stabilizing moment of the shadowed segment of the Apollo face. The effect of the retro-pack is sensitive to location of the center of gravity offset since this moment is largely a function of the induced C_N of the retropack cylinder and the loss in C_D due to shadowing of part of the face of the Apollo by the cylinder at angle of attack. In other words, the change in trim angle shown on the figure is valid for the indicated center of gravity location and would vary somewhat with varying center of gravity location. The effect of the front face of the cylinder was neglected

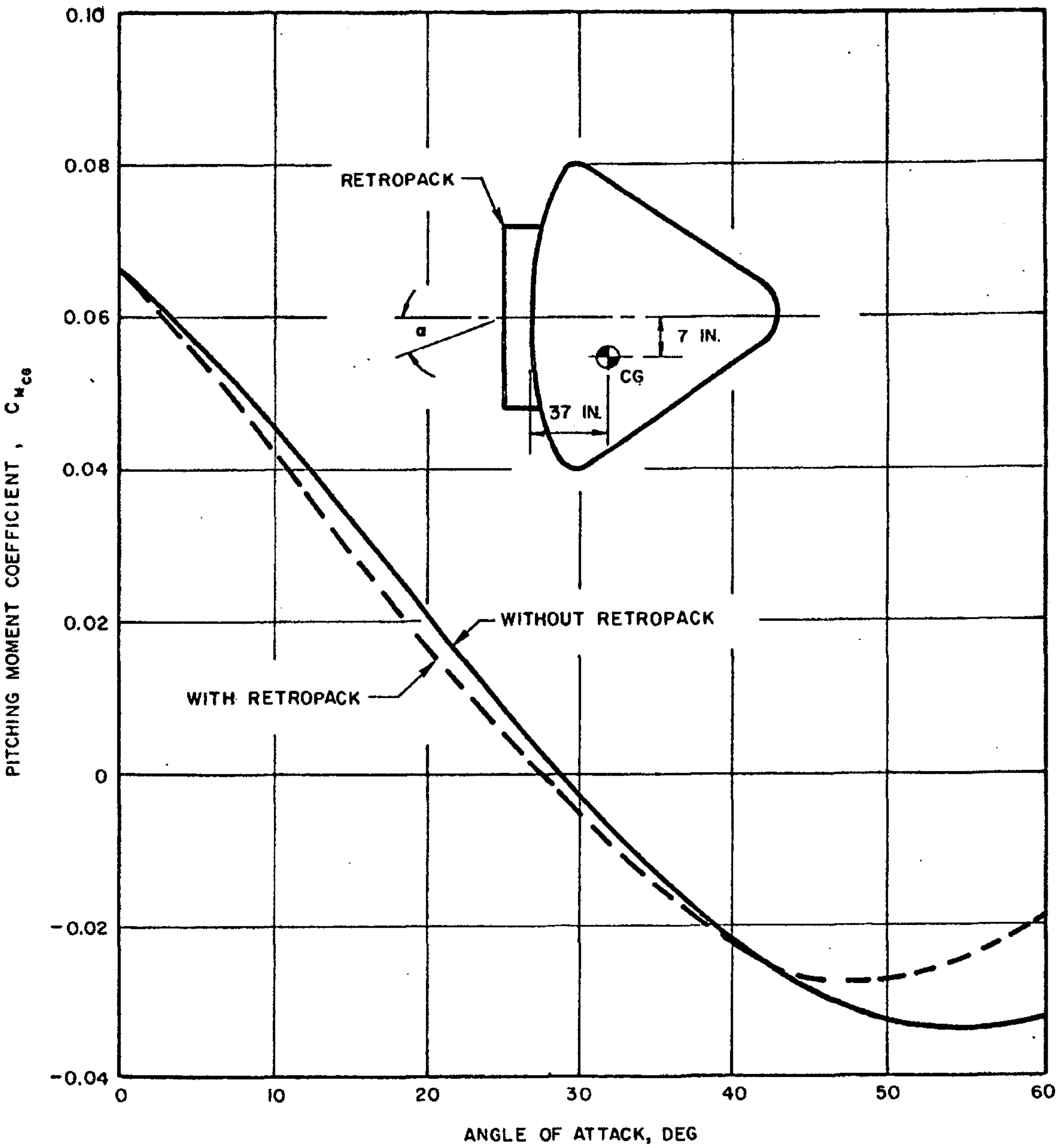


Figure 10-1. Effect of Retropack on Static Stability and Trim Characteristics of Apollo Re-entry Configuration.

since it was assumed to have about the same contribution as that part of the Apollo face to which it is attached and the effect of that part of the face of the Apollo has been reflected in the basic data (solid curve).

10.2 CAPABILITIES OF THE APOLLO EARTH LANDING SYSTEM FOR HANDLING INCREASED WEIGHT

10.2.1 Description of Apollo Earth Landing System

The Apollo earth landing system consists of a stabilization and deceleration subsystem, recovery subsystem, impact attenuation subsystem, and three subordinate subsystems. Together these subsystems provide for recovery of the Apollo command module any time from pad abort through mission completion.

10.2.1.1 Stabilization and Deceleration Subsystem

At 25,000 feet altitude a baroswitch closes, jettisoning the apex cover.[1] Two seconds later two ribbon-type parachutes are deployed in a reefed condition by mortar action. After six seconds, the drogue chutes are disreefed to their full 13.7-foot diameter. At 15,000 feet altitude, a baroswitch closes causing the disconnection of the two drogue chutes and the firing of the main chute pilot mortars. The drogues can be deployed at higher altitudes by crew command for stabilization of descent if the return flight becomes unstable.

10.2.1.2 Recovery System

The three main parachutes are 88-foot diameter ringsails each deployed by its own 7.2-foot diameter flat ringslot pilot chute. The three pilot chutes, each deployed by a mortar, inflate and deploy the three main chutes in a reefed condition. The main chutes disreef six seconds later by pyrotechnic cutter. The spacecraft descends at 25 fps, and at landing the main chutes are disconnected by crew command or inertia switch in unmanned flights. For abort conditions, the sequencing is different depending on the altitude, and an escape rocket, tower, and canards are utilized.

[1] This operation may no longer be necessary; the apex cover may have been jettisoned with the escape tower.

10.2.1.3 Impact Attenuation Subsystem

The Apollo impact attenuation system consists of a crushable toe and crushable seat struts for the command module, thus requiring a 30° pitch angle at impact.

10.2.1.4 Supplementary Subsystems

These are the sequence controller, ordnance and pyrotechnic devices, and search and location aids.

10.2.2 Effects of Increasing Allowable Landing Weights on Terminal Velocity Parachute Parameters

Figure 10-2 illustrates the change in descent velocity with increasing weight of the Apollo command module using the present main parachute system. However, not all of the weight range may be available. Once a parachute has been designed for a given recovery weight, the strength of the parachute defines the allowable weight increase because opening loads will increase with increasing descent velocities. For instance, if the design strength of the present main parachute is utilized during deployment, then only by allowing the factor of safety to be reduced could the gross weight be increased. For such a case the elimination of a factor of safety of 1.5 would allow the weight to increase from 9000 to 13,500 pounds. For an increase in weight of 5000 pounds an increase of 5.6 fps in descent velocity occurs. This may seem to be a small penalty; however, a higher velocity at a higher weight loading means a much larger increase in energy to be absorbed by the impact attenuation system. Since there now is some indication that the present impact attenuation system for the Apollo command module is marginal, any increase in weight would most likely require a new design for the impact attenuation system.

10.2.3 Maintaining the Descent Velocity of 24 FPS by Changing Parachute Characteristics

Figure 10-3 illustrates for three different main parachute clusters the parachute diameters required for various weights maintaining a descent velocity of 24 fps. While this figure shows how to maintain a specific descent velocity, any increase in weight increases the energy to be absorbed by the impact attenuation system. Thus, redesign of the impact attenuation system may be inevitable with any increase in command module weight if the system is marginal.

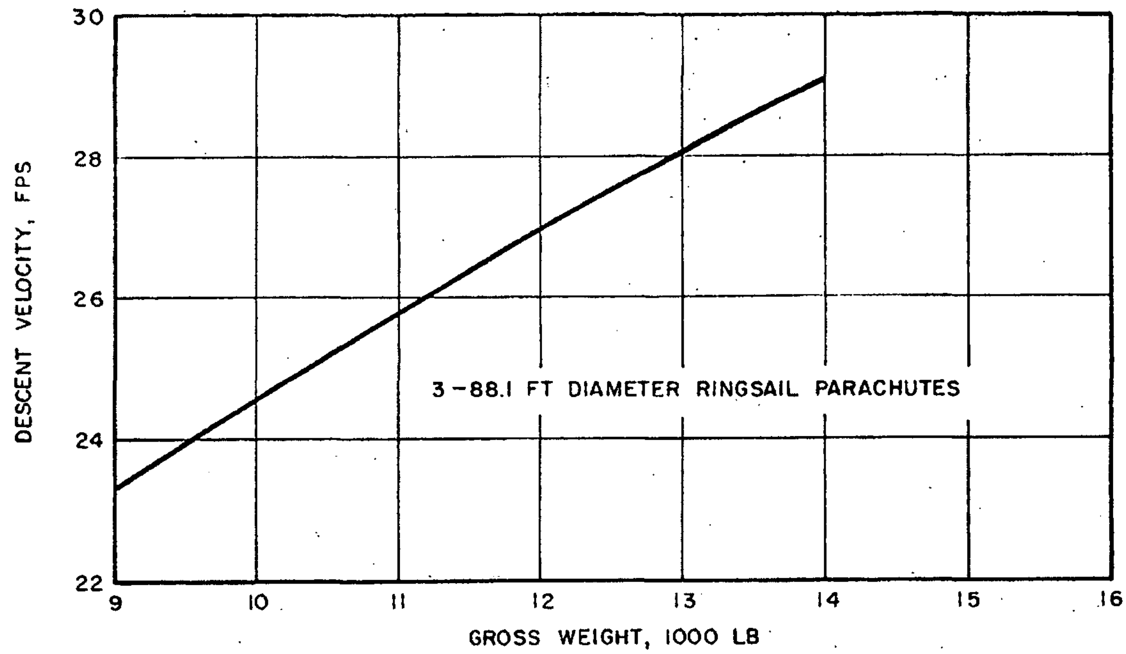


Figure 10-2. Descent Velocity Vs Command Module Gross Weight for Apollo Earth Landing Main Parachute Clusters

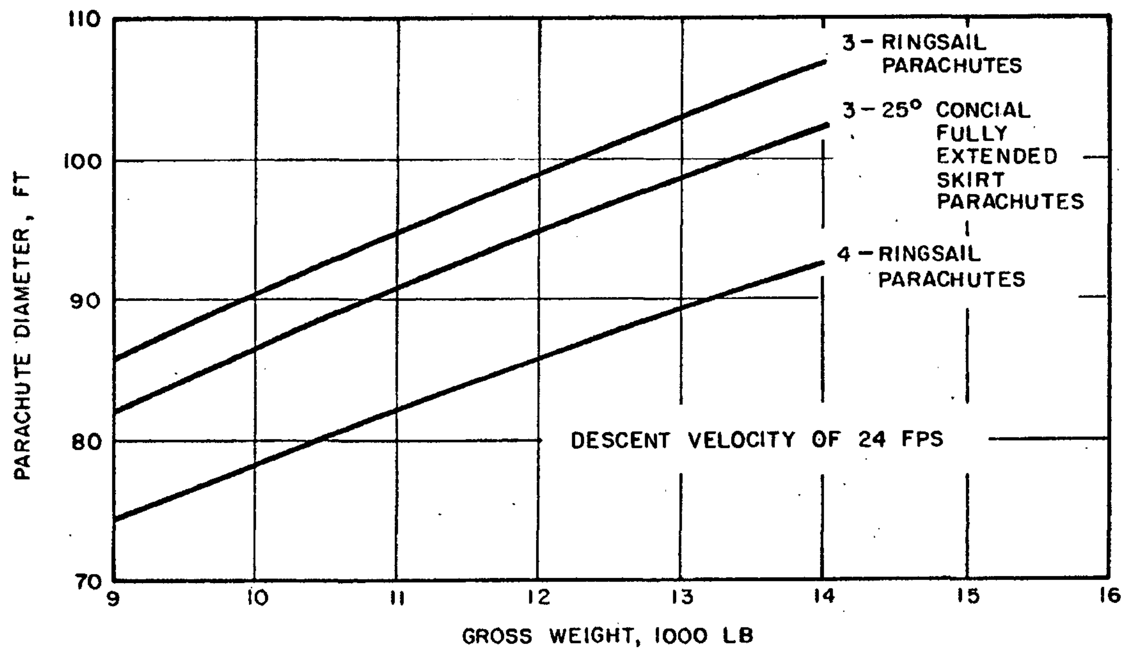


Figure 10-3. Parachute Diameter Vs Command Module Gross Weight for Different Main Parachute Clusters

While the 25° conical fully extended skirt parachute shows better drag performance than a ringsail parachute, the stability of a ringsail parachute is better than that of a 25° conical fully extended skirt.

10.2.4 Capabilities of the Apollo Earth Landing System

Table 10-1 and Figure 10-4 from Reference 10-3, indicate the earth landing system design criteria for the command module of the Apollo Program. This system meets the following gross requirements after normal re-entry or during abort:

a. Stabilization
b. Velocity control
c. Impact attenuation
d. Land or water touchdown

To meet similar gross requirements for increased command module weights of a couple of thousand pounds is within the state of the art, but would require another comprehensive design and development program. Both the drogue chutes and main parachutes would require re-sizing and possibly a change to a different canopy design to maintain the same descent velocities. The impact attenuation system would require a new design to meet similar g-load requirements. There is a trade-off between the main parachute system (rate of descent) and the impact attenuation system; however, the consideration of minimum weight would most likely make the utilization of the combination of velocity and impact attenuation system more desirable. For instance, a velocity attenuation consisting of retro rockets could be activated prior to ground impact reducing the command modules final rate of descent. If the Apollo impact attenuation system of crushable toe and crushable seat is not adequate, then the heat shield could be designed to extend and shock struts used to absorb the impact.

10.2.5 Major Problem Areas with Apollo Earth Landing System

Major problem areas associated with the design and operation of the deceleration, stabilization and recovery systems are:

a. Recovery system weight and its effect on command module stability.
b. The inter-relation of command module dynamic stability and drogue parachute design.
c. The problem of main parachute non-uniform cluster operation.

Table 10-1. Earth Landing System Design Criteria.

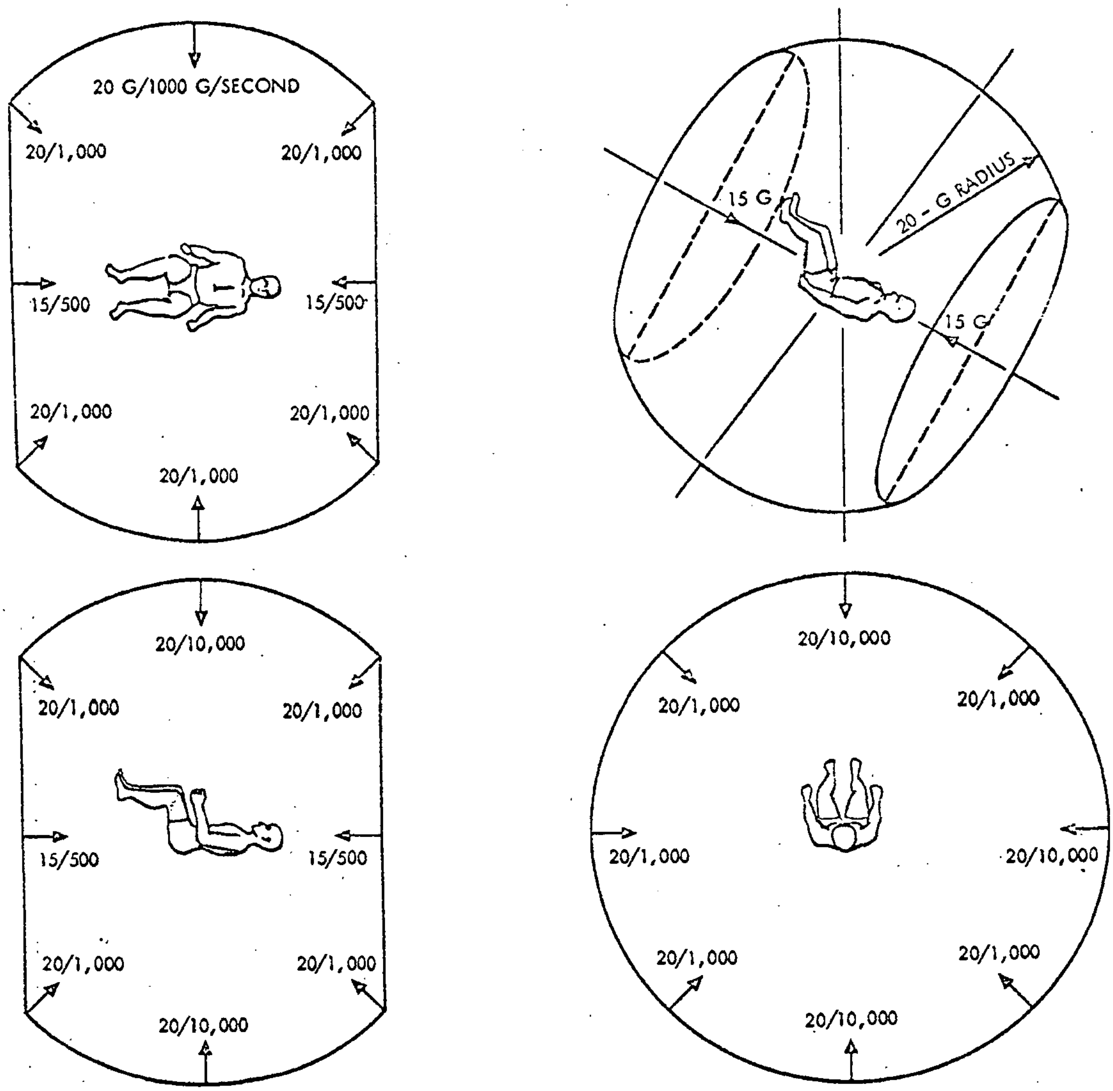
Subsystem	Condition	Limits	
		Nominal	Emergency
Launch escape (pad or boost abort)	Crew module and launch system gross boosted weight	16,300 pounds	-
	Crew module gross boosted weight	9700 pounds	-
	Required boost altitude	4500 feet (pad abort)	-
	Required boost range	5000 feet (pad abort)	-
	Crew acceleration limits	As given in Figures 39 and 40 and in accompanying text	
Crew module stabilization and deceleration	Crew module attitude at parachute deployment	Tumbling in any direction at a rate up to 150 degrees per second	-
	Deployment altitude	4500 to 25,000 feet	Up to 40,000 feet
	Deployment dynamic pressure	14 to 75 pounds per square foot	140 pounds per square foot
	Maximum allowable parachute drag and mortar reaction force	10,000 pounds	-
	Maximum allowable dynamic pressure at main parachute deployment	64 pounds per square foot	-
Parachute recovery	Crew module gross recovery weight	9000 pounds	9400 pounds
	Landing altitude	Sea level	2000 feet
	Maximum allowable parachute oscillation	±5 degrees	±15 degrees (water) ±5 degrees (land)
	Vertical descent velocity at landing	24 feet per second (3 parachutes operational)	28 feet per second (2 parachutes operational)

[redacted]

Table 10-1. Earth Landing System Design Criteria (Continued).

Subsystem	Condition	Limits	
		Nominal	Emergency
Impact attenuation	Landing surface	Water	Land
	Sea state	Sea state 3	Sea state 4
	Wave slope	5 degrees	10 degrees
	Ground slope	-	15 degrees
	Wind-drift velocity at landing	0 to 28 feet per second	0 to 34 feet per second
	Crew acceleration limits	As given in Figures 39 and 40 and in accompanying text	
	Crew module structure bending failure	2.5g's	2.5g's

[redacted]



NOTE: RESTRAINT AND SUPPORT MUST BE SIMILAR TO
THAT ILLUSTRATED IN THIS SECTION

Figure 10-4. Normal-Mission g Limits.

d. The design loads for the parachute attachment points on the command module.

e. The marginality of the present impact attenuation system.

10.3 MISCELLANEOUS FLUID MECHANICS DATA

Figure 10-5 shows the estimated drag of the Titan IIIC at $\Delta = 0$ for Mach Numbers from zero to four. This estimate pertains to the configuration shown in the figure and is based on wind tunnel tests of a bulbous payload fairing. It is recommended that, in order to minimize adverse transonic aerodynamic effects due to the hammerhead payload, the command module should be followed by a cylindrical section terminated by an abrupt (Δ 30°) reduction in diameter to the 120 inch core dimension. The drag characteristics shown in Figure 10-5 are adequate for performance calculations for all such configurations with or without the Apollo escape tower.

Hypersonic aerodynamics of the command module at trim angle of attack (Reference 10-1) were supplied for the re-entry analysis presented in Section 5. The center of gravity of the capsule is assumed to lie 8 to 10 inches off the axis of symmetry and no further than 49 inches aft of the blunt face of the heat shield. This location produces acceptable vehicle dynamics (according to NAA) and hypersonic L/D between 0.4 and 0.5. It is not yet known if the MOL version of the capsule can be balanced in this range.

The question of aerodynamic control of the re-entry trajectory was discussed with NAA during inspection of the mock-up. The current plan is to use the lifting capability of the capsule during return from the lunar mission only as a means of staying within the entry corridor defined by overshoot, heating and deceleration limits. However, in return from earth orbit, this capability could be used to steer out impact dispersions due to deboost and atmospheric uncertainties with a resulting decrease in the required recovery force. Investigation of this possibility might be desirable in the Aerospace study if cost or availability of the recovery force becomes critical.

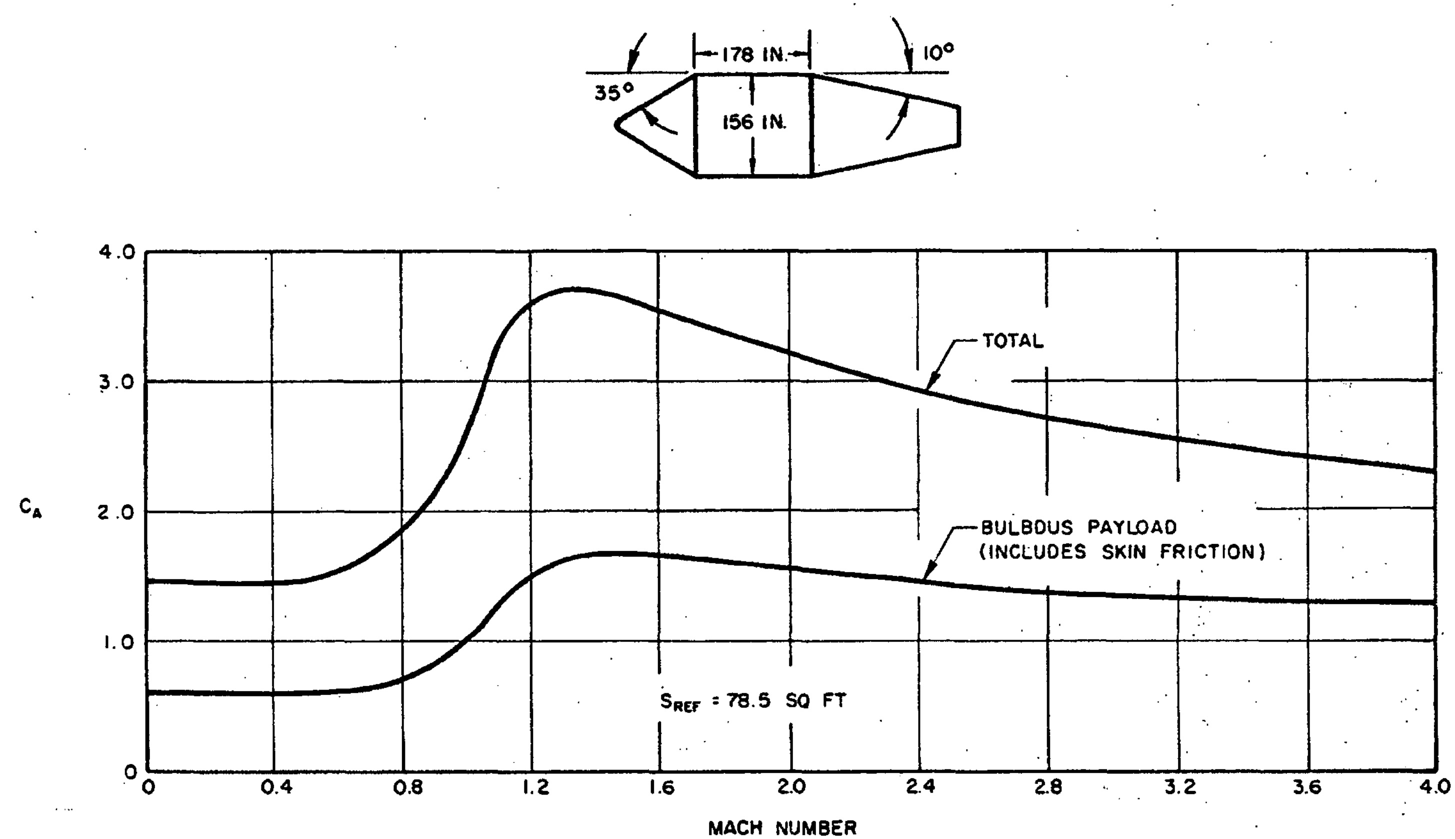


Figure 10-5. Titan III - Booster Plus 156 Inch Diameter Bulbous Payload Axial Force Coefficient

10.4 CONCLUSIONS

For the specified location of the center of gravity, the retropack would have a very small effect on the static stability and trim characteristics of the re-entry vehicle.

The Apollo Earth Landing System cannot handle significantly increased command module weights without a comprehensive additional design and development program.

REFERENCES

10-1 "Static Longitudinal Aerodynamic Characteristics of a 0.07 Scale Model of a Proposed Apollo Spacecraft at Mach Numbers of 1.57 to 4.65," by J.R. Morgan and R.H. Fournier, NASA TM X-603, dated September 1961.

10-2 "Equations and Charts for the Evaluation of the Hypersonic Aerodynamic Characteristics of Lifting Configurations by the Newtonian Theory," by E. L. Clark and L. L. Trimmer, AEDC TDR 64-25, dated March 1964.

10-3 "Modified Apollo Logistics Spacecraft Study," North American Aviation SID 64-1461-4, Final Report, Vol. IV, Earth Landing System, 30 December 1963.

SECTION 11

PROPULSION

M. J. Russi

SUMMARY

The propulsion requirements are studied for abort, de-orbit, crew module reaction control, laboratory reaction control, and laboratory main propulsion. Appropriate propulsion systems which will meet the various requirements are identified as tentative choices. Specific propulsion devices currently under development are chosen in each case, but none of them (with the possible exception of the escape tower) can be considered "off the shelf" for the MOL mission. In each case, additional development and qualification will be required to meet the MOL requirements. As the mission duration is extended, the confidence in these choices decreases.

CONTENTS

ILLUSTRATIONS

11. PROPULSION

11.1 INTRODUCTION

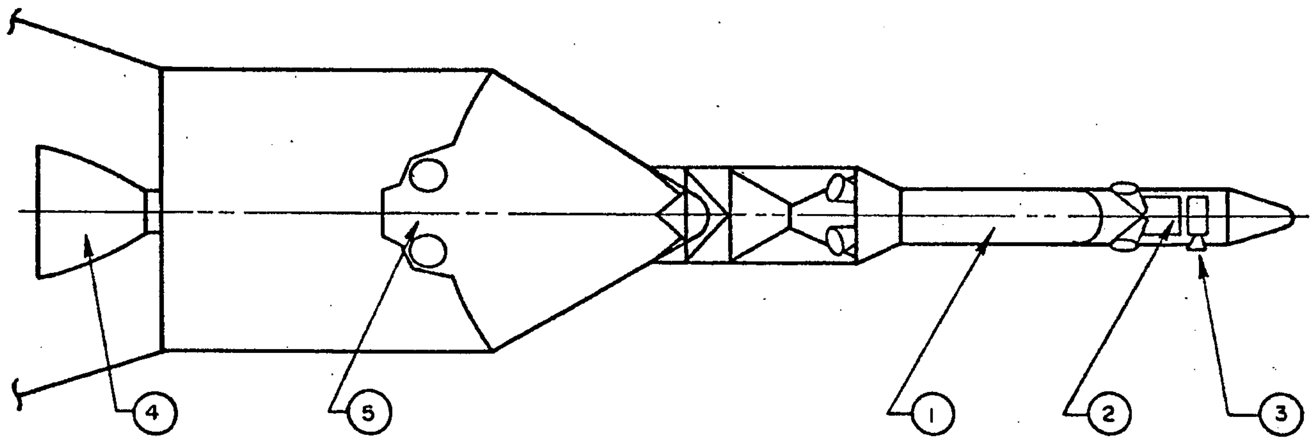
In choosing propulsion systems for Apollo/MOL, the NAA extended Apollo Configuration II was used as a baseline. This configuration makes every effort to retain current propulsion subsystems. The functions to be performed by propulsion include abort, de-orbit, crew module reaction control, laboratory reaction control, and laboratory main propulsion for final orbit injection and maneuvering.

11.2 ABORT PROPULSION

The various abort propulsion possibilities for the Apollo/MOL are depicted schematically in Figure 11-1. For abort within the atmosphere, the present launch escape tower is used and appears to be satisfactory. It can be used up to an altitude of about 300,000 feet and is then jettisoned.

For abort outside the atmosphere, the present technique is to use the 21,900-pound thrust service module engine. In the NAA studies, the LEM descent engine (10,500-pound thrust) was chosen to replace the 21,900-pound thrust motor for main propulsion and abort. The main consideration was a minimum acceleration of 0.3 to 0.4 g established by NAA for abort, which eliminated lower thrust motors. However, NAA did not consider the possibility of using the solid retro motors as posigrade abort propulsion. In this case, a minimum thrust of 3500 pounds would be required since only the crew module would have to be accelerated. The NAA retropack configuration (described below) has a thrust level of 6600 pounds and burning time of 34 seconds in salvo which appears to be more than adequate for the abort requirements.

A trade-off study is required to determine the relative advantages of using either or both the retro motors and the main propulsion engine for posigrade abort outside the atmosphere.



ATMOSPHERIC ABORT- USE ESCAPE TOWER

1. LAUNCH ESCAPE MOTOR
2. TOWER JETTISON MOTOR
3. PITCH MOTOR

EXOATMOSPHERIC ABORT- USE MAIN ENGINE OR SOLID RETRO ENGINES (TOWER JETTISONED)

4. LEM DESCENT ENGINE
5. SOLID RETROS (PROPOSED FOR DEBOOST PHASE)

Figure 11-1. Abort Propulsion.

11.3 DE-ORBIT PROPULSION

The two alternatives considered for providing retrograde propulsion are (1) the use of the main liquid propulsion engine (service module) and (2) the use of a newly designed solid propellant system (as in Mercury and Gemini). The latter approach is preferred for several reasons; lower vehicle weight, higher reliability and its capability to also perform the abort function.

Lower vehicle weight results from the fact that a solid motor system would have to de-orbit only the crew capsule whereas the main propulsion system would have to de-orbit the service module as well. Furthermore, the relatively small ΔV required for de-orbit and the relatively small difference in specific impulse between the liquid propellant system and the solid propellant system tend to favor the latter. The retrograde velocity increment requirement involves a trade-off between system weight and landing accuracy. Figure 11-2 shows NAA data for estimated heat shield weight and retro rocket weight versus retro velocity increment. The current Apollo heat shield weight is about 1300 pounds and is overdesigned for re-entry from low orbital altitude. Several hundred pounds weight could be shaved from the heat shield, but it would require requalification and additional flight tests. From Figure 11-2, for the current heat shield, the combined weight is a minimum for a retro velocity increment of about 300 fps. However, from Figure 11-2 which also shows touchdown range sensitivity vs. retro velocity increment, it can be seen that a minimum retro velocity increment of 500 fps is needed to avoid large landing inaccuracies (NAA). For this reason 500 fps was selected as a design point.

The NAA design approach for the retro-pack is shown in Figure 11-3 together with data for the selected solid motor and an overall weight estimate. Six spherical motors (XM-85 flight proven) are used in a cluster. Five motors are required and an additional one is included for redundancy.

The use of a cluster of solid motors for de-orbit is recommended rather than the liquid propellant main propulsion system. In addition to a lower weight system, the former approach offers greater reliability. Although

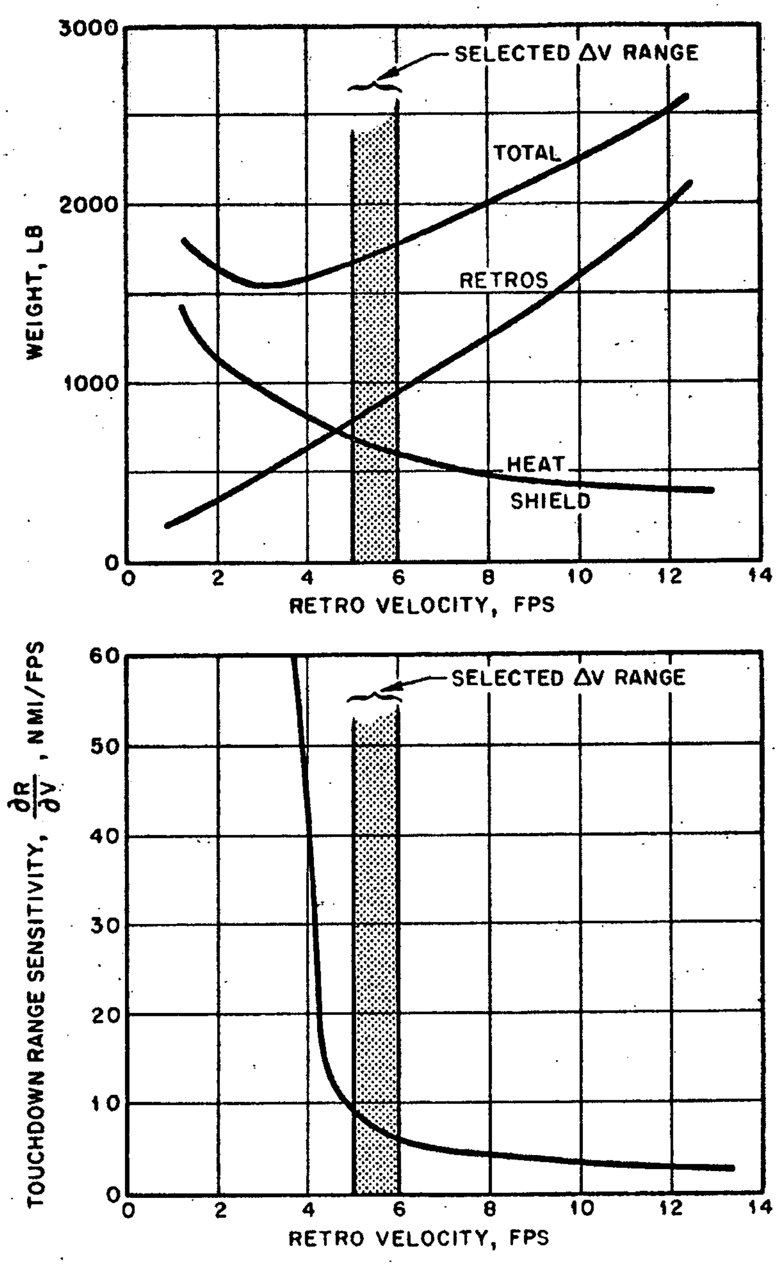
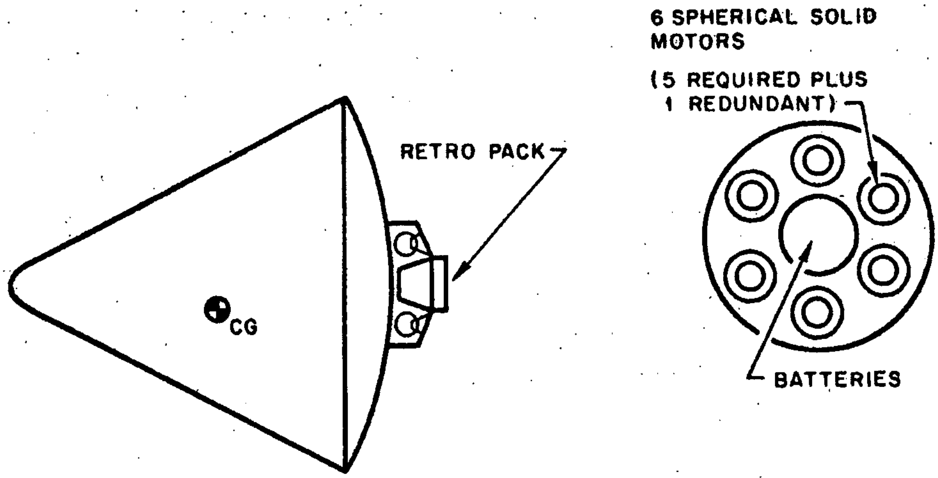


Figure 11-2. Retrograde Propulsion Requirements.



MOTOR DATA	
DESIGNATION	XM-85
THRUST	1100 LB
BURN TIME	34 SEC
SPECIFIC IMPULSE	270 SEC
PROPELLANT WEIGHT	136 LB
TOTAL WEIGHT	155.6 LB
LENGTH	17.3 IN.
DIAMETER	17.2 IN.
WEIGHTS (NAA)	
STRUCTURE	102 LB
MOTORS	936 LB
BATTERIES	232 LB
TOTAL	1,270 LB

Figure 11-3. Retrograde Propulsion Configuration.

neither liquid propellant engines nor solid propellant motors have been stored in space for 30 days or more, it is likely that the solid motors will be easier to store. If need be, the solid motors can be completely sealed until needed for re-entry. On the other hand, the liquid propellant main propulsion would be used for orbit injection and on-orbit maneuvering and would be subject to prior malfunction and leakage. Furthermore, the multiple solid motor concept offers some redundancy and can accept a single motore failure. It has been used successfully on Mercury, will be used on Gemini and appears to be a logical choice. There are, however, several areas regarding the selection of solid motors for de-orbit which require further study:

11.3.1 Ground Systems

The solid motors will have to be installed during vehicle buildup. Usual procedures prefer to leave the propellant loading until final checkout to keep the vehicle in an inert state. The motor selected by NAA (XM-85) has a beryllium loaded propellant which may not be compatible with safety regulations. An aluminized version of the motor exists (NOTS-100A) but would require modification to reach the total impulse of the XM-85.

11.3.2 Orbital Storage

The effects of long term space storability (30 days +) on solid propellant motors are not fully understood. Temperature cycling, vacuum, and radiation effects must be explored.

11.3.3 Firing Temperature

The total impulse produced by a solid motor is affected by its firing temperature and landing accuracies are sensitive to motor total impulse. An active temperature control will probably be required to keep the propellant within acceptable temperature limits. Emergency return to earth may require continuous temperature control.

11.3.4 Qualification

The selected solid motor (XM-85 or equivalent) will probably require requalification as a result of any design changes and confirmation complete retro system compatibility with launch vehicle, operational environment and performance.

11.4 REACTION CONTROL SYSTEMS

11.4.1 Crew Module

The Apollo command module reaction control system (RCS) is shown schematically in Figure 11-4, together with pertinent characteristics. The system consists of two independent equally capable, identical systems, each with six thrusters. Both systems function in normal operation. The design is such that failure of either one of the systems is acceptable and the mission can be completed with the remaining system. The system, which is sealed until time for re-entry, provides 3-axis re-entry orientation of the command module after separation from the service module.

The Apollo command module RCS is basically suitable for the Apollo/MOL application. However, there are several considerations which may require design changes that must be investigated.

11.4.1.1 System Capacity

The current Apollo command module RCS is designed for use only during re-entry. Use in the Apollo/MOL mission will require control before, during, and after retro firing prior to re-entry. Additional propellant will be required of the RCS for offsetting thrust misalignment during retro firing. Compensating for this, to some extent, is the shorter re-entry time from earth orbit than from lunar return. In any case, the total propellant requirement must be re-examined for the Apollo/MOL command module. If the propellant quantity is increased, the thrust chamber rated lifetime must be increased also which will require re-qualification to the new duty cycle.

11.4.1.2 Module Control

The current Apollo command module RCS was designed for use without a retrorocket attached. The thrust level and location of the reaction jets will have to be re-examined to determine adequacy of the system during firing of the retrorockets (thrust misalignment) and after retropack jettison (center of gravity shift). Re-sizing the thrust chambers would require additional development and re-qualification.

SYSTEM CHARACTERISTICS

PROPELLANT
- OXIDIZER - N_2O_4
- FUEL - MMH

PRESSURE FED WITH HELIUM

ABLATIVE THRUST CHAMBERS

THRUST - 91 LB EACH

RATED DURATION - 130 SEC

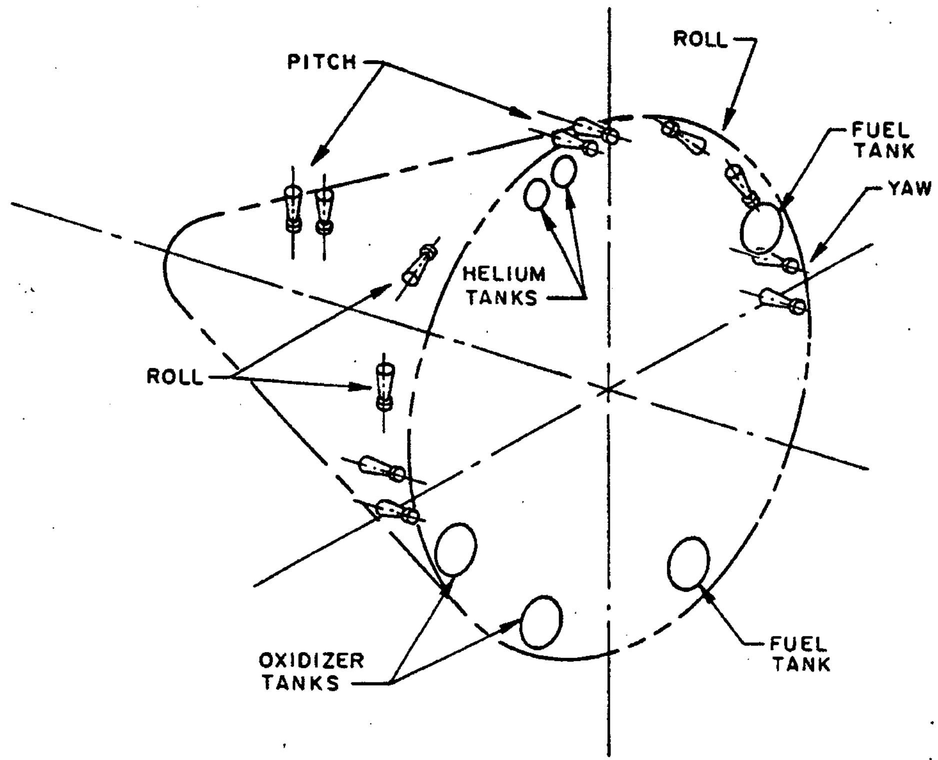


Figure 11-4. Apollo Command Module Reaction Control System Description.

11.4.1.3 Space Storability

The current Apollo systems are designed for 14 day storage in space. Propellant pressurization system leakage rates must be examined for the extended duration. Temperature control of propellant and valves is necessary to keep the system in a ready condition. An active thermal control system is probably required (electrical heaters have been suggested by NAA). The problem of storability increases with time and if growth beyond the 30 day mission is anticipated, initial design review studies must allow for it.

11.4.2 Laboratory Module - (MOL)

The Apollo service module RCS is the basis for a minimum modification approach to a MOL RCS and is shown schematically in Figure 11-5. Also listed are the system weight and characteristics. Basically, the service module RCS consists of four independent, identical clusters of four thrusters each located as shown. The hypergolic propellants, N_2O_4/AZ-50, are pressure fed by helium and expelled by Teflon bladders. The thrust chambers are fabricated from coated molybdenum and are cooled through radiation.

The above system appears to most nearly meet the MOL requirements. However, there are several factors which need additional study before it can be concluded that the system will be suitable for the MOL mission:

11.4.2.1 Design Basis

The current Apollo service module RCS design is based on a very small percentage of limit cycle operation. The design was influenced strongly by rendezvous and translation requirements. As a result, the thrusters were sized (100 pound thrust) for the latter requirements and the minimum impulse bit which is important for limit cycle propellant consumption was not optimized. The main function of the MOL RCS will be to provide long term limit cycle operation and the non-optimized minimum impulse bit may be too costly in terms of propellant consumption, particularly for

[redacted]

SYSTEM CHARACTERISTICS

THRUST	- 100 LB (EACH)
RATED DURATION	- 1000 SEC
INERT WEIGHT	- 580 LB (TOTAL 4 SYSTEMS)
PROPELLANT	- 838 LB (TOTAL 4 SYSTEMS)
OXIDIZER	- N_2O_4
FUEL	- AZ50
TOTAL RCS.WT	- 1418 LB

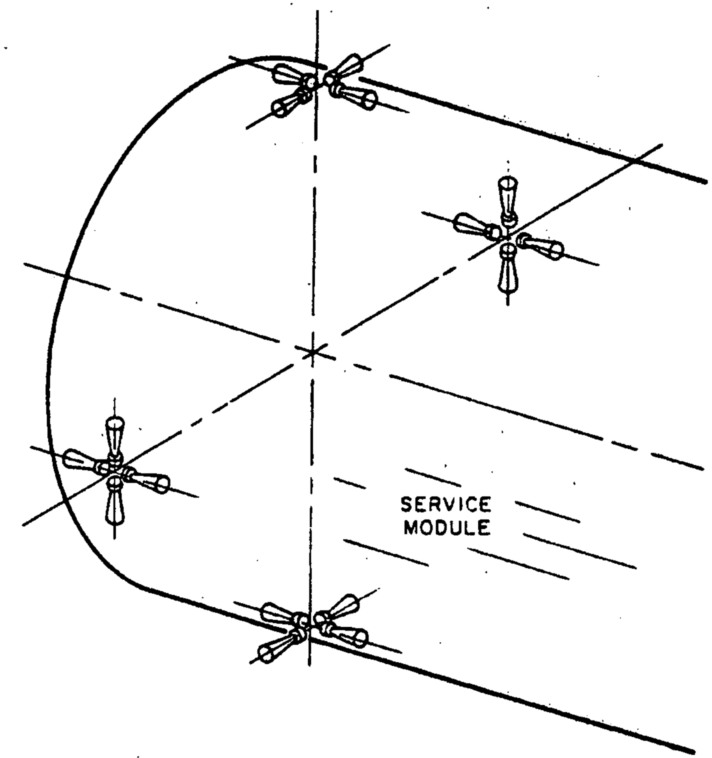


Figure 11-5. Apollo Service Module Reaction Control System.

[redacted]

missions longer than 30 days. A smaller thruster (smaller impulse bit) may be required for limit cycle operation which would require further development and qualification.

11.4.2.2 Thruster Lifetime

The current service module RCS is designed for 10,000 on-off cycles and 1000 seconds firing time. The MOL requirements will exceed the lifetime requirements for the current Apollo service module RCS because of the longer mission time. The radiation cooled thrust chamber is potentially capable of longer lifetime than required by the current Apollo mission but considerable development and qualification will be required.

11.4.2.3 Space Storability

The Apollo service module RCS is designed for a 14-day mission. The effects of pressurization gas leakage rates will have to be determined for longer missions. In addition, the current propellants must be kept within rather narrow temperature limits to remain in the liquid phase and within the design pressure limits. Electrical heaters have been considered by NAA for this purpose and a 150 watt requirement has been estimated. The Teflon bladders used for positive expulsion of the N_2O_4/AZ-50 propellants may be marginal for a 30-day mission. Other techniques such as metallic bellows should be examined.

11.5 MAIN PROPULSION

For the concept under consideration, a propulsion system larger than the RCS is required to place the Apollo/MOL on station and to provide on-orbit maneuvering such as rendezvous or as dictated by on-board experiments. In the NAA Extended Apollo approach, the service module main propulsion system (21,900 pound thrust) is replaced by the smaller, variable thrust LEM descent engine which has a maximum thrust of 10,000 pounds. In this study the requirements are first examined, candidate propulsion systems are considered, selection of the most promising system is made and problems associated with it are discussed.

11.5.1 Requirements

The assumption is made that the Apollo/MOL is boosted into an elliptical orbit. The MOL main propulsion system is then used to circularize the orbit at apogee. This requires a ΔV of about 200 - 250 fps. No restart is required and the thrust level should be several thousand pounds to minimize engine burn time (lifetime) and gravity losses (propellant). For rendezvous, the NAA study indicates additional ΔV of 1300 - 1400 fps is required and multiple restart capability is necessary.

[redacted]

In summary, preliminary estimates for main propulsion indicate a desired ΔV capability of about 2200 fps. A thrust of several thousand pounds is desired and multiple start capability with accurate control appears mandatory.

11.5.2 Candidate Propulsion Systems

The desire for multiple starts and mission flexibility indicates a liquid propellant rocket for main propulsion. The basic ground rule of minimum modification and high confidence level limits the number of engines to be evaluated to a relative few. Cryogenic propellants offer the highest specific impulse and the RL-10 engine (LOX/LH_2 - 15,000 pounds thrust) is a potential candidate. However, little performance gain would be realized at the low ΔV level required and extensive redesign of the Apollo service module would be required to use it. The agena engine (IRFNA/UDMH - 16,000 pounds thrust-pump-fed) would also require extensive service module re-design. The remaining candidates are the current Apollo service module engine, LEM descent engine, LEM ascent engine and the Titan III transtage engine. The characteristics of the above engines are listed below:

CANDIDATE ENGINES

	APOLLO SM	LEM DESCENT	LEM ASCENT	T-III TRANSTAGE
Propellants	N_2O_4/A_Z-50	N_2O_4/A_Z-50	N_2O_4/A_Z-50	N_2O_4/A_Z-50
Thrust, Lb.	21,900	10,500*	3500	8000
Rated Burn Time, Sec.	730	730	385	440
Specific Impulse, Sec.	307	305	305	300
Restart	YES	YES	YES	YES
Engine Weight, Lb.	692	360	145	200
Length, In.	160	75	--	80
Diameter, In.	100	54	31	49

*10 to 1 throttling available

The service module engine size and weight make it unattractive to this application since high thrust is not required. Its use would obviously require the least modification to the service module but would severely restrict the volume and weight available for MOL related equipment.

The LEM descent engine is about 1/2 the size and weight of the service module engine. It has adequate thrust and its throttling capability is very attractive for rendezvous and other maneuvers requiring accurate control. Relatively minor modification to the service module will be required to incorporate the LEM descent engine.

The LEM ascent engine is the most compact of the candidate engines. However, its thrust level is marginally small for orbit changes and it would exceed its rated burn time for ΔV's greater than 1500 fps.

The Titan III transtage engine is adequate as far as thrust level, size and weight are concerned. It is not as easily integrated with other NASA systems.

11.5.3 Main Propulsion Selection

From the above discussion it appears that the NAA selection of the LEM descent engine is a reasonable choice. Its reduced size and weight allows room in the service module for MOL equipment. It offers the greatest flexibility (throttling) for the MOL mission and will be limitedly qualified in the current Apollo program.

11.6 AREAS REQUIRING FURTHER STUDY

The full growth potential offered by the liquid propellant engine depends on its capability to be stored in space for long periods of time while providing many restarts. The current Apollo program will not qualify or demonstrate space storability of the LEM descent engine beyond 14 days. Furthermore, this storability will be in an unfired condition, i.e., the engine is not fired until near the end of its storage time. The MOL mission requires the main propulsion system to put the MOL on station and then to be stored for 30 days or more during which time it will be fired several times. It is possible that initial MOL missions could be programmed to perform all main propulsion functions immediately following final orbit injection thereby avoiding long time system storage problems. However, this approach would limit the flexibility and usefulness of MOL.

The problems to be solved for the LEM descent engine long space storage are somewhat common to the reaction control system. Since the propellants are the same, the thermal environment must be controlled to present freezing or tank overpressure. Propellant orientation control will be necessary under zero "g" conditions to minimize center of gravity shift and to aid in propellant temperature control. Present techniques (bladders) have shortcomings for long term storage systems. Passive systems such as surface tension control are promising but need further development.

The lifetime of thrust chamber ablative materials under extreme vacuum conditions at high temperature is not well known. This condition occurs on engine shut down in vacuum. Likewise, the lifetime capability of coated refractory metals used for radiation cooled thrusters is not well known for extreme vacuum conditions.

11.7 CONCLUSIONS

While specific propulsion systems currently under development have been identified for each MOL requirement, none of these (with the possible exception of the escape tower) can be considered "off-the-shelf" systems for the MOL mission. The degree of confidence with which each selected propulsion subsystem can be made to meet MOL requirements, with additional development and qualification, decreases with the amount of time it is required to function under space environment. The following conclusions hold for the 30-day mission.

11.7.1 Escape Tower

Since the escape tower for providing abort within the atmosphere is only required to operate for a very short time, the current Apollo system is satisfactory for MOL.

11.7.2 De-Orbit

The XM-85 flight proven solid motor appears to be the best choice for the retro pack. Some development is required to incorporate it into the necessary cluster of six motors and to provide necessary thermal control for the 30-day mission.

11.7.3 Crew Module Reaction Control System

The current Apollo command module RCS appears to be basically satisfactory for MOL. The system is not required to operate until just prior to re-entry and can be stored in a passive state. The thrust level and location of the thrusters will have to be examined to insure adequate control during the retro firing.

11.7.4 Laboratory Reaction Control System

The current Apollo service module RCS appears to be the best choice. However, additional development and requalification of the system will be required to provide the longer life and space storage required in the MOL mission.

11.7.5 Main Propulsion

The LEM descent engine represents a reasonable choice for MOL main propulsion. Additional development and requalification will be required to insure adequate space storability for the MOL missions.

SECTION 12

SOLID MECHANICS

J. Hook
D. E. Hargis
G. B. Fox
A. J. Victor

SUMMARY

The following comments on the structural feasibility of the proposed Apollo/MOL configurations are submitted. Based on the results of the NAA XMAS study, the Saturn IB configurations appear to be structurally feasible in all respects. In the Titan IIIC configuration, the spacecraft structures appear to be feasible with the weights that were considered. The launch configuration appears to be structurally feasible as far as most loads are concerned, but wind tunnel tests would be required to determine the effects of wind-induced oscillations and transonic buffeting.

CONTENTS

12. SOLID MECHANICS

12.1 TITAN IIIC CONFIGURATION

12.1.1 Spacecraft

A preliminary design study has been performed to ascertain the structural feasibility of the proposed Apollo/MOL spacecraft configuration. Critical loads were estimated and the structure of the "service" module between the transtage and the command module was sized accordingly. A conservative analysis indicates that the allowance, provided in Section 2 of this report, of 1400 pounds for the structural portions of this module should be entirely adequate. In fact, this weight might be significantly reduced by a structural optimization procedure. A comparison of the critical loads used in this study with those given in the NAA XMAS Study indicates that the command module also will be structurally adequate for this mission.

12.1.2 Booster

Several loads studies have been conducted by Martin-Marietta Corporation (Denver) during the Task #2 MOL Payload Constraints effort which involved a matrix of 35 configurations with payload ranges between 15K and 25K, laboratory diameters between 120 inches and 156 inches, and payload lengths between 30 feet and 50 feet. In addition, Aerospace Corporation has performed a loads study for a 156-inch diameter, 40-foot payload length, and 21,000 pound payload on Titan IIIC. Based on the results of these studies, it is felt generally that loads imposed on the Titan IIIC by the 154-inch diameter/21K Apollo/MOL configuration will be well within the Titan IIIC design allowable loads.

With regard to dynamic loads, however, there are two important areas of uncertainty that will require testing before their effects can be known with confidence. These are (1) ground winds and wind-induced oscillations and (2) transonic buffeting.

12.1.2.1 Ground Winds and Wind-Induced Oscillations

Results from a 7.5 per cent 624A Ground Wind and Wind-Induced Oscillation Wind Tunnel Test indicated a critical load condition and large dynamic response of the vehicle for one of the bulbous configurations tested in the presence of the ITL electrical masts. Since the ground winds problem is extremely sensitive to changes in the upper body (payload) configuration, and the interference effects of the ITL masts on the vehicle response are not known, it will be necessary to conduct additional ground winds tests on the Apollo/MOL/Titan III configuration. The Titan III 7.5 per cent booster model, ITL electrical masts, ITL umbilical tower, transporter stand, and turntable should already be available for this test. Construction of a 7.5 per cent Apollo/MOL capsule will be required. In order to utilize the existing model and turntable, the test must be conducted in the NASA Langley 16-foot transonic dynamic tunnel.

12.1.2.2 Transonic Buffeting

Transonic buffeting can generally be divided into two areas of concern. Local effects are usually related to buffet excitation frequencies above 20 cps and apply to skin panel response and component reliability. Gross vehicle effects are related to overall vehicle response to buffet excitation in the frequency range below 20 cps. It is now known, based on wind tunnel tests, that bulbous payload shapes introduce large gross vehicle buffet loads. In the Task #2 MOL Payload Constraint loads studies noted above, the portion of the total load due to buffet was based on T-III wind tunnel buffet tests and T-II Gemini wind tunnel buffet tests; however, considerable interpretation and modification of the test data was required to provide buffet loads for these studies. Due to the possible severe buffet loads induced by the bulbous Apollo/MOL configuration, verification of the buffet loads by wind tunnel tests should be included in the Apollo/MOL program.

The above comments apply to a configuration without stabilizing fins. We do not have enough information to enable us to assess the loads situation if fins are to be used. The necessity for fins and their effect on structural loads can be determined only through a further study program.

It has been possible to obtain very little information on the loads analyses performed by NAA in connection with the T-IIIC booster. The data we do have seem to agree generally with Aerospace results generated for similar configurations (e. g., bending moment diagram PS 70612 of NAA document PS 64-117, "Apollo Applicability to MOL").

12. 2 SATURN IB CONFIGURATION

The configurations studied for Apollo/MOL are within the envelope of those investigated extensively by NAA in the XMAS Study, in which feasibility was established for a mission having requirements generally similar to those of Apollo/MOL. For the purposes of the present study, it can therefore be concluded that the Apollo/MOL/SIB configuration is feasible from a structural standpoint.

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