COST STUDIES OF MULTIPURPOSE LARGE LAUNCH VEHICLES

VOLUME VII

ADVANCED TECHNOLOGY IMPLICATIONS

OCT 196
RECEI
NASA STI F
INPUT BR

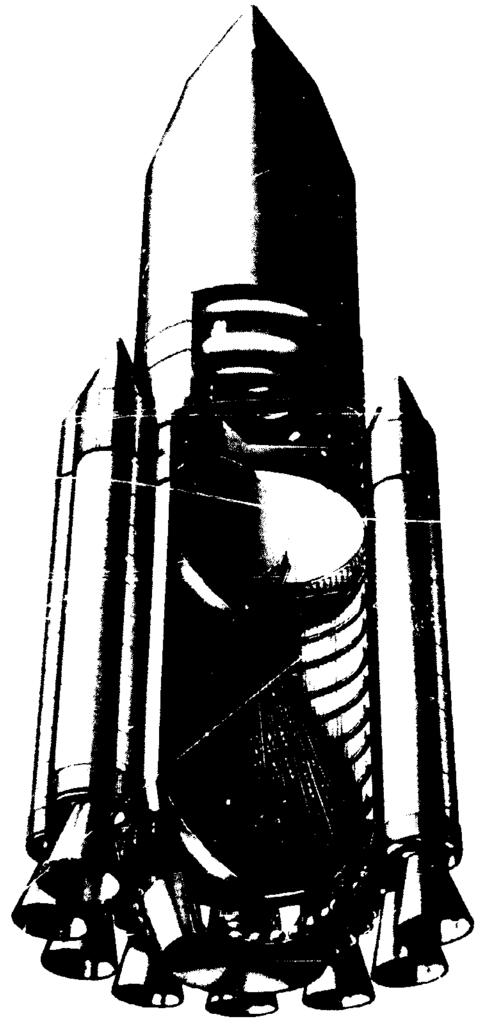
NASA

FINAL REPORT

SEPTEMBER 15, 1969

NASA DOCUMENT NO.
CR-73334

H-7 ~~C.J. Donlan~~ D.R. Lord ~~MD-T~~ MTD



PREPARED UNDER CONTRACT
NAS 2-5056

BY THE BOEING COMPANY
AEROSPACE GROUP
SOUTHEAST DIVISION

(BOEING DOCUMENT NO.
D5-13463-7)

N69-39580
(ACCESSION NUMBER) (THRU)
38 1
(PAGES) (CODE)
CR-73334 31
(NASA CR OR TMX OR AD NUMBER) (CATEGORY)
FACILITY FORM 602

D5-13463-7

FINAL REPORT
FOR
COST STUDIES OF MULTIPURPOSE
LARGE LAUNCH VEHICLES

VOLUME VII
ADVANCED TECHNOLOGY IMPLICATIONS

PREPARED UNDER CONTRACT NAS2-5056
FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
OFFICE OF ADVANCE RESEARCH AND TECHNOLOGY
MISSION ANALYSIS DIVISION
SEPTEMBER 15, 1969

PREPARED BY

Charles J. Corso
CHARLES J. CORSO

APPROVED BY

Joseph W. Monroe
JOSEPH W. MONROE

THE BOEING COMPANY
SOUTHEAST DIVISION
HUNTSVILLE OPERATION
HUNTSVILLE, ALABAMA

PRECEDING PAGE BLANK NOT FILMED.

ABSTRACT

Nine volumes including this volume present the final report documentation outlining the accomplishments for the "Cost Studies of the Multipurpose Large Launch Vehicles" (MLLV), NASA/OART Contract NAS2-5056. This Advanced Technology Implications Volume presents those areas where a need for further research and/or technology investigations was indicated by the vehicle study program.

The MLLV family will consist of a single-stage-to-orbit configuration plus other configurations consisting of a main stage (as used for the single-stage-to-orbit configuration) with various quantities of 260 inch diameter solid rocket motor (SRM) strap-on stages and/or injection stage modules. The main stage will employ LOX/LH_2 propellant with either a multichamber/plug or toroidal/aerospike engine system. The single-stage-to-orbit configuration will have a payload capability of approximately 500,000 pounds to a 100 nautical mile earth orbit. With the addition of the strap-on SRM stages and/or LOX/LH_2 injection stage modules, this payload capability can be increased incrementally to as much as 1,850,000 pounds.

The contract consisted of four study phases. The Phase I activity was a detailed cost analysis of an Advanced Multipurpose Large Launch Vehicle (AMLLV) family as previously defined in NASA/OART Contract NAS2-4079. Costs for vehicle design, test, transportation, manufacture and launch were defined. Resource implications for the AMLLV configurations were determined to support the cost analysis.

The Phase II study activity consisted of the conceptual design and resource analysis of a smaller or half size Multipurpose Large Launch Vehicle (MLLV) family.

The Phase III activity consisted of a detailed cost analysis of the smaller Multipurpose Large Launch Vehicle configurations as defined in Phase II. Costs for vehicle design, test, transportation, manufacture and launch were determined.

The Phase IV activity assessed the results of the study including the implications on performance, resources and cost of vehicle size, program options, and vehicle configuration options. The study results provided data in sufficient depth to permit analysis of the cost/performance potential of the various options and/or advanced technologies.

ABSTRACT (Continued)

KEY WORDS

Advanced Multipurpose Large Launch Vehicles (AMLLV)

Half Size Multipurpose Large Launch Vehicles (MLLV)

Single-Stage-to-Orbit

Multichamber/Plug Engine System

Toroidal/Aerospike Engine System

260 Inch Solid Propellant Rocket Motor (SRM)

Orbital Injection Stage

Contract NAS2-4079

Contract NAS2-5056

Payload to 100 NM Orbit

Cost

Resources

Zero Stage Vehicles

Parallel Stage Vehicles

Main Stage Throttling

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

FOREWORD

This volume, Advanced Technology Implications, is one of nine volumes documenting the results of a twelve month study program "Cost Studies of Multipurpose Large Launch Vehicles," NASA/OART Contract NAS2-5056. The objective of this study was to define cost, cost sensitivities, and cost/size sensitivities of potential future launch vehicles to aid in the guidance of current and future technology programs. The baseline vehicles utilized to make this assessment were:

a. The Advanced Multipurpose Large Launch Vehicles (AMLLV) as defined under NASA/OART Contract NAS2-4079.

b. The Multipurpose Large Launch Vehicles (MLLV) as defined under this contract and described in Document D5-13463-2, "Half Size Vehicle (MLLV) Conceptual Design."

The program documentation includes this "Advanced Technology Implications Volume" Volume 7 plus a Summary Volume, a Design Volume, a Resources Volume, Cost Volumes, Cost Implications Volume, and Appendices Volumes. Individual designations for these volumes are as follows:

Volume I	Summary
Volume II	Half Size Vehicle (MLLV) Conceptual Design
Volume III	Resource Implications
Volume IV	Baseline AMLLV Costs
Volume V	Baseline MLLV Costs
Volume VI	Cost Implications of Vehicle Size, Technology Configurations, and Program Options
Volume VII	Advanced Technology Implications
Volume VIII	Flight Control and Separation, and Stress Analysis (Unclassified Appendices)
Volume IX	Propulsion Data and Trajectories (Classified Appendices)

Foreword (Continued)

Data on the 260 inch diameter solid propellant rocket motor were obtained from the Aerojet General Corporation. Data on the multichamber/plug propulsion system were obtained from the Pratt and Whitney Division of the United Aircraft Corporation and the Rocketdyne Division of the North American Rockwell Corporation. Data on the toroidal/aerospike propulsion system were obtained from the Rocketdyne Division of the North American Rockwell Corporation.

These propulsion data were obtained from the propulsion contractors at no cost to the contract. The material received encompassed not only the technical data, but resources, schedules, cost, and advanced technology information. This support materially aided The Boeing Company in the preparation of a complete and meaningful study and is gratefully acknowledged.

This study was administered under the direction of NASA/OART Mission Analysis Division, Ames Research Center, Moffett Field, California under the direction of the technical monitor, Mr. Edward W. Gomersall.

1.0 INTRODUCTION

This report summarizes the Advanced Technology Implications evolving from the NASA/OART Contract NAS2-5056, "Cost Studies of Multipurpose Large Launch Vehicles." This study was conducted to define costs and cost related sensitivities of conceptual launch vehicle systems, and to determine the economic potential of indicated technological advantages of such systems. An additional objective was to identify the future technology requirements that have promise for further improving the economics of the concepts. Maximum use was made of the conceptual designs employed on the Advanced Multipurpose Large Launch Vehicle as defined by the prior NASA/OART Contract NAS2-4079. This prior study developed a single-stage-to-orbit vehicle concept which would be capable of placing 1,000,000 pounds of payload into a 100 NM orbit. With the addition of injection stage modules and strap-on stages, this payload capability could be increased to approximately 3.8 million pounds. For this current study, a half size (MLLV) single-stage-to-orbit concept capable of placing 500,000 pounds into a 100 NM orbit was identified. Addition of injection stage modules and strap-on stages will increase this payload capability to approximately 1.85 million pounds.

The first phase of this study determined, collected and collated the costs of the full size AMLLV configurations. The Phase II activities conducted vehicle trades, provided a conceptual design, and identified the resources necessary to produce the half size vehicle. In the third phase, the half size vehicle costs were determined. In the last phase of the study, a comparison of vehicle costs collected in Phases I and III were conducted. In this analysis, cost sensitivities, and cost/size sensitivities were determined as well as the cost effectiveness of program and configuration options and of advanced technology applications.

In general, the concepts for both the full size and the half size vehicles can employ existing design concepts and materials. No major unique item of tooling or capital equipment was identified which could not be developed within the existing state-of-the-art.

Specific technology areas and/or advanced technology alternatives indicated for further study are presented in Sections 2.0 through 7.0 of this volume. These are divided into systems, aerodynamics, design, vehicle environments, manufacturing technology, and launch facility implications. In addition to those areas identified by The Boeing Company, the propulsion contractors supported the activity with their recommendations for propulsion systems advanced technologies.

PRECEDING PAGE BLANK NOT FILMED

2.0 SYSTEMS IMPLICATIONS

2.1 ALTITUDE IGNITION OF THE MAIN STAGE ENGINE

2.1.1 Problem Definition

The baseline vehicle is a single-stage-to-orbit vehicle which requires ignition of the main stage engine at lift-off. When strap-on solid motor stages are utilized with the main stage, those vehicle configurations employing more than two strap-on solid stages employ a zero stage launch mode. The SRM stages are ignited at lift-off and the main stage engine is ignited at altitude. The development of an engine system capable of both sea level and altitude start will require that the engine and stage hardware be more complex and heavier than if only one ignition mode were required.

The design considerations involved in this dual ignition capability are prevention of damage to the main stage engine nozzle from the strap-on stages exhaust gases during the zero stage phase, development of an engine operation sequence for both ground launch and altitude start, development of a thrust chamber prefill system for both ground launch and altitude ignition and development of electrical and hydraulic systems which can be used for both the ground launch and altitude ignition modes.

2.1.2 Solution Approach

An analysis must be made of the variable environment that exists when strap-on stages are added to the main stage. Since the number of strap-on SRM stages will vary from 2 to 8, the thermal, acoustic and atmospheric environments encountered will be significantly different. Insulation for the protection of the module nozzles in the multichamber/plug propulsion system and for the nozzle lip on the toroidal/aerospike propulsion system must be examined. As an alternative to altitude ignition, it may be desirable to operate the main stage engine at a very low throttled condition through the strap-on stage operation and thus eliminate the need for altitude ignition, thermal protection, and additional electrical and hydraulic systems. Performance impact and economic considerations should be examined.

2.2 LARGE SOLID ROCKET MOTOR THRUST VECTOR CONTROL SYSTEMS DEVELOPMENT

2.2.1 Program Definition

The baseline vehicle as defined by this study requires 3.9 degrees of gimbal angle capability per solid motor to maintain control of the vehicle during the time of maximum dynamic pressure. This gimbal angle capability has been demonstrated

2.2.1 (Continued)

by the flexible seal nozzle concept in smaller solid motor applications. The flexible seal concept has not been applied to nozzles of the size required for this application.

2.2.2 Solution Approach

A 260 inch nozzle, incorporating a flexible seal concept with a 3.9 degree of gimbal capability (plus an additional degree as a safety factor) should be designed, built and tested in the actual firing of a 260 inch solid diameter motor. This test in addition to verifying the design and demonstrating the feasibility of the nozzle system will also provide data on the actuation force requirements and the cycle rate.

2.3 MULTICHAMBER/PLUG PROPULSION SYSTEM - NOZZLE DESIGN

2.3.1 Problem Definition

The existing multichamber/plug propulsion system design utilizes a single position nozzle. This nozzle is positioned axially until it flows full at altitude and then it is canted inward towards the plug. As a result, during the initial portion of the launch, the nozzles are overexpanded and degradation of engine performance occurs. As an alternative, a two position multichamber/plug cluster configuration can be designed which will operate more efficiently and will weigh less than the multichamber/plug single position nozzle.

2.3.2 Solution Approach

With the multichamber/plug two position nozzle concept, the same engine power package consisting of: turbopumps, preburner, and main burner, can be used with nozzle skirts of different expansion ratios. This will permit each skirt to be tailor-made to provide the optimum performance for a specific altitude regime, i.e.: the primary or "fixed" nozzle would have a low expansion ratio to provide significantly improved sea level and intermediate altitude engine performance. The secondary or "moveable" nozzle would increase the overall module expansion ratio when extended to provide greater performance at higher altitudes. The two position nozzle would have a lightweight dump cooled secondary nozzle and a tube type regeneratively cooled primary nozzle. This will result in a lower thrust chamber weight (including the weight of the nozzle translating mechanism) when compared to the fully regeneratively cooled tube type single position nozzle. (The above advanced technology implication was provided by the Pratt and Whitney Division of United Aircraft Corporation.) Section 7.1.5 of Volume VI discusses the cost/performance advantages of this concept.

2.4 MULTICHAMBER/PLUG PROPULSION SYSTEM - PLUG DESIGN

2.4.1 Problem Definition

The multichamber/plug propulsion system consists of the large number of engine modules mounted circumferentially around a plug. This plug during single-stage-to-orbit vehicle operation must be cooled to prevent structural damage. When used with the vehicle configurations employing strap-on stages, the plug must be further protected by an ablative coating to prevent the heat from the strap-on stages from destroying the plug. The plug itself weighs approximately seven percent of the total weight of the propulsion system. The use of the plug results in a loss of payload capability.

2.4.2 Solution Approach

With the two position nozzle, the engine modules have excellent performance at both sea level and vacuum. It may be possible to eliminate the base plug. The centerlines of the engine modules would then be parallel to the vehicle centerline throughout the flight. The structural advantage of mounting the engines in a ring around the peripheral of the vehicle would be maintained. The engine module specific impulse would increase about two to three percent because the requirements for base pressurization would be eliminated. Further, the base plug structure would be deleted and the need for the gas generators for base pressure would be eliminated. Mission trade studies would be required to determine possible losses in overall mission performance because of the lower specific impulse during the time when the modules would normally be tilted against the plug. (This advanced technology implication was provided by the Pratt and Whitney Division of United Aircraft Corporation.)

2.5 LOW PRESSURE TOROIDAL/AEROSPIKE PROPULSION SYSTEM DEVELOPMENT

2.5.1 Problem Definitions

For the 2000 psia chamber pressure toroidal/aerospike propulsion system, it will be necessary to develop advanced high pressure turbopump machinery. This machinery would require time for development and would increase the propulsion system development costs. An alternative is a 1200 psia chamber pressure toroidal/aerospike propulsion system using the J-2S turbopump machinery which will have already undergone development and may share common usage with another vehicle system. The weight of the system would decrease approximately 25 percent by reducing from the 2000 to 1200 psia operating pressure. These advantages, however, would be somewhat offset by the loss in performance which will occur with the lower operating pressure. The cost effectiveness analyses discussed in Section 5.2.2 of Volume VI indicate that this approach is the most cost effective for small programs with few operational flights. For larger programs the

2.5.1 (Continued)

lesser performance at the lower pressure will override the weight (and engine cost) advantages such that the 1200 psia system will probably not be as cost effective as the 2000 psia system.

2.5.2 Solution Approach

Comparisons with a low pressure propulsion system using new low pressure turbopump machinery, and a higher operating pressure (2000 psia) propulsion system using new turbopump machinery should be reviewed and explored in more detail. (This advanced technology implication was provided by the Rocketdyne Division of North American Rockwell Corporation.)

2.6 ON-BOARD TEST AND CHECKOUT SYSTEM

2.6.1 Problem Definition

The present operational launch vehicles do not employ completely automated on-board test and checkout systems. As a result, each system of the vehicle is subjected to various test and checkouts throughout its manufacture, assembly, pre-launch and operation. Use of on-board test and checkout systems on each specific stage may reduce the test and checkout time sequence at the launch site, provide for greater reproducibility, permit interim checkout of subsystems and systems (by their integral test sets) at the manufacturing site, reduce the required functions for ground support equipment at the launch site and reduce the number of functions required of the instrument unit of the vehicle. (An on-board test and checkout system is discussed in further detail in Section 4.3.6 of Volume II.)

2.6.2 Solution Approach

An on-board test and checkout system and its individual elements should be defined in greater detail. Trades should be conducted to determine the desired design and operational features. Advantages in terms of performance, cost and schedules of these systems should then be compared to the existing ground test and checkout systems.

3.0 AERODYNAMIC IMPLICATIONS

3.1 UNCONTROLLED DIVERGENCE RATES

3.1.1 Problem Definition

The MLLV configurations will not have sufficient time to double amplitude to permit the pilot to abort. Several alternatives are available to provide sufficient time. These alternatives include a flared aft skirt or fins, or an automatic system which will eliminate man from the loop. The response time of this latter system should be capable of reducing the abort time requirements to an acceptable level.

3.1.2 Solution Approach

Each of the above approaches should be analyzed in terms of effect on vehicle performance and costs. An automatic system appears to be the most desirable system from the standpoint of lower weight and cost than the other alternatives proposed. This type of approach could be used with existing vehicles so as to reduce the procurement costs.

3.2 AERODYNAMIC STATIC STABILITY

Vehicles that use strap-on stage components are difficult to analyze because of the complex flow field about the main stage and the strap-on stages. Estimates of the total normal force contribution to the core vehicle are difficult to make because of the carry-over from the strap-on stages to the core vehicle and the carry-over from the core vehicle to the strap-on stages. The effective gap between the core vehicle and the strap-on stages and the effect of engine throttling are also difficult to determine. Lack of experimental data and inability of the various theoretical methods add to this difficulty. (Data from previous NASA sponsored tests of models with strap-on stages were used to improve the analytical techniques for the MLLV. This data, however, was not directly applicable because of geometric variations.)

3.2.3 Solution Approach

It is recommended that wind tunnel tests be conducted to determine the effects of these strap-on stages on the vehicle aerodynamic coefficients. These tests should be done with the number of strap-on stages varied between the lower and the upper limit.

3.3 AERODYNAMIC HEATING

3.3.1 Problem Definition

The complex configuration created by the addition of strap-on stages to the vehicle will create localized aerodynamic heating effects where the shock waves from the SRM stage noze cone impact the main stage forward skirt. The vehicle configuration must consider modification to the forward skirt design to compensate for this effect.

3.3.2 Solution Approach

Wind tunnel tests should be conducted to define these localized effects throughout the flight trajectory. The full range of strap-on stages should be examined.

3.4 STRAP-ON STAGE SEPARATION

3.4.1 Problem Definition

The solid propellant rocket motor (SRM) strap-on stages will be separated from the core vehicle during SRM tail-off. The separation trajectories of these expended SRM stages relative to the core veh[illegible]le must be such as they will not endanger the continuing flight vehicle.

3.4.2 Solution Approach

Wind tunnel model tests should be conducted for the expended SRM stages. Trajectories should be defined, and design concept or features altered if problems exist.

4.0 ADVANCED CONFIGURATION DESIGN IMPLICATIONS

4.1 ALTERNATIVES TO ALUMINUM STRUCTURES

4.1.1 Problem Definition

The half size MLLV configuration is based on proven materials and manufacturing techniques. Advanced technology to provide better materials and manufacturing techniques may be desired.

4.1.2 Solution Approach

Composite or sandwich structures offer the potential for structural weight reduction and consequently vehicle size reduction. Figure 4.1.2-1, Sheet 1, shows a design using aluminum honeycomb with a titanium face sheet for the hydrogen tank structure. The oxygen tank structure is an integral aluminum skin stringer configuration identical to the design presently proposed. No change is made due to the poor compatibility between liquid oxygen and titanium. As the cylindrical section of the LOX tank is short, very little weight savings is possible. No internal rings are necessary in either tank except for the junction rings. The forward and aft skirts are bonded aluminum/honeycomb sandwich construction. The thrust posts are bolted onto honeycomb panel edgings as shown in Section A of Figure 4.1.2-1, Sheet 2.

Beryllium or titanium truss core sandwich construction may be even more promising for tank walls as indicated by Figure 4.1.2-2. In addition to the weight reduction offered by truss core construction of liquid hydrogen tank, the need for insulation may be eliminated by proper design of the skin panels.

Filament wound tanks using a roving winder can reduce the tank weight by as much as 50% if a metal or elastomer liner can be developed to withstand the fatigue caused by numerous pressure cycles. (A liner is necessary due to the porosity of the epoxy binder.)

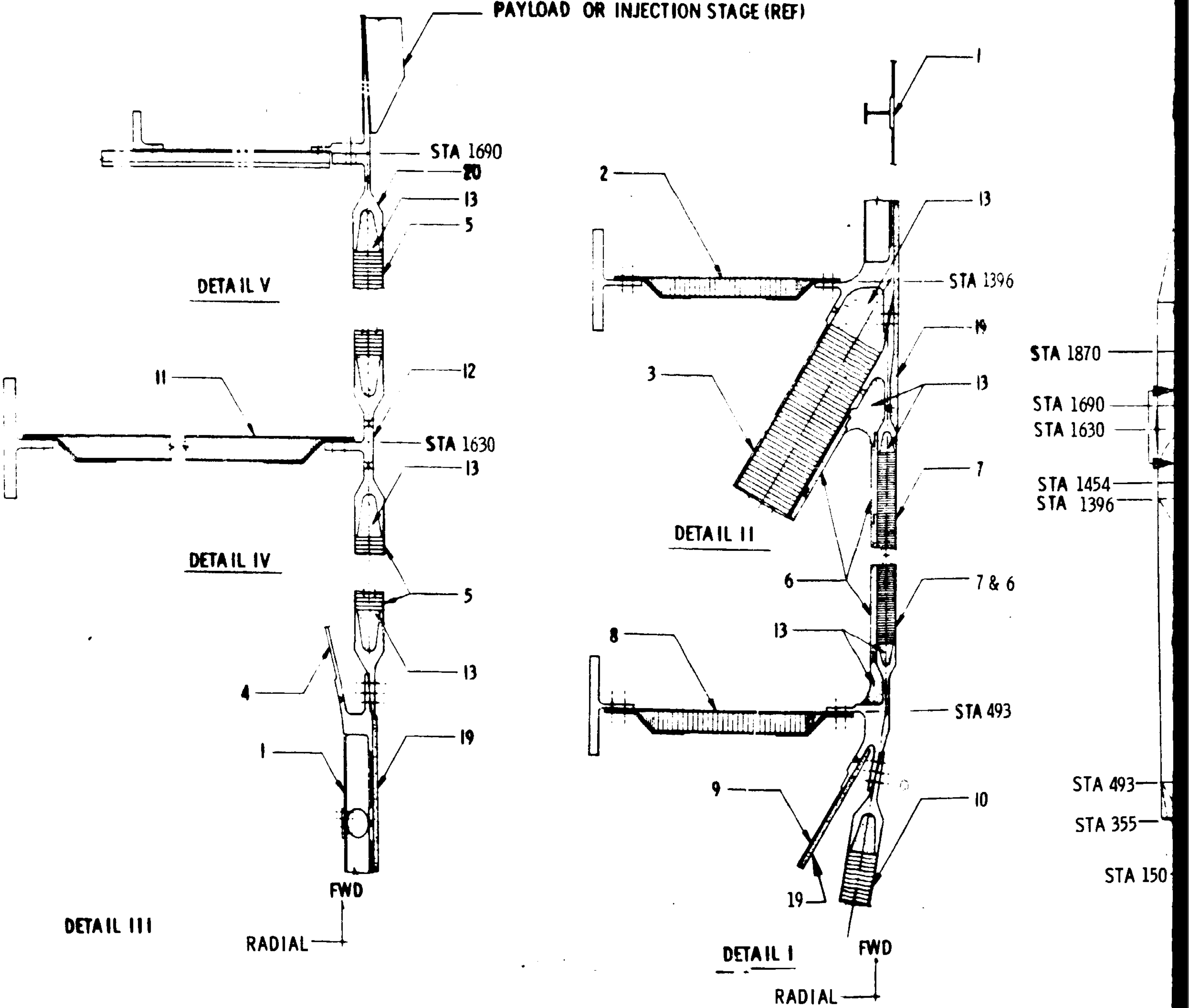
All of the above structural options should be studies both technically for performance characteristics and resource-wise to determine fabrication and cost implications

4.2 STABILITY CRITICAL STIFFENED CYLINDRICAL SHELL DESIGN

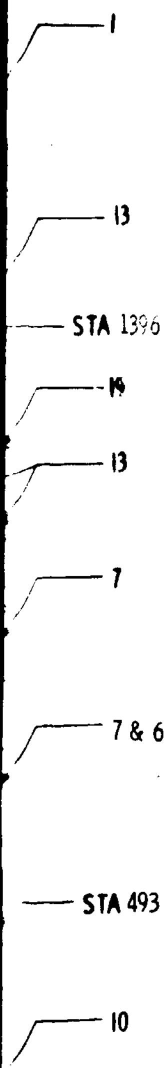
4.2.1 Problem Definition

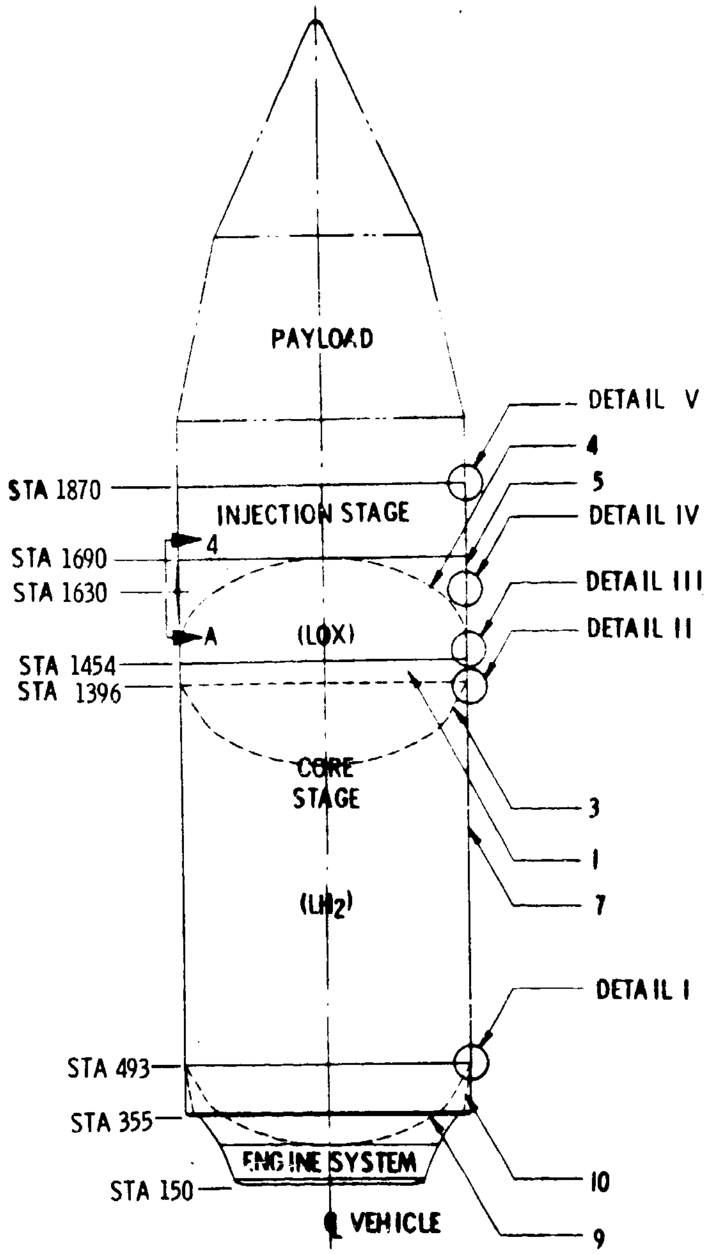
Several methods of analysis have been developed to more accurately predict the load carrying capability of stability-critical axial compressed stiffened cylinders.

PRECEDING PAGE BLANK NOT FILMED.



FOLDOUT FRAME 1

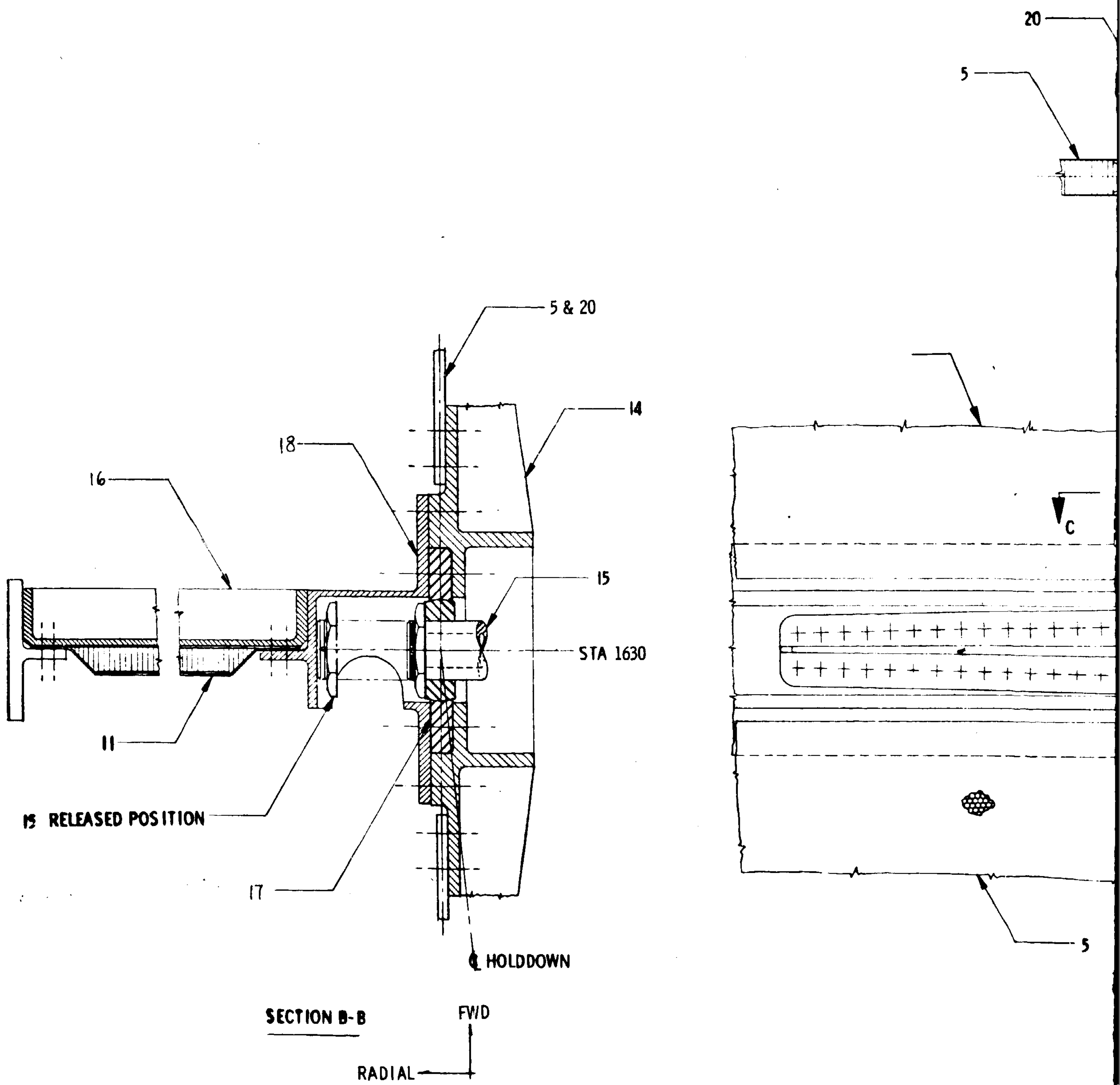




SIDE VIEW

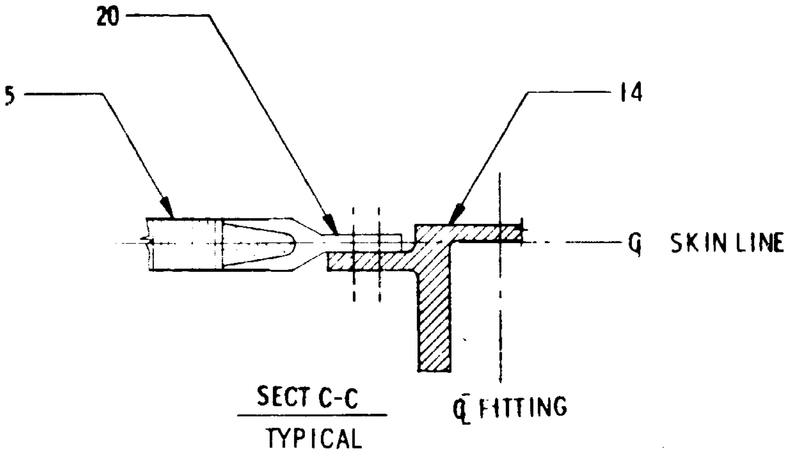
1. LOX CYLINDER. 2219-T87 AL INTEGRAL SKIN-STRINGER.
2. JUNCTION RING FRAME, COMMON BULKHEAD. HONEYCOMB CONSTRUCTION, BRAZED 2219-T87 AL FACING WITH 5052 AL CORE.
3. COMMON BULKHEAD. 2219-T87 AL FACING WITH 5052 AL CORE BRAZED OR ADHESIVELY BONDED.
4. UPPER BULKHEAD. MILLED 2219-T87 AL MONOCOQUE PLATE.
5. FORWARD SKIRT. HONEYCOMB CONSTRUCTION, 6AL-4V Ti/5052 AL OR BE CORE
6. INSULATION, LH_2 TANK. POLYURETHANE FOAM TILE ADHESIVELY BONDED TO TANK.
7. LH_2 TANK CYLINDER. HONEYCOMB CONSTRUCTION, 6AL-4V TITANIUM FACING WITH 5052 AL CORE.
8. JUNCTION RING FRAME, LOWER BULKHEAD. HONEYCOMB CONSTRUCTION, 2219-T87 AL FACING WITH 5052 AL CORE.
9. LOWER BULKHEAD. MILLED 2219-T87 AL MONOCOQUE PLATE.
10. AFT SKIRT. HONEYCOMB CONSTRUCTION, 6AL-4V TITANIUM FACING WITH 5052 AL CORE, BRAZED OR BONDED
11. DEEP RING FRAME, HOLDDOWN. HONEYCOMB CONSTRUCTION, 7075-T6 AL FACING WITH 5052 AL CORE.
12. RING CAP. INTEGRAL TEE OF FORWARD SKIRT, 2219-T87 AL EXTRUSION.
13. INJECTION FOAM STABILIZER. POLYURETHANE FOAM.
14. HOLDDOWN POST. 7075-T6 AL DIE FORGING.
15. SHEAR PIN/TENSION BOLT COMBINATION, HOLDDOWN. 4340 STEEL, 270 KSI H. T.
16. BACKUP FITTING. 7075-T6 AL DIE FORGING.
17. SPECIAL SPHERICAL BEARING, HOLD DOWN. 4340 STEEL, PRESS FIT INTO 14.
18. SHEAR PIN RETAINER FITTING. 7075-T6 AL DIE FORGING.
19. INSULATION, POLYURETHANE FOAM, SPRAY-ON TYPE.
20. TUNING FORK EDGE FITTING. 2219-T87 AL EXTRUSION, WELDED OR ADHESIVELY BONDED TO HONEYCOMB PANELS.

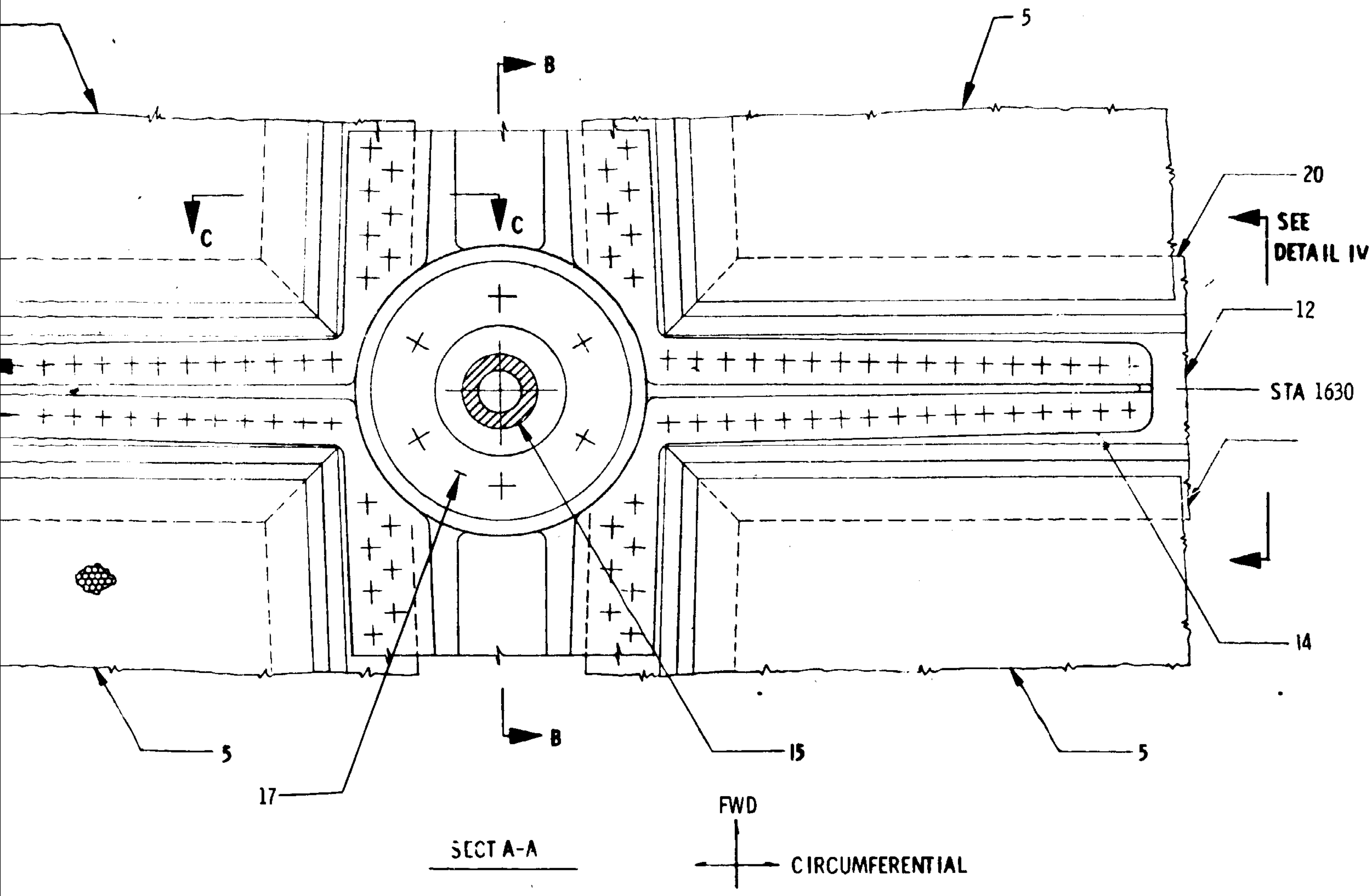
ADVANCED STRUCTURES
FIGURE 4.1.2-1 SHEET 1



SECTION B-B

FOLDOUT FRAME





ADVANCED STRUCTURES

FOLDOUT FRAME FIGURE 4.1.2-1 SHEET 2

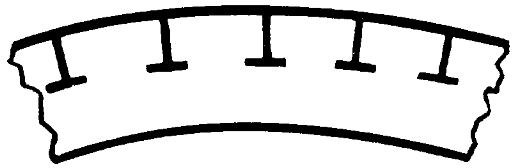
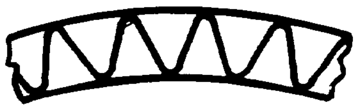
FOR SPECIFIC DESIGN POINT (50 psig, Nc = 10,000 lbs./inch)		INDICATED WEIGHT SAVING
	ALUMINUM SKIN-RING STIFFENER CONSTRUCTION (BASELINE CONSTRUCTION	0%
	ALL ALUMINUM TRUSS CORE CONSTRUCTION	20%
	ALL TITANIUM TRUSS CORE CONSTRUCTION	37%
	TITANIUM FACE SHEET BERYLLIUM CORE	50%

FIGURE 4.1.2-2 WEIGHT COMPARISON FOR PROPELLANT TANK WALLS

4.2.1 (Continued)

However, none of these methods are reliable for all stiffened cylinder configurations. One of these methods, developed by Boeing, and as shown in Reference 4 2.1-1 considers all three possible modes of instability failure and has been partially verified with results from the Saturn V/S-IC corrugated intertank test program and from various test data in the literature. This method should be extended to account for increase in load carrying capability due to pressure stabilization

4.2.2 Solution Approach

The validity and range of applications of applicable analytical procedures, including the methods developed by Boeing, should be verified by further tests and analyses The results of several different methods should be compared and the best of these results applied to each type stiffened cylinder on the MLLV for possible weight savings and increased reliability.

4.3 STRUCTURAL DESIGN AND TEST PHILOSOPHY, GROUND RULES AND ASSUMPTIONS

4.3.1 Problem Definition

Previous vehicle designs have utilized design and loads criteria, safety factors, material allowables and test procedures which create large weight penalties. These criteria are ground ruled into future vehicle designs because of their proven performance on test design vehicles. To get the weights into a more desirable range, it has been necessary to approach the vehicle configurations by the use of ingenious design techniques to remove weight rather than accepting more realistic design criteria.

4.3.2 Solution Approach

All previous design criteria should be re-evaluated in light of the advances made in the state-of-the-art and empirical results. Both higher and lower safety factors should be evaluated in terms of performance and economic considerations. Tests should be reviewed to determine the need for the test. Points at which the tests may be removed from the program and/or substitution of new testing techniques or procedures to improve economy without reducing reliability should be defined.

4.2.1-1 D5-13272, "Analysis of Stability Critical Orthotropic Cylinders Subjected to Axial Compression", The Boeing Company, September 14, 1966.

5.0 VEHICLE ENVIRONMENT

5.1 ACOUSTIC ENVIRONMENT

5.1.1 Problem Definition

Two severe acoustic environmental problems will occur as a result of the addition of strap-on stages to the core vehicle. One of these problems will be that the shock wave off the nose cone of the strap-on stages will impact the main stage creating an acoustic spike at the point of impact. This acoustic overall sound pressure level cannot be readily determined with existing mathematical techniques and computer programs. The second problem will be the high acoustic sound levels at lift-off caused by the reflected acoustic ground wave which will reinforce the primary acoustic wave thus creating a high acoustic level at the base region of the vehicle. The resulting acoustic level will exceed the levels obtained in past or current vehicles. No method is available for assuring that the components packaged in this area are acoustically acceptable without an actual test of the vehicle. The actual acoustic environments must be better defined to evaluate the above problems and to provide launch pad siting criteria. (See Section 7 below.)

5.1.2 Solution Approach

It will be necessary to conduct wind tunnel model tests to determine the shock waves which occur off the solid motor strap-on stages and to determine their impact points onto the main stage. By relocation of the acoustically sensitive components, it may be possible to minimize the effects of the acoustics on the vehicle. The second problem (base area acoustics) can be resolved several ways. The first way may be by conducting the acoustic tests for a longer period at a lower sound pressure level. Another technique would be to develop equipment capable of producing the necessary sound level. A third method would be to develop materials and packaging techniques which would reduce the sound pressure level that reaches the acoustically sensitive components. Tests should be developed that would determine how much a reduction in sound pressure level would occur with the proper type of materials and packaging techniques.

5.2 BASE HEATING ENVIRONMENT

5.2.1 Problem Definition

The large volume of exhaust gasses from the strap-on stages will create a severe vehicle base thermal environment. The main stage may not be functioning during this period and must be protected from the strap-on stage environment. To provide the necessary data for the design of (1) the strap-on stage shield, (2) the main stage base heat shield and (3) the main stage engine nozzle protection devices, this environment must be defined.

5.2.2 Solution Approach

Test data from large solid rocket motors must be correlated to determine the scaling factors for modeling tests. Two separate tests of test models should then be conducted, i.e.:

1. Model tests simulating vehicle lift-off from the launch pad.

2. Wind tunnel model tests simulating various flight conditions (i.e., varying mach numbers and ambient pressures).

5.3 INSULATION MATERIALS AND INSULATING ASSEMBLIES

5.3.1 Problem Definition

The aforementioned thermal environment definition (see Section 5.2) will be used for the selection of insulation materials and design of insulating elements. As this environment is prognosticated to be significant, new materials and/or new approaches to applications of these materials are indicated. Actual tests under simulated environments will be required to prove these materials and their applications.

5.3.2 Solution Approach

The candidate materials and design concepts for their applications must be selected. A materials test program should be conducted to determine the operational characteristics of the materials and the insulating assemblies under simulated use environments.

6.0 MANUFACTURING TECHNOLOGY IMPLICATIONS - MAIN STAGE

6.1 ELECTRON BEAM WELDING PROCESS

6.1.1 Problem Definition

Out-of-vacuum electron beam (EB) welding is the process proposed for welding of the various components of the MLLV since it is deemed to be a feasible, reliable, and cost effective welding method which possesses certain advantages over the more conventional inert gas (TIG or MIG) processes. However, since no flight hardware has been produced by this process, a certain amount of development effort is required to perfect the optimum weld schedules and methods. Two of the principal advantages expected to accrue to the electron beam process are: less distortion in the joint and adjacent structure; and a lower sensitivity of weld quality to weld heat deflection.

6.1.2 Solution Approach

Tests should be conducted on hardware using materials, construction methods and dimensions typical of launch vehicles. Weld specifications should be prepared. The welds should be inspected by various non-destructive test techniques. Data banks should be developed on weld defects, porosity, etc., and tabulated. Techniques and inspection specifications for non-destructive testing of electron beam welds should be developed.

6.2 BULKHEAD GORE SEGMENT FORMING

6.2.1 Problem Definition

The problems inherent in the forming of compound contoured bulkhead segments on the S-IC will also exist on the MLLV main stage, but will be magnified in some degree due to the increased size. It is believed, however, that this forming operation is well within the scope of present day manufacturing technology. The gore segments will be bulged or formed depending primarily on skin thickness selected. If the skins are not too thick (1/4 inch or less) the segments might lend themselves more readily to stretch forming since the skins will be constant thickness (no waffle pattern for added stiffness) except for weld lands on the periphery.

6.2.2 Solution Approach

Gore segments should be formed using both of the above techniques (with the thicknesses envisioned for the MLLV gores) to define the problems associated with the forming and to identify the necessary tool and capital equipment requirements. The same specimens used for electron beam welding tests could

6.2.2 (Continued)

be used for gore segment forming tests.

6.3 FABRICATION OF THE COMMON BULKHEAD

6.3.1 Problem Definition

Fabrication and assembly of the common bulkhead will impose a number of problems due primarily to the size of the structure. A brief outline of the assembly sequence will permit an insight to the fabrication problems expected. The upper (LOX) common bulkhead skin will be fabricated (convex side up) in a manner similar to that employed in the fabrication of upper and lower bulkheads. The common fitting ring will then be welded to the skirt of this skin, utilizing methods resembling those used in joining the Y-rings to the upper bulkhead. Subsequently, this skin and ring assembly will be cleaned, etched, and primed on the convex side. Adhesive film will then be placed on this surface and tacked. Flexcore aluminum honeycomb (purchased in the largest sections available) will be located on the adhesive film and core splice adhesive applied at the core section butt joints. The adhesive film and core will be installed by separate crews of personnel, in that sequence, working simultaneously. The lower (liquid hydrogen) skin which will also have been formed in apex and base gore segments, will be positioned and installed on the segments.

Prior to assembly, the apex and the base gore segments, polar cap, and Y-ring will be cleaned, etched, and primed as before. Adhesive film will then be tacked onto the interior surfaces. The base gore segments will be placed in position, fitted, and temporarily fastened in place. The apex gore segments will then be located and fastened in the same manner. The common bulkhead assembly, as constructed to this point, will be placed in the autoclave and cured. Following removal from the autoclave, the polar cap, skin splice doublers and Y-ring will be placed again in the autoclave for final cure. After the final cure, foam will be pumped into the void bounded by the Y-ring, honeycomb and common fitting ring. The Y-ring will be mechanically fastened to the common fitting ring, through their bonded interface, by bolts. Commencing with the upper skin cleaning operation, all subsequent tasks will be performed in a controlled atmosphere or positive pressure area to minimize contamination.

The principal problems foreseen in the construction of the common bulkhead involve: handling; adhesives; nondestructive testing and distortion. Each of these problem areas are discussed in the solution approach below.

6.3.2 Solution Approach

Handling - This problem pertains to the actual lifting and moving of the lower (apex and base gore segments and the polar cap) skin parts of the common bulk-

6.3.2 (Continued)

head which form the side of the common bulkhead facing the liquid hydrogen tank. The problem also is that of locating and fitting of those parts without dislocating the core or adhesive film. A likely solution to the former problem would be the use of hoisting tool with vacuum cups attached. The latter problem can conceivably be solved by the exercising of extreme care by the personnel engaged in fitting the liquid hydrogen skin segments in the core and adhesive film. An alternative method of assembly could be implemented. This method would consist of laying up the adhesive film and aluminum core on the liquid oxygen common bulkhead skin surface and applying the core splice adhesive to the core butt joints as before but then curing in the autoclave. Subsequent operations would then involve cleaning, application of the adhesive film, placing and fitting of the liquid hydrogen skin segments, curing, installation of polar cap and skin splice doublers, attachment of the Y-ring and the final cure in the autoclave.

Adhesive - The problems pertaining to adhesives as related to the main stage are problems that exist currently, but that may not exist in the time period when the MLLV is built. Specifically, the problems involve the need for: A cryogenic compatible adhesive; a LOX compatible adhesive; and a liquid hydrogen compatible adhesive. The need exists also for these adhesives to possess longer "open" assembly time to provide sufficient time for lay-up of all details in the operational sequence previously outlined. A study to develop the compatible adhesive should be initiated in the near term future to ensure its availability when required.

Non-destructive testing - This problem involves the need for a completely reliable and cost-effective method of non-destructive testing which will determine the quality of bonds within the common bulkhead. This is not a new problem since it existed and still exists within the Quality and Reliability Assurance phase of the Saturn V program. The need is now accentuated because of the size of the MLLV common bulkhead. Therefore, increased emphasis should be placed on non-destructive test programs to determine the quality of bonds in honeycomb structure. All of the current non-destructive methods are likely candidates for the initial phase of this study. New techniques involving ultrasonic and nuclear inspection should be included in the second phase of the advanced non-destructive testing program.

Distortion - This consideration relates to the problems of distortion in the LOX common bulkhead assembly and/or the liquid hydrogen skin segments or polar cap. If such distortion does exist, it could conceivably be removed, depending on the degree and extent (area) of distortion, by one of two methods. Certain types of distortion would be best suited for removal by the use of the Porta Power tool (magnetomotive hammer) which was used quite successfully in the S-IC program.

6.3.2 (Continued)

If the distortion is general or covers a large area, it could best be corrected by age forming (stress relieving) in a restrained aging fixture contoured for that particular part. These same distortion correction methods can also be utilized in correcting distortion problems on the upper and lower bulkheads.

6.4 CYLINDRICAL SKIN SECTIONS

6.4.1 Problem Definition

The proposed method for forming of the approximately 8 by 38-foot liquid hydrogen tank sections is as follows: Using a numerically controlled milling machine, mill the aluminum sheet in the flat to leave weld lands; stretch-straighten the separate extruded T-sections; electron beam weld T-sections to the machined weld lands while in the flat; age form in heat treat furnace; trim and check. The same procedure would be utilized for the shorter LOX tank skins. This method of age forming after welding would relieve stresses and distortion induced by the welding operation. In the event that electron beam welding process should be deemed undesirable, the T-stiffeners could be TIG or MIG welded, or mechanically fastened to the skins in a similar sequence.

Forming the skins to contour will be accomplished in a restrained aging fixture using a typical heat treat furnace. The configurations of the skins section (length and width versus nominal skin thickness) should readily lend itself to this forming operation. The skin section will be placed in an appropriately designed fixture and restrained at approximately 12 to 8 inch intervals across its entire surface and clamped to the fixture around the pheriphery. This section will be contoured by means of applying mechanical force, at the previously stated spacings. Force will be applied through large vertically mounted screws terminating in pressure pads in direct contact with the skin. With only a nominal development effort to determine the degree of springback that will be encountered, this process should consistently produce the desired results.

An alternative procedure would be to mill, form roll and then weld the T-sections to the skin sections. It is believed, however, that the skins will form just as readily with the T-sections already attached.

Another problem concerning fabrication and assembly of the liquid hydrogen tank skins will be that of handling. The fixture tooling and handling methods employed will have to be such that the skin sections will not be allowed to bend or deform past their elastic limits.

6.4.2 Solution Approach

Two proposed methods for forming the skin sections were presented above. Each of these methods should be analyzed to determine the more acceptable approach. Development efforts should be undertaken to determine the springbacks which will occur in skin contour forming. Tests to determine tooling and handling design criteria so that large structures can be handled without bending or deformation should be developed.

6.5 FOAM INSULATION - PROPELLANT TANKS AND LOX TUNNELS

6.5.1 Problem Definition

The exteriors of the LOX tank, liquid hydrogen tank, LOX tunnels and the surface of the common bulkhead facing the hydrogen tank will require a layer of foam insulation. The exterior cylindrical surface of the LOX tank, liquid hydrogen side of the common bulkhead and the LOX tunnels will require approximately one inch of insulation while the liquid hydrogen tank will require approximately two inches of insulation. The insulation will be a polyurethane foam with Freon, which should cryopump when the tanks are filled, thus providing excellent insulative properties. The polyurethane foam in each instance will receive a final sealing coat of nonfoam polyurethane after the initial foam is sanded smooth.

The insulation would be applied to the bulkhead as outlined in the previous paragraph. It will be applied to the exterior LOX and the liquid hydrogen cylindrical skin tank area with a tank assembly rotating on a turntable while in a vertical position. The tank would rotate at a peripheral speed of approximately one foot per second while four rack-mounted spray heads located 90 degrees apart travel vertically at a speed of apporximately 1/2 foot per second. This operation would be repeated, working from the top to the bottom until the entire skin section area has been covered to the required thickness.

The LOX tunnel would be foam insulated in approximately 40 foot sections in a similar manner as the cylindrical skin section except that it will be accomplished with one or two spray heads traveling laterally while the tunnel sections rotate horizontally about their centerlines.

6.5.2 Solution Approach

Study activities should be undertaken to determine if the above insulation application techniques are feasible and can be adapted to the MLLV components for the desired thickness.

PRECEDING PAGE BLANK NOT FILMED.

7.0 MANUFACTURING TECHNOLOGY IMPLICATIONS - INJECTION STAGE

7.1 TOROIDAL TANK FABRICATION AND ASSEMBLY

7.1.1 Problem Definition

The injection stage tankage will consist of two toroidal tanks per module. The tank skins will be fabricated from 2219-T87 aluminum alloy torus segments consisting of an upper and lower half. The torus segment halves will be bulge formed.

A disadvantage of this fabrication concept is that the weld seams will be located in the area of highest stress. A second disadvantage may be the nonavailability of aluminum sheet stock in sizes commensurate with the concept of 8 torus segments for the liquid hydrogen tank. For the MLLV, the outer circumference for the major diameter (57 feet) of the tank will be 179 feet and the circumference of the cross-section of this 8 foot diameter tank would be 25.2 feet. With the currently available aluminum sheet sizes, the tank would have to be made in approximately 15 or 16 torus segments entailing many more fabrication, assembly and welding operations.

An alternative method of producing the torus tanks would be to bulge form the torus skin segments in an inner/outer half configuration thus placing the weld seams in the area of least stress. The only major disadvantage as opposed to the former method would be the requirement for two bulge formed dies rather than for one. The problem of nonavailability of large sheets would be the same as for the previously stated method.

A third alternative method would be explosive forming of the torus segments from 2219-T87 aluminum cylindrical tubing. Tubing of the size required is now available from the industry and current studies have indicated that this fabrication method is feasible and would be the most desirable of the methods outlined. The torus segments would be welded together by an out-of-vacuum electron beam method. Shear webs to maintain the crosshead sections circularity and provide internal structural stiffness would be incorporated much the same manner as outlined previously. It would be possible to fabricate the cylindrical tubing, prior to the torus forming operations, by roll-forming aluminum sheet into the cylindrical form, then electron beam welding it.

The explosive form die would be made into two halves. The cylindrical tube with outside diameter somewhat less than the inner diameter of the closed explosive form die, would be placed in the open die, then one-half of the die would be hydraulically or mechanically closed against the stationery half. The tube would thus be forced to conform generally to the curvature of the die in the major circumferential direction. The tube would be flattened to some degree in the cross-section plane. An explosive charge would be placed within the tube and detonated thereby forcing the tube to assume the shape of the inner configuration of the die. The segments

7.1.1 (Continued)

could be probably fabricated in at least 30 degree segments. A number of developmental problems would be encountered and would have to be resolved. These would include but not necessarily be limited to: The determination of the initial metal condition or temper required; and ascertainment of the tube wall thickness, diameter, and length required to obtain the desired finished torus segment wall thickness, configuration and length.

7.1.2 Solution Approach

All three of the above approaches can be used to successfully fabricate the torus tankage for the injection stage. However, as the explosive forming technique appears to be the most desirable, a study to determine the effects of explosive forming on the material properties should be undertaken. Pre-forming diameters, length, thickness, etc, required to achieve the final diameters with the desired lengths after forming should be studied in small scale testing.

7.2 TOROIDAL TANK INSULATION

7.2.1 Problem Definition

The LOX and liquid hydrogen toroidal tanks will require insulation to prevent external icing during pre-launch operations and to prevent excessive boil-off during both earthbound and orbital conditions. Type and amount of insulation will be dictated by the mission requirements. Foamed insulation applied on both tanks should suffice for one or two orbits. If the injection stage is scheduled to complete more than three or four orbits, a blanket of approximately 3/4 of an inch to an inch thickness of super insulation applied to the liquid hydrogen tank and approximately one inch of polyurethane foam applied to the LOX tank will be necessary. If the injection stage is scheduled for a longer orbital mission, the thickness of insulation on each tank will be increased but not necessarily in direct proportion of the number of orbits.

A three-fourths inch thickness of super insulation would consist of approximately 60 alternating layers of aluminized mylar and nylon net sewn together to form a blanket. Blanket sections would be fitted onto the exterior of the completed liquid hydrogen tanks in sections and sewn to each other. Fasteners located in the corners of the blankets would be adhesively bonded to the exterior of the tank skin. The polyurethane foam would be applied to the LOX tank exterior. The configuration of the tank would be the principal factor affecting the method of application. The suggested method for applying the foam would be to use two or three spray heads mounted adjacent to each other with the LOX tank rotating, about its centerline, on a turntable in a position parallel to the floor and beneath the spray heads. After the tank completes one revolution, the spray heads will be moved to another point next to

7.2.1 (Continued)

the tank skin (around the cross-sectional circumference) and the spray foaming operation repeated until the entire exterior is covered with foam. The same operating sequence will be repeated for the non-foaming polyurethane finished coat.

7.2.2 Solution Approach

Each of the above methods discussed are satisfactory to meet the insulation requirements for the potential mission applications for the injection stage. However, tests must be undertaken to determine the insulation capability of these materials under the temperatures and environments created by liquid hydrogen and oxygen. In addition, manufacturing techniques to fabricate the insulation and to spray the foam insulation are required. These tests would include rate of application, cure times and temperatures, methods of handling, methods of processing, evenness of applied coats, etc.

PRECEDING PAGE BLANK NOT FILMED.

8.0 LAUNCH FACILITY IMPLICATIONS

8.1 EXHAUST GASES HANDLING AND THERMAL PROTECTION AT LAUNCH

8.1.1 Problem Definition

The large volume of solid rocket motor stage exhaust gases (lift-off thrust equals approximately 51,700,000 pounds) will present major problems during the initial launch sequence of the vehicle, i.e., (1) handling and disposal of the exhaust gases with deflectors and trenches, (2) protection of deflectors and trenches from the thermal environment, and (3) protection of the launch complex and associated GSE from the pressures and thermal environment. Pressure and thermal environment during lift-off must be defined such that the launch facility GSE items can be adequately designed to operate at this environment. Model tests must be conducted to verify the design of these items.

8.1.2 Solution Approach

Existing data should be correlated and evaluated. Three series of tests should then be conducted:

a. Tests to define the thermal environment.

b. Materials tests to define and test candidate materials for launch facility and GSE protection.

c. Model tests which simulate the vehicle lift-off accelerations and thermal and pressure environments using scale models of the MLLV configurations, facility and GSE designs.

8.2 LAUNCH ACOUSTICS

8.2.1 Problem Definition

The magnitude of the thrust at lift-off (approximately 51,700,000 pounds) of the largest vehicle in the MLLV family will create a severe acoustic environment. This environment will impact personnel and certain facilities within the launch area. Further, there is a potential hazard to personnel and buildings in inhabited areas adjacent to the launch complex. The actual expected acoustic environment must be better defined to fully evaluate this problem and to provide specific criteria for siting of the launch pads. The acoustics during launch are extremely directional depending on the orientation of the flame trench and specific earth and hardware deflectors, protuberances, etc. Because of this directionality, specific orientation of the launch pad will have a significant effect on the acoustic

8.2.1 (Continued)

hazard.

8.2.2 Solution Approach

The existing acoustic data must be correlated and analyzed. Additional data from actual firings should be acquired. Model tests should be conducted to determine the directionality factors associated with the flame trench, etc. This data must be combined into an acoustic profile which will define the acoustic environment from ignition through flight ascent to an altitude of approximately 1,000 feet. This profile can then be used for specific siting and orientation of the launch pad and for the establishment of safety procedures during launch. It may be necessary that these launch sites be located off-shore as a result of the acoustic criteria. Therefore, every effort should be undertaken to minimize the acoustic effects through site location and direction.

8.3 SITING CRITERIA - TNT EQUIVALENCIES

8.3.1 Problem Definition

The current NASA safety criteria rates the solid rocket motor propellant as 100 percent equivalent to TNT when the solid rocket motors are located adjacent to the fueled vehicle. This equivalency factor when combined with the 0.4 psi design overpressure for launch facilities creates a major problem (and/or waivers on safety) for siting of the facility items (specifically launch pads). Actual tests to date of similar propellants have indicated that this equivalent factor is grossly exaggerated for solid propellant rocket motors. Existing test data indicates a more reasonable factor is on the order of 10 percent TNT equivalency or less. (Reference 8.3.1-1). Similarly, the TNT equivalency factor of 60 percent for LOX/Hydrogen appears unreasonably large considering the degree of mixing of the propellant constituents required to cause detonation on the order indicated.

These equivalency factors should be reviewed with regard to current and future test data to allow for more realistic safety requirements.

8.3.2 Solution Approach

Existing data must be correlated and a test program defined which will more realistically determine the actual TNT equivalent of the separate propellant components and for combinations of these propellant components.

8.3.1-1 CPIA Publication No. 167, "Explosive Hazards of Composite Solid Propellants", Billings Brown, Institute of Defense Analyses, April 1968.

8.4 SRM STAGE HANDLING

8.4.1 Problem Definition

Unlike the core vehicle stages, the solid motor stage will contain its propellant when delivered to the launch site. Its weight of over 3,000,000 pounds will present a unique handling and operational sequencing problem.

8.4.2 Solution Approach

There have been several studies accomplished to date on the handling of large solid rocket motors. All of these study programs involved a specific vehicle configuration and did not have SRM handling and launch operations as a primary objective. It will be necessary to conduct a study to determine the optimum method of handling the SRM stage at the launch site. Methods of supporting the SRM stage prior to launch and at lift-off should be an integral part of any study.

8.5 IMPACT OF ON-BOARD TEST AND CHECKOUT SYSTEM ON LAUNCH FACILITY GSE

8.5.1 Problem Definition

Studies have been conducted to evaluate the advantages of an on-board test and checkout system on launch vehicles. These studies have indicated that an on-board test and checkout system is a highly desirable feature. These studies, however, have not analyzed the impact of the on-board test and checkout onto the launch facility. Areas to be investigated include (1) methods of adapting the on-board test and checkout system to the existing launch facilities, (2) reduction in the existing GSE equipment requirements, (3) reduction in on-pad time as a result of the on-board test and checkout system, and (4) the impact on the on-board test and checkout system on the launch facility and its cost.

8.5.2 Solution Approach

A specific analysis of the effect of on-board test and checkout of the vehicle onto the launch facility GSE requirements, launch facility timelines, overall schedules and on launch costs should be conducted.

PRECEDING PAGE BLANK NOT FILMED

9.0 MAIN STAGE RECOVERY AND REUSE

9.1 RECOVERY AND RE-USE

9.1.1 Problem Definition

Analysis of the cost distribution show that approximately 50 percent of the operational cost of the single-stage-to-orbit vehicles is attributable to the cost of the hardware which will be expended. Preliminary estimates indicate that operational program costs could be reduced by 30 to 50 percent if this hardware could be recovered and reused.

9.1.2 Solution Approach

Preliminary design studies of the AMLLV vehicle family (in the previous study) indicated that a recoverable and reusable single-stage-to-orbit vehicle, using the AMLLV design concepts, was feasible. Such a system would use a ballistic re-entry mode with aerodynamic decelerators and would land on water. As the stage would be called down on command from orbit, landing could be made in the near vicinity of the launch facility to minimize recovery costs.

Additional studies should investigate methods for recovery of the main stage from orbit and evaluate the cost effectiveness of potential recovery modes. While operational savings are indicated, the R&D costs for implementation of the recovery mode must be assessed relative to overall program costs.

END

DATE

FILMED

NOV 21 1969