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PROPULSION STUDY FOR SMALL PROBES

Prepared For

National Aeronautics and Space Administration
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NASA Headquarters, Washington, D.C.

Prepared by

J.P. Taylor

AEROJET-GENERAL CORPORATION

SPACE DIVISION

9200 E. Flair Drive

El Monte, California

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Industry

Propulsion system data were generated by the following:

Hybrid System - Mr. J. Hammer and Mr. P. LaForce, under the direction of Mr. R. Taylor of the United Technology Center, Sunnyvale, Calif.

Liquid System - Mr. E. Latin of Liquid Rocket Operations, Aerojet-General Corp. - Propulsion Division, Sacramento, California

Solid System - Mr. L. Briggs, under the direction of Mr. P. Browne of Solid Rocket Operations, Aerojet-General Corp. - Propulsion Division, Sacramento, California

Fluid-Controlled Solid System - Mr. N. Cohen of the Lockheed Propulsion Company, Redlands, California.

Monopropellant System - Mr. J. J. Galbreath of Rocket Research Corporation, Seattle, Washington.

FOREWORD

The report is submitted as complete fulfillment of Contract NAS 7-550.

Prepared by:

J. P. Taylor, Technical Specialist
Advanced Technology Center

Approved by:

R. B. Jenkins, Program Manager
Flight Vehicles Section

ABSTRACT

The study reported herein is concerned with an investigation into the propulsion requirements of a small rocket motor suitable for future D-region and meteorological sounding applications. The main thrust of the study is the establishment of a propulsion system concept which has a good potential for low unit cost in high production quantities.

Three motor concepts are finally developed. These motors are: a hybrid unit, a semi-self-pressurizing liquid unit, and a slow burning solid unit. All offer good potential for the stated mission and cost objectives. Additionally, all merit special attention from the standpoint of propulsion technology development.

In addition, a fluid power control unit is included as Appendix A. This concept is included as a separate section simply to indicate that it represents a Phase I effort begun after all other Phase I work had been completed, the Mid-term Presentation made, and Phase II was well underway. Sufficient time was not, therefore, available for design iteration. The concept is rather unique and deemed worthy of notice, and therefore, is included in this report.

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1.0 INTRODUCTION

This study in advanced propulsion techniques as applied to small rocket motors represents a follow-on effort to work performed under NASA Contract NAS 7-413," Application of Advanced Solid and Hybrid Motors to Sounding Rockets," that identified mission requirements for which existing propulsion units did not provide acceptable performance. The area selected for this study represented the definition of a propulsion unit suitable for low cost, high reliability soundings in the near earth region of 50-140 km (D-region physics and synoptic meteorological applications).

Existing propulsion units capable of functioning in this region are either excessively large or expensive or both. The attention of this study was, therefore, directed to the application of new propulsion concepts to a motor(s) which could be efficient cost wise and performance wise when applied to the chosen mission.

It should be noted that while the emphasis of this study was propulsion technology it was necessary to evaluate the mission requirements and various systems aspects of the problem. To be effective, the propulsion system has to be considered as a component in a system designed to meet specific mission requirements. Too often in times past, the mission has evolved around the available propulsion, thus resulting in inefficiency and unnecessary expense. As these systems aspects are considered to be of interest, they are included as Section 3.4 of this report.

It was recognized at the initiation of this study that conventional propulsion techniques did not provide an adequate solution to the accomplishment of the desired mission. As a result, emphasis was placed on the investigation of new propulsion concepts. The work reported in Ref. 3.0.1 indicated that the hybrid propulsion concept might be readily adaptable to the present application and, thus, the initial orientation of the study was toward the hybrid motor. As the study progressed, investigations into a number of propulsion concepts (self-pressurizing liquid, long-burning solid, monopropellant and fluid-controlled solid) were undertaken with the result that

several novel conepts, not only for the present mission but various other missions, were identified.

It is felt the results of this study can be used as a starting point for the development of the propulsion system for the small probe or for demonstration tests of the propulsion concepts identified herein from which other applications may be found.

2.0 CONCLUSIONS AND RECOMMENDATIONS

The following conclusions and recommendations are presented as they relate to the propulsion technology investigated. As this study was, in part, applied technology, the relationship of the propulsion system to the mission application assumed colors the conclusions and recommendations.

This orientation should not be construed to mean that if a particular motor concept is not adaptable to the current system, it is without merit. On the contrary, the advanced propulsion concepts herein described should be considered for broader applications than were possible within the scope of this study.

2.1 CONCLUSIONS

a. The following unique propulsion concepts were established as feasible and carried through conceptual design:

- Hybrid
- Self-pressurizing liquid (VāPak)
- Monopropellant
- Fluid-controlled solid
- Long-burning solid, cartridge-loaded
- Long-burning solid, HTPB

While all were not determined to be cost effective for the current application, each was considered to have technical merit and worthy of further development.

b. Contrary to an assumption made at the beginning of this study, total impulse content is not necessarily a reliable guide to the cost of a propulsion unit in the category of interest herein. For example, the VāPak liquid motor, and to a lesser extent to the hybrid cost is in the motor case and valve assemblies, not in the propellant. Thus, a motor yielding half the impulse would only cost fractionally less. Likewise, in the solid motors, fabrication techniques (i.e., cartridge versus case-bonded) affect the cost as much as the propellant.

c. This study demonstrates the existence of a wide range of design capability and imagination in the field of propulsion technology which can be brought to bear on specific requirements. The current application of solid liquid and hybrid technology to the small-probe concept confirms previous indications that if current propulsion technology is brought to bear on a specific objective, a more cost-effective, yet more sophisticated design can be obtained. Even those Phase I designs not carried to the Phase II level (i.e., the monopropellant and the RSVP design) reflect a high level of competence and originality. Those interested in the applications aspect should note that the Phase II designs more than complete with existing equipment in that the projected costs of the advanced designs approach those of the cheapest systems now available (Dart Vehicles) yet exceed the capabilities of the more expensive of the current systems. This was projected to be the case by a previous study which is effectively summarized in a propulsion system economics study by Stanford Research Institute for NASA/OART (see Section 3.3). In addition, the new designs are also more sophisticated, and are potentially more adaptable to other applications.

d. Any of the Phase II motor design concepts are capable of providing the required performance for the specific missions envisioned (i.e., D-region physics and synoptic meteorology) in a cost-effective manner. Of the systems studied, one does not appear to have a clear-cut advantage over the others. The accompanying chart is a generalized comparison of the propulsion systems studied as applied to the basic study ground rules. The determination of that concept best suited to this mission will probably not depend entirely upon cost or performance capabilities, but will require consideration of requirements/restrictions/compatibility with a final overall system the definition of which was beyond the orientation and scope of this study. In this regard see Section 3.4 and the recommendations below.

e. The future propulsion requirements for the small probe mission were defined based on requirements established through a user survey. These requirements were, if anything, somewhat stringent; nevertheless the final motor configuration should be a result of integration with the total

system. Reconciliation of the candidate vehicles with a final system design should be easily accomplished, and would be expected to have little effect on the estimated cost.

2.2 RECOMMENDATIONS

a. Propulsion System Application.--The propulsion technology outlined herein should be examined for application to other missions, both military and space oriented. The propulsion concepts investigated (hybrid, VāPak, monopropellant and RSVP and the HTPB propellant identified with the solid) are not in general use in any operational motors. However, they appear to offer significant advantages for replacement of existing systems or application to new systems.

b. Propulsion System Demonstration.--Consideration should be given to funding propulsion demonstration programs based on the designs and concepts generated herein. The final test of the feasibility of a propulsion system must come as a hardware test. In view of the anticipated production quantitites and system useful life, the expenditure of relatively small amounts of money for flight weight propulsion demonstration tests would be justified. An additional bonus of the demonstration of new propulsion technology with broad potential applications would be realized.

The following recommendations relate to the application of the propulsion system to the meteorological probe. While not in the prime orientation of this study they were identified as a result of this work and are deemed to merit attention:

c. System Integration Study.--A specific follow-on effort is recommended in the area of system operation and integration study. A final meteorological system configuration or specification based on the advanced propulsion system of this study is not now on hand. Although general operational considerations are discussed in Section 3.4, additional effort, in terms of overall trade-off evaluations, are required to establish the meteorological system requirements and thus the final propulsion system details.

d. Recovery System Study.--The advent of a country and/or world wide meteorological rocket network will bring the falling mass hazard problem sharply into focus. It is demonstrated in Section 3.4 that real estate costs for rocket ranges – even ones for gun probe accuracy – will probably be prohibitive. The concept of a simple parawing fly-back system (Section 3.4) appears to be worth additional effort and should be subject of a study in the near future. While this system has the advantage of greatly reducing real estate costs, it has a potential for some economic recovery through the reuse of payloads and possibly other vehicle components. The propulsion concept of this study has the payload volume and weight carrying capacity to make such a recovery concept feasible.

SMALL PROBE PROPULSION CAPABILITY COMPARISON

Item	MOTOR					
	Hybrid	Liquid	Long-Burning Solid	HTPB Solid	Monopro-pellant	RSVP
Payload Capacity (Volume)	+	+	+	+	+	+
Altitude Capacity (10 lb to 140 km) (40 lb to 100 km)	+	+	+	+	+	+
Acceleration Environment (30 g max.)	+	+	+	-	+	+
Estimated Wind Sensitivity-Launch	+	-	+	+	+	+
Launch Weight - Portability (180 lb max)	+	+	+	+	-	+
Fuel Toxicity	-	-	+	+	+	-
Storage and Operating Environment ($-65^{\circ}F$ - $+135^{\circ}F$)	+	+	+	+	+	+
Explosion Hazard	+	-	+	+	+	+
Estimated Cost - Small Quantities (Rank)	2	1	4	3	Not Given	5
Estimated Cost - Large Quantities (Rank)	1	3	4	2	5	6

3.0 RESULTS AND DISCUSSION

This study originated as a follow on to the study reported in Reference 3.0.1 and had as its prime objective the application of advanced propulsion concepts to the definition of a motor for use in a small D-region or meteorological probe vehicle. The results of this study are discussed in detail in the subsequent sections. This study was performed in two phases which, for continuity, are reported separately.

The Phase I study involves the establishment of the basic ground rules and investigation of the various propulsion concepts from a technical and a cost-effective point of view. Finally, the Phase I results, conclusions and recommendations as reported in the mid-term presentation are given.

Phase II emphasis was placed on conceptual designs of three basic propulsion techniques, the hybrid, long burning solid and self pressurizing liquid. A detailed analysis of each of the propulsion systems including the economics of development and production are given. These propulsion systems are compared against the ground rules established under Phase I as they relate to the anticipated mission.

Also included is an evaluation of the systems aspects of the rocket motor in the context with the D-region and meteorological application. It was necessary to investigate these areas in order to establish realistic propulsion systems requirements. These data are included under the section on System Consideration for the interest of those concerned with propulsion system applications. To complete the correlation, analysis of this projected development programs for the system as well as for the propulsion units is included.

Two advanced propulsion techniques were identified at a late stage in the study. The first of these relates to the use of a hydroxy-terminated polybutadiene (HTPB) propellants to the long-burning solid. Preliminary data on this application as applied to the current propulsion concept are included in the section on the long-burning solid.

The second propulsion technique identified was that of the fluid-controlled solid propulsion technique. Data on this design were not received

until the final report was in the publication phases. Therefore, it is included as an appendix and only limited comparisons are made between this system and the other candidate propulsion systems.

Appendix B presents the computer printout for the final trajectories showing performance comparison of each of the propulsion units considered in this study. In making these trajectories all units were assumed to have the same basic drag characteristics and inert ancillary hardware weights.

3.1 PHASE I - PRELIMINARY PROPULSION SYSTEMS EVALUATIONS

3.1.1 PRELIMINARY GROUND RULES

Basically the study was limited to considering rocket propulsion devices, and, although the initial emphasis was to be on the potentialities of hybrid techniques, competing systems (particularly conventional liquid and solid techniques) were also to be considered, at least from the standpoint of cost-effectiveness. In the general category of rocket propulsion units there are a large number of devices; therefore, calling upon general knowledge of systems aspects into play, certain preliminary ground rules were established. Briefly, the initial ground rules were as follows:

- Single stage, dual thrusting propulsion.
- Complete vehicle weight, in meteorological configuration, not to exceed 200 lb.
- Gross payload weight of 20 pounds to get a net useful 10 lb for the meteorological configuration.
- Fineness ratio (i.e., length/diameter) on the order of 15, and not over 20.
- Acceleration level (axial) not more than 25 g's.

These ground rules were, in some cases, somewhat arbitrary but had as their basis experience. They were, therefore, considered as somewhat flexible, rather than as concrete. Throughout the program the overriding consideration was system cost-effectiveness as well as it could be projected at that time.

The single stage, dual-thrusting requirement was arrived at by a process of elimination. Logical rocket propulsion systems which would be candidates for this mission are; boosted systems having either an inert or active second stage, or single stage systems having either single or dual-thrust capability.

The dual-stage, dual-thrusting concept was eliminated on the grounds of inherent cost of two propulsive stages, coupled with the added weight of interstage and reduced overall reliability of the more complex unit.

The dual-stage, single-thrusting concept is presently used quite extensively in the so-called boosted-dart systems. In this concept, the booster unit delivers its impulse in a short duration, high-level thrust. The second stage is an inert dart configuration, having a relatively high mass density and low drag configuration which minimizes aerodynamic drag losses. The objective is to minimize the system's wind sensitivity. However, this approach imposes very high environmental conditions (acceleration, and aerodynamic structural and heating loads) on the system. In addition, the available payload volume is restrictive, and does not appear to offer good growth potential. The gun-boosted systems which have received some attention recently might be considered as belonging to this family; however, the remarks made concerning the boosted-dart system also apply except that environmental conditions are orders of magnitude more severe. Both systems are limited to largely inert type payloads, although some electronics have been "hardened" against the severe environment. Indications to date are, however, that there are difficulties being experienced. Based on these considerations, these concepts were eliminated from the study.

A major consideration in sounding rocket usage is trajectory dispersion due to winds. As pointed out above, relief from this condition is the reason behind the boosted dart systems. The larger sounding systems having propulsive upper stages also use a booster to quickly increase off the launcher velocity, thus reducing wind sensitivity. For meteorological application, wind sensitivity will be a major consideration. Thus, the single stage, dual thrusting concept was selected as the concept most likely to fulfill the system requirements within the prescribed constraints.

With regard to the weight limitation, both meteorological and scientific operations may be expected to occur, using this system, in remote regions where handling conditions/equipment will be less than ideal. In fact, there are strong indications that portability would be a highly desirable feature of any new system. See the discussion concerning questionnaires used in this section. The prescribed weight limit, 200 lb, is about all a 2 or 3 man expedition could cope with manually.

The gross payload weight was simply an estimate based on 10 pounds of active (useful) or net payload, with 10 pounds of ancillary structure and hardware.

A restriction on the motor fineness ratio has as its basis, experience. That is, past experience indicates that as a typical motor fineness ratio (the ratio of the motor length to its nominal diameter) goes over 18 to 20, that either the potential for encountering elastic problems greatly increase, or the structural weight becomes excessive.

The restriction on acceleration level also has as its basis experience. That is, scientific instruments, as would be used in the scientific mission, are usually somewhat acceleration sensitive. Also, it is anticipated that advanced meteorological payloads will contain additional instrumentation, as yet unspecified, which may also be acceleration sensitive.

As pointed out above, these preliminary ground rules were based on both anticipated system requirements/constraints, and past experience.

To assess the reasonableness or desirability of the ground rules, and to be as responsive to actual user requirements as possible, a questionnaire was distributed to typical users such as; NASA field centers; military missile/weather/launch/research centers; Governmental agencies such as ESSA, Weather Bureau; institutions such as NCAR, Southwest Research Institute, etc.; those individuals listed as members of appropriate committees from National Academy of Science, Woods Hole Meeting of 1965 (Reference 3.1.1.1). The questionnaire contained a request for comments as follows:

1) Are the payload weights reasonable for the type of missions your organization is considering? If not, please make recommendations.

2) Indicate payload volume requirements and limiting linear dimensions (e.g., minimum acceptable package diameter or length as dictated by a particular instrument or group or instruments likely to be in the payload makeup for missions being considered).

3) Indicate likely environmental limitations imposed on the system by payload instruments.

4) Indicate special auxiliary onboard systems requirements. Payload separation system.

5) Indicate likelihood of requiring special features in the payload housing structure such as access doors, externally mounted antennae, etc.

6) Indicate trajectory accuracy requirements and probable requirement for "quick" launch capability.

7) Comment on desirable geographical launch locations.

8) Comment on likely requirements for range support if this is to be required (i.e., established range versus complete portability).

9) Please record any additional comments which are felt to be pertinent to the successful development of an integrated system to accomplish the indicated missions.

EXPERIMENTER/USER _____

ORGANIZATION _____

EXPERIMENTAL INTERESTS _____

Twelve (12) replies were made, in some cases by individuals, but generally as a representative or collective response of an organization. Those responding covered fairly well both the D-region scientific interest and the meteorological interest, and included those who would be anticipated to have a vested interest in these fields.

Typical replies to the questions were:

1) Payload weights and attitudes are reasonable target areas (one dissent).

2) Payload volume requirements indicated simply volume of 300 cu. in. or more and minimum linear dimension of 4.5 inches.

3) Environmental limitations likely are mostly from scientific requirement of non-magnetic material and minimum motor outgassing.

4) Onboard systems are likely, particularly payload separation and descent systems.

5) Special features, most for scientific purposes, were access to the payload and external antennae.

6) Trajectory accuracy requirements of simple good repeatability to $\pm$ 1 kilometer in altitude.

7) Geographic location desires for both scientific and meteorological use include remoted site (i.e., wilderness or limited access) operation.

8) The range support requirements ranged from major range data/track capability, to remote site operation using standard GMD and/or phototheodolite equipment.

9) Further comments could be summarized as indicating an obvious desire for a low cost, highly reliable vehicle of the general category being considered (i.e., small probe).

With these results in mind, Phase I then moved toward a preliminary evaluation of candidate propulsion concepts.

3.1.2 PRELIMINARY ESTIMATES OF PROPULSION SYSTEM REQUIREMENTS

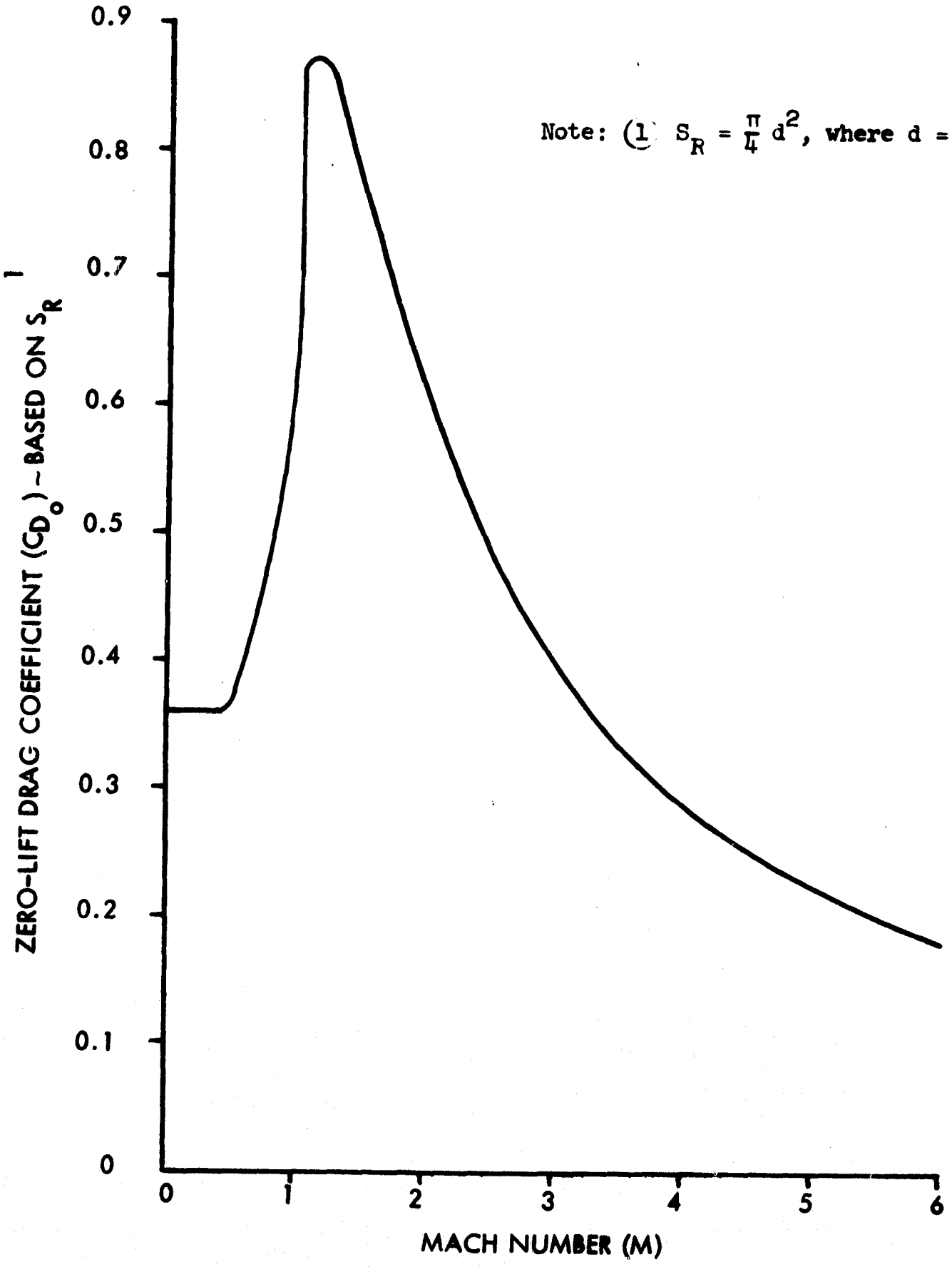
It was expected that, all else being equal, a propulsion unit's cost would be strongly related to its total impulse content. Since propulsion system costs were expected to play such a strong part in the overall system economies, the initial propulsion system requirements were based on minimizing total impulse requirements, hopefully resulting in minimum or near-minimum unit costs. The results were then to be refined and, reconsidered with overall system requirements and constraints. Also, with regard to system constraints and economics, exotic designs were eliminated from consideration as having low economic potential and/or potentially unsafe/toxic operational characteristics. Either the economic or operational defects would make such a motor completely unacceptable for the application of interest here.

As stated earlier, the main target for this study was to place 10 pounds of meteorological equipment at an altitude of 120 kilometers. During this very first phase the target was raised to 140 kilometers to be conservative in that, if the payload is to be separated and activated by the time it reaches 120 kilometers, on the downward leg, the overshoot would allow ample time plus a margin for operational variations.

As a starting point, it was assumed that the vehicle would be a very low aerodynamic drag configuration having the characteristics illustrated in Figure 3.1.2.1. As for the motor, typical values of the parameters used initial:

- Delivered specific impulse (I_{sp}) $\approx$ 210 sec - 220 sec @ sea level.
- Mass fraction (m_f) $\approx$ 0.75 - 0.80
- Burn rate = as low as 0.005 in/sec
- Limiting diameter $\approx$ 4.5 inches
- Average unit density $\approx$ 0.05 lb/in^3.

PHASE I DRAG REPRESENTATION



Note: (1) $S_R = \frac{\pi}{4} d^2$, where d = motor diameter

These values were obtained from UTC as being typical of a hybrid unit. Throttling ratios on the order of 10:1 were considered as easily attainable.

Evaluation of impulse delivery mode was made using a two-degree-of-freedom trajectory simulation programmed for the digital computer. Initial calculations made it immediately clear that several of the mission objectives and constraints would have a significant affect on the motor sizing. That is, for the given mission objectives in terms of payload and attitude, and motor fineness ratio constraint, coupled with probable motor packaging density and mass fraction, placed the required motor diameter very near 6.0 inches. A subsequent survey of metal tubing suppliers by the motor manufacturer indicated that 6.0 inch diameter tubing of the variety likely to be used in the motor casing was standard, and the most readily available, and therefore potentially the most economical. On this basis, the 6.0 inch diameter was adapted for the rest of the study. Additionally, should subsequent work indicate that requirements could be lowered (e.g., lower altitude or advanced design, lower weight payload), it felt that downward scaling, particularly costwise, would be somewhat easier than the reverse.

Selection of the impulse delivery mode was based on a number of trajectory calculations in which the thrust curve shape was varied parametrically. The shaping was quite simply two consecutive, constant level steps to simulate a boost and a sustain phase. The time increment of the boost phase was varied, as was the total burning time. This is illustrated in the following diagram (Figure 3.1.2.2).

While allowing the shaping variations, the total area under the curve, i.e., the total impulse was held constant. Also, the specific impulse was an average value over the powered phase of the trajectory. The results are illustrated in Figure 3.1.2.3.

There are several items of interest here. First, the lower dashed curve represents one limit in the parametric variation - boost and sustain thrust levels are equal; i.e., a single thrust level motor. The other curves then represent the advantage or penalty for going to the dual thrust mode.

GENERALIZED THRUST CURVE
DUAL THRUST LEVEL

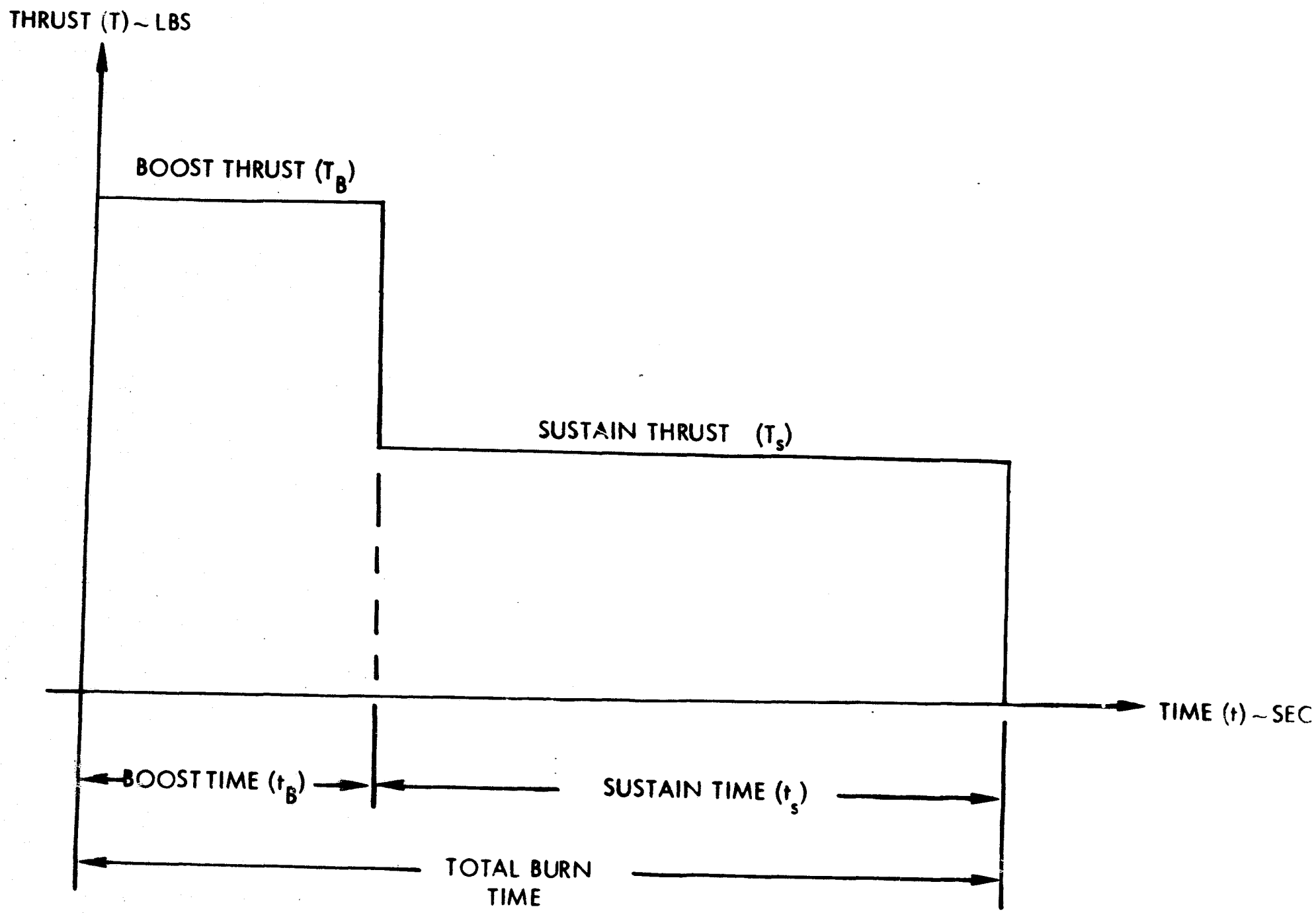


Figure 3.1.2.2

VEHICLE APOGEE PERFORMANCE

Note: Initial Vehicle Weight = 170 lb
Final Vehicle Weight = 54 lb
Motor Diameter (d) = 6.0 inches
Total Delivered Impulse = 25,500 lb-sec
Effective Delivered Specific Impulse = 210.0 sec.
Effective Launcher Length = 15 ft.

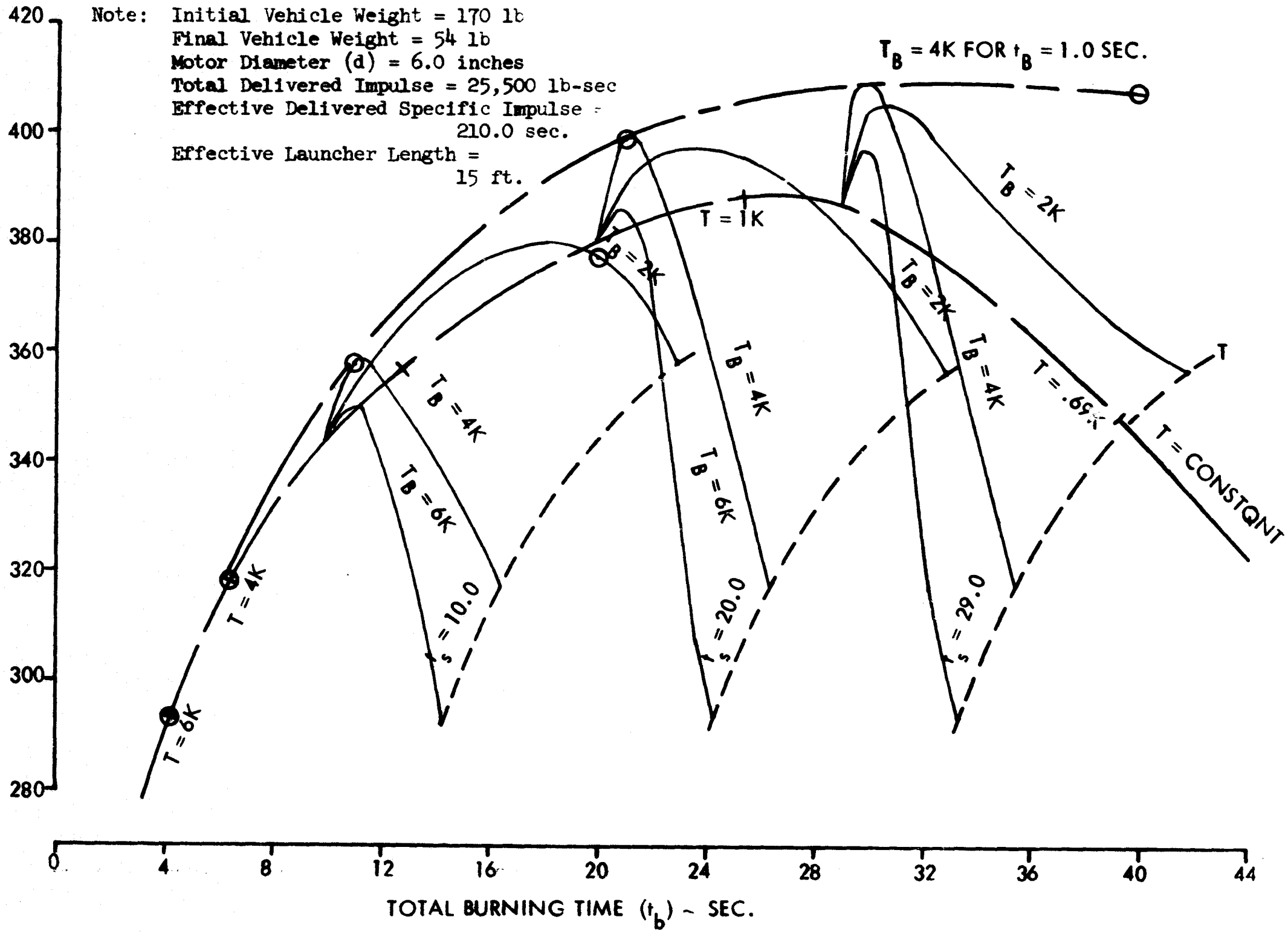


Figure 3.1.2.3

The families of curves were constructed from the actual trajectory computation points, which are indicated on the figure. The curves should be interpreted in the following manner. Consider the family which originates from the t_b = 29.0 sec point (i.e., total burning time = 29.0 sec) on the constant thrust curve, and terminates on the dashed line labeled T-T and noted as t_s = 29.0 sec. The note, t_s = 29.0, on the terminator of this family means that for the family the sustaining phase has a duration of 29.0 sec (see generalized curve Figure 3.1.2.2). Now note that each individual curve of the family is labeled T_B = nK, which means that the boost thrust is a constant at n-thousand pounds. Follow now, the curve labeled T_B = 4K (i.e., boost thrust level = 4,000 lb). At the t_b = 29.0 sec point, where the curves of this family originate, the sustainer burning time, (t_s), is identical to the total burning time (t_b). The booster operating time is zero, therefore, even at T_B = 4,000 lb, there is no boost phase; i.e., the single thrust level. Thus the entire total impulse available is delivered at a single thrust level over 29.0 sec. Move to the point along the T_B = 4K curve to t_b = 30.0 sec. For t_b = 30.0 sec and t_s = 29.0 sec, the booster operating time is 1.0 sec. The total impulse delivered during this phase is 4000 lb-sec, thus leaving 21,500 lb-sec to be delivered over the 29.0 seconds of the sustain-phase. Now, follow the T_B = 4K curve to the family terminator. At this point, the entire total impulse of 25,500 lb-sec is delivered during the boost phase. Thus, even though the sustainer time for the family is 29.0 sec, the thrust level is zero. If the logic is followed through, it is realized that the terminator of each family is only that portion of the constant thrust curve that corresponds to the boost thrust level, shifted in time by the sustainer operating time.

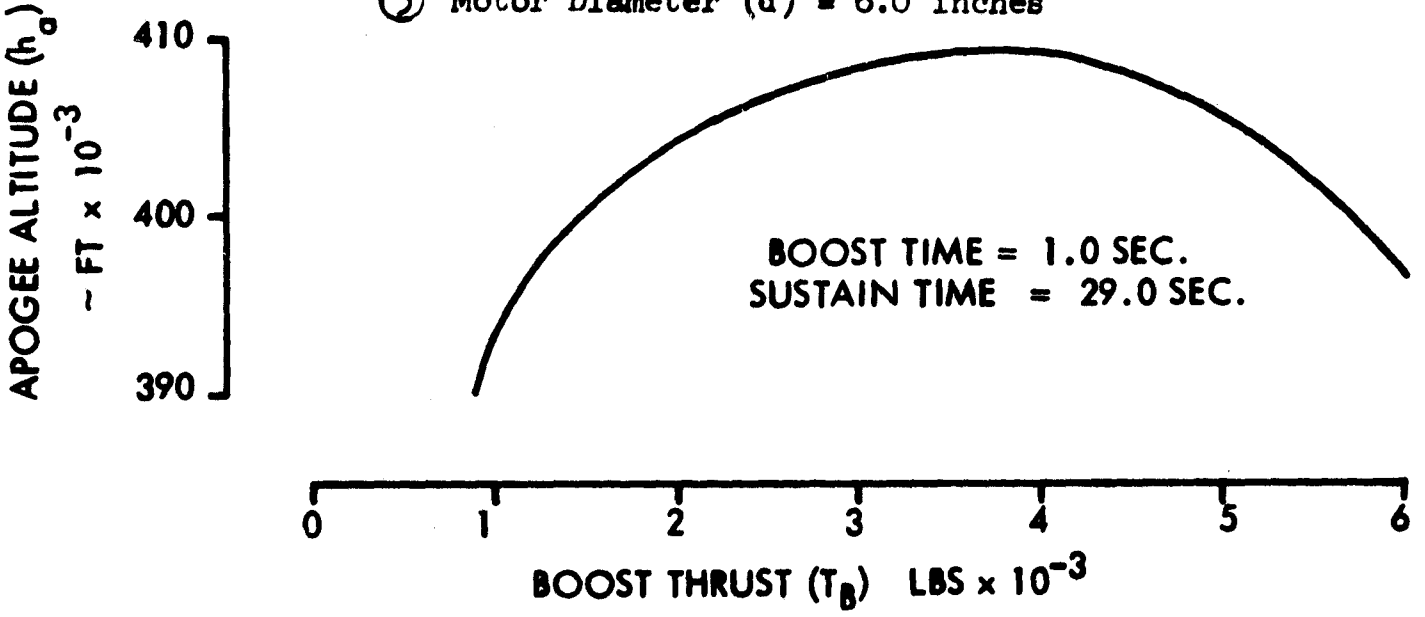
Thus optimization of the usage of available impulse is indicated by the approximate boundary line labeled T_B = 4K for t_B = 1.0 sec. That the best burning time for a single thrust level small probe motor is around 20 - 30 seconds is not new or surprising. Nor is the fact that a high initial or boost level does have some benefit performance-wise (see References 3.1.2.1 through 3.1.2.3).

The characteristics shown in Figure 3.1.2.3, however, should not be taken as universally applicable. The limits of scaleability were not determined since this was not the purpose of the study. Also, the results are of the form of generalized delivered thrust curve, at an assumed (but typical) specific impulse level, and, therefore, must be reconciled with actual motor parameters, particularly with regard to dual thrusting (e.g., extremely high chamber pressure for high boost thrust, in order to get good specific impulse at the lower sustain thrust with the same nozzle geometry). Fortunately, it appears that an extreme in boost-to-sustain thrust ratios is not required. Looking at Figure 3.1.2.3, it appears that there is a marginal gain in having a dual thrust ratio for burning times much less than 20 seconds, and no big gain unless the time is larger than approximately 30 seconds. However, other operation/system requirement must be kept in mind. For the particular case in point, reducing the vehicle wind sensitivity is of primary concern, so dual thrusting is deemed mandatory in any case; therefore, the fact that there is some performance benefit is simply a bonus. The question of burn time also requires consideration of operational requirements/constraints. As the burn time is shortened the flight environment becomes more severe. On the other hand, the longer the burn time, the further down range apogee attitude is attained, hence the more dispersion prone the vehicle becomes. At this point, the area around 30 sec burn time appeared the most attractive. Performance characteristics in this area are shown in more detail in Figure 3.2.1.4. Maximum performance occurs in the area of 4000 lb of boost thrust for 1.0 second, with the remainder of the total impulse being delivered in 29.0 seconds, at the lower sustain thrust level, resulting in a total burning time of 30 seconds.

The data for evaluating impulse delivery mode, as discussed above, were generated for a typical vehicle having fixed characteristics, particularly weight, size, and propulsion representation; i.e.,

EFFECTS OF THRUST LEVEL AND BURN TIME
ON APOGEE PERFORMANCE

Note: (1) Initial Weight (W_o) = 170.0 lb
(2) Total Delivered Impulse (I_T) = 25,500 lb-sec
(3) Effective Delivered Specific Impulse (I_{sp_o}) = 210.0 sec
(4) Effective Launcher Length = 15.0 ft.
(5) Motor Diameter (d) = 6.0 inches



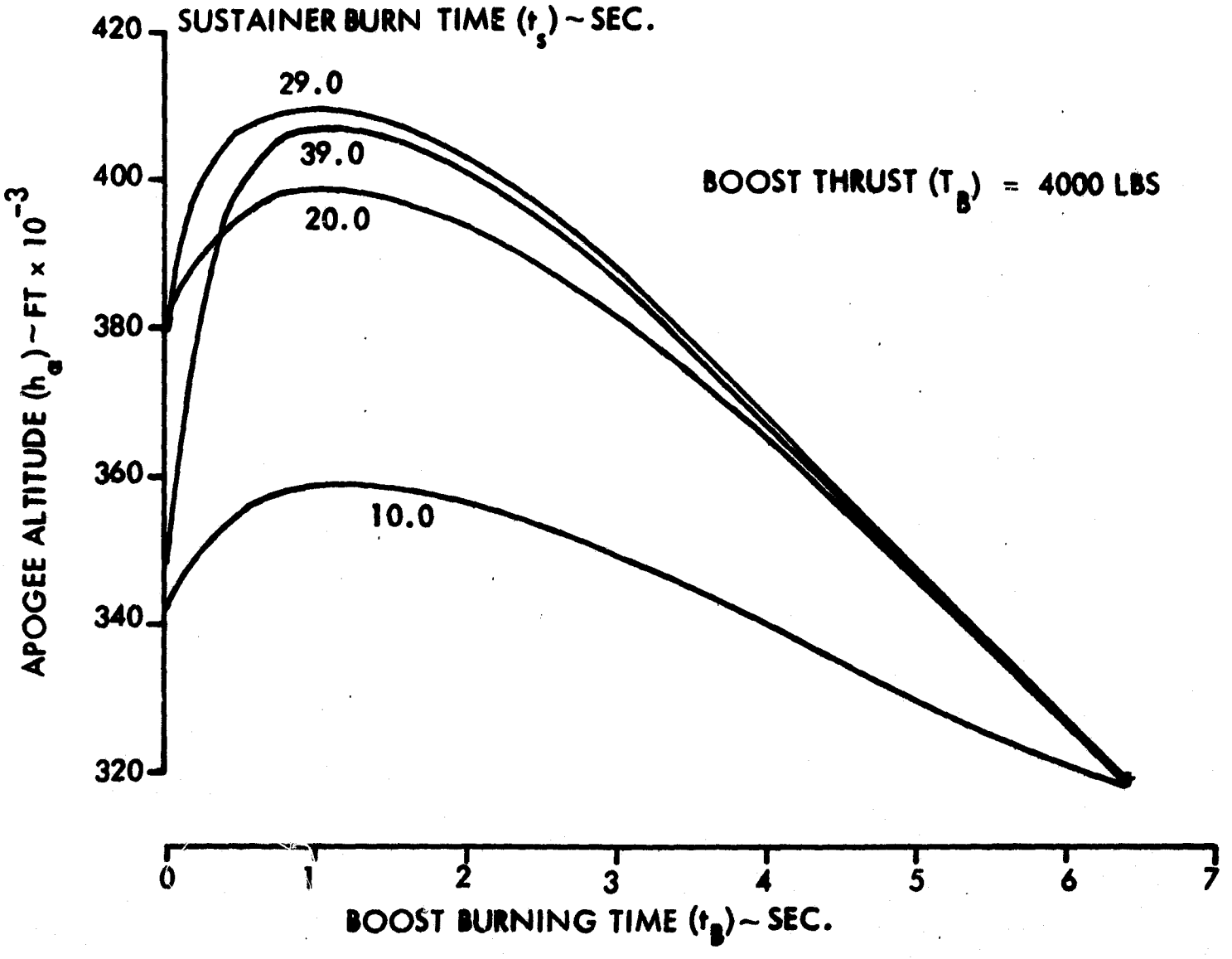


Figure 3.1.2.4

W_o = 170.0 lb

$I_{T_{del}}$ = 25,500 lb-sec

$I_{sp_{del}}$ = 220.0 sec

W_f = 54.0 lb

Using the 4000 lb, 1.0 sec boost phase, and 29.0 seconds of sustain time, the variation of apogee attitude with total delivered impulse and final vehicle weight was determined for a 6.0 inch diameter vehicle, as shown in Figure 3.1.2.5. The results were used to guide the motor designers.

3.1.3 PHASE I SUMMARY, CONCLUSIONS, AND RECOMMENDATIONS

It should again be noted that this study was aimed initially at developing a hybrid system. This resulted from the fact that the previous study (Reference 3.0.1), which indicated the potential need for the vehicle system considered herein, also indicated that the hybrid concept had excellent possibilities from low unit cost at high rates of production. It was noted, however, that the previous study was of a general nature and covered the whole spectrum of sounding rockets. Since this study was concerned with a relatively narrow portion of that spectrum and would be more detailed in nature, the sponsor and program manager agreed that, in this first phase, other concepts potentially applicable to this mission should at least be assessed from the standpoint of potential cost-effectiveness.

Based on the foregoing discussion, the impulse delivery mode investigation, and the system objectives/constraints, motor conceptual design were generated by various propulsion system designer/manufacturers. These candidate designs were examined from the standpoint of conforming to the system requirements, and the resultant vehicle performance checked using the trajectory computer program. After several iterations, the candidate systems were then subjected to a cost analysis by their designers. Costs considered were RDT and E costs, and production costs at various production rates.

APOGEE PERFORMANCE

EFFECTS OF TOTAL DELIVERED IMPULSE AND FINAL VEHICLE WEIGHT

Note: (1) Average Delivered Specific Impulse = 210.0 sec
(2) Effective Launcher Length = 15 ft.
(3) Motor Diameter (d) = 6.0 inches

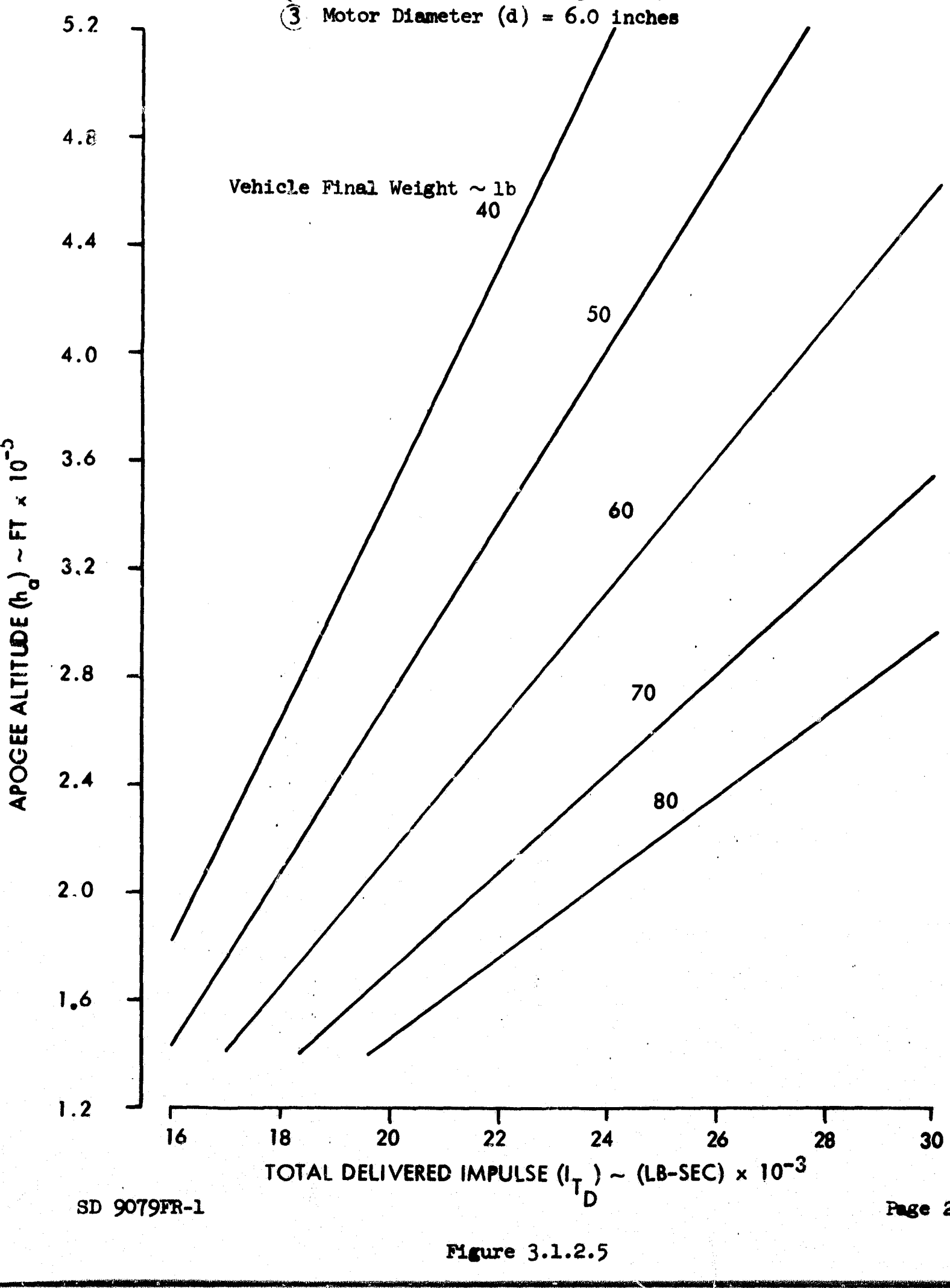


Figure 3.1.2.5

The conceptual designs generated in the first phase effort are described in the following paragraphs. Their physical characteristics and cost data are summarized at the end of the following discussion.

It is again noted that at the outset of this study, the hybrid system was to receive the major consideration. It was, therefore, the only design effort specifically funded in support of this study during the first phase. All other Phase I designs were solicited and provided on a voluntary basis.

Hybrid System - designed specifically for this study by United Technology Corporation, Sunnyvale, California

This system is based on the hybrid concept; i.e., the oxidizer is in liquid form, while the fuel is in solid form. Dual thrust capability was attained, in this design, by adding a conventional solid propellant charge to the basic hybrid propulsion unit. This charge is located in the chamber between the aft end of the hybrid solid fuel grain and the entrance to the exhaust nozzle. It produces a boost thrust of 4000 pounds for 1.0 second. The boost charge also serves two other functions. First, it preheats the hybrid grain which is a prerequisite to hypergolic ignition of the hybrid propellants. This is a safety feature in that, if hybrid oxidizer is introduced into the combustion chamber, either by leakage or by accidental pressurization of the oxidizer tank, there will be no ignition. Second, the volume occupied by the boost charge becomes a mixing chamber after booster burnout; thereby increasing combustion efficiency during hybrid operation.

The hybrid motor consisted of a liquid oxidizer, nitric acid (IRFNA) contained in a forward tank, and a solid fuel, aluminized polybutadiene (Al + PBAN) located in the chamber. Shortly after booster ignition a solid cartridge gas generator is actuated (either by a circuit delay device or by a launcher motion switch to insure booster operation), to pressurize the oxidizer tank. As chamber pressure drops below oxidizer tank pressure, during boost tail-off, a diaphragm ruptures allowing oxidizer to flow from the tank to a small plenum, and hence to injector nozzles for injection into the solid

fuel grain parts. The design operating time of the sustain phase was 29.0 seconds. The thrust level was approximately 900 pounds under ambient condition corresponding to approximately 25,000 feet altitude; i.e., average over the trajectory. Trajectory calculations indicated that the performance level was adequate.

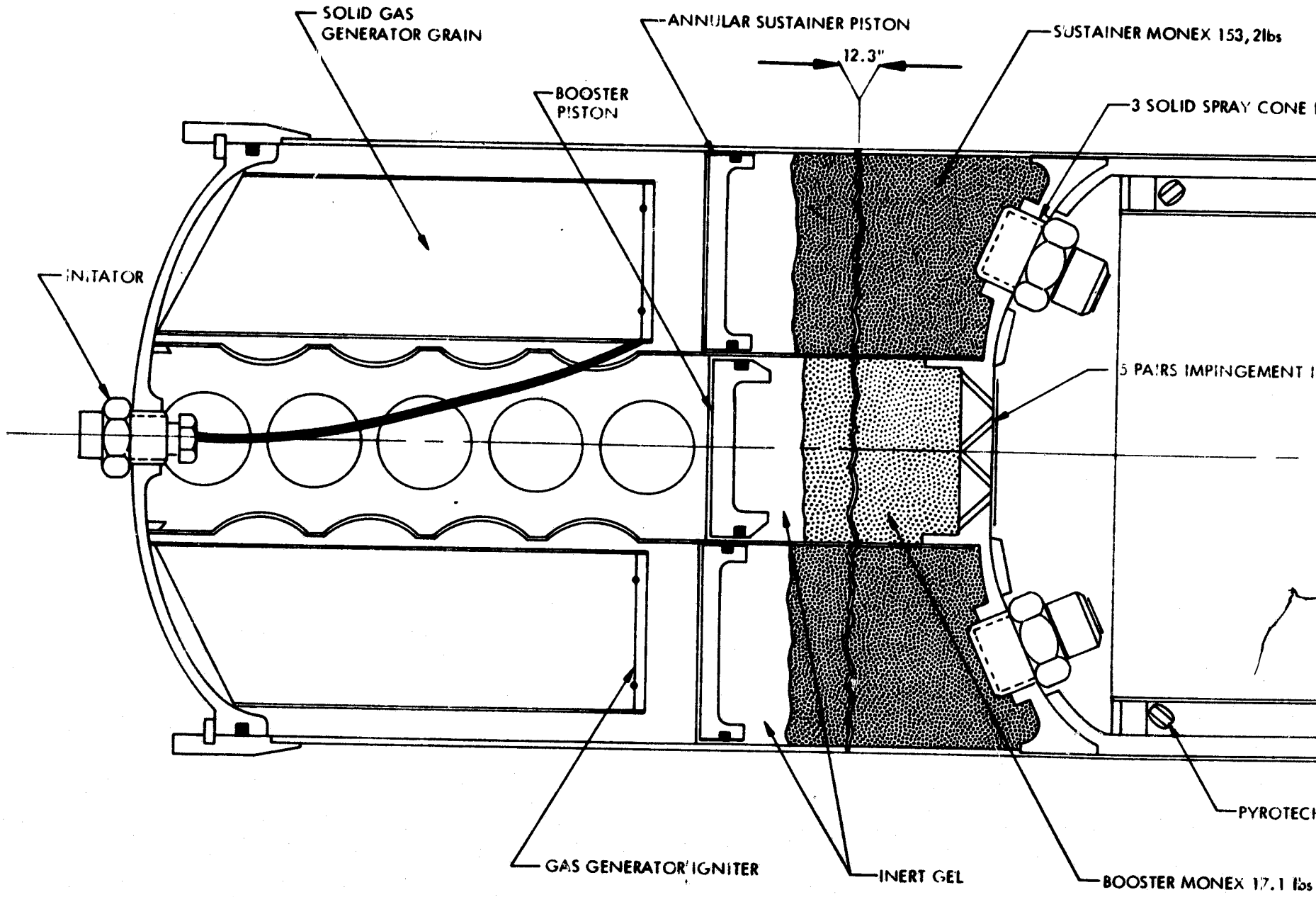
The design was considered to be very attractive from an operational standpoint, and initial estimates indicated excellent economic potential.

Monopropellant System - designed by Rocket Research Corporation, Seattle, Washington

This system was based on a proprietary monopropellant liquid propellant, Monex. The concept is illustrated in Figure 3.1.3.1. A 4000 pound boost phase thrust of 1.0 second duration is provided by the center segment of a variable rate piston pressurizing system, powered by a solid cartridge gas generator. This is followed by a 29.0 second sustain phase. The sustain thrust level is 1033 pounds at conditions corresponding to $\approx$5000 feet altitude, again a representative value over the trajectory. As can be noted from Table 3.1.3.1, the unit developed exceeds several preliminary constraints such as weight and fineness ratio. Also, from the preliminary cost data the economic potential did not appear to be too good.

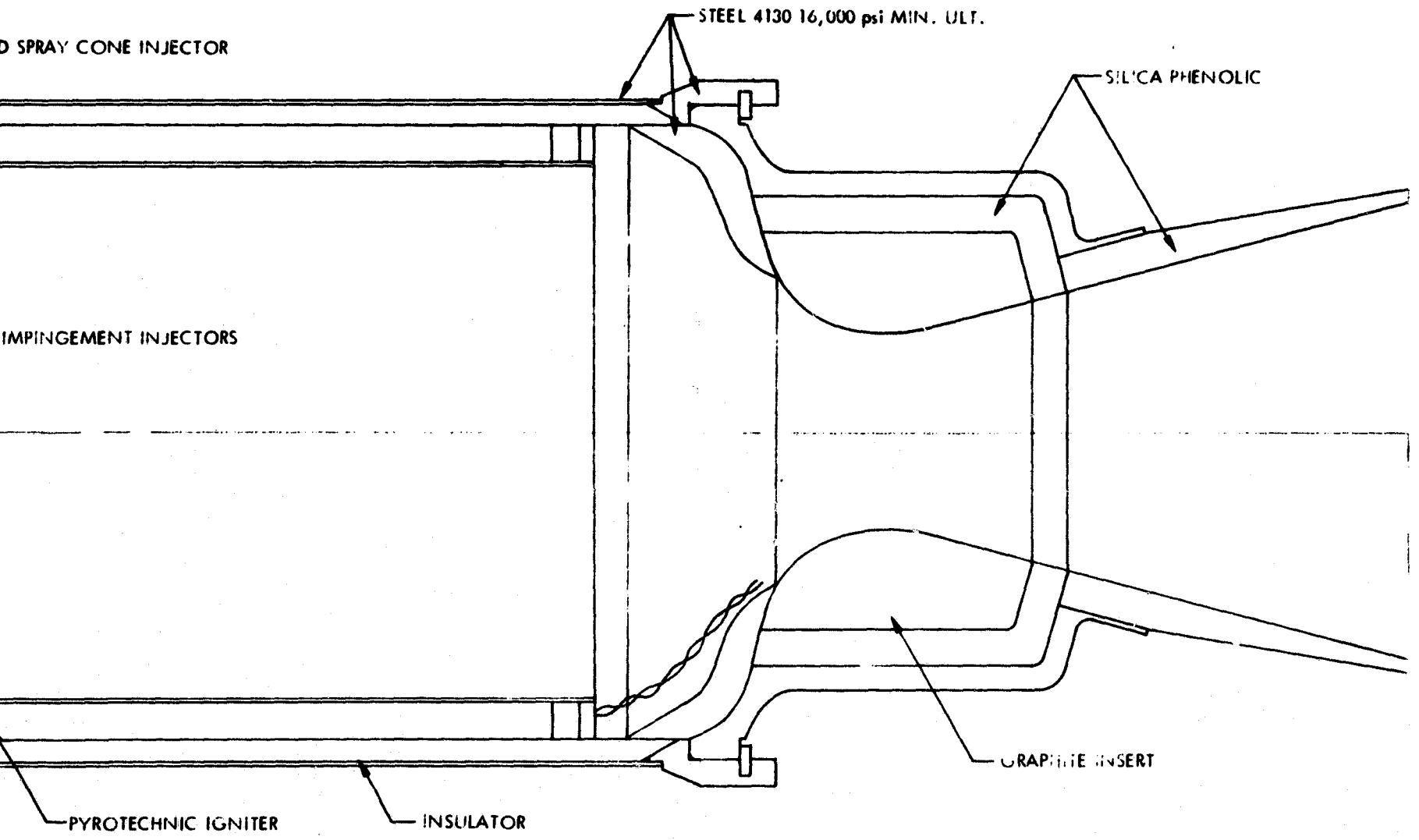
VaPak Liquid System - designed by Liquid Rocket Operations, Propulsion Division, Aerojet-General Corp., Sacramento, California

This system is based on a rather unusual concept which is particularly applicable to the size of vehicle considered here. It is a liquid motor, but lacks the usual costly items characteristic of liquid systems; i.e., pressurization system, regulation and complex injection system. The propellants used in this motor are preheated by a certain temperature level in the tanks by an integral heater element. The resultant propellant vapors then become the pressurization system. As the propellants are expelled from the tanks, the vapors and propellants remaining in the tank maintain an equilibrium state, thus there is little change in pressure level from start to finish. In fact,



FOLDOUT FRAME 1

EX 153, 2lbs



MONEX 17.1 lbs

Figure 3.1.3.1

MONEX SOUNDING ROCKET

FOLDOUT FRAME 2

the vapors themselves are consumed. As far as the injector geometry is concerned, the propellants are, by the nature of the manner of pressurization, saturated liquids. Experiments indicate that saturated solutions do not need to be broken down mechanically, rather they vaporize readily upon expansion from a nozzle. Based on this fact the injection system is a single impinging triplet face of very simple design.

As in the hybrid unit, the VaPak gets its boost thrust from a conventional solid propellant charge which is cartridge loaded into the combustion chamber. This is not a completely new design, and in fact, the VaPak concept is covered by a U.S. Government patent held by Aerojet-General Corporation. Some previous work on a smaller vehicle of less performance capability, has been accomplished including some testing. Also, some attempts at much larger scale application have been made.

Whereas, at the beginning, it was anticipated that liquid systems complexities would put them completely out of the picture costwise, the data submitted with this design indicate good potential for a low unit cost at high production rates. According to the designers this is attributed directly to the VaPak concept.

Long Duration Solid System - designed by Solid Rocket Operations, Propulsion Division, Aerojet-General Corporation, Sacramento, California

At the start of this study, it was generally thought that a solid system of long duration (e.g., t_b > 10-15 seconds) would have to be an end burner for the size motor of interest here (i.e., motor diameter $\approx$ 5-6 inches). This was considered to translate into a relatively expensive manufacturing process. However, Aerojet has a proprietary additive that allows relatively low burning at moderate chamber pressures with good specific impulse capabilities. A propellant formulation using this additive will allow a large duration radial burning unit, a capability long sought in the sounding rocket field. This capability is particularly attractive from the standpoint of the dual thrusting requirement. That is, a high boost thrust level can be

obtained from a high burn-rate boost charge, at a reasonably high pressure level (e.g., 1500-2000 psi). The lower sustain phase thrust level, at the lower pressure level, does not, in this case, translate into poor specific impulse. Further, the approach is more than conceptual. It is now operational in several weapons systems of somewhat larger diameters.

This concept appeared to hold a very large potential for application to the sounding rocket field in general, particularly the larger, more specialized category. Although the initial cost estimates made for the special applications considered in this study indicate higher costs for the long duration solid motor than either the hybrid or VaPak liquid concepts, it should be recommended for further consideration in other sounding rocket applications.

UTC Comparison Solid Systems - designed by the Solid Propulsion Div., of United Technology, Sunnyvale, Calif.

Purely for cost comparison purposes, the UTC solid motor design group was asked by the UTC hybrid group to generate a solid unit of comparable capability. Actually two units were designed; one with the objective being absolute minimum of costs, and the other with an objective of approximating the physical and dimensional requirements at low cost.

The system requirements are presented in the following table with a comparison of the predicted performance for the two systems:

Comparison of Required and Predicted Performance at Sea Level, 70°F

Parameter	Specified Value	Aluminum Case	Maraging Steel Case
Boost Thrust, lbf	4000	3940	3940
Average Sustain Thrust, lbf	725	720	720
Boost Duration, sec	1.0	1.0	1.0
Sustain Duration, sec	29.0	29.2	29.2
Max. Overall L/D	18.0	6.7	7.1
Target System Weight, lbm	150	165.1	150.0
Motor O.D., in.	Min	8.25	7.80

TABLE 3.1.3.1

SOUNDING ROCKET PERFORMANCE SUMMARY

$70^{\circ}F$ Sea Level

$\epsilon = 7.0$

Parameter	Value
Total impulse, lb-sec	25,138
Total duration, sec	31.3
Boost	
Impulse, lb-sec	3940
Web duration, sec	1.0
Average thrust, lbf	3940
Maximum thrust, lbf	4100
Average pressure, psia	2524
Maximum pressure, psia	2632
Sustain	
Impulse, lb-sec	21,036
Web duration, sec	29.2
Average thrust, lbf	720
Maximum thrust, lbf	733
Average pressure, psia	435
Maximum pressure, psia	511
MEOP, psia	2990*
Nozzle throat diameter, in.	1.12
Propellant weight, lbm	111.8
Standard Delivered specific impulse, sec	241
Propellant burn rate at 1000 psia, in/sec	1.65
Propellant pressure exponent, in.	0.27

*Based on $160^{\circ}F$

SOUNDING ROCKET PROPULSION
THRUST VS TIME HISTORY
70°F Sea Level
ε = 7.0

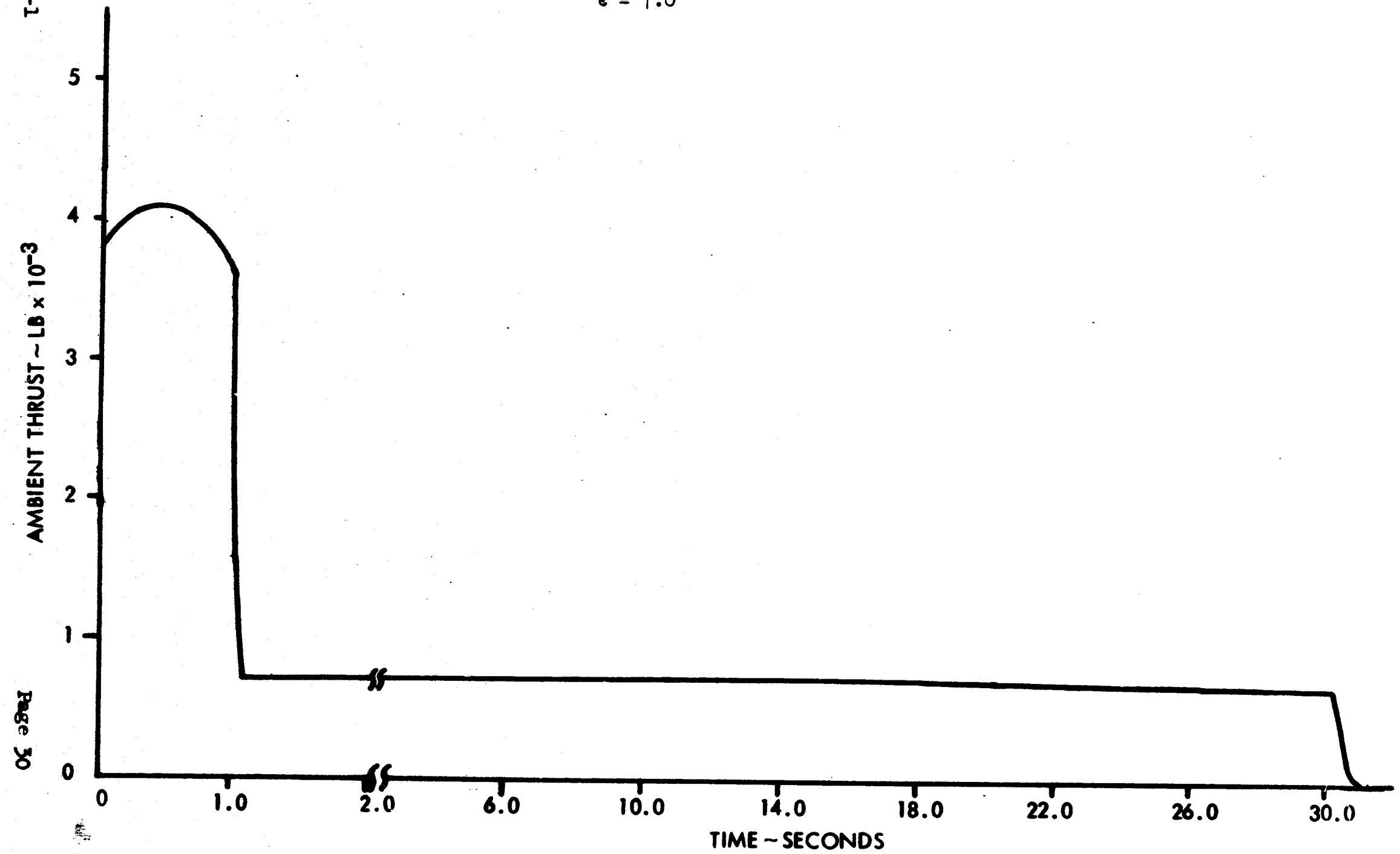


Figure 3.1.3.2

SOUNDING ROCKET PROPULSION
NOZZLE STAGNATION PRESSURE VS TIME
70°F

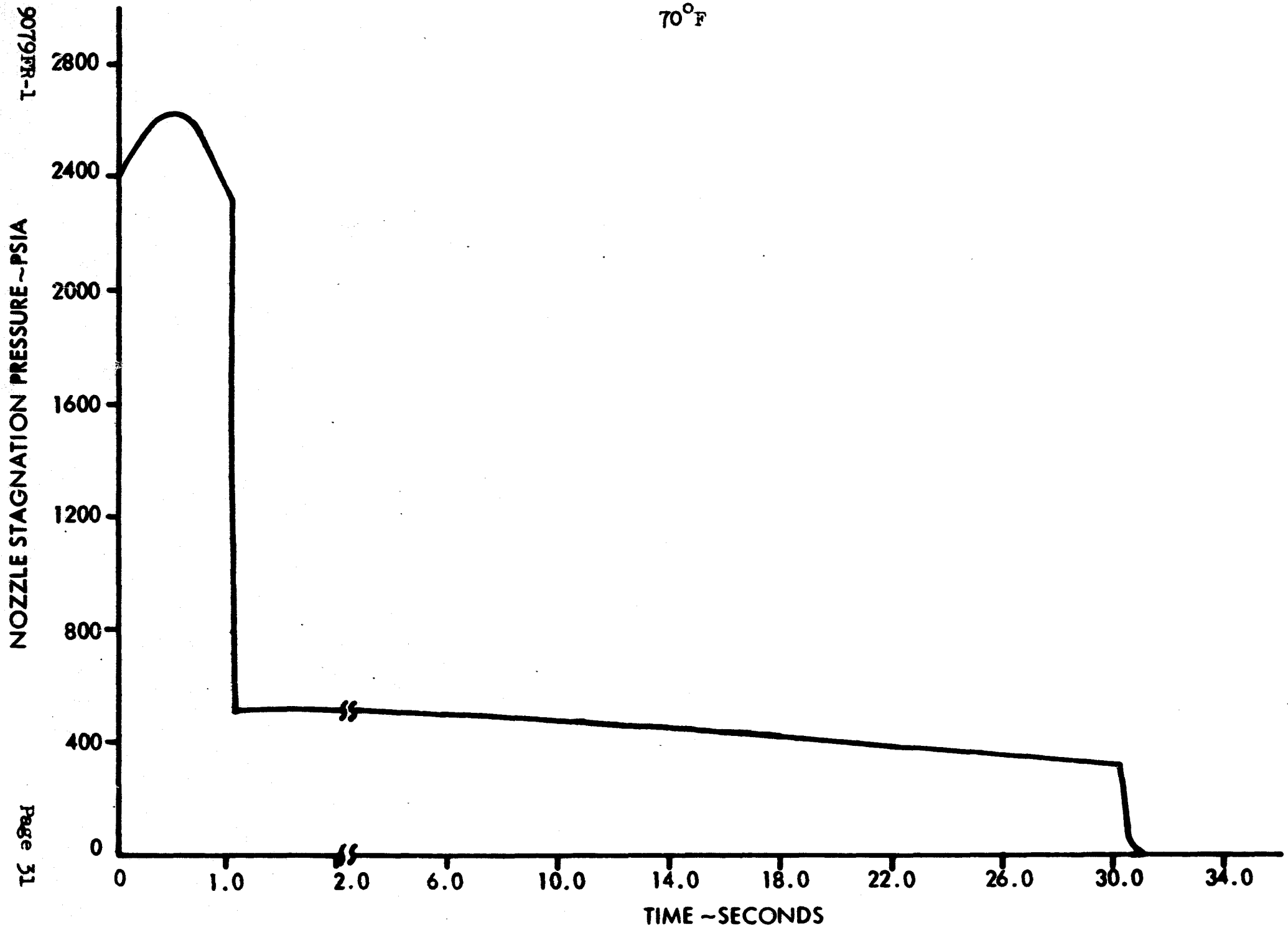


Figure 3.1.3.3

TABLE 3.1.3.2

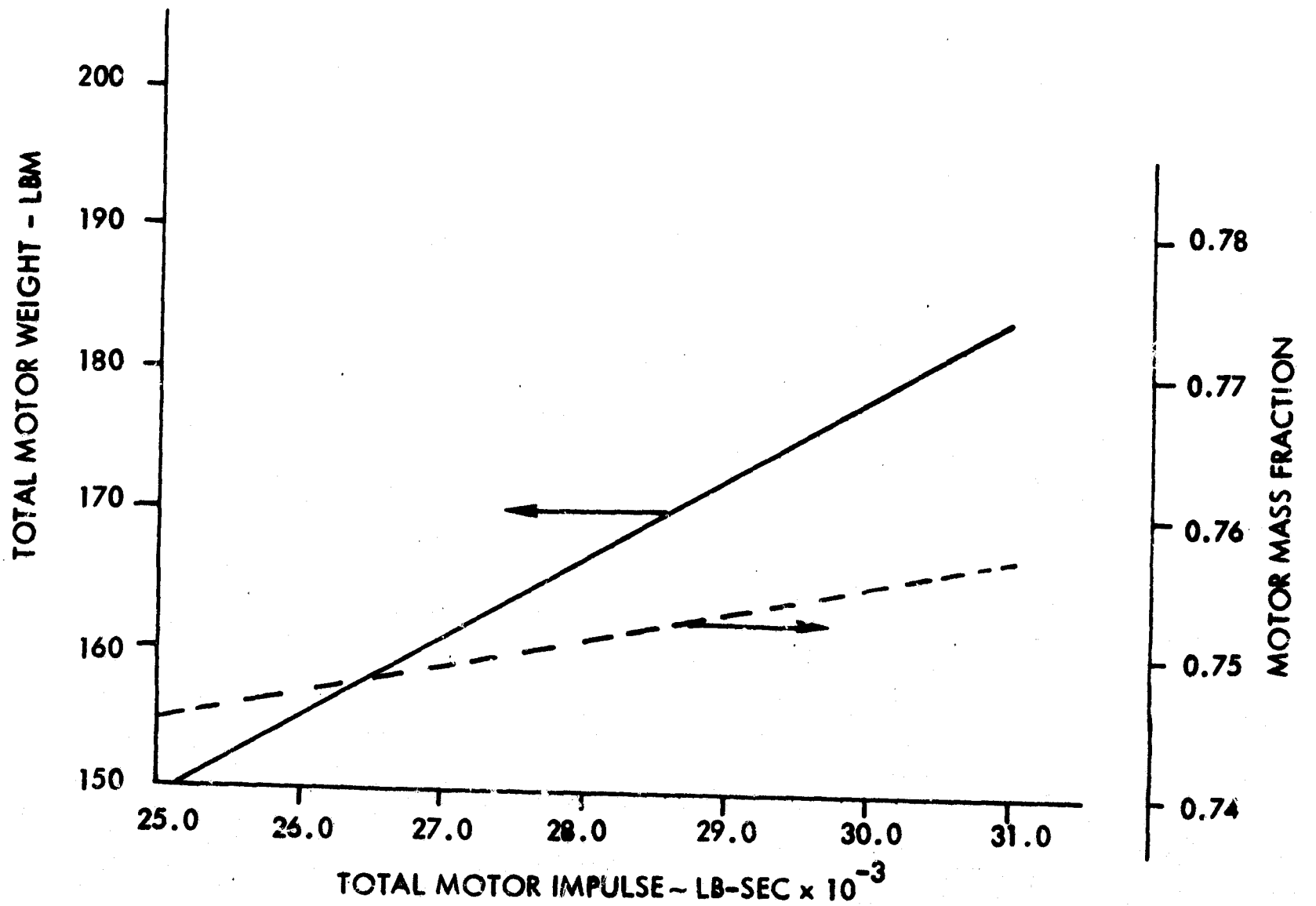
MOTOR WEIGHT TABULATION

Item		Aluminum Case	Steel Case
Forward Closure		3.7	1.1
Case		35.1	22.6
Insulation and Liner		9.9	9.9
Glass Phenolic	1.8		
Aluminum	2.4		
Igniter		0.4	0.4
Total Inert		53.3	38.2
Propellant		111.8	111.8
Total Motor		165.1	150.0
Mass Fraction		0.68	0.75

The performance of both systems equal or exceed the requirements listed previously. The system weight of the aluminum case exceeds the target weight by 15.1 lbm whereas the steel design equals the target value of 160 lbm. The performance as summarized in Table 3.1.3.1 is applicable to both systems. Figures 3.1.3.2 and 3.1.3.3 present the thrust and nozzle stagnation pressure versus time for both systems. The motor weights are tabulated in Table 3.1.3.2. The sustain impulse can be increased by lengthening the sustain grain (duration increases) which results in a weight penalty. Figure 3.1.3.4 presents the weight penalty and resulting mass fraction as a function of total motor impulse for the steel design. As shown the mass fraction improves since the effect of exit nozzle, forward closure, etc., becomes less.

The propellant proposed is in 82% solids CT-3 binder system with 16% aluminum. This propellant has a burning rate capability of 1.65 in/sec at 1000 psia. The standard delivered I_{sp} for small motors is 241 sec. This propellant formulation has been cast and tested in subscale motors in the Research Department of UTC. Further documentation work is required for a production system. The grain design proposed is a simultaneously burning dual grain consisting of an end-burning sustainer with a cylindrical perforation boost grain. The sustain duration (29 sec) dictated an end burning grain since the burn rate requirements for an internal perforated configuration in a small diameter motor would be considerably below the present day propellant capability. The end burning grain then dictated the motor diameter when using the maximum available UTC burn rate of 1.65 in/sec at 1000 psia. The minimum sustain grain diameter is 7.03 inches and increases due to the tapering of the insulation as the web decreases. The sustain grain is 41.3 inches long to obtain the required impulse and duration. The grain is case-bonded which presents stress problems in an end burner. To alleviate this, an inhibited stress relief port is used. The configuration of this port was based on a preliminary grain stress analysis for the -65°F cooldown plus pressurization. The boost grain is a simple cylindrical tube grain with a b/a of 2.5. The grain is 5.56 inches long. The forward face of the grain is inhibited to help neutralize boost operation.

EFFECT ON MOTOR WEIGHT AND MASS FRACTION FOR INCREASED TOTAL IMPULSE DUE TO LENGTHENING THE SUSTAIN GRAIN



Two design approaches were used for the inert hardware. Initially, low cost appeared to be the main objective. A design was made using an aluminum case with minimum of machining. The fore and aft closures attach to the case with Ortman keys. The case wall is thick enough for the key slot as the MEOP of 2990 psia dictates a nominal wall thickness of a 0.28 inch when using 2014-T6 aluminum with an ultimate strength level of 60,000 psi. The case outer diameter is 8.25 inches, the case barrel is 50.1 inches long. The fore head is modified flat plate to reduce production costs.

The case is insulated with 0.30 inch insulation in the boost portion. The sustainer insulation is tapered from 0.30 to 0.050 inch at the forward head. A liner thickness of 0.030 is used to bond the grain to the insulation.

The exit nozzle has a cast aluminum housing which fastens to the case with a full diameter Ortman key. Silica phenolic is used for insulation in the convergent section and exit liner. The throat insert is ATJ graphite. The expansion ratio is limited at 7.0 as this is near optimum for sea level due to the large effect of sustain operation that is at low operating pressure. If a larger expansion ratio is required for altitude no problems are encountered other than a small weight penalty.

A smaller size, lower weight unit was designed using a steel case. This design is identical to the one previously discussed except the motor case is fabricated from 300K psi maraging steel. The case wall is decreased to 0.055 inch nominal which gives an outside case diameter of 7.80 inches. The case ends are upset to thicken the material so that heat treating is not required after welding. The forward closure is a formed 2:1 ellipse and welded directly to the case. The aft end of the case is either upset to the required thickness for the Ortman key groove or a thicker ring is welded on. The remaining design is identical to that previously given. The exit nozzle casting will be cadmium plated at the closure mating joint to eliminate corrosion due to dissimilar metals.

Fluid Power Control System - designed by Lockheed Propulsion Company, Redlands, California

Subsequent to the completion of the Phase I effort, Mid-term Presentation, and Phase II start-up, Lockheed Propulsion Company indicated a desire to have included a design based on their fluid-controlled solid propellant rocket motor concept. Accordingly, they were given the basic ground rules and requirements; however, due to the state to which the program had advanced by that time, and the unfunded time that LPC could allot to this effort, there was no iteration or other adjustments possible with the LPC design, as there was with the others. Since this places the LPC design in a status equivalent to that of early Phase I (i.e., without the benefits of feedback) it is included separately as Appendix A to this report.

The physical characteristics of the Phase I preliminary designs are summarized in Table 3.1.3.3. Note that the UTC solid designs A and B, while having order of magnitude impulse content, and are within specified weight and fineness ratio limitations, they have relatively large diameters that would probably prevent them from meeting the mission. They are included primarily to provide comparative solid system cost data. Note also that the weight and fineness ratio of the Rocket Research "Mon A" system both fall outside the ground rules established for this study.

Motor cost estimates, based on the Phase I preliminary designs, are given in the upper portion of Table 3.1.3.4. Included are both RDT and E cost estimates, as well as unit costs for various production rates. It is of some interest to note the relatively large variance in both categories; however, it should again be emphasized that only the UTC hybrid design effort was funded. All other data were ROM estimates based on voluntary design efforts. Shown in the lower portion of Figure 3.1.3.4 are estimates of RDT & E and ancillary hardware required to transform a design motor to a flight vehicle. The two sets of cost data were added to obtain the complete vehicle cost estimates as given in Figure 3.1.3.5.

Given in Table 3.1.3.5 are cost estimates based on existing systems which potentially could be either boosted or ballasted to meet the specified

Table 3.1.3.3

PHYSICAL CHARACTERISTICS OF CANDIDATE MOTORS

Characteristic \ Unit	UTC Solid A	UTC Solid B	UTC Hybrid	AGC/LRO VaPac	AGC/SRO Solid	RR Mon A	Dimensions
Length	55.3	55.4	120.3	120.0	94.5	145.0	in.
Diameter	8.25	7.0	6.0	6.0	6.0	6.0	in.
Initial Weight	165.1	150.0	173.6	128.0	156.9	228.0	lbs.
Mass Fraction	0.68	0.75	0.761	0.78	0.79	0.75	-
Total Impulse	25,138*	25,138*	30,100	26,500*	30,000	35,000	lb-sec.
Specific Impulse	241*	241*	248/236	250/266*	255/240	235/202	sec.
Boost Thrust	3,940	3,940	4,000	2,000	4,000	4,000	lbs.
Boost Time	1.0	1.0	1.0	1.0	1.0	1.0	sec.
Total Burning Time	30.2	30.2	30.0	49.0	28.0	31.0	sec.

* Sea Level; all others are average between sea-level and approximately 70,000 feet.

Note: Solid A and Solid B above were generated for cost comparison purposes only. Even though they are within weight and length limitations, and have ROM impulse content, it is doubtful they could meet the mission due to diameter.

Table 3.1.3.4

COST DATA FOR CANDIDATE MOTORS

NEW UNIT

ITEM \ Unit		UTC[1] Solid A[2]	UTC Solid B[3]	UTC Hybrid[4]	AGC/LRO[5] VaPak[6]	AGC/SRO[7] Solid	RR[8] Mon A[9]
RDT&E	$	312,000	376,000	354,000	593,000	675,000	900,000
Unit[10] @ 100/yr	$	2,800	4,260	1,760	-	-	-
Unit @ 1000/yr	$	2,240	3,360	917	986	2,285	-
Unit @ 2000/yr	$	-	-	-	838	-	-
Unit @ 5000/yr	$	2,020	2,910	756	737	1,775	1,360
Unit @ 10,000/yr	$	-	-	-	695	1,596	@5-10K/yr

1. United Technology Center.
2. Minimized Cost } Although both units package specified range of I_T and are within weight and
3. Maximize Objective } length constraints, diameters are such that mission probably could not be met.
4. Candidate System - Liquid ox. and solid fuel.
5. Aerojet-General Corporation/Liquid Rocket Operations.
6. Self-pressurizing liquids after initial heating.
7. Aerojet-General Corporation/Solid Rocket Operations.
8. Rocket Research Corporation
9. Mono-propellant system.
10. Cost/unit at yearly rate of production indicated.

Note: The above costs are for propulsion unit only and do not include payload housing, nozzle shroud and fins. For comparison purposes the following estimates of ancillary hardware and vehicle RDT&E cost will be added.

Rate	$	100	1,000	2,000	5,000	10,000
Unit Cost	$	260	185	165	142	128
RDT&E	$	300,000	300,000	300,000	300,000	300,000

Table 3.1.3.5

COST DATA FOR COMPARISON EXISTING MOTORS

EXISTING UNITS

ITEM / UNIT		ARC ARCAS	ARC S-Boosted-HV Arcas (2)	Thiokol Nike Cajun/Apache	Thiokol Tomahawk	Thiokol Iroquois	Canadian Bristol BB III
Unit Cost (1)	$	$2,000	$4,400	$6,600	$7,000	$5,000	$9,000

(1) Estimated cost/unit at present rates of production based on available general industrial knowledge. The prices include payload housing nozzle shroud and fins. These items have been in production for a few years to many years at rates from a few hundred to a few thousand per year.

(2) S-Boosted; Sidewinder or Sparrow.

Note: As a basis for comparison a 10% reduction will be assumed for production rates in the order of 5,000/year.

Item / UNIT		ARCAS	S-Boosted-Arcas	Nike-Cajun/Apache	Tomahawk	Iroquois	BB III
Unit Cost	$	$1,800	$3,960	$6,000	$6,300	$4,500	$8,100

Note: Existing systems used in this comparison are simply those whose proven operational characteristics can be matched to the mission requirements (by ballasting or otherwise.) In most cases, however, these systems will exceed preliminary ground rules in terms of initial weight, motor dimensions, etc.

TOTAL COST OF CARRIER VEHICLES[1] FOR A 10 YEAR OPERATIONAL PROGRAM

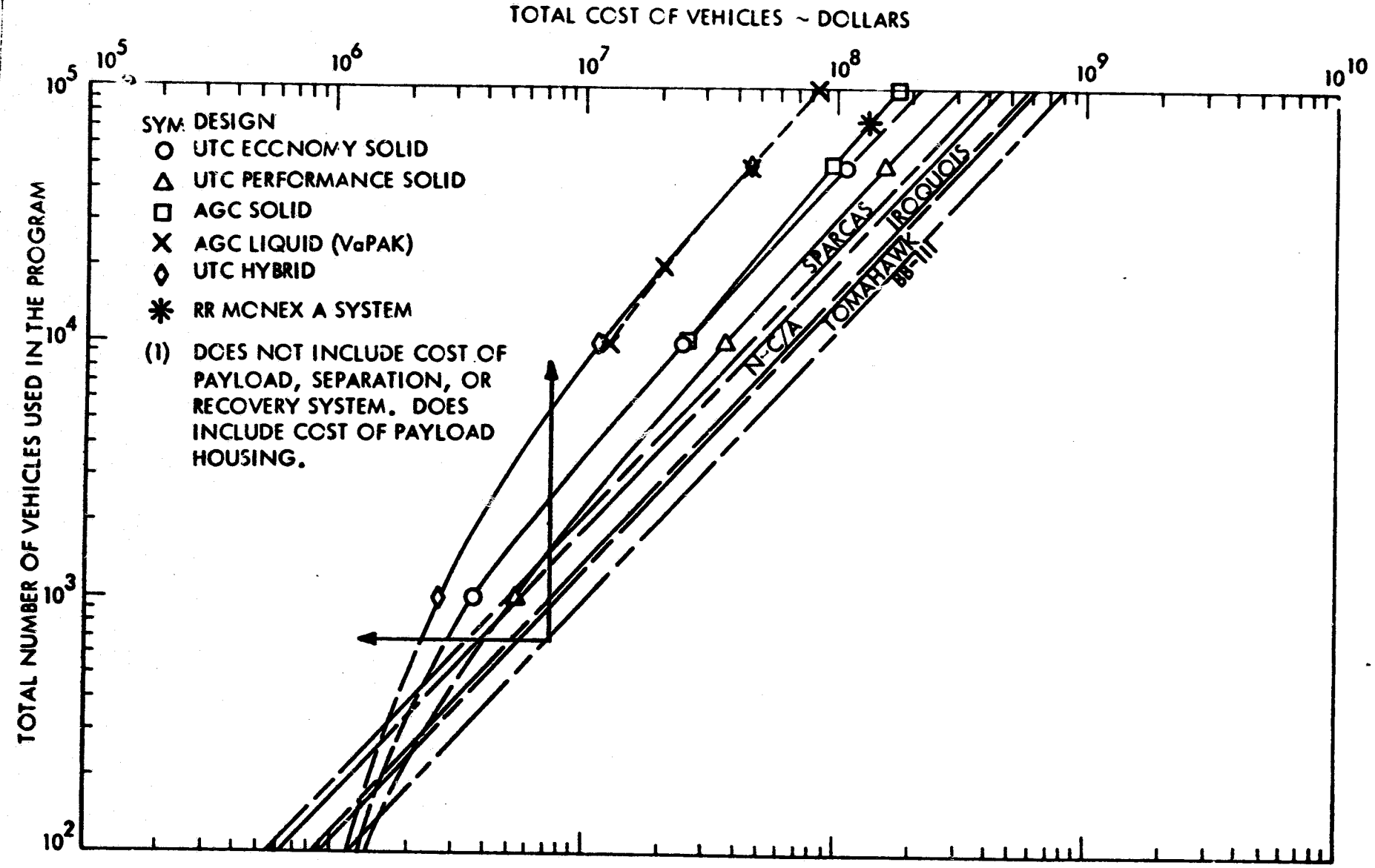


Figure 3.1.3.5
Page 40

missions. The upper portion of Table 3.1.3.5 gives estimated cost of systems as now in general use with the current usage rate varying between a few hundred to a few thousand per year. These systems have been in existence for from a few to almost ten years. The lower portion of the table gives the unit cost of the existing systems used for comparison purposes. The costs in the lower portion reflect a 10% discount from those in the upper portion, as an adjustment for higher than current usage rate. The 10% adjustment was deemed to be realistic for these units since their development and production has been established for at least a few years; therefore, even at elevated production rates, it is doubtful that significant "learning" effects or other production techniques would further reduce unit cost significantly. Further, shifting to other fabrication methods would require additional costs in the nature of R and D and/or tooling. Also, most of these existing systems would fall outside of the study ground rules in terms of weight and/or size, and are included for cost comparisons only.

In order to make a comparison, on a realistic basis, it was assumed that for the missions considered, that the carrier vehicle would have a service life on the order of ten years (a reasonable assumption based on past experience, and current expectations). Figure 3.1.3.5 shows the total cost for carrier vehicles as a function of the number required for the ten-year program. The area of prime interest is the upper portion of Figure 3.1.3.5. That is, for the meteorological mission, the projected usage rate is of the order of 1,000-10,000 units/year or, for the 10-year program, 10,000 to 100,000 units total. As can be seen in Figure 3.1.3.5, all the candidate designs would be cheaper than using existing units by as much as an order of magnitude, e.g., at 10^5 units for the 10-year program both the VaPak and UTC hybrid projected costs are on the order of 8.5×10^7 dollars vs 8.0×10^8 dollars for a Black Brant III based system. Even at the lower level (i.e., 10^2 - 10^3 units), which corresponds to the D-region probe mission, the new units would be theoretically competitive with existing units. The curvature of those curves corresponding to the new vehicles, where they cross those corresponding

to existing vehicle, simply represent the burden of RDT and E cost that are associated with the new vehicles. It is not surprising that a relatively small number of new vehicles are required for their costs to "break even" with existing vehicle costs, for it must be remembered that the existing vehicles were not designed for the missions considered.

Based on the results graphically presented in Figure 3.1.3.5, it was recommended that Phase II of the program be carried out as planned. That is, the UTC hybrid unit be subjected to detailed design and analysis within the context of an overall meteorological system concept. In addition, since costs were so close, it was recommended that some consideration be given to refining the design of the VaPak system also.

3.2 PHASE II - CONCEPTUAL DESIGNS

As noted in the closing of the previous section, the preliminary propulsion system evaluations of Phase I indicated good economic potential for both the UTC Hybrid and the AGC VaPak motor concepts when employed in a high usage rate meteorological system. In fact, both looked competitive with existing adaptable systems at lower usage rate, in the range probable for a D-region probe mission.

It was, therefore, the recommendation that the Phase II effort proceed as originally planned. That is, to define the hybrid unit in much greater detail; to generate an overall system concept based on a carrier rocket system built around the hybrid propulsion unit; and to define a development program from the complete system, including estimated system development and motor production costs. In addition, it was recommended that consideration be given to refining the design and cost data for the AGC VaPak unit since its estimated costs were comparable to the hybrid unit. It could thus be considered as a back-up design.

After the Phase I presentation, the program sponsor and technical director took the recommendations under advisement, and raised an additional question with respect to the original premise. That is, it was expected that the hybrid unit would be attractive in the small motor sizes in that the propellant cost would be low like those of liquid units, but have a packaging density approaching that of a solid. The potential was thus thought to be for a small, light, compact propulsion unit, coupled with low cost, particularly at high rates of production. Indeed, this potential was indicated in the Phase I design effort. Additionally, the VaPak concept seems to allow a liquid unit of similar characteristics. The question raised was "why were the solids so much more expensive?" Contact with the AGC solid unit designers indicated that "since they were unfunded and allowed little time, that their estimates were purposely conservative to begin with, and that full military Q.C. and other specs were assumed to apply, which add greatly to the unit costs." They were of the opinion that given some funding and time, (to produce a detailed design and to subject that design to a detailed cost and requirements analysis), that projected costs could be substantially reduced.

It was the decision of the sponsor and technical director that a small increment in funding be added, and that both the VaPak and AGC long-burning solid concept be carried into the second phase effort.

3.2.1 HYBRID MOTOR

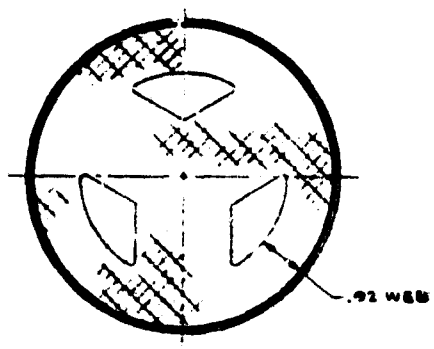
3.2.1.1 SUMMARY AND INTRODUCTION

A propulsion system design has been completed which conforms to requirements for a small meteorological and D-Region scientific probe vehicle. These requirements include two thrust levels (boost and sustain) and are as follows:

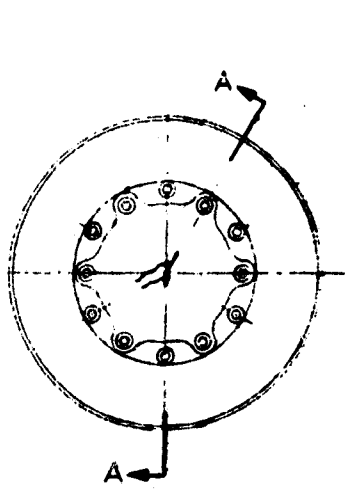
Parameter	Value
Boost Thrust (Sea Level), lb	4000
Sustain Thrust (average), lb	900
Boost Duration, sec	1
Sustain Duration, sec	29
Overall Length-to-Diameter Ratio	20
Maximum Overall System Weight, lb	200

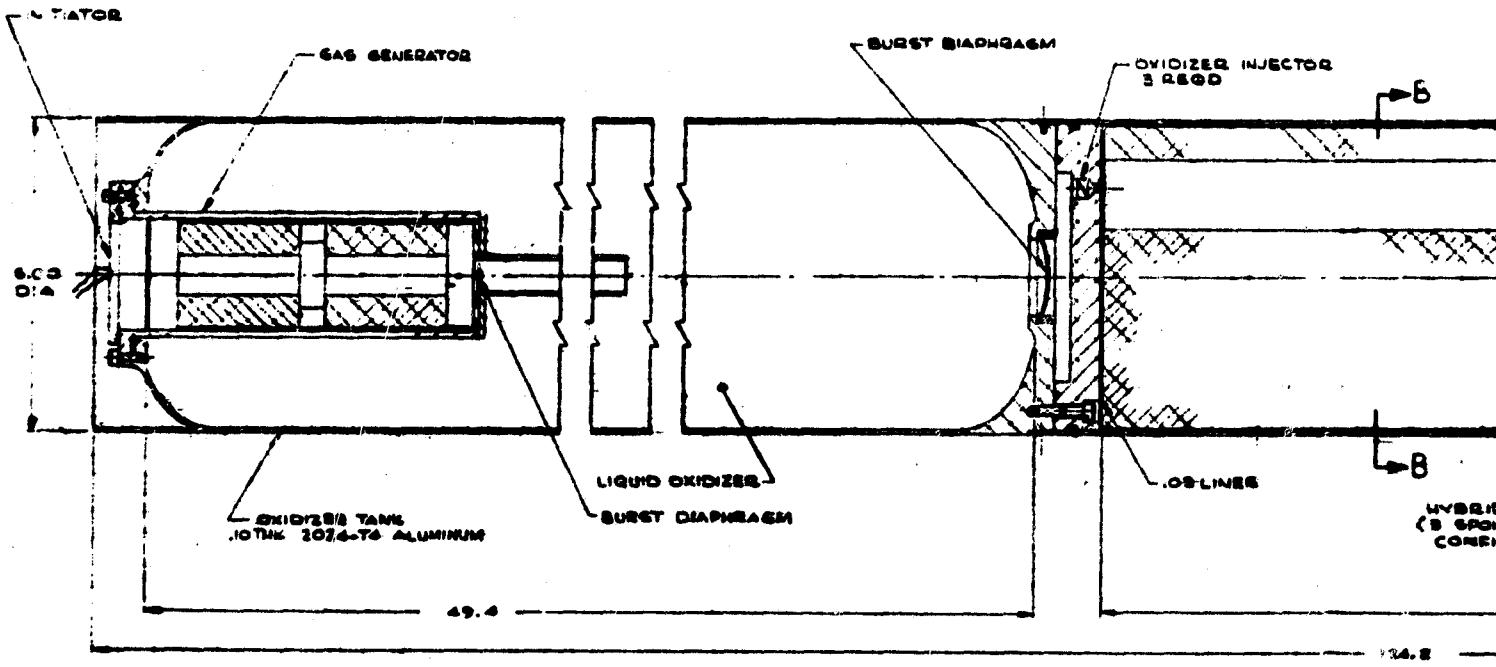
The design philosophy was conservative with emphasis on economy and packaging within a minimum practical diameter while not exceeding an overall length-to-diameter ratio of 20. The resultant configuration (shown in Figure 3.2.1.1) is a 6.0-in. diameter unit, approximately 123-in. long (an L/D of about 20) and weighing about 171 lb. A slightly smaller diameter can be utilized but at the expense of departing from a standard hardware size and thus incurring additional expense.

The unit has been designed as two mechanically joined sections; an oxidizer tank and a thrust chamber assembly. This approach provides for economy of fabrication and maintenance (casting of the hybrid fuel grain and maintenance of the internal O-ring surface), ease of service (replacement of oxidizer injectors, burst diaphragm, etc.), and in addition, makes the unit easier to handle in the field (two pieces instead of one). As an option (dependent upon customer preference), with minor design changes the sections could be welded and supplied as an integral unit.



SECTION B-B

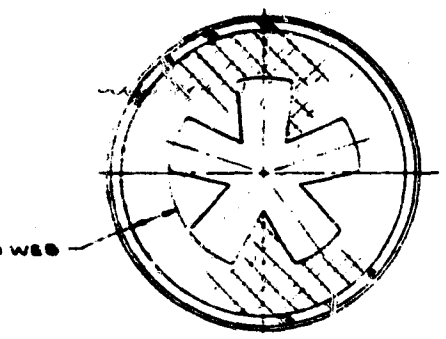




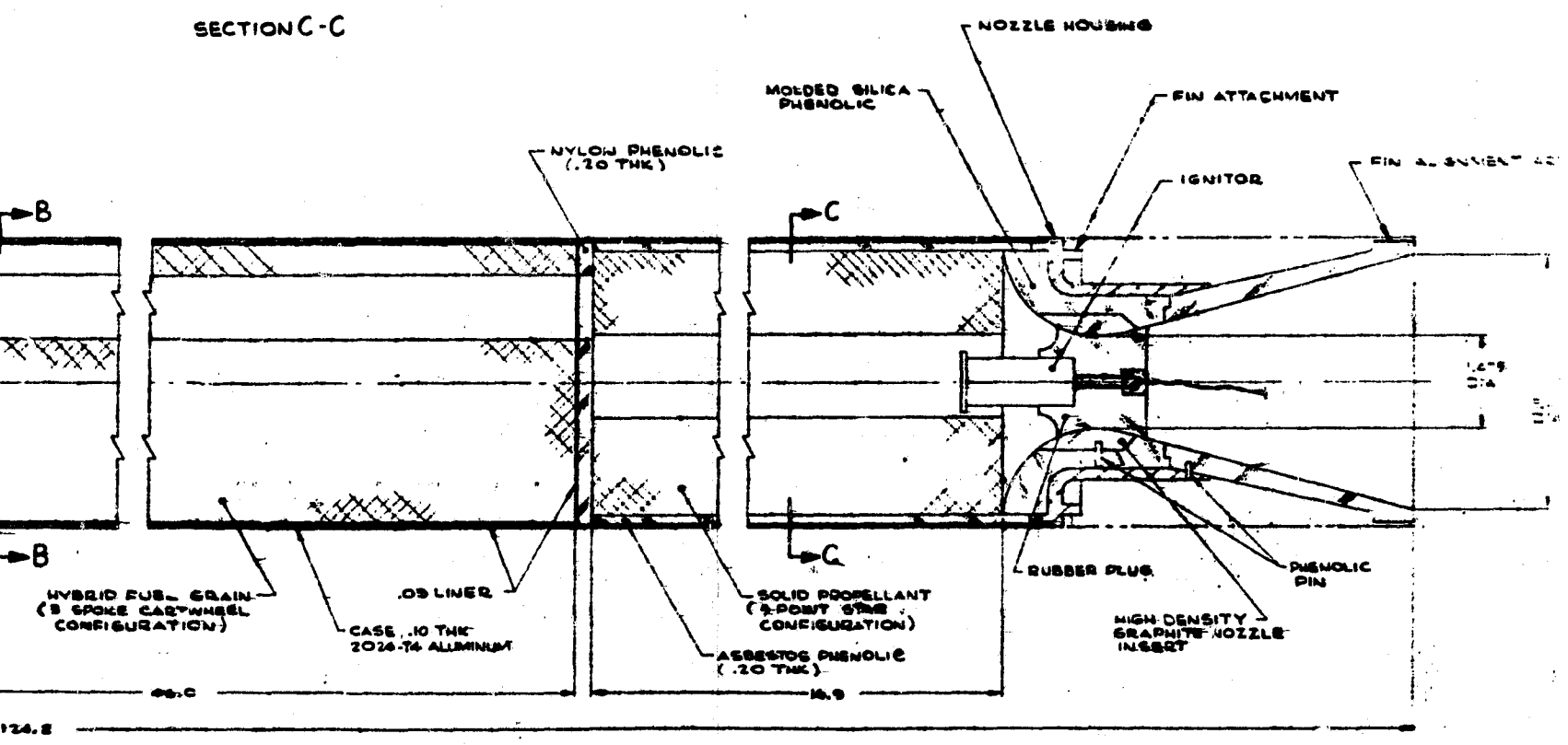
SECTION A-A

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FOLDOUT FRAME 1



SECTION C-C



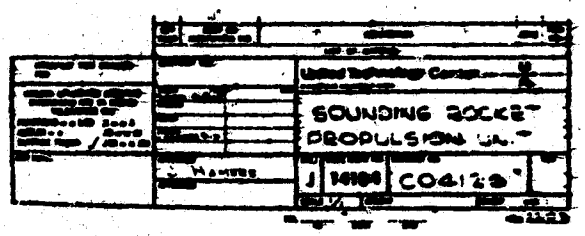


Figure 3.2.1.1
Sounding Rocket
Propulsion System

FOLDOUT FRAME 2

3.2.1.2 MOTOR CONCEPT

BOOST PHASE PROPULSION

The propulsion system employs a solid propellant charge for the 1-sec, 4,000 lb thrust level boost phase and a pressure-fed hybrid system for the 29-sec sustain phase. The solid propellant, which consists of aluminum, ammonium perchlorate, and binder, is in a five-point internal-burning star geometry with no end-burning. A chamber pressure of 1500 psia was selected as the practical maximum. At this pressure and with a nozzle area ratio of 13.4:1 (maximum nozzle expansion within dimensional restrictions), this propellant delivers a specific impulse of 248 sec and a regression rate of 0.9 in/sec. Ignition is accomplished by an initiator located in the exhaust nozzle throat; the initiator is expelled with an increase in internal pressure. The ignitor will use standard squibs and propellants requiring approximately 2.0 amp D.C. for ignition. The exact electrical requirements will be determined when field operating and safety requirements are defined.

SUSTAIN PHASE PROPULSION

The hybrid portion of the system employs a 45% aluminum/55% PBAN solid fuel oxidized with IRFNA to deliver an average specific impulse of 236 sec at a propellant mixture ratio of 1.5:1 and a chamber pressure of 330 psia (which resulted from sizing the nozzle throat for the boost thrust conditions) when expanded 13.4:1.

PRESSURIZATION

The liquid oxidizer tank is pressurized and maintained at 430 psi after the one-second boost phase is completed with a gas generator located in the forward end of the tank. The gas generator is an internal end burning cylindrical grain using a standard Olin[(1)] propellant system. An initiator is used to ignite the gas generator which pressurized the oxidizer tank. Subsequent rupture of the burst disc (located at the aft end of the oxidizer tank) allows oxidizer to flow through the injectors and onto the hybrid fuel grain. Thermal feedback from combustion of the boost propellant

(1) Olin Mathieson Chemical Corporation

will have preheated the hybrid fuel and provides ignition of the hybrid propellants. Phenolic caps placed over the outlets of the oxidizer injectors prevent contamination of the oxidizer feed and injection system during boost operation.

SOLID GRAIN DESIGN

A three-spoke cartwheel fuel grain geometry coupled with the plenum chamber created by consumption of the boost propellant will induce thorough mixing of the hybrid propellant gases and maintain high combustion efficiency. The aft end of the hybrid fuel grain is inhibited from burning by an 0.2-in. piece of nylon phenolic which is bonded to the grain surface and which will be consumed during the sustain firing. An 0.2-in. thick asbestos phenolic sleeve is used to insulate the chamber walls and the nozzle and to provide the cartridge in which the solid propellant charge is loaded.

The hybrid fuel grain will be cast (case-bonded) into the prelined case, cured and trimmed to length. The aft grain insulation will be bonded to the aft end of the grain with liner and then the cartridge loaded solid charge will be placed in the case adjacent to the nylon insulation. The nozzle assembly will be attached to the motor case with screws. The solid initiator can be installed by compressing it or by inserting it into the chamber prior to nozzle installation and then pulling it into the nozzle throat after the nozzle has been secured.

TANK DESIGN

Both the IRFNA tank and the thrust chamber case will be fabricated of 2024-T4 aluminum. The oxidizer tank consists of a tube with welded ends and has a burst diaphragm secured in the aft end of the tank with an internal wrenching nut. Fill and drain of the tank is accomplished with the gas generator removed and is done through the same opening. An extended flange is provided for payload attachment. The gas generator assembly consisting of the grain, case, and burst diaphragm is inserted through the tank opening and bolted to the aft end of the oxidizer tank.

INJECTION SYSTEM DESIGN

The oxidizer injectors consist of an orifice with a spinner upstream to achieve the desired spray angle and pattern. The bulkhead containing the oxidizer injectors will be bolted to the oxidizer tank and the two assemblies (oxidizer tank assembly and the thrust chamber assembly) mated and secured with screws. The gas generator initiator can be installed at any point during the assembly or in the field. If the initiator would become prematurely active while installed, the unit would not become propulsive because the hybrid propellants are not hypergolic without first preheating the fuel grain. A standard initiator will be used requiring approximately 2.0 amp. D.C. to activate.

OTHER SYSTEM CONSIDERATIONS

At present, there is no indication that any special tooling or handling equipment will be required for the motor assembly in field operation. All propellants will meet the operating temperature requirement of -65°F to +165°F. The specific shelf life, storage environment, inspection and service requirements will be determined during the development program.

The gas combustion products for the hybrid propellant sustain phase and solid propellant boost phase are presented in Tables 3.2.1.1 and 3.2.1.2, respectively.

DESIGN SUMMARY

A design summary and a weight breakdown of the individual components is presented in Tables 3.2.1.3 and 3.2.1.4. A time history of the motor weight and center of gravity location is presented in Figure 3.2.1.2.

3.2.1.3 PROPULSION SYSTEM DEVELOPMENT AND QUALIFICATION OUTLINE

INTRODUCTION

The technical approach, program plan, and schedule to develop and qualify a hybrid propulsion system suitable for the small-probe vehicle

Table 3.2.1.1

HYBRID PROPELLANT GAS PRODUCTS
PROPELLANT: Al/PBAN (45/55)
IRFNA

Species	Product Composition	Moles/100g	Products less than 4×10^{-4} moles/100g
	Chamber	Exhaust	
CO	1.2499	1.2053	AlF_2
H_2	0.7280	0.8061	AlF_3
H_2O	0.5191	0.5701	AlH
N_2	0.5118	0.5172	AlN
Al_2O_3 (liq.)	0.3168	0.2642	Al_2O
H	0.1998	0.1717	C
CO_2	0.1073	0.1529	CF
OH	0.0792	0.0021	CH
HF	0.0152	0.0184	CH_2
O	0.0121	0.0000	CH_3
NO	0.0105	0.0000	CH_4
O_2	0.0034	0.0000	F
AlF	0.0031	0.0000	HCN
Al	0.0020	0.0000	N
AlO	0.0017	0.0000	NH
HCO	0.0011	0.0000	NH_2
Al_2O_3 (solid)	0.0000	0.0057	NH_3

Table 3.2.1.2

SOLID PROPELLANT GAS PRODUCTS

PROPELLANT: UTX 10146

Species	Product Composition, Moles/100g		Species less than 5×16^{-4}
	Chamber	Exhaust	moles/100g propellant
H_2O	1.1984	1.1997	Cl_2
CO	0.8266	0.8163	Cl
HCl	0.6437	0.6556	NH_2
H_2	0.6191	0.6243	NH_3
N_2	0.3422	0.3428	Al
CO_2	0.2190	0.2294	HCN
Al_2O_3 (liq.)	0.0912	0.0219	N
H	0.0347	0.0215	NH
Cl	0.0246	0.0157	
OH	0.0234	0.0118	
$FeCl_3$	0.0024	0.0024	
Al_2Cl_2	0.0021	0.0010	
NO	0.0019	0.0008	
O	0.0008	0.0003	
O_2	0.0006	0.0002	
AlCl	0.0006	0.0002	
$AlCl_3$	0.0004	0.0002	
HCO	0.0003	0.0002	
Fe	0.0001	0.0000	

Table 3.2.1.3
DESIGN SUMMARY

A. Boost Phase (Solid Propellant)	
Propellant	UTX 10146*
Regression Rate @ 1500 psia, in/sec	0.9
Density, lb/in^3	.0638
Performance	
Average Thrust (Delivered), lb	4000
Burn Time, sec	1
Average Chamber Pressure, psia	1500
Specific Impulse (Delivered), sec	248
Grain Design	
Grain Shape	Five Point Star
Grain Diameter, in.	5.48
Grain Length, in.	16.9
Port Area-Chamber Area Ratio	0.234
Fuel Area-Chamber Area Ratio	0.704
Sliver Area-Chamber Area Ratio	0.062**
Port Area-Nozzle Area Ratio	3.23

*Consisting of Aluminum, Ammonium Perchlorate, and Binder

**Solid propellant sliver will be consumed during sustain burning phase.

Table 3.2.1.3 Continued

B. Sustain Phase (Hybrid Propellants)

Fuel	AL/PBAN (45/55)
Oxidizer	IRFNA
Specific Impulse (Theoretical), sec	262
Propellant Mixture Ratio	1.5
Regression Rate Relationship	$\dot{r} = .065\ G^{1/2}$
Fuel Density, lb/in^3	.0517
Oxidizer Density, lb/in^3	.0560

Performance

Average Thrust (Delivered), lb	900
Burn Time, sec	29
Average Chamber Pressure, psia	330
Specific Impulse (Delivered), sec	236
Specific Impulse Efficiency, %	90
Characteristic Exhaust Velocity Efficiency, %	96

Grain Design

Grain Shape	Three Spoke Cartwheel
Grain Diameter, in.	5.74
Grain Length, in.	46.0
Port Area-Chamber Area Ratio	0.19
Fuel Area-Chamber Area Ratio	0.75
Sliver Area-Chamber Area Ratio	0.06
Port Area-Nozzle Throat Area Ratio	2.88
Oxidizer Mass Flux (G_o) Range, $lb/sec\text{-}in^2$	0.463 to 0.088

Table 3.2.1.3 Continued

C. Nozzle Data	
Throat Diameter, in.	1.475
Exit Diameter, in.	5.40
Area Ratio	13.4
Half Angle	15^{o}
D. Combustion Chamber	
Material	2024-T4 Aluminum
Wall Thickness, in.	0.10
E. Oxidizer Tank	
Material	2024-T4 Aluminum
Wall Thickness, in.	0.10
Length, in.	49.4
F. Total System	
Length, in.	123.5
G. Gas Generator	
Type	OMAX Ammonium Nitrate base
Grain Shape	Cylinder with end burning
Grain Length (Two Segments)	4.68
Grain Diameter	2.00
Grain Weight, lb.	0.77

Table 3.2.1.4

WEIGHT SUMMARY

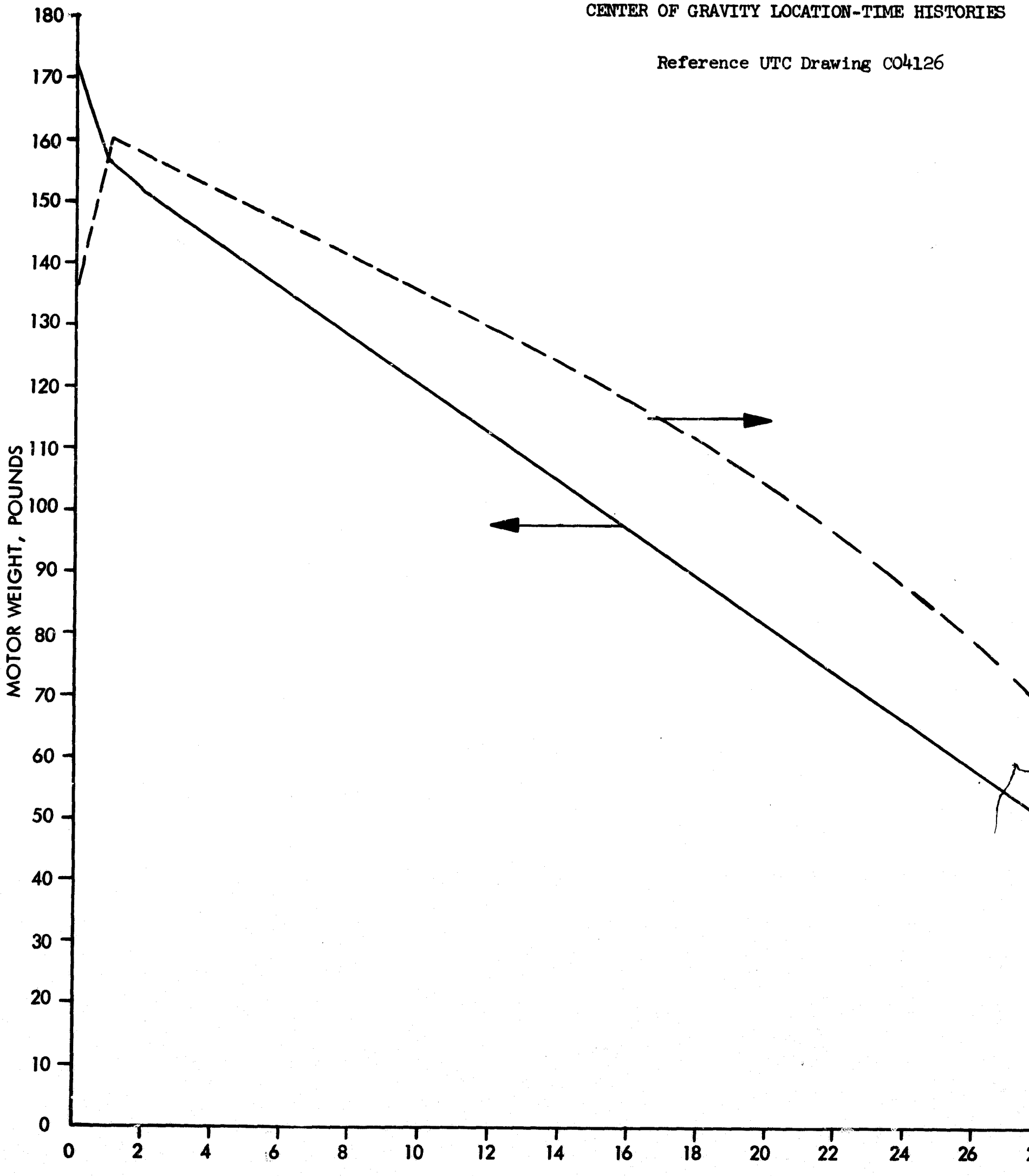
Item	Weight, Lb
Solid Propellant	16.5
Hybrid Fuel	49.0
Hybrid Oxidizer	66.1
Nozzle	4.1
Aft Grain Insulation	0.16
Aft Plenum and Nozzles Insulation	5.4
Case	14.04
Oxidizer Tank	8.9
Bulkhead and Injectors	3.0
Igniter Forward	0.5
Igniter Aft	1.0
Gas Generator	1.3
Hybrid Fuel Liner	1.0
	171.00

$$\text{Mass Fraction} = \frac{132.06}{171.00} = 0.772$$

MOTOR WEIGHT

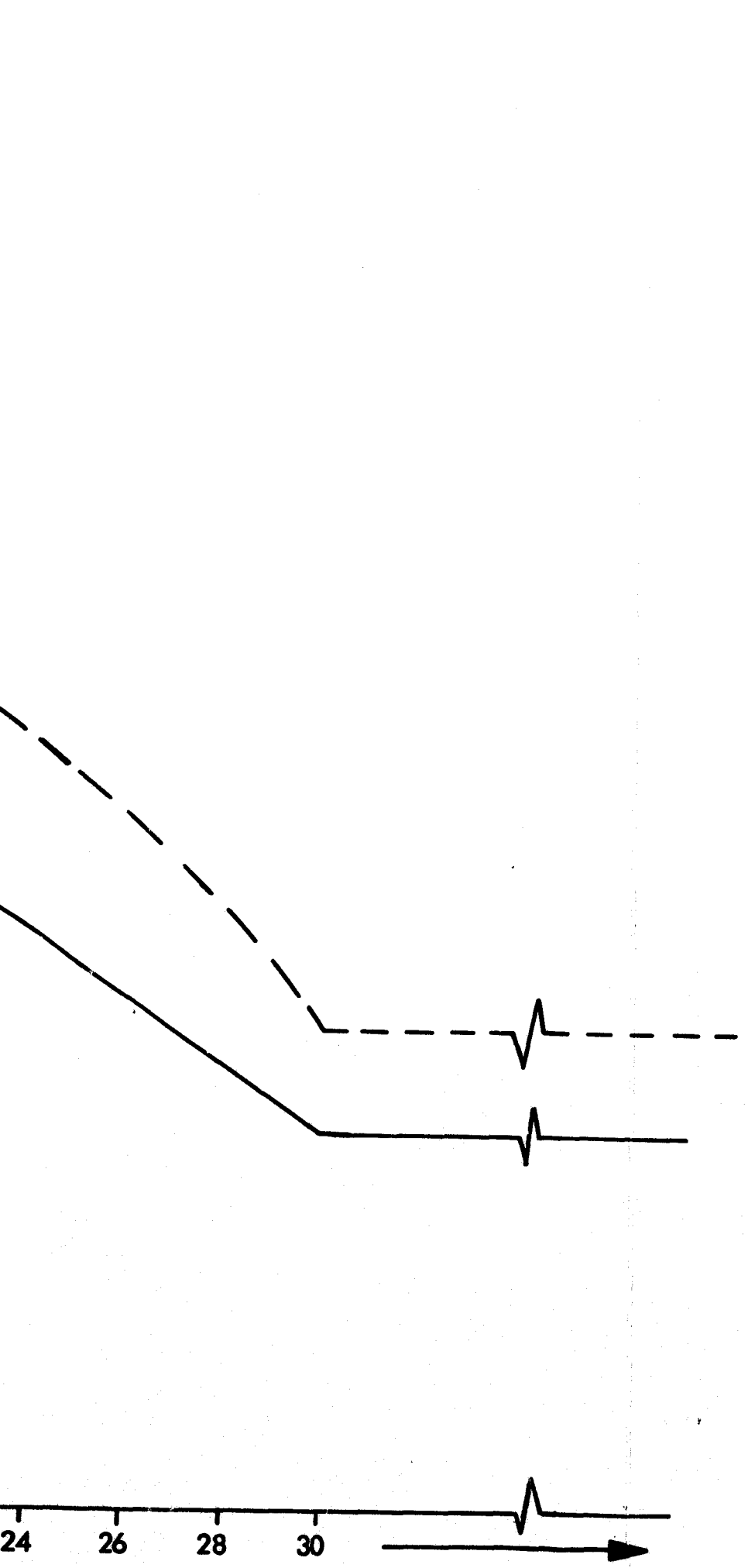
CENTER OF GRAVITY LOCATION-TIME HISTORIES

Reference UTC Drawing C04126



SD 9079FR-1

FOLDOUT FRAME 1



FOLDOUT FRAME 2 Figure 3.2.1.2

application are outlined in this section. The program objective is to contrive and demonstrate a reliable, low-cost propulsion unit capable of operation under diverse conditions; the approach suggested is to design a complete propulsion unit, fabricate and test the components, first individually and then sequenced into an integral unit, and then qualify the unit through a series of prototype tests.

APPROACH

The approach may be accomplished in the following manner (based on the preliminary propulsion system design presented in UTC drawing C04126, (Figure 3.2.1.1). The propulsion unit will be separated into five major areas of evaluation (as shown in Figure 3.2.1.3); (1) the solid propellant area, (2) the hybrid fuel area, (3) the complete thrust chamber assembly (TCA), (4) the liquid oxidizer section, and (5) the integral prototype unit (first using heavyweight hardware and then as a prototype unit).

TEST PROGRAM

Initial testing will be performed with the solid propellant phase (solid propellant, nozzle, insulation, and initiator) in a thickwall housing (Section A). This section, less the solid propellant, will then be attached to an additional thickwall housing loaded with hybrid fuel (Section B). This configuration constitutes a thickwall hybrid TCA, the second major section for evaluation.

Complete TCA testing (including the solid propellant) will follow as the third area of evaluation. This assembly will allow full duty cycle testing of the TCA components using thickwall hardware and a facility oxidizer supply and feed system.

The oxidizer tank and a flight configuration TCA case will be developed concurrently and will be integrated for prototype and subsequent testing. The oxidizer tank will be subjected to cold flows and structural tests and then mated with the thickwall TCA and test fired. The flight

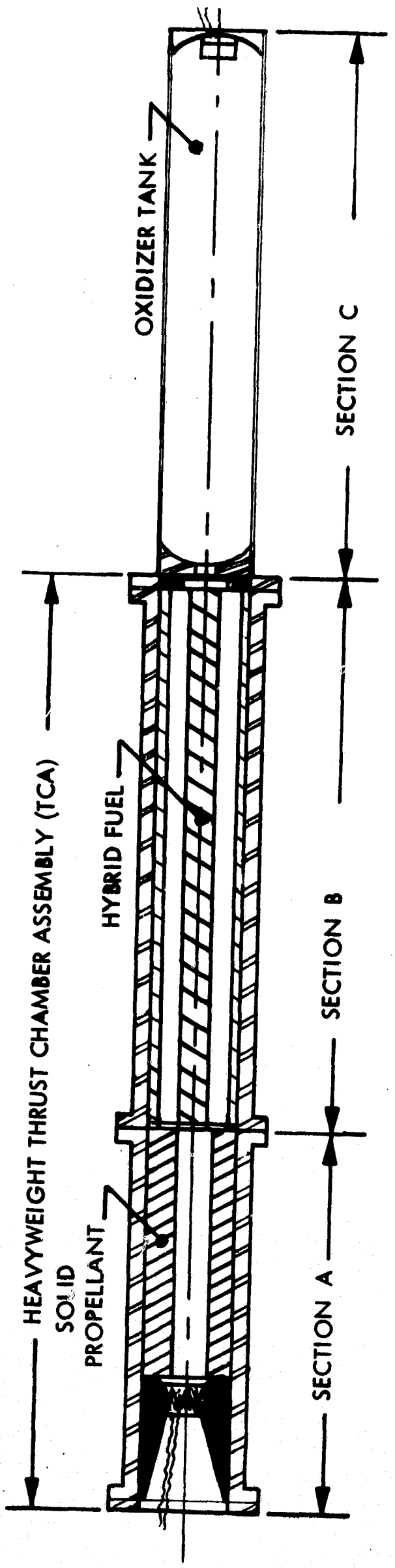

OXIDIZER TANK
SECTION C
HEAVYWEIGHT THRUST CHAMBER ASSEMBLY (TCA)
HYBRID FUEL
SECTION B
SOLID PROPELLANT
SECTION A

Figure 3.2.1.3

configuration TCA will also be structurally evaluated and then loaded and test fired using the facility oxidizer supply and feed system. Once satisfactory performance and confidence in the independent systems has been achieved, the systems will be integrated for verification testing as a unit.

PROGRAM DEFINITION AND SCHEDULE

As indicated in the enclosed program schedule (Figure 3.2.1.4), approximately thirty static test firings over an 18-month period are required to accomplish the development phase; an additional thirty prototype assemblies and six months will be needed to fully qualify the system. The program phases have been defined as follows:

I	Program Management
II	Design and Analyses
III	Procurement and Assembly
IV	Component Evaluation and Design Refinement
V	Prototype Testing (Integration and Verification)
VI	Qualification Testing
VII	Reporting and Travel
VIII	Manufacture and Delivery

Details of these phases are presented in the following paragraphs.

Phase I - Program Management

Program management will be a continuous function throughout the program. Among the tasks assigned to this phase will be preparation of a progress plan to accomplish the established schedule. Design review and approval, activity coordination and general supervision. Program management will also provide the technical liaison function between the customer and UTC and will be responsible for preparation of all reports.

Phase II - Design and Analyses

System analyses and detailed system, component, and tooling design will be completed in this activity. This phase will also be a relatively

PROGRAM PLAN

PROPULSION SYSTEM DEVELOPMENT AND QUALIFICATION

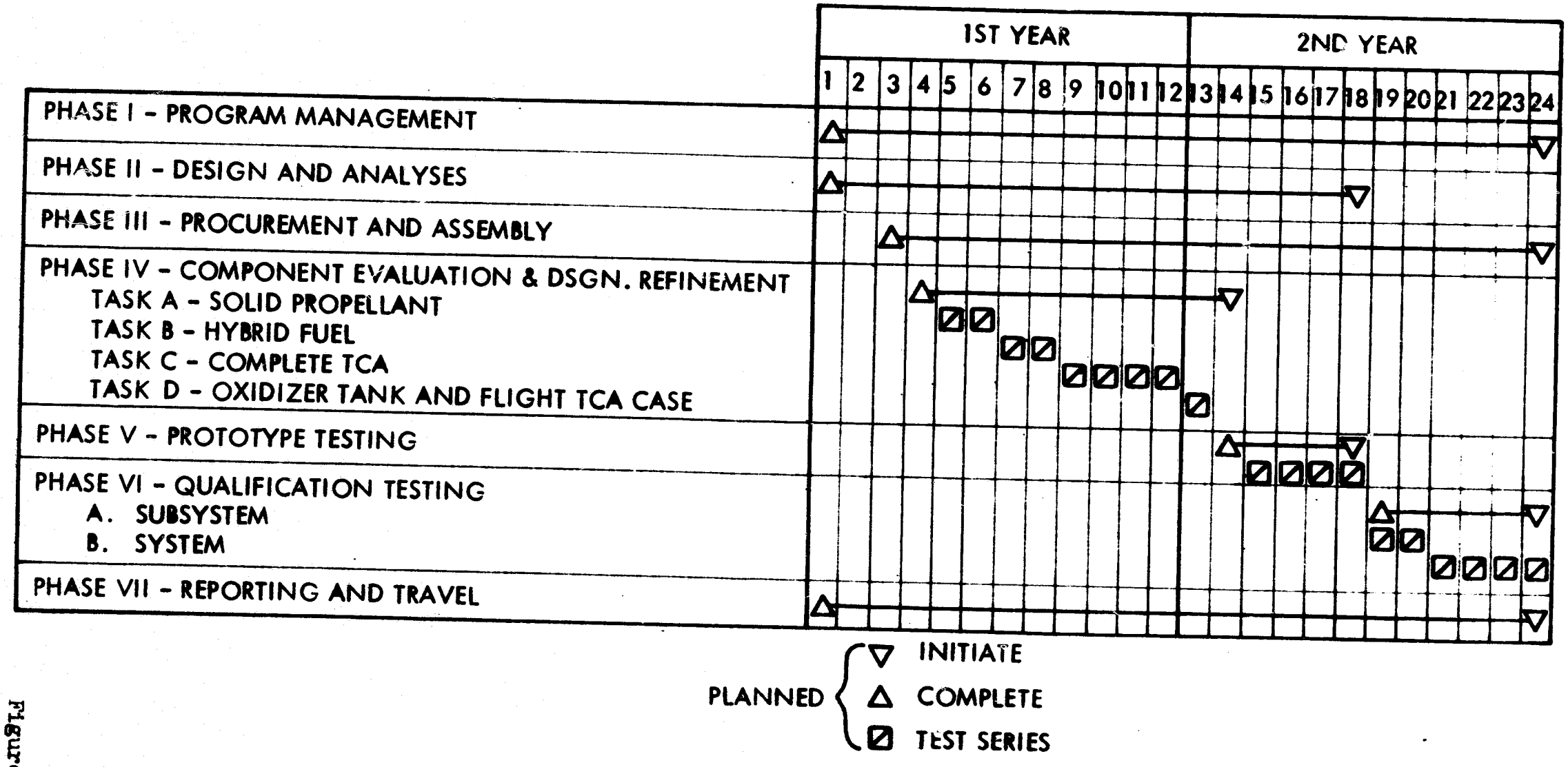


Figure 3.2.1.4

continuous function during development to incorporate modifications indicated by the experimental effort.

System analyses will result in propellant selection, grain geometry optimization, selection of a method of oxidizer tank pressurization, and determination of the internal ballistics. Structural and thermal analyses will be performed when applicable, predicted operational characteristics will be determined, and fabrication and assembly procedures will be established.

Assembly drawings will be prepared of the major subassemblies as well as the complete engine assembly. In addition, detailed drawings will be prepared of fabrication and process tooling, handling equipment, and the static test fixtures. Standard component items such as initiators, tubing, and seals will be utilized where possible. The designs will be reviewed and approved by project management to insure functionality, simplicity and economy of fabrication, handling, loading, component accessibility and practicality.

Phase III - Procurement and Assembly

Procurement or fabrication and assembly of the components, subassemblies, and complete units will be completed in this phase. Standard procurement procedures will be utilized throughout the activity. Fabrication and assembly will be performed in accordance with the procedures established in the design and analyses phase and will be supervised by program management.

Phase IV - Component Evaluation and Design Refinement

The development testing of the program will be conducted in this phase. The major systems previously discussed will be developed independently through a series of cold flow, structural and hot firing tests. Design modifications will be incorporated as required.

The tasks of this phase have been summarized in the following outline.

TASK A. Solid Propellant Phase (Section A, Refer to Attached Sketch) Approx. Five Firings with Heavyweight Hardware to Evaluate:

1. Solid Propellant - Performance, Regression, Grain Geometry, Ignition
2. Nozzle - Performance
3. Insulation - Performance
4. Initiator - Capability and Function

TASK B. Hybrid Fuel Phase (Sections A and B Less Solid Propellant) Cold Flows and Approx. Five Firings with Heavyweight Hardware and Facility Oxidizer Tank to Evaluate:

1. Hybrid Fuel - Performance, Regression, Combustion Efficiency, Grain Geometry
2. Nozzle
3. Insulation (Aft Grain and Aft Plenum)
4. Oxidizer Injectors - Uniformity, Spray Pattern and Distribution, Fuel Utilization

TASK C. Complete TCA (Sections A and B) - About Ten Firings with Heavyweight Hardware and Facility Oxidizer Tank to Evaluate:

1. Hybrid and Solid Propellant Ignition
2. Nozzle, Insulation, and Injector Performance Over a Complete Duty Cycle
3. Performance Efficiency of Total System

TASK D. Flight Configuration TCA Case and Oxidizer Tank (Section C) Cold Flows, TCA and Gas Generator Firings to Evaluate:

1. Flight Configuration TCA Case - Structural (Hydrotest), Functional (Firings)
2. Flight Configuration Oxidizer Tank - Structural (Hydrotest)
3. Gas Generator (First Using Water in the Oxidizer Tank and then IRFNA)
4. Burst Diaphragm

Phase V - Prototype Testing (Integration and Verification)

Approximately eight static firings will be conducted in the integration of components into a flight configuration unit and to verify the system operating characteristics and function. Initial firings will be performed with the oxidizer tank mated to the thickwall TCA. Later firings will include the complete flight propulsion unit, i.e., the oxidizer tank mated with the flight configuration TCA. Full duty-cycle verification tests will be performed with this configuration to demonstrate component durability, total system performance, and system characteristics.

Phase VI - Qualification Testing

Upon completion of the design, development, and testing phase, system and component qualification will be initiated.

A. Subsystem Qualification

Motor components and subassemblies will be subjected to environmental test conditions anticipated to be encountered during handling, staging, and flight. Since the environmental test parameters are not defined at this time, exact test procedures will be established at a later date. However, typical environmental tests to be conducted will be as follows:

Vibration - Motor components will be subjected to vibration tests. Vibration will be induced in the component in three mutually perpendicular axes to the specified vibration limits and duration. Where applicable, vibration will be induced at temperature.

Shock - Motor components will be qualified to withstand shock in all axes. Shock magnitude, duration and number of shocks to be applied in each axis will be defined by the customer.

Temperature - Where temperature becomes a factor in the design of certain components, these components will be qualified to the required temperature extremes.

Acceleration - All motor components including attachment interfaces will be qualified to withstand the maximum anticipated acceleration on all axes.

Structural - Burst and drop tests will be performed to qualify the structural integrity of the unit and individual components when assembled.

B. System Qualification

Static firings will be conducted using prototype engines to qualify the unit with respect to combustion efficiency, fuel utilization, component durability and performance, and reproducibility and reliability. The firings will also produce operational data usable in range qualification systems. Conditioning environments will be established in conjunction with the customer's requirements.

Approximately ten flight motors will be delivered to the customer for use in practical application and flight tests. Data from previous static tests will be supplied with the engines to establish statistical reliability and confidence values for the flight units. Technical assistance will also be provided.

Phase VII - Reporting and Travel

Complete program status reviews will be made and reported in a monthly progress and status letter report. Quarterly reports summarizing the program development and system operating data will be prepared and submitted as well as a complete final report at the conclusion of the development and qualification effort which will detail all aspects of the program.

Trips will be made as required in the coordination of the program.

Phase VIII - Manufacture and Delivery

This phase represents the follow-on activity for units according to customer requirements. Since no unit quantity needs have been established at this time, the phase significance is principally that of indicating lead time and the planning needed to provide the vehicles.

3.2.1.4 PROGRAM COSTS

FR4236/84

GOVT TLX PD 42 33 SUNNYVALE CALIF 03 APRIL R 032155Z APR 68

FM UNITED TECHNOLOGY CENTER/SANNYVALE CALIF.

TO: RUMJRGA/AEROJET GENERAL/SPACE GENERAL/EL MONTE CALIF

AF GRNC

RT

UNCLAS GRNC ATTN: MESSRS R JENKINS/J TAYLOR

SUBJ: BUDGETARY COST SUMMARY

THE FOLLOWING BUDGETARY COST DATA ARE SUBMITTED PER YOUR REQUEST FOR USE IN THE NASA SMALL-PROBE VEHICLE STUDY. THESE BUDGETARY COSTS ARE APPLICABLE TO THE FINAL HYBRID MOTOR DESIGNS BEING FORWARDED UNDER SEPARATE COVER. 845 APR 4

DEVELOPMENT AND QUALIFICATION	$ 360,000 /"/
PRODUCTION AT 100 UNITS/YR	180,000 /"/ /"/
PRODUCTION AT 1,000 UNITS/YR	940,000 /"/ /"/
PRODUCTION AT 5,000 UNITS/YR	3,850,000 /"/ /"/
PRODUCTION AT 10,000 UNITS/YR	5,500,000 /"/ /"/

/"/ BASED ON TIME PERIOD JULY 1968 TO JUNE 1970

/"/-/"/ BASED ON PRODUCTION IN 1970

D ALTMAN

DIVISION VICE PRESIDENT

RT

3.2.2 LONG BURNING SOLID MOTOR

3.2.2.1 INTRODUCTION

This report describes the study effort conducted by the Solid Rocket Operations of the Propulsion Division of Aerojet-General Corporation to establish a design and plans for providing solid rocket motors for small probe meteorological rockets.

Included in this report in accordance with program requirements is the design of a motor with a maximum diameter of 6 in. and a minimum total impulse of 30,000 lbf-sec delivered during a 4000 lbf thrust boost phase plus a 1000 lbf sustain phase and a 27 to 30 sec total duration. Design of the motor was engineered so that it could be mass produced economically.

A development plan was prepared that would provide a fully demonstrated, reliable motor design. This plan with the associated schedules and test matrices is incorporated in Section 3.2.2.7 of the report. Costs for conducting the development program were calculated and are included in Section 3.2.2.8.

A comprehensive study of manufacturing procedures and associated costs was made for production rates of 100, 1000, 5000, and 10,000 motors per year. The unit cost for each of these production rates is presented in Section 3.2.2.8.

Additionally, alternate design data for each major motor component, are presented, as well as a limited amount of data concerning a very attractive alternate complete motor design. This alternate motor design was not made the prime design motor due to the fact that as yet the propellant characterization is not complete, and the currently established burn rate results in an objectionably high acceleration level. Data for this design are included due to the fact that both limitations listed above are currently being resolved under other programs, and the fact that a preliminary cost analysis indicates extremely attractive low manufacturing cost potential.

3.2.2.2 SUMMARY

Design criteria goals were established and coordinated with the Space Division, as were initial motor configuration and performance studies. A motor design was selected that would meet the performance requirements of the program and provide low production costs. The selected design includes a 6 in. diameter by 104 in. long chamber of AISI 4130 steel with a welded-on forward head. The desired ballistic performance in an internal burning grain design was achieved by use of a very low burning rate Aerojet propellant which provides a unique capability to optimize sounding rocket ballistics. A cartridge-loaded grain design was used that consisted of 10 slow burning rate (0.08 in/sec) sustainer grains, each 9.02 in. long, and one fast-burning rate (1.8 in/sec) booster grain 6.02 in. long. This design delivers 33,000 lbf-sec of impulse over a 27.2 sec duration; a 1.2 sec, 3500 lbf thrust boost phase; and a 26 sec, 1100 lbf thrust sustain phase. The grains are cast in phenolic sleeves which also act as insulation for the case. A glass-phenolic nozzle with an ATJ graphite throat insert is held in place by a metal retaining ring that is welded to the chamber after assembly of the propellant grains. The total motor weight is 178 lb, consisting of 136.7 lb of propellant and 41.3 lb of inert components. The motor total length is 110 in.

A plan was prepared for developing and demonstrating the selected design so that it could be produced for flight use with a high reliability and minimum production costs. This plan consists of conducting 9 flight-weight motor tests to demonstrate materials, performance and environmental capability. In addition, six flight tests would be conducted to demonstrate the motor capability in its intended application. Supporting component effort such as case burst tests, igniter demonstration tests, and propellant bond tests would also be conducted.

A cost was estimated for conducting the proposed development plan and is presented in Section 3.2.2.8. Also, an analysis of procedures, tooling requirements and facility requirements for production of the motor in quantities of 100, 1000, 5000, and 10,000 per year was made. The motor unit price for each of these quantities was estimated and is included in Section 3.2.2.8.

Additionally, a limited amount of data are included relative to a very preliminary design which appears quite attractive from the production cost standpoint. This alternate design was not made the prime design for this study due to two considerations. First, the design is based on a propellant that is currently under development; therefore, its properties are not completely determined. The second consideration, which is related to the first, is that the burning rate of the present formulation results in a shorter than desirable burning time. This, in turn, results in a relatively "hot" trajectory with very large acceleration levels, at least in a six-inch diameter motor. On the other hand, two other considerations offer potential for relieving this design's shortcomings. First, a program is under way to produce a full characterization of this propellant. Second, very preliminary tests indicate that by use of additives, the present burning rate can be substantially reduced. The latter two considerations, coupled with a particularly attractive economic potential, and the fact that it has a better specific impulse potential than the prime design, prompted the inclusion of the data related to the alternate design.

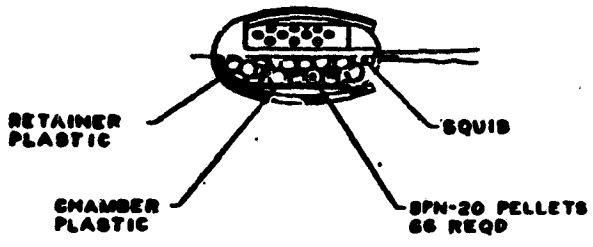
3.2.2.3 MOTOR DESIGN

OVERALL MOTOR DESCRIPTION

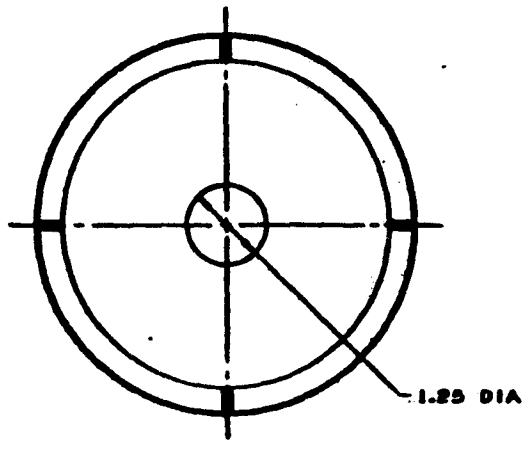
The selected design configuration of the Small Probe Solid Rocket Motor is shown in Figure 3.2.2.1. The motor is 110.13 in. long and has a maximum diameter of 6.0 in. The propellant grains are cartridge loaded into the chamber and provide a dual-thrust capability. The nominal boost thrust is 3500 lbf, and the nominal sustain thrust is 1100 lbf. The total nominal duration is 27.2 sec, of which boost thrust would account for the first 1.2 second of motor operation. The motor weight is 178 lb of which 136.7 lb consists of propellant and 41.3 lb consists of inert components. A summary of motor performance and weight characteristics is shown in Table 3.2.2.1. Estimated nominal pressure curves at -65, +70, and +165°F are shown in Figure 3.2.2.2 and the corresponding vacuum thrust curves are shown in Figure 3.2.2.3. Postfire motor weight is 39.6 lb. Nominal motor mass properties showing the motor weight, center of gravity, and moments of inertia variation during motor operation are presented in Figure 3.2.2.4.

COMPONENT DESCRIPTION

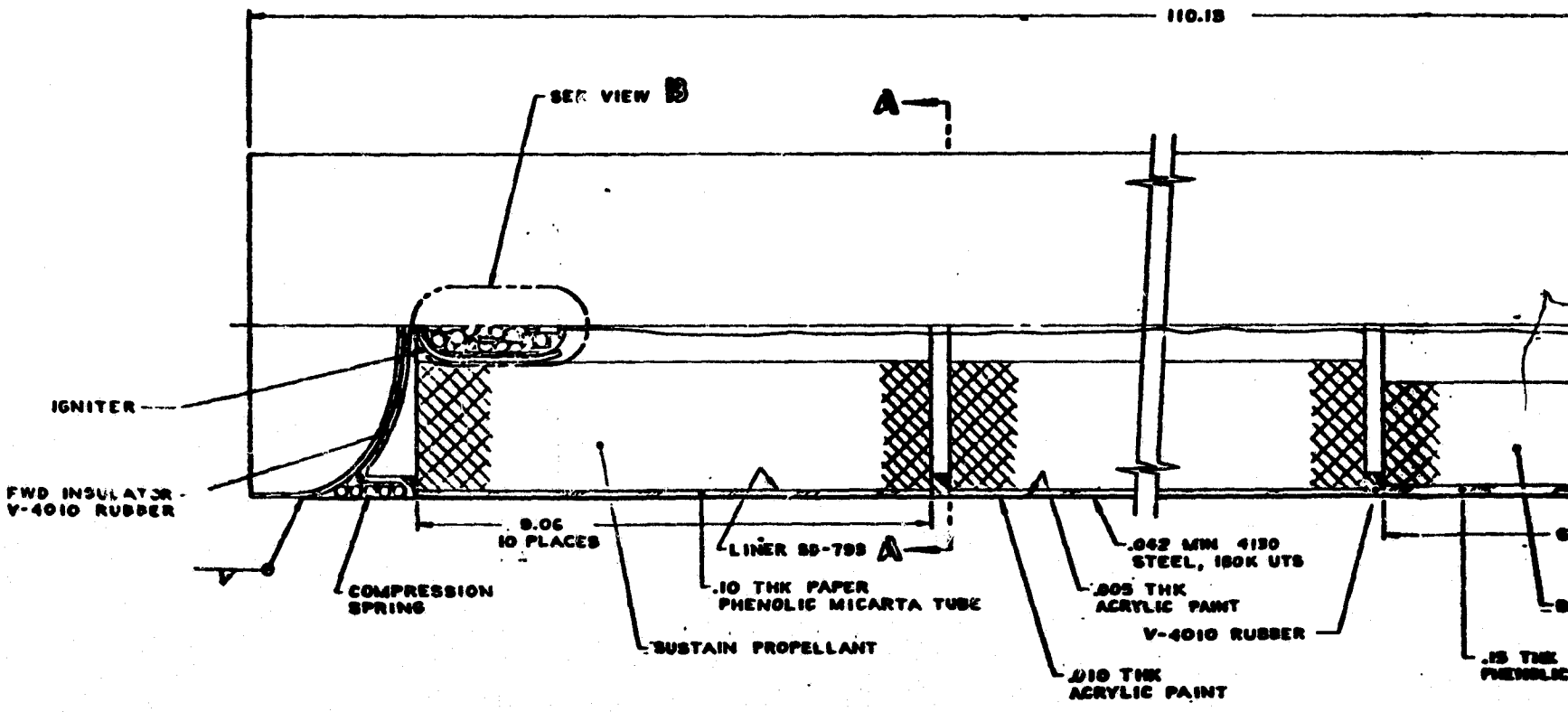
Propellant Grain Design - To provide a solid propellant rocket motor having a wide operating and storage temperature range and at the same time low in cost, a cartridge loaded motor design was selected. The propellant is cast into 4-ft long (approximately) Micarta (paper-phenolic) sleeves with the center core in position; after the propellant is cured the sleeves are cut to the desired length. Ten sustain grains, each 9.06 in. long, are used in the motor. The sustain propellant is an off-the-shelf formulation, which is in production use in other small motor programs. To provide for the dual-thrust capability, one boost grain is cast into a Micarta sleeve and cartridge loaded in the aft end of the motor. The boost grain is a high-burning-rate propellant, 1.8 in/sec at 2000 psia, and burns out during the first second of motor operation, providing a nominal boost thrust of 3500 lbf. The sustain thrust, 1100 lbf, is maintained by the sustain propellant, which has a burning rate of 0.08 in/sec



VIEW B

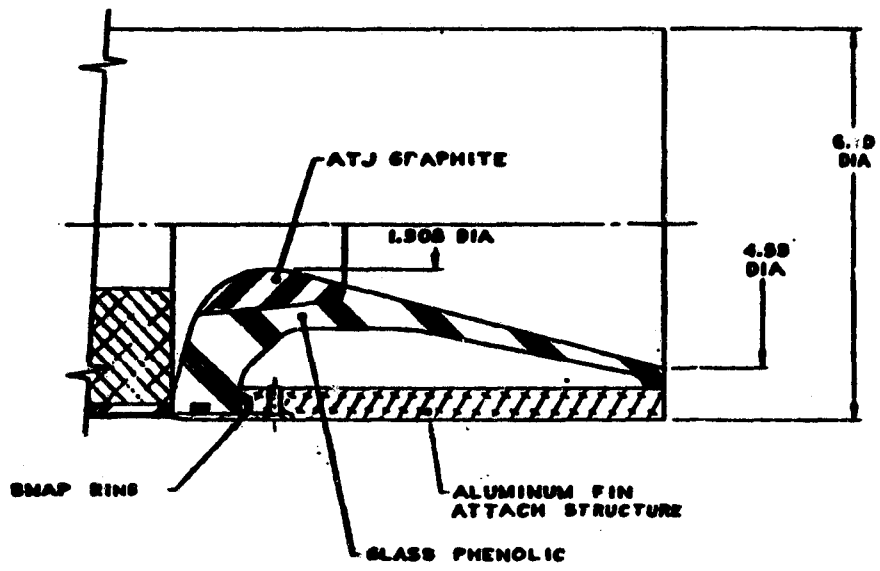


SECTION A-A

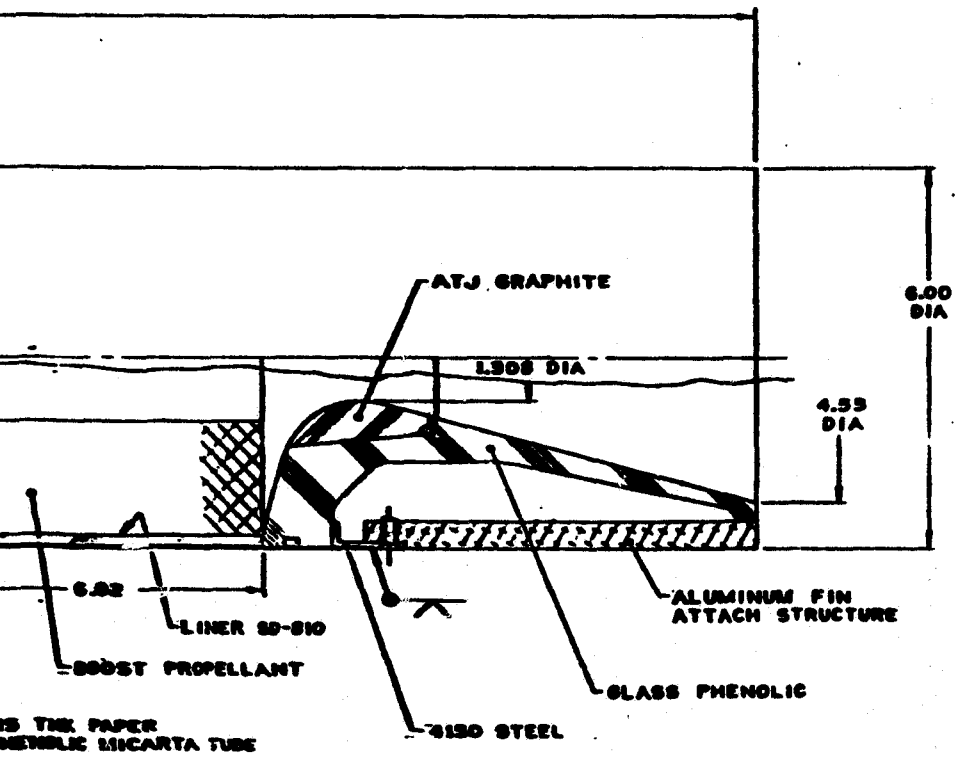


SD 9079FR-1

FOLDOUT FRAME 1



ALTERNATE NOZZLE DESIGN



SMALL PROBE ROCKET MOTOR

13310 1146410

Figure 3.2.2.1

FOLDOUT FRAME 2

Predicted Motor Performance at 70°F and Vacuum

	Boost	Sustain
Firing Duration, sec	1.2	26.0
Total Impulse, lb-sec	4226	28,938
Average Thrust, lbf	3520	1113
Propellant Specific Impulse, lbf-sec/lbm	270	240.8
Chamber Pressure, avg, psia	1600	500
Chamber Pressure, max nominal, psia	1800	545
Chamber Pressure, MEOP @ 165°F, psia	2180	670

Propellant Ballistic Properties

Std I_s at 1000/14.7 psia, opt exp lbf-sec/lbm	244.0	217.6
Burning Rate, in./sec	1.8 @ 2000 psia	0.08 @ 500 psia
Burning Rate Exponent	0.47	0.09
Propellant Density, lb/in.^{3}	0.0663	0.0591

Design Data

Throat Area, in.^{2}	1.344	1.344
Port-to-Throat Ratio	2.33	1.33
Expansion Ratio	12:1	12:1
*Web Thickness, in.	1.80	2.08
Motor Length, in. (face of fwd skirt to exit plane)	110.13	110.13
Motor Dia, in.	6.00	6.00
Exit Dia, in.	5.0	5.0

Weight Summary, lb

Chamber	26.2
Nozzle	3.5
Insulation and Liner	10.1
External Paint	0.8
Compression Spring	0.7
Total Inerts	41.3
**Propellant	136.7
TOTAL MOTOR	178.0
Mass Fraction	0.768

Note: * Initial sustain propellant web is 2.21 in. thick, however, 0.13 in. are consumed during the boost phase.

** Propellant weight is comprised of 7.85 lb of boost propellant and 128.85 lb of sustainer propellant. During boost, 7.85 lb of boost propellant is consumed as well as 8.7 lb of sustainer propellant.

Table 3.2.2.1

ESTIMATED NOMINAL PRESSURE

SMALL PROBE ROCKET MOTOR

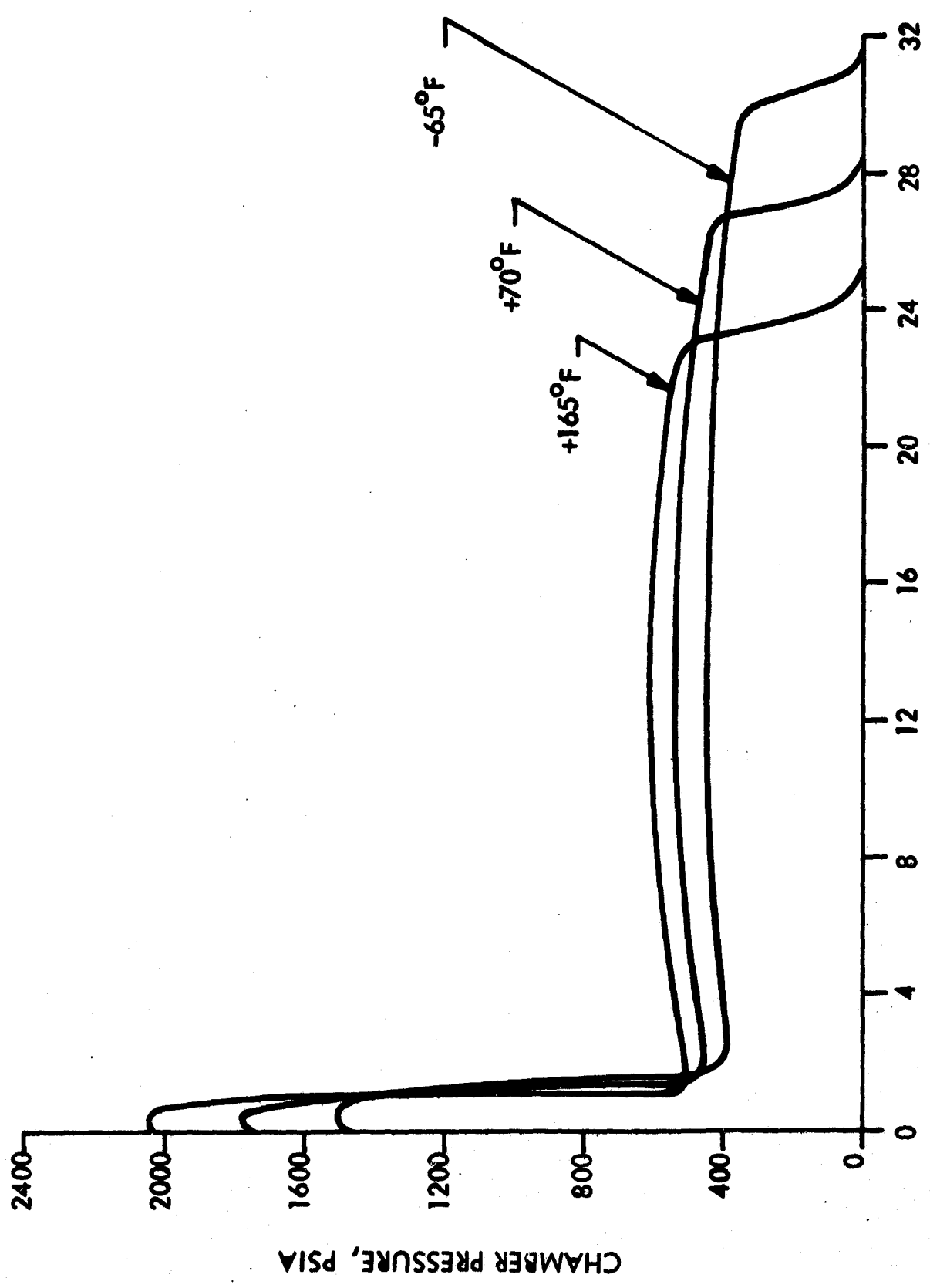


Figure 3.2.2.2

ESTIMATED NOMINAL VACUUM THRUST
SMALL PROBE ROCKET MOTOR

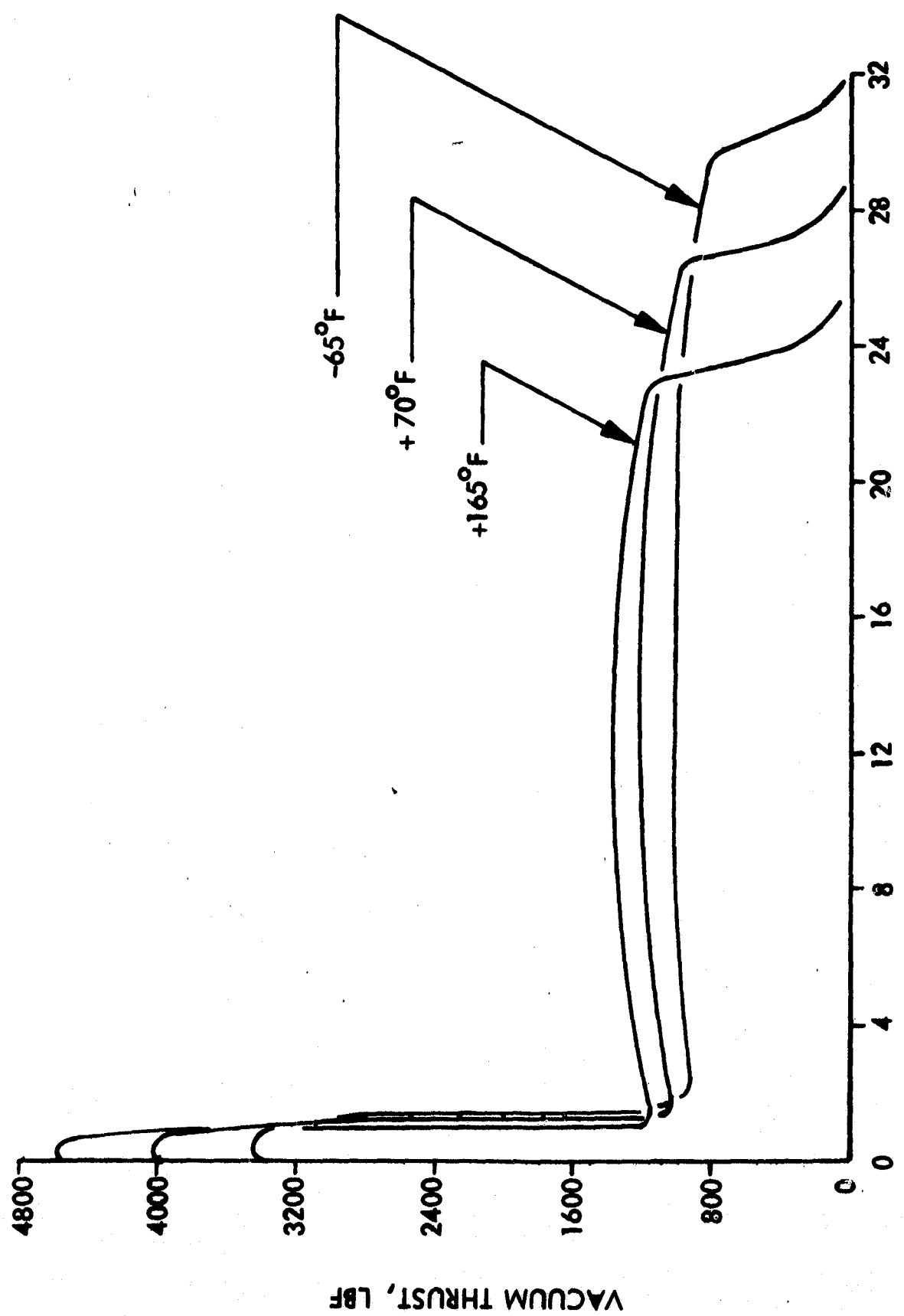
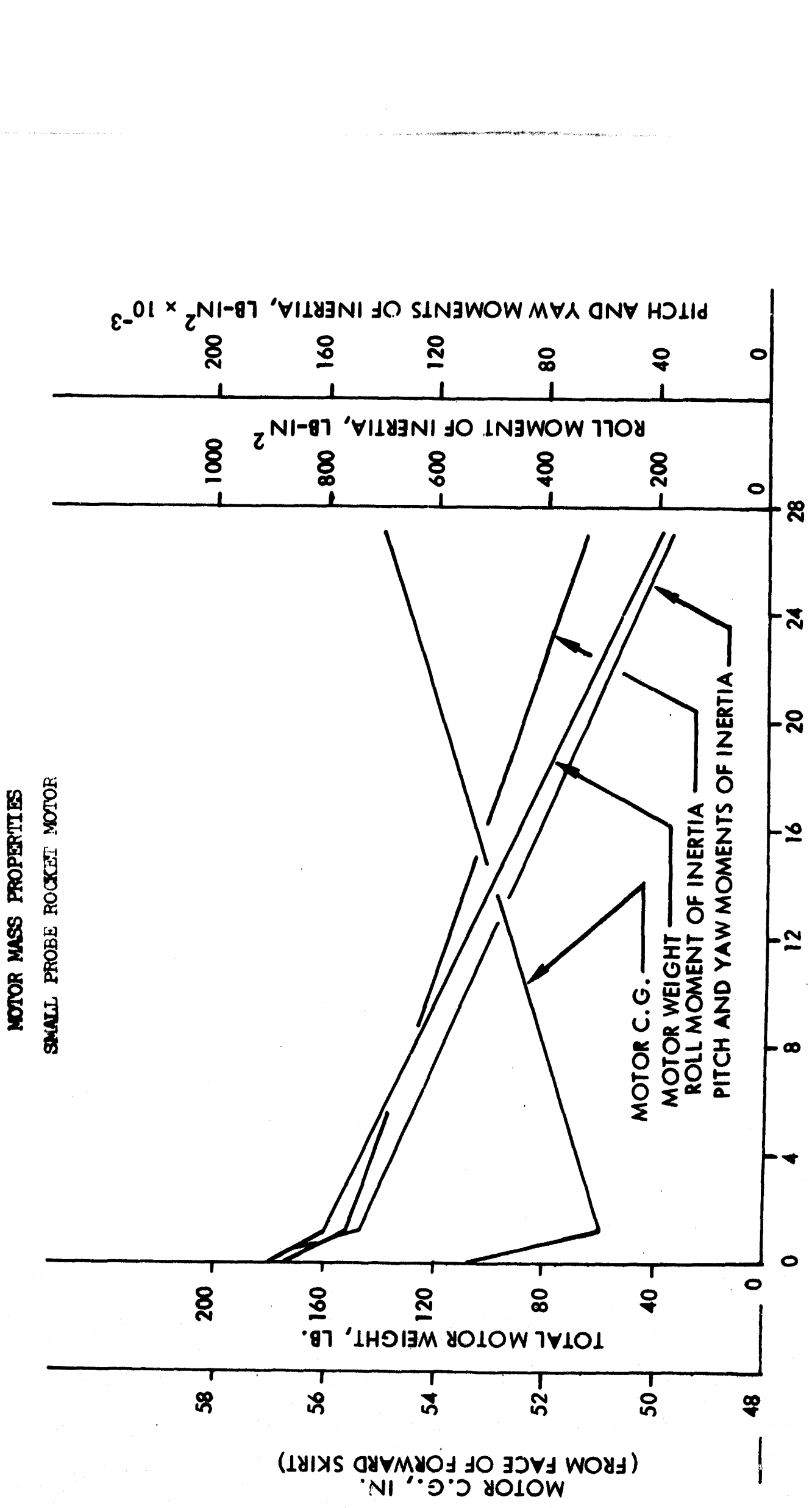


Figure 3.2.2.3


MOTOR MASS PROPERTIES
SMALL PROBE ROCKET MOTOR
MOTOR C.G. (FROM FACE OF FORWARD SKIRT), IN.
TOTAL MOTOR WEIGHT, LB.
ROLL MOMENT OF INERTIA, LB-IN²
PITCH AND YAW MOMENTS OF INERTIA, LB-IN² × 10⁻³
MOTOR C.G.
MOTOR WEIGHT
ROLL MOMENT OF INERTIA
PITCH AND YAW MOMENTS OF INERTIA
TIME, SEC.

at 500 psia. A detailed discussion of the selected propellants is presented in Section 3.2.2.3 of this report. A preliminary stress analysis indicates that the grains would withstand strains resulting from a storage temperature of -65°F.

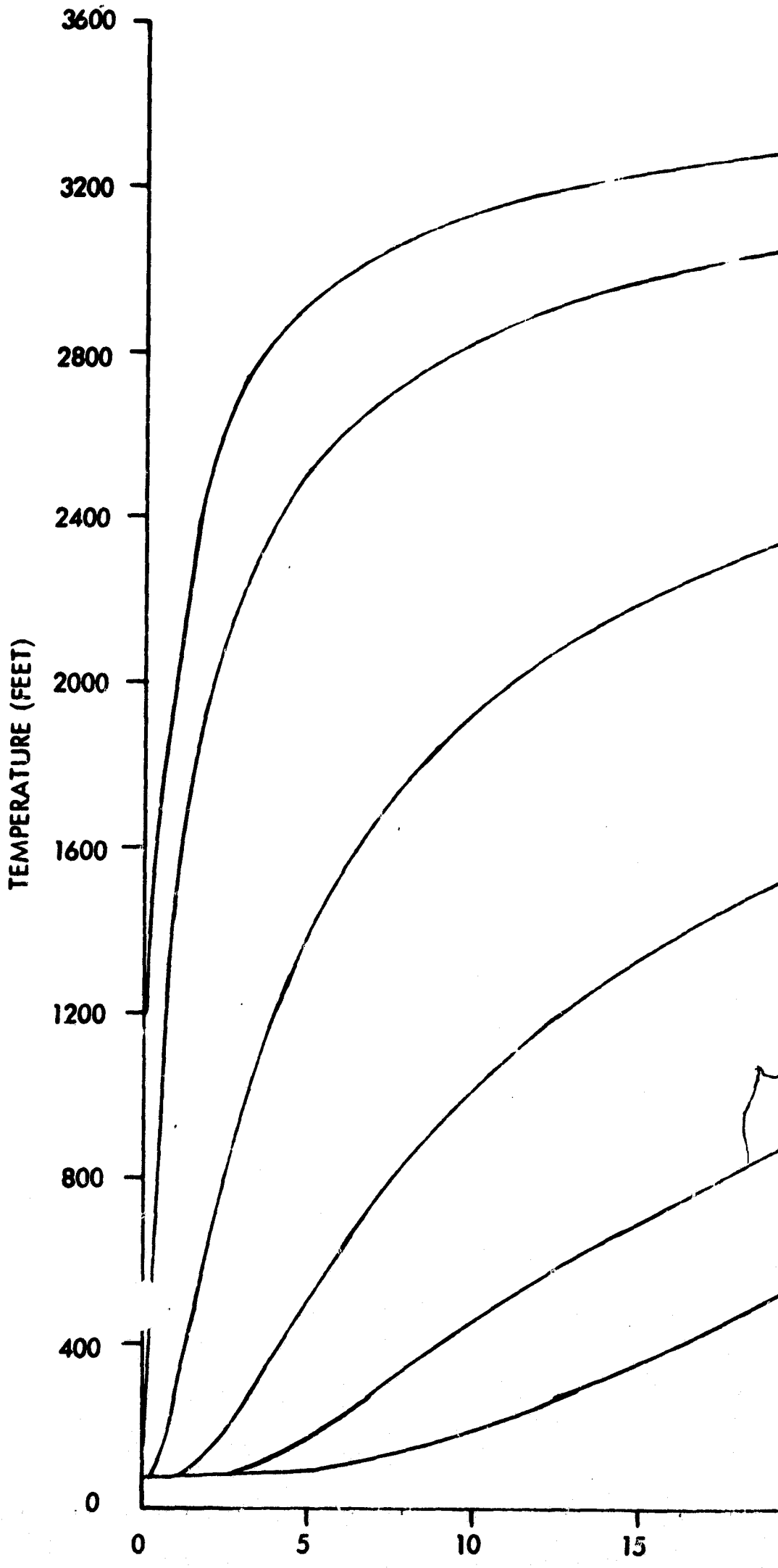
Insulation - The primary insulation material is the Micarta (paper-phenolic) sleeves in which the propellant is cast. These cartridge-loaded grains are separated by V-4010 rubber spacers. Each of the spacers has grooves cut into the aft face to allow pressurization between the steel chamber and the Micarta sleeves. This prevents excessively high grain strains upon pressurization. The chamber interior is protected with paint to protect the surface from rust during storage and to minimize the heat transfer from the exhaust gases that pressurize between the Micarta sleeves and case wall. A V-4010 rubber insulator is used at the forward end of the motor to protect the forward head during firing. A compression spring is also used at the forward end of the motor to assure that the propellant grains are tightly packed together and to compensate for differential thermal expansion and contraction. An acrylic paint is used on the exterior of the motor to protect it from the elements and to provide partial insulation from the aero-dynamic heating that occurs during flight.

A preliminary thermal analysis was performed of the proposed motor design. Figures 3.2.2.5, 3.2.2.6, and 3.2.2.7 show the results of the analysis in terms of maximum temperature vs time at various depths within the Micarta and for the steel skin. Aerodynamic heating was considered in this analysis using the trajectory given in Table 3.2.2.2. Figure 3.2.2.5 shows the temper-ature profile if no exterior insulation paint is used. Figures 3.2.2.6 and 3.2.2.7 show the temperature profiles if exterior paint thicknesses of 0.010 and 0.020 in., respectively, are used. Without exterior insulating paint, the metal skin temperature reached 960°F (Figure 3.2.2.5) at the end of the 27-sec nominal burning duration. With the paint thicknesses of 0.010 and 0.020 in., the corresponding temperatures are 870 and 800°F. Because the metal chamber is sized for the nominal boost pressure of 2000 psia, the

X-SCALE = 1/1.00

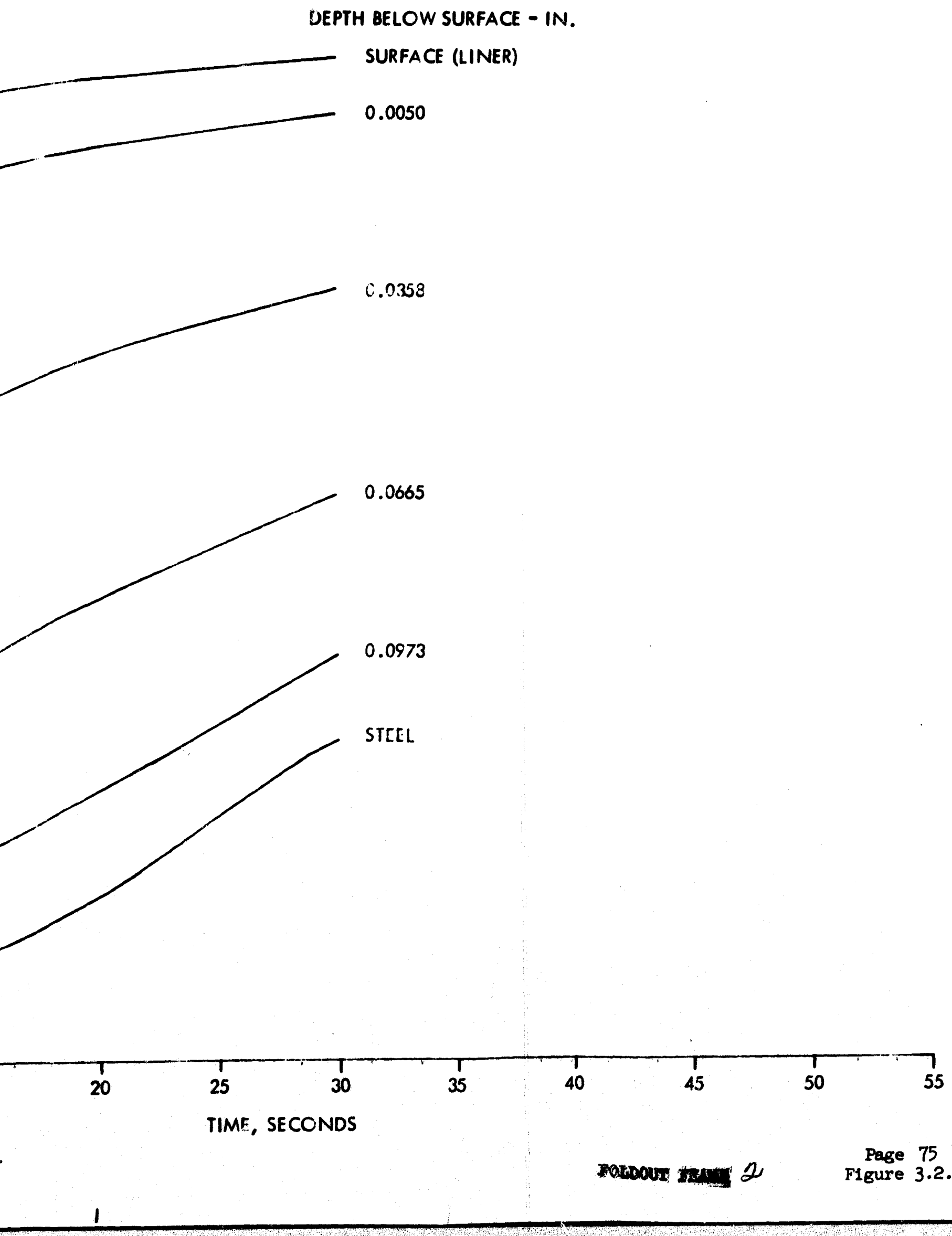
Y-SCALE = 1/1.00

NODE	SYMBOL
1	1
4	2
8	3
12	4
16	5



FOLDOUT FRAME 1

INSULATION TEMPERATURE GRADIENT



FOLDOUT FRAME 2

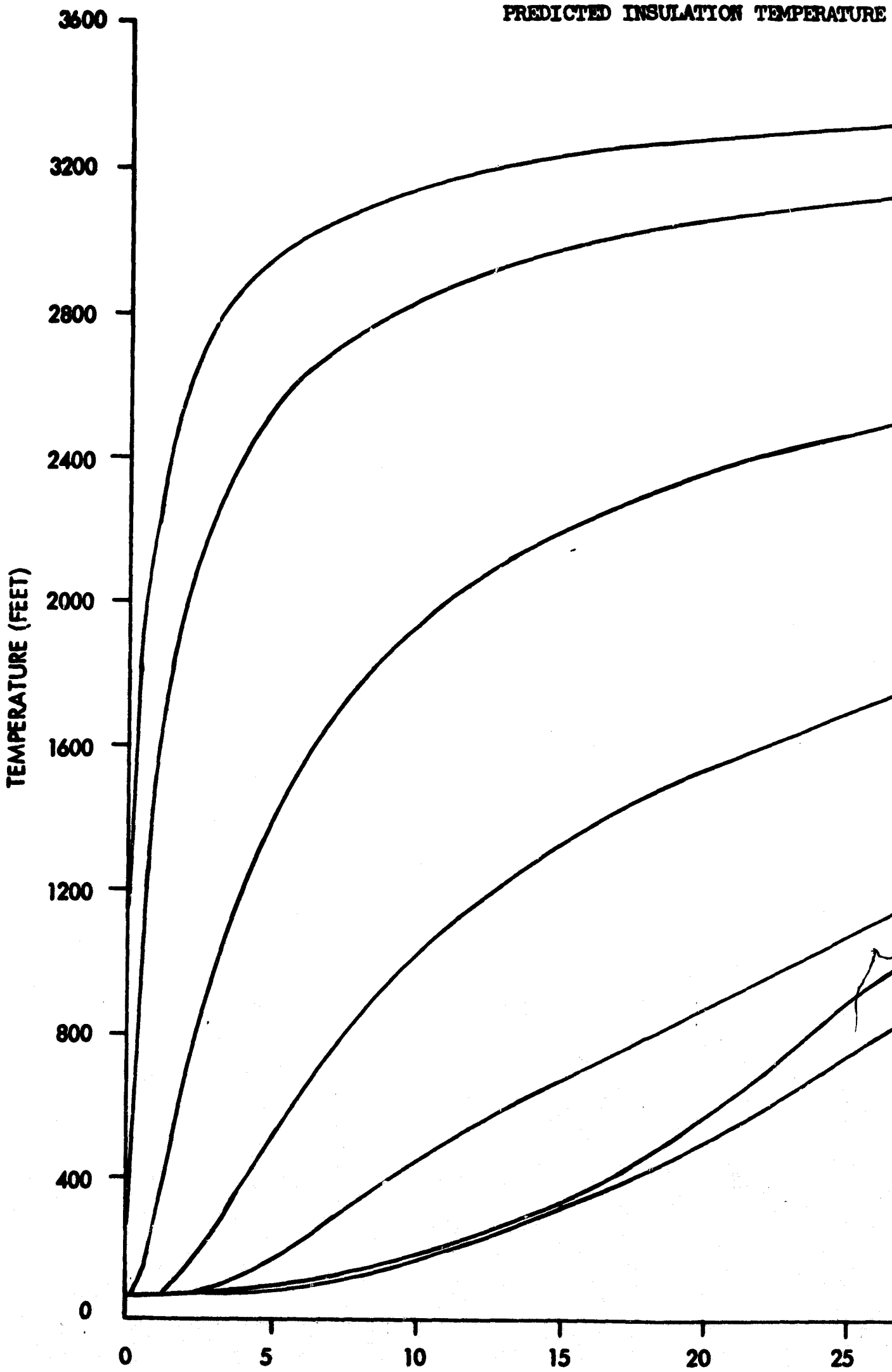
Figure 3.2.2.5

PREDICTED INSULATION TEMPERATURE

X-SCALE = 1/1.00

Y-SCALE = 1/1.00

NODE	SYMBOL
1	1
4	2
8	3
12	4
16	5



NI 90'9 E-1

FOLDOUT FRAME 1

PERATURE GRADIENT

DEPTH BELOW SURFACE - IN.

SURFACE (LINER)

0.0050

0.0258

0.0665

0.0973

SURFACE (PAINT)
STEEL

25 30 35 40 45 50 55

TIME, SECONDS

FOLDOUT FRAME 2

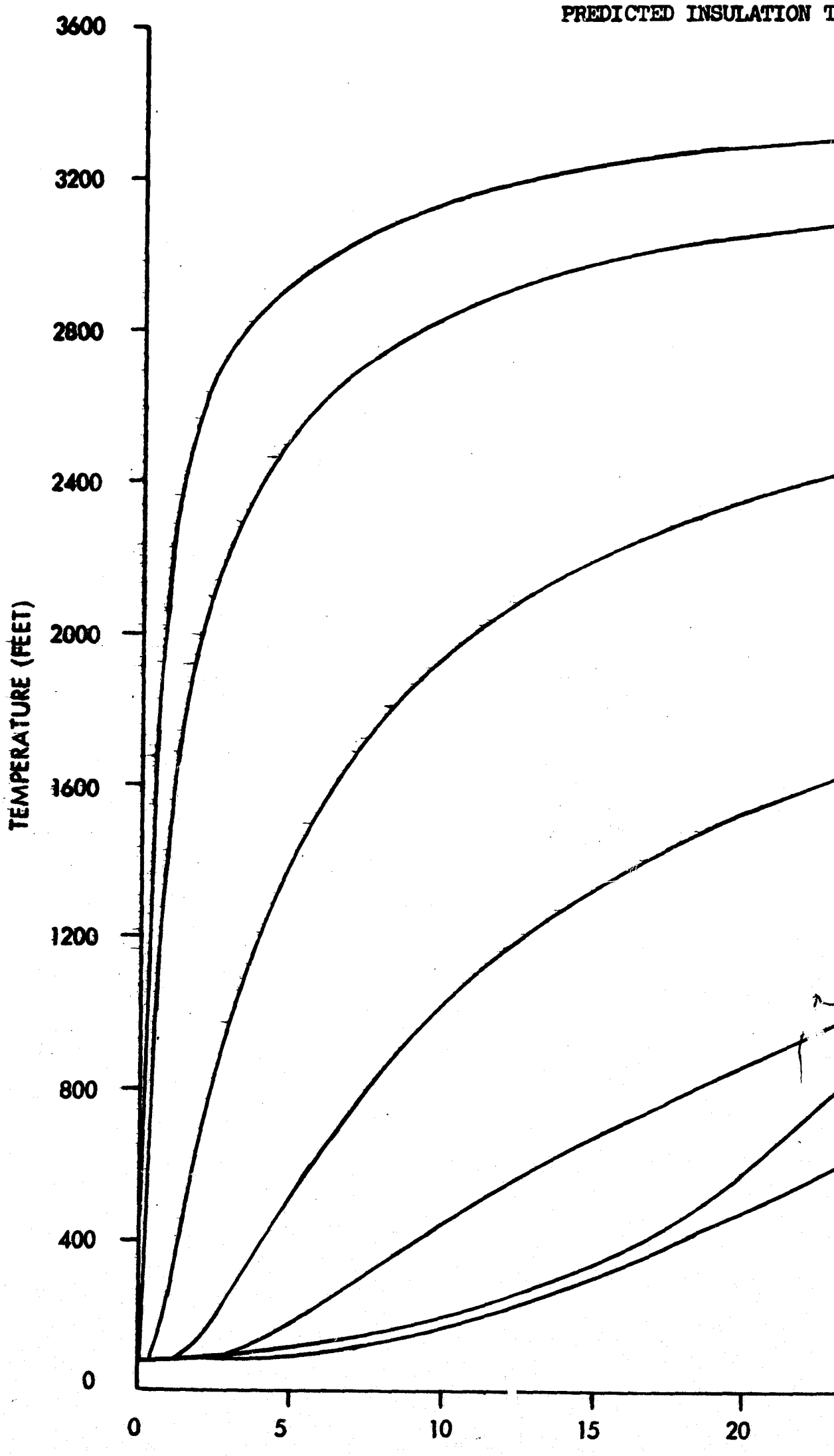
Page 76
Figure 3.2.2.6

PREDICTED INSULATION T

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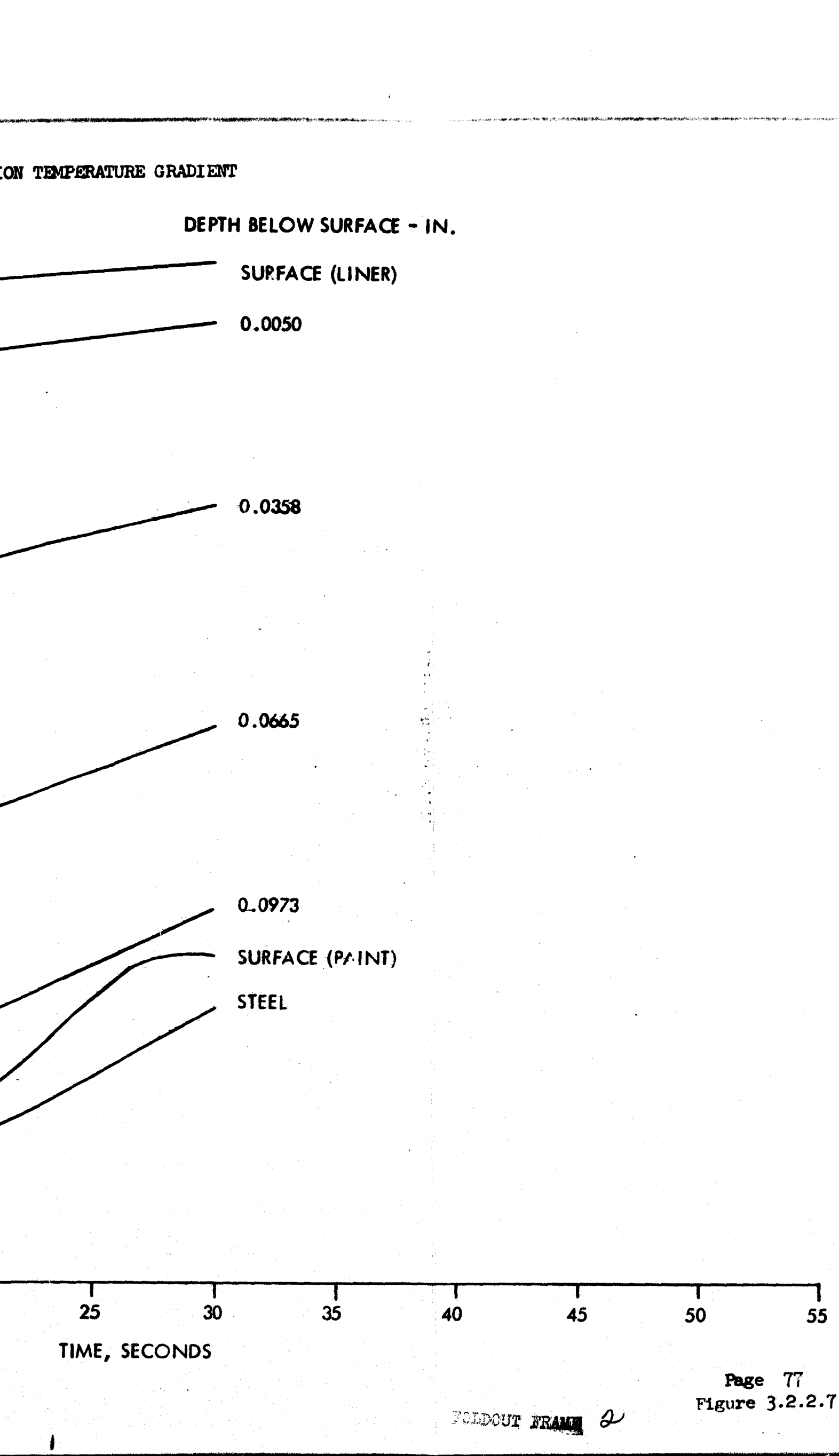
Y-SCALE = 1/1.00

NODE	SYMBOL
1	1
4	2
8	3
12	4
16	5



FOLDOUT FRAME /

ION TEMPERATURE GRADIENT



Page 77
Figure 3.2.2.7

FOLDOUT FRAME 2

Alt-Ft	Vel.-Ft/Sec	Time-Sec	Mach. No.
320	640	1	.57
2748	986	4	.89
16185	2065	13	1.96
31255	3070	19	3.10
64002	5525	27	5.71
114884	4896	37	4.84
208896	4208	58	4.09
421874	2097	128	
483952	754	191.6	

Meteorological Rocket Trajectory
Small Probe Rocket Motor

Table 3.2.2.2

IGNITION TRANSIENT
SMALL PROBE ROCKET MOTOR

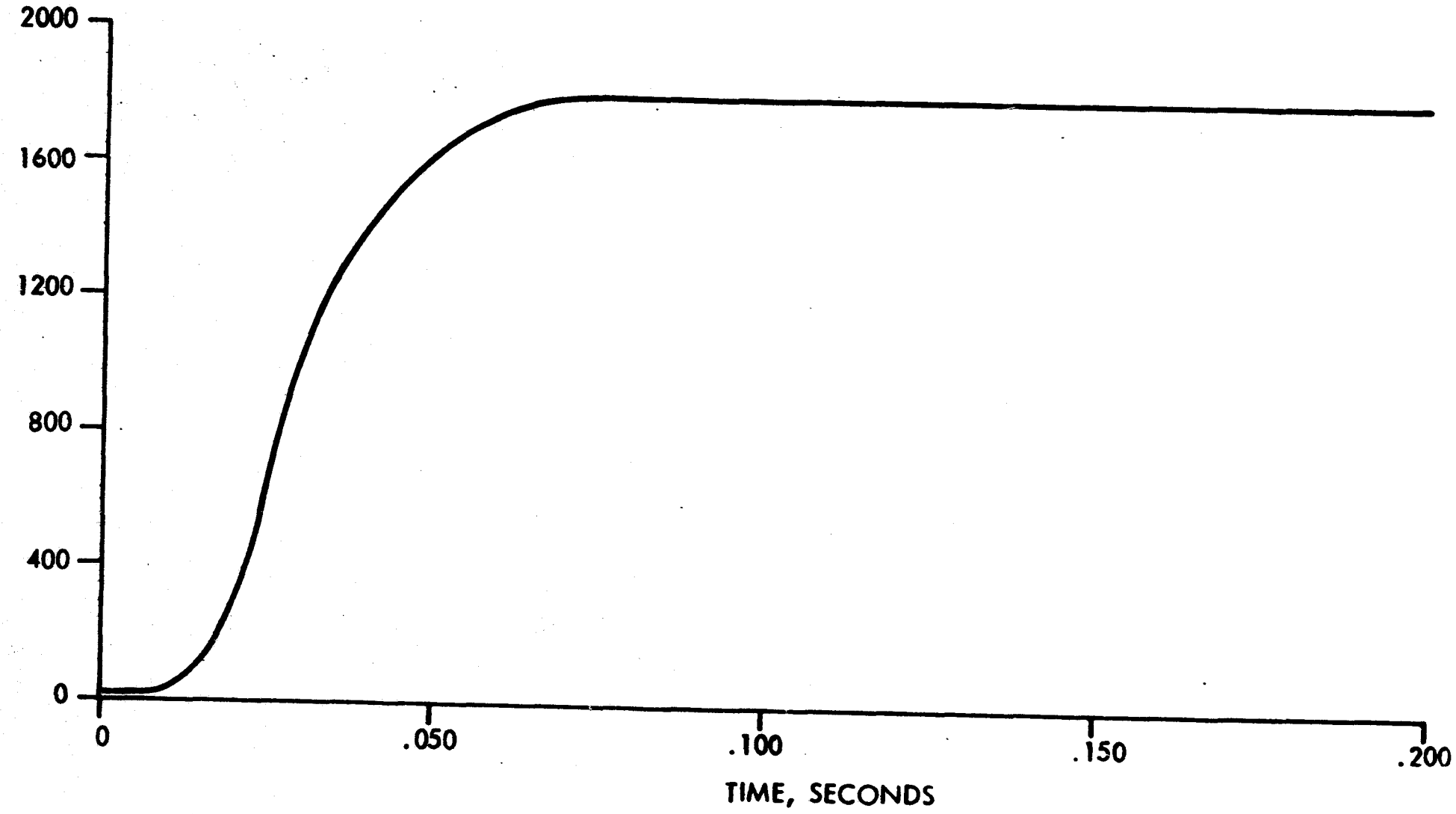


Figure 3.2.2.8

chamber temperature can reach 1100°F before the 500 psia sustain pressure requirement exceeds the material strength properties of AISI 4130 steel at 1100°F. Therefore, an exterior paint coat applied to protect the rocket motor from the weather, will also suffice to protect the chamber from excessive aerodynamic heating and keep the maximum temperature limit within the design allowables.

Chamber - The material selected for the chamber is AISI 4130 steel, heat-treated to an ultimate tensile strength of 180,000 psi. This material was selected on the basis of low cost and high strength at elevated temperatures. The chamber is 6.00 in. O.D. with a 0.042 in. minimum wall thickness. This wall thickness is based on a 1.15 safety factor above MEOP at 165°F. Chamber length is 104.2 in. A stamped, dished dome is welded into the forward end of the motor. This forward weld must then be heat-treated as a result of discontinuity stresses. After the propellant grains have been cartridge-loaded into the chamber and the nozzle inserted, a steel retaining ring is circumferentially welded in place at the aft end of the chamber. This aft closure weld will work satisfactorily in the normalized condition and does not require heat-treatment. Another way to attach the aft closure is to spot weld the steel retaining ring in place instead of the circumferential weld. This method will be evaluated during the development program.

An alternative aft-closure assembly is shown in Figure 3.2.2.1. This design uses a snap ring to retain the nozzle. The disadvantage of this design is that the chamber diameter must be increased locally to 6.1 in. to provide material thickness for the snap ring joint.

A motor design incorporating an aluminum chamber was investigated, using aluminum alloy 2014-T6 having an ultimate tensile strength of 60,000 psi. Because of the lower volume loading resulting from the thicker chamber wall required, the motor length would increase 6 in. over the proposed baseline design. Motor inert weight would increase 2.2 lb because of the longer chamber and the lower strength-density ratio of the aluminum alloy that was evaluated as compared to the 4130 steel. An additional inert weight penalty would also

be incurred because of the heavier insulation required to prevent the aluminum skin temperature from exceeding allowable values.

Nozzle - The nozzle consists of a glass phenolic housing and exit cone with an ATJ graphite throat insert for erosion resistance. The throat diameter is 1.308 in., and the nozzle expansion ratio is 12:1. Further analysis is required to determine the optimum expansion ratio for this motor. A one-piece fin attachment structure is recommended as shown in Figure 3.2.2.1. This structure would slip over the exit cone and be held in place by radial screws. It is recommended that this housing not be attached to the nozzle exit cone because of the stresses incurred as a result of thermal expansion upon firing.

Igniter - The igniter consists of 10 gm of BPN pellets contained within a molded plastic housing. This housing hermetically seals the igniter prior to firing. Upon initiation of the 1-watt, 1-amp, no-fire squib, initial pressurization causes the radially oriented ports to burst, pressurizing the motor bore and providing sufficient heat flux to igniter the propellant grains. The remainder of the plastic housing will melt and disintegrate prior to being expelled from the motor. The igniter is held in place by a "four-fingered" plastic retainer which applies pressure against the grain bore providing a friction fit. Because the grain bore diameter (1.25 in.) is smaller than the throat diameter (1.308 in.) passing the igniter through the throat poses no problem. An ignition analysis shows that the fastest ignition is achieved if the igniter is located at the forward end of the motor. The predicted transient is shown in Figure 3.2.2.8. The ignition delay (time to reach 90% of initial steady-state pressure) is estimated to be 0.050 sec.

PROPELLANT AND LINER

Basis for Selection - Selection of propellants for use in the booster and sustainer grains of the proposed motor was based on fulfillment of the following general requirements in addition to the ballistic performance requirements obtained from preliminary grain designs:

Low raw material costs

Low processing costs

Use of only fully developed and well-characterized formulations

A storage temperature range capability of -40 to +165°F

An operational temperature range capability of -65 to +165°F

A Class 2 explosive hazard rating

Existing production formulations and propellants ready for production were evaluated in terms of the stated requirements. Propellants formulated with carboxy-terminated polybutadiene binders were eliminated from consideration, principally because of high raw materials costs. Formulations based on PBAN binders, although very low cost systems, were rejected because no existing propellants exhibited ballistic properties within or near the ranges indicated as necessary by analysis of preliminary grain designs. Propellant selection was, therefore, made from the large family of polyurethane formulations available at Aerojet. A high burning rate ammonium perchlorate/aluminum/polyurethane formulation developed for use in an air-launched rocket was selected as the boost propellant, and a very low burning rate ammonium perchlorate/nitroguanidine/polyurethane production propellant was chosen for use in the sustain grains of the proposed motor. These Class 2 propellants, which exhibit the desired combination of low cost and satisfactory mechanical properties over a wide temperature range, are characterized by the following ballistic properties:

	Booster Propellant	Sustainer Propellant
Delivered Specific Impulse, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia	244.0	217.6
Burning Rate, in/sec at 80°F		
Boost Phase at 2000 psia	1.8-2.0	0.14
Sustain Phase at 500 psia	N.A.	0.08
Burning Rate Pressure Exponent, n		
Below 500 psia	0.47	0.09
Above 500 psia	0.47	0.37
Temperature Sensitivity, π_K, %/°F	0.225	0.14
Density, lb/cu. in.	0.0663	0.0591

Motor grain designs based on these values proved satisfactory with respect to motor performance and the propellants were selected as final candidates.

A preliminary stress analysis of the final grain design indicated that the propellants had mechanical properties which would assure grain stability under all storage and operational environments. The stress analysis also indicated that the SD-886 polyurethane liner used with the booster propellant and the SD-793 polyurethane liner used with the sustainer propellant provided bond strengths in excess of those required to provide propellant-liner bond integrity. In early phases of the program, SD-886 and SD-793 liners will be used with the booster and sustainer propellants, respectively, to fully utilize past experience and assure strong system bonds. During the development program effort will be expended to provide a more efficient system.

Booster Propellant - The booster propellant is a C-1 polyurethane formulation developed for use in an air-launched missile designed for operation over a temperature range of -65 to +160°F. This high burning rate formulation has the following composition:

	Wt.,%
Ammonium Perchlorate	68.00
Aluminum	18.20
Copper Chromite	2.00
C-1 Polyurethane Binder	11.80
	100.00

The use of ammonium perchlorate of small average particle size and 2 wt% of the burning rate catalyst copper chromite permits attainment of a 2.0 in/sec burning rate at 2000 psia. The C-1 polyurethane binder provides propellants with outstanding mechanical properties, particularly at low temperatures. When C-1, the compound 2,3-dihydroxypropyl bis-(2-cyanoethyl) amine, is incorporated into the polyurethane binder structure at a low concentration, the oxidizer/binder bond is strengthened because dewetting is reduced. As a result, both propellant tensile strength and elongation are increased significantly. The low temperature mechanical properties of C-1 polyurethane propellants are not approached by those of any other available propellant system.

The propellant selected is a fully developed formulation which has been well characterized with respect to ballistic, mechanical, storage, processing, and safety properties. Mechanical property characterization was extensive and included tests under a variety of conditions. The propellant is readily processible, has a low initial viscosity, a long potlife, and reaches an equilibrium cure in only 36 hr at 135°F. Safety tests have shown the propellant to be a Class 2 explosive hazard formulation with an I.C.C. shipping classification of Class B. Propellant properties are summarized in Table 3.2.2.3. The propellant will be used without modification in the cartridge-loaded booster grain of the proposed motor.

Experience in a number of motor programs has demonstrated the performance capabilities, reproducibility, and reliability of C-1 polyurethane propellants containing from 82 to 88.2 wt% ballistic solids. All raw materials

Propellant Composition	Weight %
Ammonium perchlorate	68.00
Aluminum (Class 1)	18.20
Cu 0202	2.00
C-1 polyurethane binder	11.80
	100.00

Ballistic Properties

Burning rate at 2000 psi, in./sec	
liquid strands	2.0
3KS-500 motors	2.0
π_K, %/°F, -65°F to +160°F	0.225
I_s Standard, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia	244.0
Mass flow coefficient, C_w, lbm/lbf-sec	0.00629
Flame Temperature, °F at 1000 psia	
Chamber	6048
Exhaust	3714
Propellant Density, lb/in.^{3}	0.0663

Mechanical Properties	Temp, °F	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
	-65	450	15	34	4973
	+77	132	26	28	696
	+160	83	25	27	453

Bonding Properties to SD-886 Liner	Temp, °F	Tensile, psi	Shear, psi
	-65	585	493
	+77	110	74
	+160	64	42

Safety Properties

Autoignition Temp, °F	
Copper block	519
DTA	600
Bureau of Mines Impact Sensitivity, cm/2kg (50% firepoint)	24
I.C.C. Shipping Classification	B

Summary of Booster Propellant Properties

Table 3.2.2.3

used in these propellants are available from the chemical industry in large quantity and at the required levels of purity. Raw material, processing, and acceptance specifications established for each formulation assure reproducibility of propellant properties.

SD-886 polyurethane liner which was developed for use in conjunction with the proposed booster propellant, is a sprayable formulation that contains 2 wt% titanium dioxide and 3 wt% carbon black as inert fillers. This formulation is a modification of SD-810 liner, a spinnable liner used with complete success in the Polaris A-2 first stage and other motors. The only difference between SD-886 and SD-810 liners is the presence of 3 wt% carbon black in SD-886 to prevent void formation on spraying. Bond properties of the two liners to the proposed booster propellant are comparable.

Sustainer Propellant - The sustain propellant selected for use in the cartridge-loaded grains of the proposed motor is a production formulation with the following composition:

	Wt.,%
Ammonium Perchlorate	61.00
Nitroguanidine	21.00
Polyurethane Binder	18.00
	100.00

To date, over 1000 full-scale (2150-lb) batches of this low-burning-rate formulation have been produced and over 200 succsssful motor tests, both static and flight, have been conducted. These firings were conducted at temperatures between +20 and +130°F with unaged motors and motors aged over the temperature range of -10 to +130°F. The propellant has also been qualified for use in another motor with storage and operational temperature range requirements of -65 to +160°F. In both motors, the propellant is used in conjunction with SD-793 polyurethane liner, the liner selected for use in the sustain grains of the proposed motor. Propellant properties are summarized in Table 3.2.2.4.

Propellant Composition	Wt %
Ammonium Perchlorate	61.00
Nitroguanidine	21.00
Polyurethane Binder	18.00
	100.00

Ballistic Properties

Burning Rate, in./sec in 3KS-500 size motors	
at 500 psia	0.08
at 2000 psia	0.14
I_s Standard, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia	217.6
Mass Flow Coefficient, C_w, lbm/lbf-sec	0.00688
Flame Temperature, ^oF at 1000 psia	
Chamber	3694
Exhaust	1355
Density, lb/cu in.	

Mechanical Properties	Temp. ^oF	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
	-75	1160	2.4	2.9	62,500
	+77	99	17	25	834
	+160	62	13	20	647

Bonding Properties to SD-793 Liner	Temp, ^oF	Tensile, psi	Shear, psi
	-75	128	329
	+77	45	40
	+160	35	34

Safety Properties

Autoignition Temperature, ^oF	585
Bureau of Mines Impact Sensitivity, cm/2kg (50% firepoint)	41
I.C.C. Shipping Classification	B

Summary of Sustainer Propellant Properties

Table 3.2.2.4

The sustain propellant and the liner are readily processible, formulated with materials available in large quantity, well characterized, and are reproducible systems which will be used without modification. All necessary specifications, procedures, and acceptance criteria necessary for the production of reproducible and reliable sustainer grains are available and have been proven.

Propellant Liner System Evaluation - A limited effort will be required (a) to assure that the selected boost/liner and sustain/liner systems will perform satisfactorily in the cartridge-loaded grains of the proposed motor and (b) to provide the most economical cartridge lining procedure. To date, neither propellant/liner system has been extensively tested in a Micarta sleeve and the nature of the Micarta/liner/propellant bond system has not been firmly established. Experience has indicated that the bond between liners and dried Micarta sleeves is significantly increased by application of a thin coating of FM-47 primer to the Micarta prior to liner application. In addition, experience with the booster propellant/SD-886 liner system has shown that a diisocyanate-cure catalyst washcoat applied to the liner before propellant casting enhances bond strength. It will, therefore, be necessary to determine (a) if the FM-47 prime coat can be eliminated; (b) if a washcoat is required; (c) if one liner, either SD-886 or SD-793, can be used with both the booster and sustainer propellants; and (d) if the propellants can be bonded directly to primed Micarta and the use of a liner eliminated. These investigations will be conducted, for the most part, with small laboratory test specimens to permit rapid screening. Promising approaches will then be evaluated more extensively under a variety of test conditions and the best approach will be incorporated into the motor processing procedure.

Booster Propellant Properties - The boost propellant is characterized by the following properties:

a. Ballistic Properties

Delivered Specific Impulse, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia

Measured in 3KS-1000 size (16-lb grains) motors	244.0
Burning Rate, in/sec at 2000 psia and 80°F	1.8-2.0
Burning Rate Pressure Exponent, n	0.47
Temperature Sensitivity, π_K,%/°F	0.225
Flame Temperature, °F, at P_c/P_e = 1000/14.7 psia	
Chamber	6048
Exhaust	3714
Density, lb/cu in.	0.0663

Compositions of the propellant chamber and exhaust gases at P_c/P_e = 1000/14.7 psia are presented in Table 3.2.2.5. As shown above, the specific impulse of booster propellant has been measured in 3KS-1000-size test motors which are cast with a grain weighing approximately 16 lb. The results from these firings provide a good measure of the specific impulse which can be expected from the booster grain of the proposed motor. The test results for the firings conducted with the booster propellant at +80°F, the standard test temperature, and at both -65 and +160°F are presented in Table 3.2.2.6.

The burning rate, pressure exponent, and temperature sensitivity (π_K) of the booster propellant were established by 3KS-500-size (6-lb grain) motor firings at -65, 80, and 160°F. As shown in Figure 3.2.2.9, these firings were conducted over a broad pressure range and the test data demonstrate that the propellant burning rate-pressure relationships is linear at pressures well above and below the operating pressure of the proposed motor design.

b. Propellant Mechanical Properties

Extensive laboratory testing has provided a well defined characterization of the mechanical properties of the booster propellant with respect to a variety of formulation, environmental, storage, and operational parameters. Tensile properties (Table 3.2.2.7) show the good propellant mechanical properties over the temperature range of -65 to +160°F at strain

	P_c/P_e = 1000/14.7 psia	
	Chamber at 6048°F	Exhaust at 3714°F
HCl	0.4574	0.5635
N_2	0.2920	0.2951
H_2O	0.7022	0.7077
H_2	0.7776	0.8159
O_2	0.0025	-
O	0.0071	-
OH	0.0673	0.0025
Cl	0.0579	0.0087
NO	0.0061	0.0001
H	0.1464	0.0170
Cl_2	0.0001	-
CO	0.6016	0.5786
CO_2	0.0656	0.0885
AlH	0.0001	-
Al	0.0005	-
AlCl	0.0123	0.0001
$AlCl_2$	0.0233	0.0006
$AlCl_3$	0.0005	-
AlO	0.0005	-
Al_2O	0.0001	-
Al_2O_3	0.3187(s)	0.3370(s)
Cu	0.0210	0.0210
S	0.0005	0.0002
HS	0.0007	0.0005
H_2S	0.0003	0.0007
SO	0.0012	0.0008
SO_2	0.0005	0.0008
Cr	0.0017	0.0003
CrCl	0.0025	0.0030
$CrCl_2$	0.0001	0.0010
Total, mol/100g	3.2496	3.1066

Theoretical Equilibrium Gas Compositions for
Booster Propellant

IMPULSE FIRING DATA FOR BOOSTER PROPELLANT[1]

Table 3.2.2.6

Grain No.	Temperature, °F	Ave. Pressure, psia	I_s at Pressure	I_s (std)
66-517-3	70	1643	254.2	244.9
66-517-4	80	2167	258.5	244.4
66-517-6	80	2224	258.0	243.6
66-517-12	80	2173	257.8	243.7
66-517-1	-65	1278	242.6	238.8
66-605-10	-65	1606	251.9	243.6
66-517-2	160	1910	256.7	244.8
66-517-7	160	2614	261.6[2]	244.7[2]
66-517-9	160	2208	257.4	243.1

(1) 16 lb. test charge fired at 15.4/1 expansion ratio.
(2) Flexure damaged, data questionable.

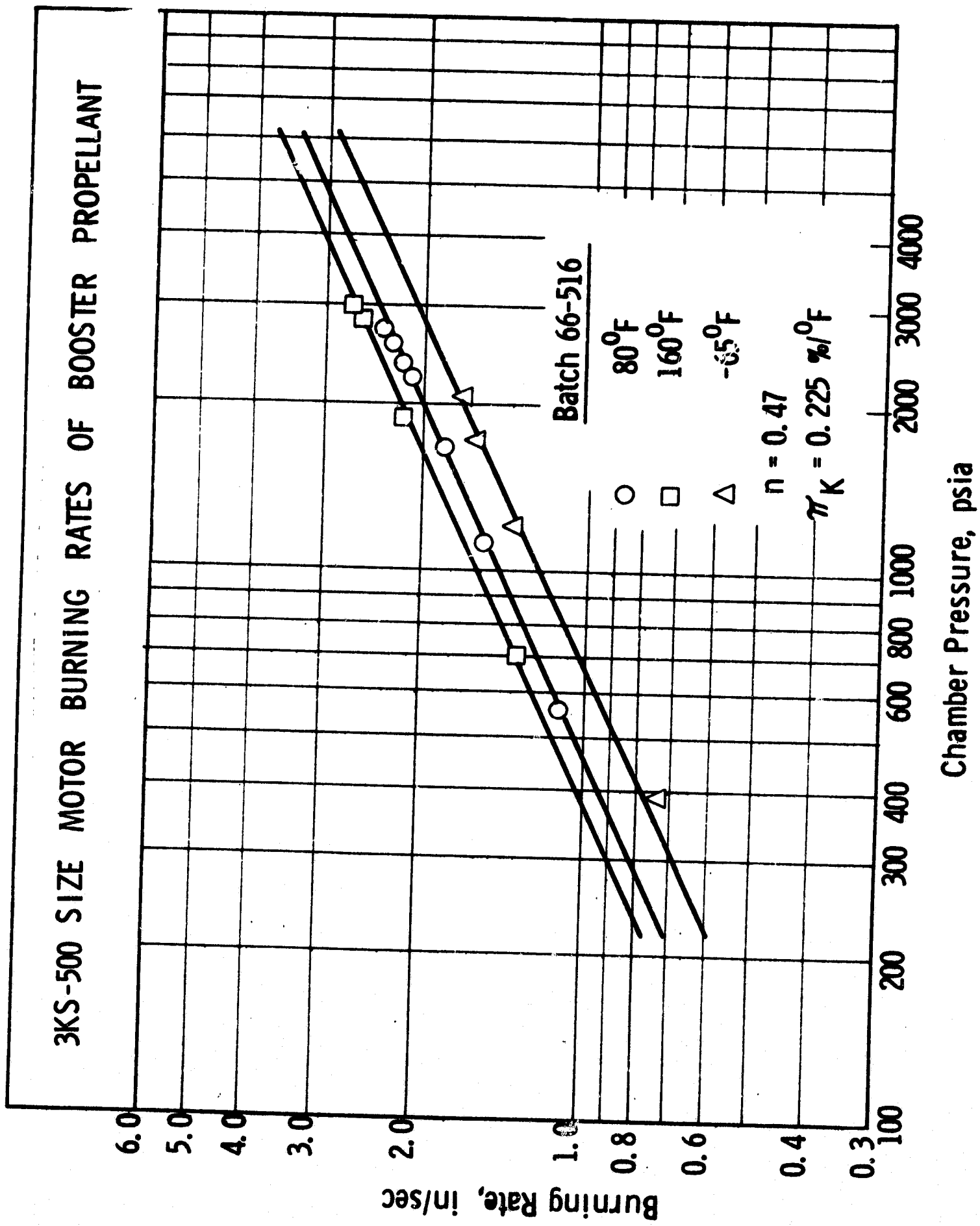

3KS-500 SIZE MOTOR BURNING RATES OF BOOSTER PROPELLANT
Batch 66-516
80°F
160°F
-65°F
n = 0.47
π_K = 0.225 %/°F
Burning Rate, in/sec
Chamber Pressure, psia
6.0
5.0
4.0
3.0
2.0
1.0
0.8
0.6
0.4
0.3
100
200
300
400
600
800
1000
2000
3000
4000

Figure 3.2.2.9

MECHANICAL PROPERTIES OF BOOSTER PROPELLANT

Test Temp, °F	Mechanical Properties at Various Strain Rates											
	0.074 min-1				0.74 min-1				7.4 min-1			
	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
-65(1)	366	15	27	3870	493	16	31	5306	736	17	26	8067
77(1)	125	22	23	681	143	26	28	751	168	25	28	1028
160(1)	74	23	25	385	95	26	29	501	114	27	29	623
-65	361	12	32	4326	450	15	34	4973	792	16	22	10350
-40	-	-	-	-	279	29	38	2612	-	-	-	-
0	-	-	-	-	193	26	33	1445	-	-	-	-
40	-	-	-	-	183	27	31	1071	-	-	-	-
77	110	23	24	570	132	26	28	696	150	28	32	863
110	-	-	-	-	115	25	25	590	-	-	-	-
160	69	22	24	383	83	25	27	453	107	24	25	612

(1) Samples were conditioned at 0% RH for 24 hours prior to testing

Table 3.2.2.7

rates between 0.074 and 7.4 min^{-1} (crosshead rates of 0.2 to 20.0 in/min). The data in Table 3.2.2.7 also show that conditioning of test specimens for 24 hr at 0% relative humidity prior to testing had no measurable effect on the properties. Normal sample conditioning is carried out at a relative humidity of approximately 30%. C-1 polyurethane propellants, like the boost propellant, typically exhibit their highest elongations at low temperatures where a premium is normally placed on high strain capability.

The maximum constant strain held in a 168-hr test at +40°F and 70% relative humidity is a criterion employed to determine the ability of a propellant to withstand the strains imposed on a grain during temperature cycling in storage and during operation. Constant strain tests conducted with the boost propellant demonstrated that propellant will withstand 20% strain for 168 hr without failure.

The most severe test of propellant strain capability is that imposed by suddenly chilling a case-bonded grain from a high temperature to a very low temperature. Boost propellant was cast into strain evaluation cylinders to give grains 5-in. in diameter and 20 in. long with a cylindrical bore. When these grains were cooled to -65°F, the inner-bore hoop strain was 10%. The grains withstood three cycles between +160 and -65°F without failure.

The boost propellant, like all C-1 polyurethane formulations, shows excellent resistance to embrittlement. This is illustrated in Table 3.2.2.8 by data from tensile specimens of the boost propellant conditioned at +77°F for two weeks at relative humidities ranging from 0 to 70% prior to testing at a strain rate of 0.74 min^{-1} and at temperatures of -65, +77, and +160°F. In a supplemental test, specimens were subjected to additional conditioning at 0°F after exposure to a humid environment and were then tested at -65°F. As shown (Table 3.2.2.7) propellant elongation (ϵ_m and ϵ_b) was unaffected by storage conditions.

EFFECT OF MOISTURE ON EMBRITTLEMENT OF BOOSTER PROPELLANT

Storage Conditions			Test Temp,	Mechanical Properties at Strain Rate of 0.7h min^{-1}			
Days	°F	% RH	°F	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
14	77	0	-65	485	13	28	6298
		30		471	12	31	6253
		50		460	11	26	7290
		70		506	12	26	7966
14	77	0	77	140	22	24	908
		30		120	20	22	769
		50		107	24	28	625
		70		95	24	27	564
14	77	0	160	90	24	26	533
		30		75	23	25	445
		50		67	23	25	394
		70		57	24	29	409
14	77	0	-65 (after 4 days storage at 0°F)	491	13	33	6137
		30		483	12	31	6701
		50		510	11	22	8173
		70		561	12	23	9460

Table 3.2.2.8

c. Propellant/Liner System Bond

The liner initially selected for use with the boost propellant was SD-810, an epoxy-polyurethane formulation. This spinnable liner bonds well to rubber insulations and to polyurethane propellants. As shown in Table 3.2.2.9, the tensile and shear strengths of the boost propellant-liner bond are good over the temperature range of -65 to +160°F. During testing, the propellant-liner specimens broke within the body of the propellant and, therefore, the interfacial bond strength is greater than that of the propellant. The data in Table 3.2.2.9 also show that the bond strength generally increased on aging at +180°F for up to 35 weeks.

To expedite the lining of chambers on a program with a high production rate, a sprayable modification of SD-810 liner was developed by adding 3 wt% of carbon black to the formulation. This modified liner, designated SD-886, provided bond strengths comparable to those of SD-810 (Table 3.2.2.10). The storage stability of SD-886 is expected to be essentially identical to that of SD-810.

d. Stability of Propellant Mechanical Properties on Long-Term Storage

Accelerated aging tests at +180°F have shown that the storage stability of the boost propellant is excellent. As illustrated in Table 3.2.2.11, the mechanical properties of the propellant at -65, +77, and +160°F are essentially unchanged after 12 wk storage at 180°F. Stability of this nature is indicative of a very long useful life at ambient temperatures.

The storage life of polyurethane propellant has been demonstrated to be in excess of 5 yr in both accelerated aging tests at high temperatures and in successful full-scale motor firings after long-term storage at ambient temperatures. The use of an Arrhenius plot for the extrapolation of data from accelerated aging tests conducted at 150, 180, and 220°F to ambient temperature is a well established industry technique for estimation of long-term storage stability at lower temperatures. As shown in Figure 3.2.2.10, successful motor firings with Polaris and E-8 Hawk motors after 5 yr or longer

EFFECT OF 180°F AGING ON BOND STRENGTH OF
LINER SD-810 TO BOOSTER PROPELLANT

Days at 180°F	Test Temp., °F	Bond Strength, psi*	
		Tensile	Shear
0	-65	565	462
	77	81	58
	160	48	32
14	-65	718	620
	77	122	78
	180	80	46
21	-65	740	679
	77	135	78
	180	80	48
28	-65	624**	571**
	77	132	68
	180	88	56
35	-65	724	646
	77	140	84
	180	95	60

* Each value average of two samples; samples were composite of steel plate, epoxy adhesive, Gen Gard V-44 rubber, SD-810, and propellant.

** Failure occurred in insulation

Table 3.2.2.9

COMPARISON OF COMPOSITIONS AND BOND PROPERTIES
OF LINERS SD-810 AND SD-886

	Liner SD-810	Liner SD-886
Composition, wt%		
Titanium Dioxide	2.00	2.00
Carbon Black	—	3.00
Binder	98.00	95.00
	100.00	100.00
Method of Application	Spin or Brush	Spray

	Bond Properties to Sustainer Propellant			
Test Temp. °F	Tensile Strength, psi	Shear Strength, psi	Tensile Strength, psi	Shear Strength, psi
-65	587	466	585	493
77	110	77	110	74
160	65	45	64	42

Table 3.2.2.10

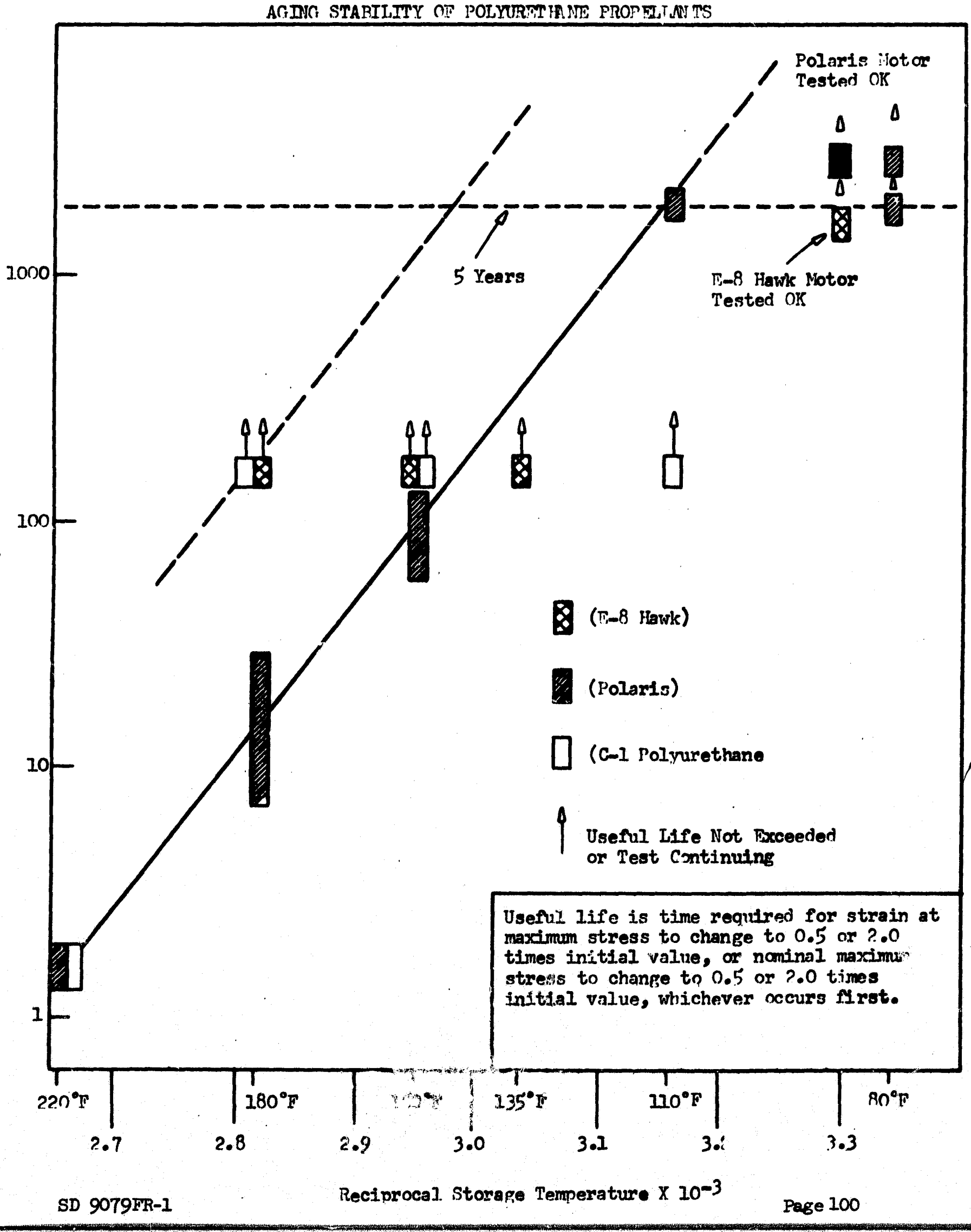
EFFECT OF 180°F AGING ON MECHANICAL PROPERTIES OF BOOSTER PROPELLANT

Test Temp, °F	Property	Weeks at 180°F 0	1	2	3	4	6	8	10	12
-65	σ_m, psi	493	496	435	433	522	457	502	537	424
	ϵ_m, %	16	17	16	16	16	15	16	16	14
	ϵ_b, %	31	29	32	36	30	32	29	29	27
	E_o, psi	5306	6118	5482	5156	7210	5486	7198	6529	5837
77	σ_m, psi	143	124	121	121	119	120	131	126	117
	ϵ_m, %	26	27	26	26	28	24	27	30	27
	ϵ_b, %	28	30	28	28	30	26	29	33	27
	E_o, psi	751	631	646	656	613	759	671	633	669
160	σ_m, psi	95	93	101	93	98	94	100	106	103
	ϵ_m, %	26	27	31	28	26	30	24	26	28
	ϵ_b, %	29	29	32	29	28	30	25	27	28
	E_o, psi	501	427	445	425	475	427	537	495	475

Strain rate of 0.74 in./in./min.

Table 3.2.2.11

AGING STABILITY OF POLYURETHANE PROPELLANTS



storage at 80°F have confirmed the estimation of storage life based on accelerated aging tests. Data shown for the C-1 polyurethane propellant, ANP-2952, which contains 85 wt% ballistic solids, demonstrated that C-1 formulations are also characterized by long-term stability and can be expected to have a service life in excess of 5 yr.

e. Adequacy of Propellant and Propellant/Liner System Mechanical Properties

Since only a very preliminary stress analysis has been carried out for the boost-grain design, no meaningful comparison of propellant and propellant-liner bond allowables with grain design requirements can be made. Such a comparison will be made during the program after a comprehensive stress analysis of the final grain design has been conducted. However, on the basis of the preliminary analysis, it appears that the allowables for the unaged and aged systems will be excess of the calculated requirements and provide very satisfactory safety margins.

f. Processing Properties

C-1 polyurethane formulations are processed with simple procedures developed to provide maximum system reproducibility and economy of production. Before processing, all raw materials are fully characterized through extensive laboratory qualification tests and a master-batch of premix is prepared and qualified. The premix contains the aluminum, ballistic additives, and all binder ingredients other than the diisocyanate and cure catalyst. Qualification criteria for the raw materials and the premix are derived from analyses of the effects of component purity and minor premix composition variables on propellant properties. Propellant processing is carried out by (1) charging the mixer with premix, (2) addition of the oxidizer, and (3) addition of mixture of the isocyanate and the cure catalyst. The reproducibility inherent in the raw materials employed, and the simple processing procedures that can be used lead to a highly reliable and reproducible product. Acceptance specifications for all of the raw materials used in the booster propellant have been fully defined, as have the basic processing procedures and product-acceptance criteria.

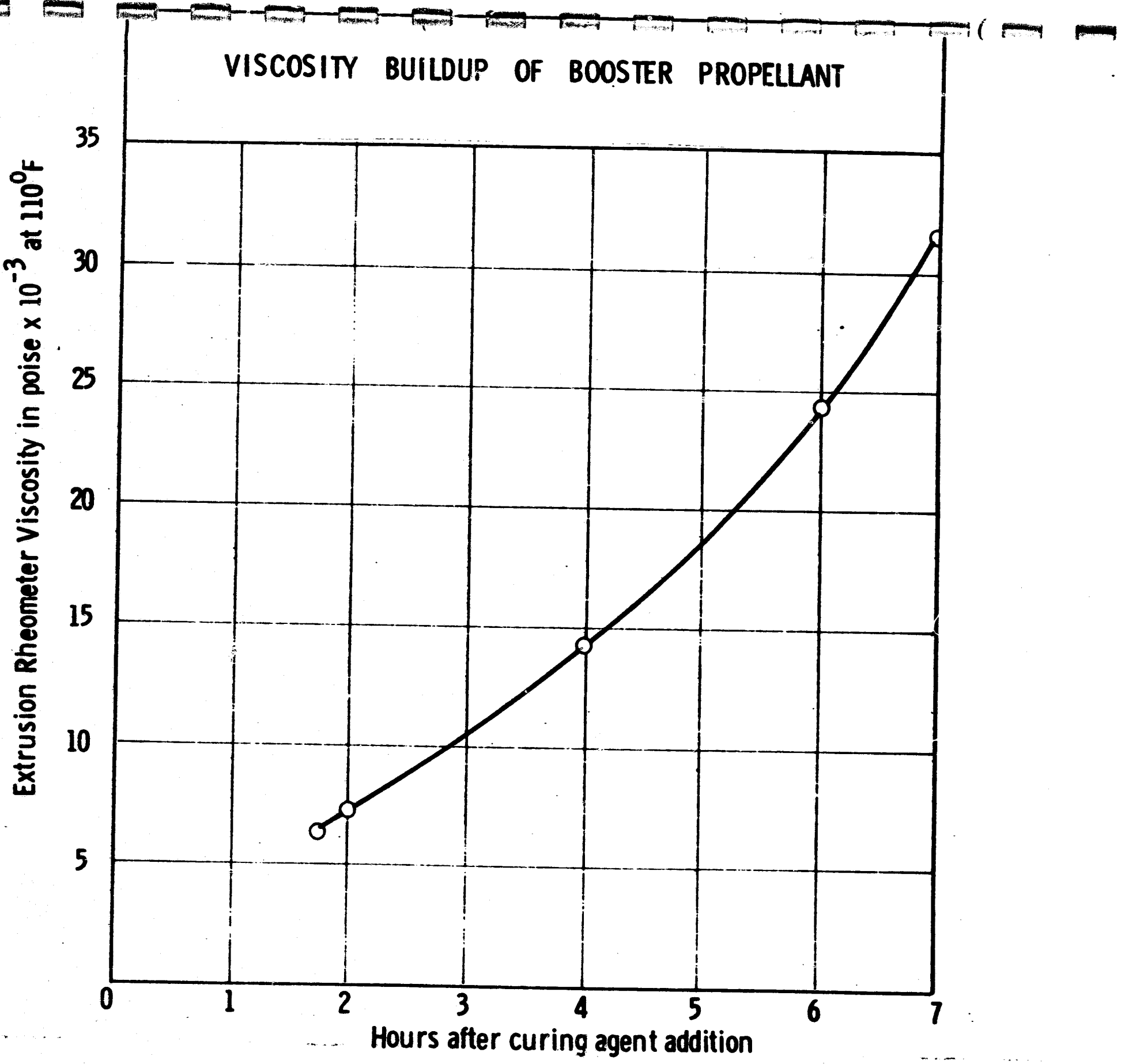


Figure 3.2.2.11

REPRODUCIBILITY OF BOOSTER PROPELLANT PROPERTIES

Batch No.	Liquid Strand Burning Rate at 2250 psig in./sec Rate	n	Mechanical Properties at Strain Rate of 0.74 min^{-1} -65°F σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi	77°F σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi	160°F σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
800-lb*														
66-516	2.10	0.55	450	15	34	4973	132	26	28	696	83	25	27	453
66-517	2.11	0.56	472	14	31	5287	131	26	28	679	76	27	31	384
66-605	2.10	0.52	512	15	22	5700	155	24	26	870	83	24	27	479
67-312	Not Measured		609	16	20	8037	154	19	20	967	84	23	24	490

* AP blend was 60/40 MA/Ung

The boost propellant possesses remarkable flow properties in spite of the high solids loading (88.2 wt%) and the high proportion (60%) of the oxidizer that is Mikro-Atomizer ground. The viscosity buildup as measured in the extrusion rheometer is shown in Figure 3.2.2.11. The initial reading 1 3/4 hr after the curing agent was added was only 6,200 poises at 110°F; at 6 hr, the viscosity was still a relatively low 25,000 poises. Despite the low viscosity and the 6- to 8-hr pot life, the propellant reached a Shore "A" hardness of 25 at 135°F after only a 12-hr cure. An equilibrium hardness of 55 to 60 was reached after 36 hr at 135°F.

The reproducibility of the burning rate and mechanical properties of booster prepared in 800-lb batches is shown in Table 3.2.2.12. In the three batches prepared for evaluation of the propellant during the characterization phase of another program (batches 66-516, 66-517, and 66-605), the liquid-strand burning rates at the target pressure of 2250 psig were identical. There was a slight variation in slope (n). The low-temperature (-65°F) elongation (ϵ_m) was essentially identical not only for the three batches prepared during the program, but also for Batch 67-312, which was prepared a year later. Burning rates were not determined for this latter batch.

g. Safety Characteristics

Explosive hazard characteristics of the booster propellant have been well defined. The propellant is non-detonable, is formulated with raw materials in wide field use, and is rated as a Class 2 material. An I.C.C. shipping classification of B has also been established for the propellant. The propellant has an impact stability (50% firepoint) of 24 cm/2Kg, a copper block (heating rate of 20°F/min) autoignition temperature of 519°F, an autoignition temperature of 600°F as determined by differential thermal analysis (DTA) at a heating rate of 9°F/min, and even with no attenuation (0 cards) does not detonate in an NOL card gap test. Safety properties of the propellant are summarized in Table 3.2.2.13.

SAFETY AND HAZARD TEST DATA FOR
BOOSTER PROPELLANT

1. ICC Tests	Results
Impact stability, Bureau of Explosives	
drop of 3-3/4-in.	80% positive
drop of 10-in.	100% positive
Thermal stability (48 hrs at 167 to 180°F)	No change
Detonability (2-in. cube, No. 8 blasting cap)	Burns
Unconfined burning (2-in. cube)	Normal
I.C.C. classification	Class B
2. Hazard Classification Tests	
Autoignition temperature, DTA, °F (heating rate 9°F/min)	600
Copper block, °F (heating rate, 20°F/min)	519
Bureau of Mines impact, cm/2kg, 50% point	24
NOL card-gap, 160g Pentolite booster with 0-in. attenuation	Negative (4 tests)
Expected Hazard Classification	Class 2

Table 3.2.2.13

Sustainer Propellant Properties - The sustain propellant is characterized by the following properties:

a. Ballistic Properties

Delivered Specific Impulse, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia in 10KS-2500 size (100-lb grain) motors	217.6
Burning Rate, in/sec	
at 500 psia	0.08
at 2000 psia	0.14
Burning Rate Pressure Exponent, n	
Below 500 psia	0.09
Above 500 psia	0.37
Temperature Sensitivity, π_K, %/°F	0.14
Flame Temperature, °F	
Chamber	3694
Exhaust	1355
Density, lb/cu in.	0.0591

Chamber and exhaust gas compositions for the propellant are presented in Table 3.2.2.14. Specific impulse was measured in three 10KS-2500-size (100-lb grain) motor firings on a precision Aeroscience test stand at chamber pressures close to the operating pressure of the sustain grain for which the propellant was developed. Measured values and extrapolated values at standard conditions* are shown below:

	Measured Values		
Grain	Avg. Pressure, psia	Specific Impulse, lbf-sec/lbm	Specific Impulse, lbf-sec/lbm
11A	323	191.5	215.7
11B	383	197.3	217.6
51	418	200.9	219.5
		Average	217.6

* Standard conditions are P_c/P_e = 1000/14.7 psia; a 15-degree nozzle half-angle; optimum nozzle expansion.

THEORETICAL EQUILIBRIUM GAS COMPOSITION
FOR SUSTAINER PROPELLANT
(Moles/100g Propellant)

Product	P_c/P_e = 1000/14.7 psia Chamber at 3694°F	Exhaust at 1355°F
H_2O	1.3041	0.9530
CO	0.8898	0.5316
H_2	0.7676	1.1171
N_2	0.6714	0.6703
HCl	0.5178	0.5189
CO_2	0.2683	0.6252
PN	0.0058	0.0082
PO	0.0040	-
H	0.0022	-
CH_4	-	0.0013
Cl	0.0011	-
OH	0.0006	-
SiO	0.0001	-
P_2	0.0001	0.0009
FeCl	0.0001	-
$FeCl_2$	-	0.0001
SiO_2	-	0.0001(s)
Total, mol/100g	4.4334	4.4266

Table 3.2.2.14

The burning rate and other ballistic properties of the sustainer propellant have been well established by a large number of tests in 3KS-500-size motors over broad pressure and temperature ranges. Full-scale motor firings confirmed the 3KS-500-size motor test results. As shown in Figure 3.2.2.12, the burning rate-pressure relationship for the propellant exhibits a slope break at a chamber pressure of about 500 psia. Above and below this transition temperature, the burning rate-pressure relationships are linear.

b. Propellant Mechanical Properties

The sustain propellant, like the booster formulation, has been well characterized with respect to mechanical properties. Testing conducted over a range of temperatures, at a number of strain rates, under superimposed hydrostatic pressure, and in strain evaluation cylinders provided the data required for characterization and the establishment of material allowables. Tensile properties of the propellant, as a function of temperature and strain rate, are shown in Table 3.2.2.15. A strain rate of 0.74 min^{-1} is employed for normal propellant evaluation, whereas tests at other strain rates are required for more extensive characterization and allowables determination. The mechanical properties exhibited by the sustainer propellant have been proven highly satisfactory with respect to the requirements of case-bonded booster/sustainer bipropellant grain designs (where the sustain grain is sandwiched between the liner and the booster and bonded to both) and for cartridge-loaded grains of the type to be used in the proposed motor. The propellant withstands a constant strain of 4% for 168 hr at +40°F. In strain evaluation cylinders (5 in. dia by 20 in. long with cylindrical bores), the propellant has been cycled successfully six times between +135°F and -40°F at inner-bore hoop strains of up to 15.5%.

c. Propellant/Liner System Bond

SD-793 liner, formulated with 25 wt% of iron oxide as an inert filler in a polyurethane binder, has been used in conjunction with the sustain propellant in motors having temperature range requirements of -10 to +130°F and -65 to +160°F. As shown in Table 3.2.2.16, the tensile, shear and peel

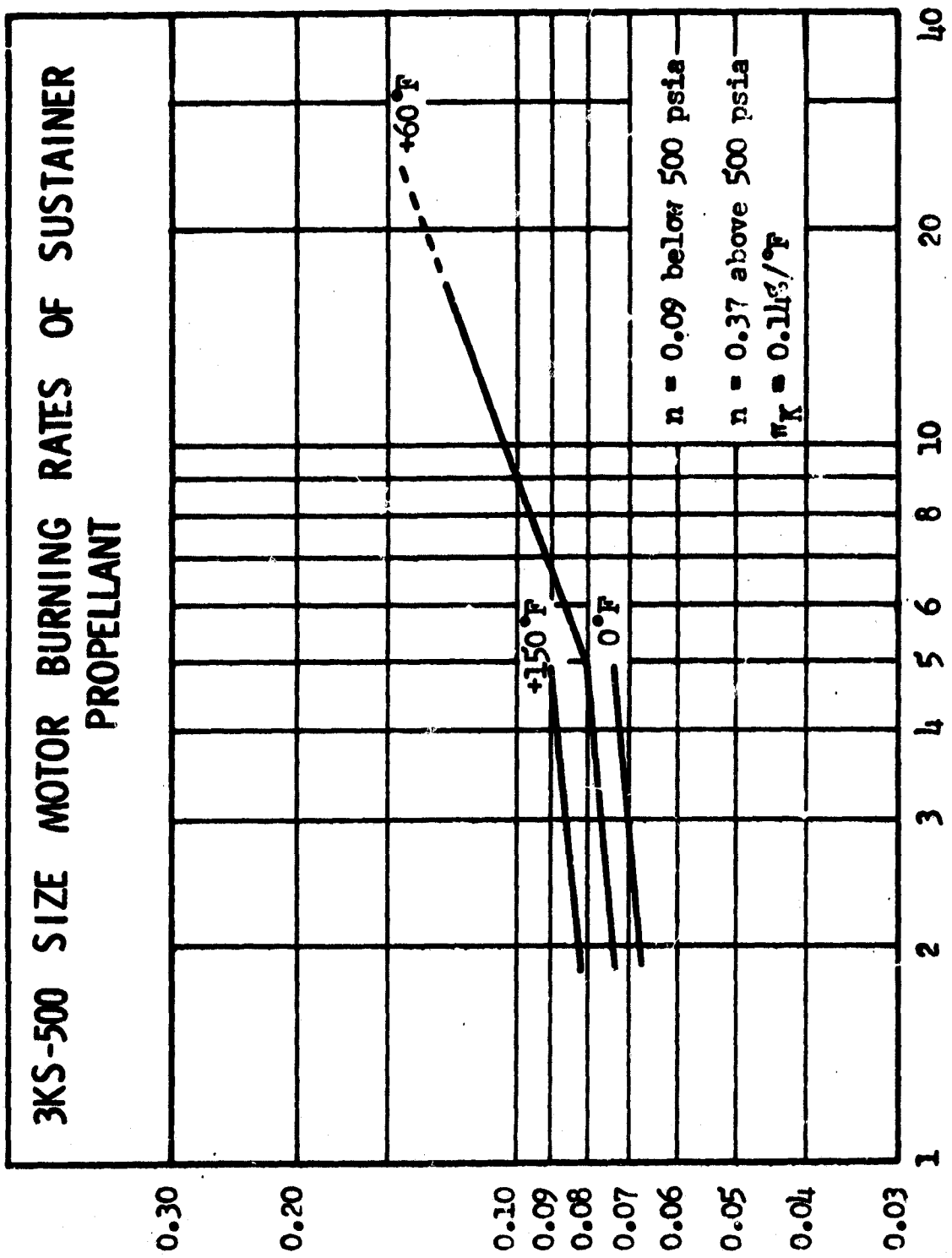


Figure 3.2.2.12

Strain Rate, min^{-1}	Test Temp., °F	Mechanical Properties			
		σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
0.0015	-75	-	-	8.6	18,422
	-40	186	15	24	1,940
0.0074	-75	733	3.7	7.6	22,885
	77	54	16	22	531
	160	33	9.4	14	478
0.074	-75	828	3.5	6.1	25,723
	77	80	15	23	738
	160	43	10	17	563
0.74	-75	1160	2.4	2.9	62,500
	-40	539	8.4	15	10,348
	0	282	20	44	3,038
	77	99	17	25	834
	110	83	17	27	770
	160	62	13	20	647
7.4	-75	1237	2.6	2.6	53,425
	77	134	22	33	1,012
	160	81	17	24	812

Mechanical Properties of Sustainer Propellant

Table 3.2.2.15

Test Temp, °F	Bond Properties		
	Tensile Strength, psi	Shear Strength, psi	Peel Strength, lb/in.
-75	128	329	17
-40	443	319	38
77	45	40	23
110	59	42	28
160	35	34	5

Bond Strength of Sustainer Propellant/SD-793 Liner System

Table 3.2.2.16

strengths of the sustain propellant-SD-793 bond are good over the temperature range of -75 to +160°F. In general, bond system failures occurred within the body of the propellant. Aging data (Table 3.2.2.17) for sustainer propellant-liner specimens stored at temperatures between +80 and +180°F show that bond strength; (1) generally increases on storage for up to 20 wk at temperatures from +80 to +160°F and up to 4 wk at +180°F, and; (2) decreases somewhat on 16 wk storage at the severe storage temperature of +180°F. The results of these laboratory tests are indicative of a long bond system storage life and have been confirmed by the field storage of full-scale motors containing the sustainer propellant/SD-793 system* for more than 6 yr.

d. Stability of Propellant Mechanical Properties on Long-Term Storage

Accelerated aging tests with the sustain propellant have demonstrated that the formulation should have a long useful life. As shown in Table 3.2.2.18, propellant properties remain acceptable even after continuous storage for 20 wk at +160°F and 16 wk at +180°F. Extrapolation of the data on an Arrhenius plot (Figure 3.2.2.13) indicates a useful life in excess of 5 yr. This extrapolation has been substantiated by field storage of the propellant for more than 6 yr.

e. Adequacy of Propellant and Propellant/Liner System Mechanical Properties

On the basis of a very preliminary stress analysis of the proposed sustainer grain configuration, it appears that the sustainer propellant and sustain propellant-SD-793 liner bond will fulfill motor performance requirements. When a comprehensive stress analysis of the final sustainer grain design is made during the motor program, it will be possible to evaluate propellant capability with respect to requirements derived from the stress analysis. The extensive testing conducted with the propellant and propellant-liner system has permitted the establishment of the allowables, for both unaged and aged materials, necessary for analysis of system capability under all motor storage and operational conditions.

---

* Motors containing an inner boost propellant grain bonded to the proposed sustain propellant which was, in turn, bonded to SD-793.

EFFECT OF STORAGE ON THE BONDING PROPERTIES OF THE SUSTAINER PROPELLANT/SD-793 LINER SYSTEM

Temp., °F	Time Weeks	Test Temperature, °F: -75 Tensile Strength, psi	-75 Shear Strength, psi	-75 Bond Strength, psi	77 Tensile Strength, psi	77 Shear Strength, psi	77 Bond Strength, psi	160 Tensile Strength, psi	160 Shear Strength, psi	160 Bond Strength, psi
80	0	128	329	17	45	40	23	35	34	5
	16	147	293	-	48	36	24	25	17	4
135	0	128	329	17	45	40	23	35	34	5
	2	-	-	-	63	39	27	-	-	-
	16	199	375	24	77	53	26	41	23	7
	20	-	-	-	64	40	27	-	-	-
150	0	128	329	17	45	40	23	35	34	5
	2	-	-	-	56	31	24	-	-	-
	16	297	485	13	76	50	25	42	24	7
160	0	128	329	17	45	40	23	35	34	5
	2	-	-	-	59	43	17	-	-	-
	16	269	461	18	65	44	26	36	20	8
	20	-	-	-	57	34	32	-	-	-
180	0	128	329	17	45	40	23	35	34	5
	2	-	-	-	53	40	19	-	-	-
	4	-	-	-	57	40	32	-	-	-
	16	285	444	21	38	30	35	21	21	9

Table 3.2.2.17

EFFECT OF AGING ON SUSTAINER PROPELLANT MECHANICAL PROPERTIES

Table 3.2.2.18

Properties Tested at 0.74 min^{-1} Strain Rate and Indicated Temperature

Temp. °F	Time Wks	-75°F σ_m, psi	-75°F ϵ_m, %	-75°F ϵ_b, %	-75°F E_o, psi	-40°F σ_m, psi	-40°F ϵ_m, %	-40°F ϵ_b, %	-40°F E_o, psi	0°F σ_m, psi	0°F ϵ_m, %	0°F ϵ_b, %	0°F E_o, psi	77°F σ_m, psi	77°F ϵ_m, %	77°F ϵ_b, %	77°F E_o, psi	160°F σ_m, psi	160°F ϵ_m, %	160°F ϵ_b, %	160°F E_o, psi
80	0	1160	2.4	2.9	62500	539	8.4	15	103500	282	20	44	3038	99	17	25	834	62	13	20	647
	8	1143	1.5	1.5	103750	589	6.2	12	25200	274	21	46	3767	99	18	23	885	59	14	20	588
	16	1056	1.7	5.0	108500	553	7.9	15	14040	225	20	38	2173	102	18	23	896	64	14	18	684
135	0	1160	2.4	2.9	62500	539	8.4	15	10350	282	20	44	3038	99	17	25	834	62	13	20	647
	2	1167	2.5	2.9	62740	530	8.6	15	10300	284	19	44	3112	101	18	27	946	61	13	21	655
	8	1185	1.9	1.9	90000	555	5.2	12	22600	244	19	44	2752	87	16	24	824	56	12	18	647
	16	1062	1.5	2.0	105000	515	6.8	16	13020	225	21	40	2211	103	19	23	915	65	14	18	691
	20	1348	1.0	1.9	171000	510	9.1	19	11550	229	19	46	2435	96	17	21	861	60	13	21	685
150	0	1160	2.4	2.9	62500	539	8.4	15	10350	282	20	44	3038	99	17	25	834	62	13	20	647
	2	1159	2.3	2.8	61990	533	8.7	15	10410	286	18	35	3096	102	18	28	867	66	14	20	665
	8	1435	1.3	1.5	146250	582	5.6	15	24900	235	21	52	2490	96	17	22	852	60	14	22	640
	16	1068	2.1	4.0	103700	528	8.7	16	11920	231	22	40	2053	106	19	25	900	67	14	18	682
160	0	1160	2.4	2.9	62500	539	8.4	15	10350	282	20	44	3038	99	17	25	834	62	13	20	647
	2	1176	2.2	2.7	62560	541	8.9	15	10290	290	19	38	3100	104	19	24	908	62	14	21	636
	8	1134	2.1	2.5	108000	568	6.2	13	25170	218	20	46	2441	87	18	28	739	59	14	22	599
	16	1022	1.9	2.3	87000	510	7.9	18	10280	204	24	52	1857	96	23	34	708	59	20	28	468
	20	1393	1.1	1.3	170000	494	9.3	22	11010	168	29	68	1616	75	32	49	426	49	28	40	311
180	0	1160	2.4	2.9	62500	539	8.4	15	10350	282	20	44	3038	99	17	25	834	62	13	20	647
	2	1164	2.0	2.6	60780	534	8.4	16	10040	282	19	37	3025	102	21	32	763	63	16	22	550
	4	1066	1.6	2.0	89300	500	8.2	18	11400	185	26	52	1541	95	24	34	656	61	20	29	490
	8	1163	1.9	2.2	106300	582	5.6	12	24800	201	25	65	1906	88	25	36	586	54	19	29	417
	16	994	2.1	4.4	81000	494	10.1	20	9540	161	46	84	1262	88	49	58	391	54	32	40	289

AGING STABILITY OF POLYURETHANE PROPELLANTS

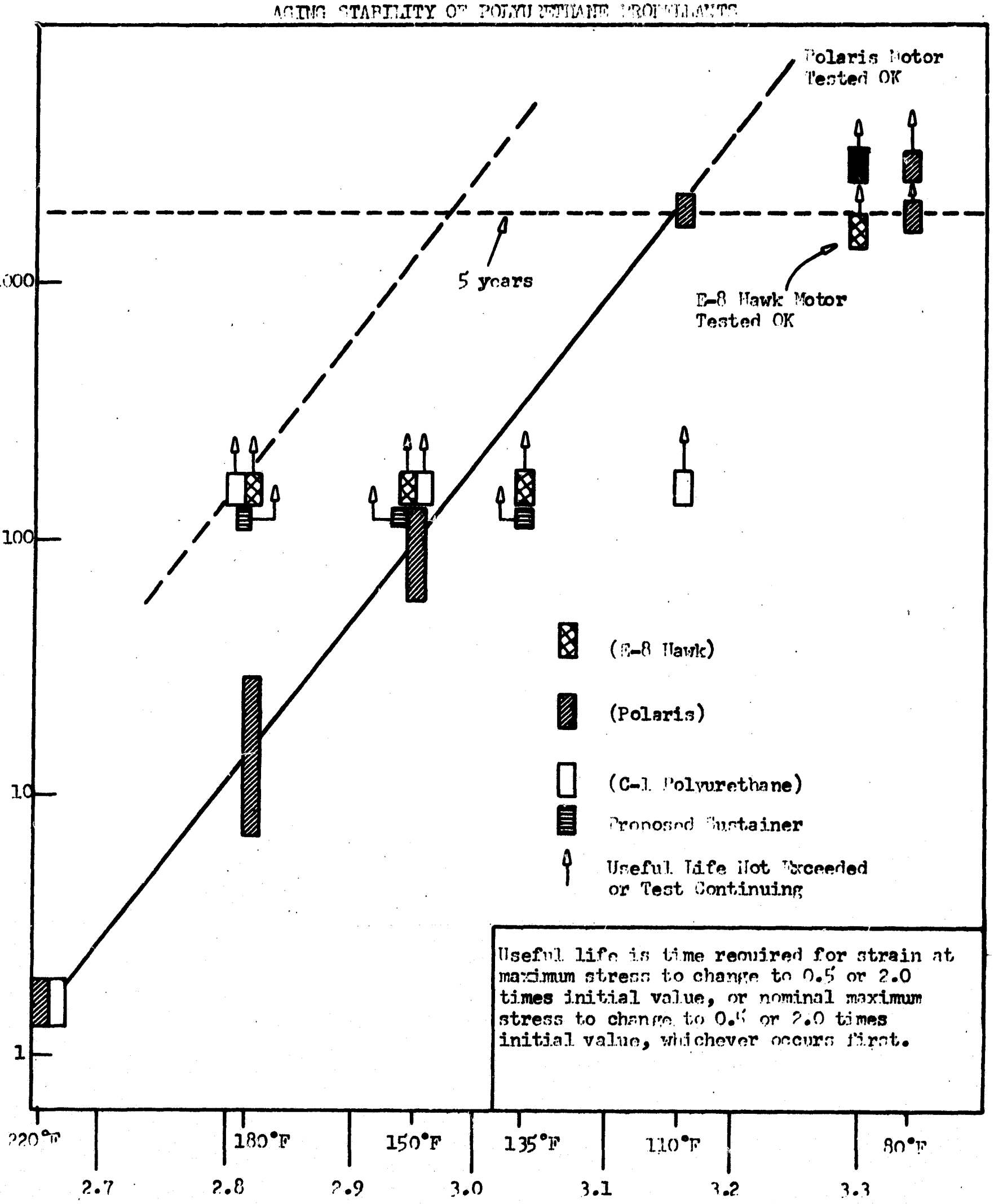


Figure 3.2.2.13

f. Processing Properties

The proposed sustainer propellant is a very fluid formulation with a low rate of viscosity buildup, a long potlife, and a cure time of 5 days at +110°F. As shown in Figure 3.2.2.1^4, propellant viscosity 2 hr after addition of the curing agent to the mixer is less than 1000 poises at +110°F and at 16 hr is only 8000 poises. The latter value is well below the initial viscosity of most propellants now in production. Low viscosity and slow rate of viscosity buildup assure adequate system fluidity during the period required for precasting batch acceptance tests and the time required for motor casting from large production batches. Low viscosity also assures the production of homogeneous, void-free grains of high quality.

As is the case with the proposed boost propellant, simple procedures are used in the production of the sustainer propellant. All raw materials are fully characterized before use, processing procedures have been well defined to provide maximum system reproducibility and economy of production, and product qualification tests have been developed to preclude the casting of an unacceptable batch. All raw materials used are available in large quantity at purity levels which fulfill Aerojet acceptance specifications. The specifications and procedures now in use for production of the sustainer propellant have been proven during the processing of over 1000 full-scale (2150-lb) batches and will be used during the proposed program. The reproducibility achieved through the use of current specifications and procedures is excellent, as illustrated by the sustainer propellant mechanical property data presented below.

	Mechanical Properties at 77°F			
	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
Average Value, $\bar{X}$	101.7	24.1	31.5	741.8
Standard Deviation	7.6	3.5	3.6	123.7
Full-Scale Batches*	160	150	160	160

* 2150-lb batches

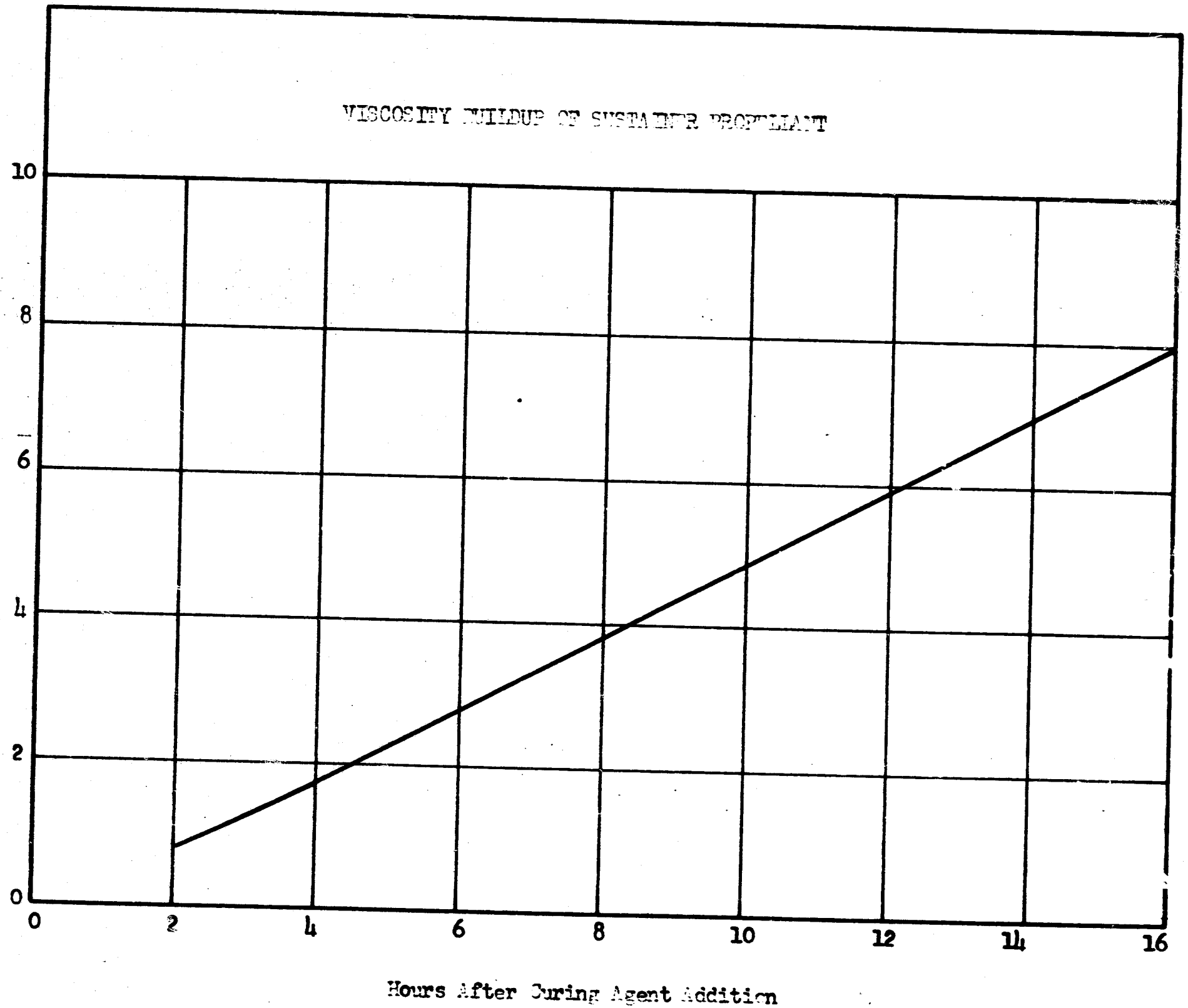


Figure 3.2.2.14

g. Safety Characteristics

The sustain propellant has been well characterized with respect to safety properties and is a nondetonable material with an explosive hazard rating of Class 2 and an I.C.C. shipping classification of B. As shown in Table 3.2.2.19, the propellant has an impact stability (50% firepoint) of 41 cm/2 kg and a copper block autoignition temperature of 585°F. The propellant does not detonate with 0-in. attenuation in an NOL card gap test or when initiated with a No. 8 blasting cap in a 2-in. cube test. All raw materials used in the formulation are in field use and are stable alone or in combination with other raw materials.

ALTERNATIVE DESIGNS

During the course of the design study several alternative designs were evaluated. The primary reasons for considering other designs were to attempt to (1) incorporate desirable features such as a nonmagnetic case, (2) provide a lighter weight motor, (3) improve motor performance, and (4) reduce motor costs. A summary of the evaluation results for each of the major concepts considered follows, including a preliminary case-bonded motor design.

Improved Propellant Formulation - Review of the current propellant capability indicated that propellants with a standard specific impulse up to 235 lbf-sec/lbm could be developed in the desired burning-rate range. There appeared to be little possibility of reducing the burning rate substantially without an associated loss in impulse. The best estimate of propellant capability with a limited development program is an impulse of 225 lbf-sec/lbm at 0.08 in/sec burning rate. A slight reduction in burning rate below 0.08 would probably be possible with slightly lower impulse. After motor performance utilizing existing propellant had been reviewed with representatives of Space Division, the decision was made not to incorporate an improved propellant in the basic design. It is desirable, however, to develop an improved propellant to provide a performance safety margin which would be useful in overcoming any potential problem and still maintain the desired performance without increasing the size of the motor.

1.	ICC Tests	Results
	Impact stability, Bureau of Explosives	
	drop of 3-3/4-in.	Negative
	drop of 10-in.	Negative
	Thermal stability (48 hrs at 167 to 180°F)	No change
	Detonability (2-in. cube, No. 8 blasting cap)	Burns
	Unconfined burning (2-in. cube)	Normal
	I.C.C. Classification	Class B

2.	Hazard Classification Tests	
	Copper block, °F (heating rate, 20°F/min)	585
	Bureau of Mines impact, cm/2kg, 50% point	41
	NOL card-gap, 160g Pentolite booster with 0-in. attenuation	Negative (4 tests)
	Expected hazard classification	Class 2

Safety and Hazard Test Data for
Sustainer Propellant

Table 3.2.2.19

Aluminum Case - During early coordination meetings it was stated that it would be desirable to incorporate a nonmagnetic case. A preliminary design tradeoff study indicated that to maintain the 6.0 in. O.D. and utilize an aluminum case, a considerable propellant volumetric loading penalty would occur because of the added case thickness and insulation requirements. This condition also reduced the burning duration below the desirable minimum of 27 sec. The motor weight would increase because of (a) the added length required to accommodate the propellant displaced by the thicker case and insulation, (b) the increased thickness of insulation required because of the low temperature capability of the aluminum, (c) the lower strength-to-density ratio (vs steel) of the aluminum alloy evaluated, and (d) the additional propellant required to overcome the previous weight penalties.

On the basis of these disadvantages the decision was made not to use an aluminum case in the basic design. It would be possible to make a motor with an aluminum case and the general characteristics of the selected design, however, it would be quite difficult to meet the selected design goals. Development of an improved propellant would partially overcome some of these disadvantages.

End Burning Grain - During the propellant selection studies it was determined that an existing propellant formulation with a 2.5 in/sec burning rate capability could deliver the desired impulse and duration in an end-burning configuration that would be somewhat shorter than the selected cartridge loaded design. One major disadvantage was that the propellant had to burn at 2000 psi to realize the 2.5 in/sec burning rate. Consequently, to achieve the 4:1 boost-to-sustain thrust ratio, more than 7000 psi operating pressure would be required during the boost phase. The corresponding case weight penalty was very high. One method of overcoming this problem would be to provide a two position nozzle that had a large throat area during boost and a small area during sustain. Aerojet recently submitted a proposal* to the Navy for development of such a nozzle which could be economically mass produced.

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* Exploratory Development of a Dual Area Nozzle for Solid Rocket Motor Applications (U) Confidential, Proposal SRO 681601, 7 December 1967.

Use of this nozzle would add substantially to the development cost and time. Manufacturing costs of the end-burning design would be substantially less than for the selected design, however, a corresponding increase in the cost of the nozzle would probably overcome any associated savings.

The use of a separate boost stage would also overcome the problem associated with the boost-sustain pressure differential. Reduced costs of processing the simpler end burning design would be lost and the final cost would probably be of the same magnitude as for the selected cartridge-loaded design but would have the advantage that the boost grain could be omitted, if desired.

An additional problem with the end-burning design is that the high-burning-rate propellant has a high oxidant value and causes severe nozzle erosion. Additionally, the high flame temperature and high operating pressure would increase the insulation requirements.

It seems apparent that an end burning design could be developed to meet the mission requirements, however the resultant design would be somewhat more complex and the associated development schedule would be longer at least for the dual area nozzle design.

Coordination with Space Division indicated that it would be preferable to avoid using an end burning design, consequently detailed design analyses and cost studies were not conducted.

3.2.2.4 ALTERNATE DESIGN, CASE BONDED HTPR GRAIN

Case Bonded Grain - Initial studies assumed the use of a case-bonded grain, which would be very desirable since propellant handling would not be required after casting. Stress analysis showed the propellant selected for the basic design experienced unacceptable stress levels due to the relatively long case length, and operational environments temperature specified. This coupled with the burning time requirements resulted in the multiple segment grain basic design,based on a well known propellant formulation.

There is, however, a relatively new propellant which has been under development by Aerojet over the past few years. The new propellant

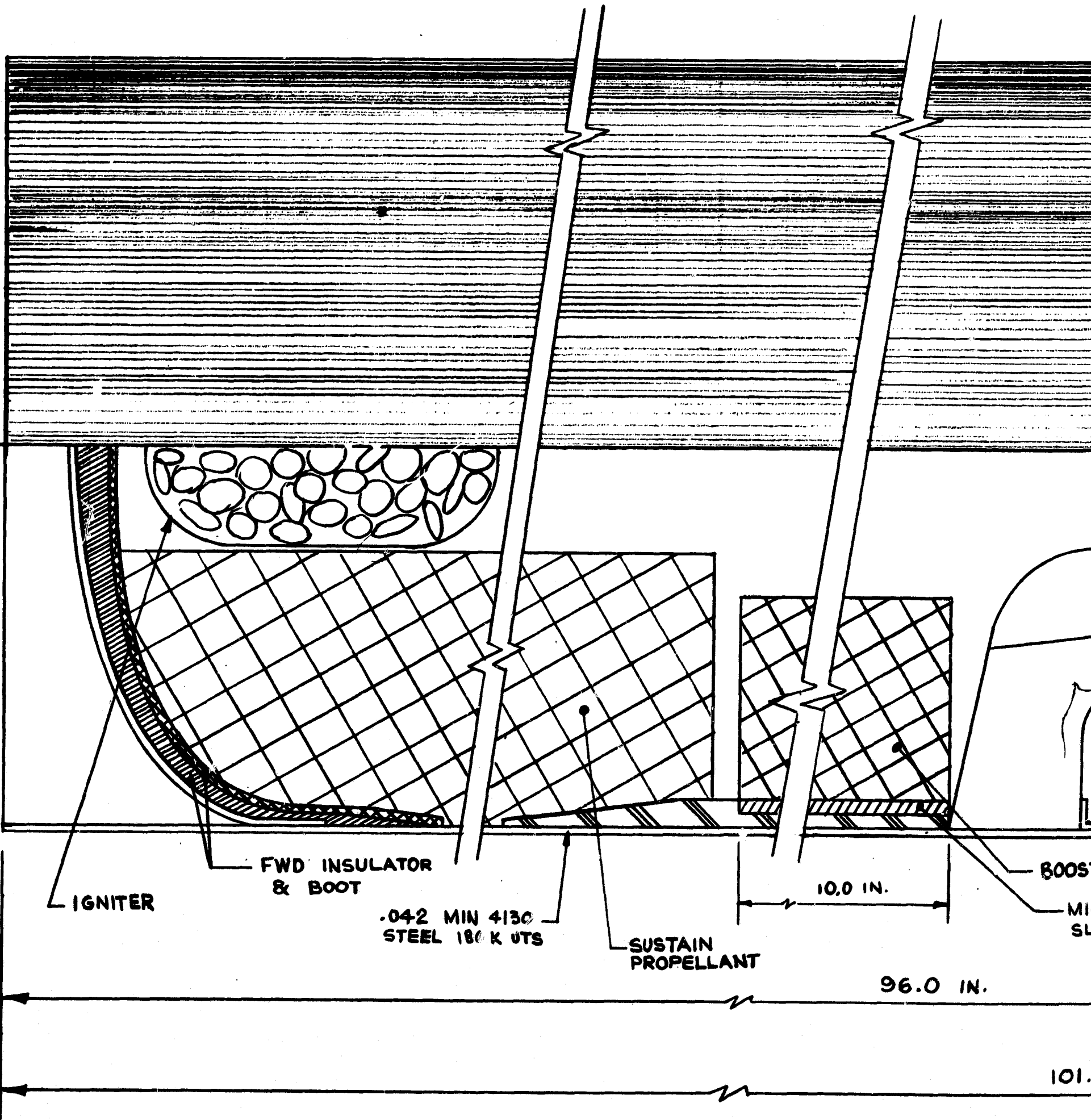
has mechanical properties which are adequate to allow bonding of the propellant to the full length of the chamber. This design would permit removal of the phenolic sleeves from the original design and create a higher propellant loading with a higher energy propellant than is utilized in the current design. The burning rate capability of this propellant demonstrated at the time of these studies would cause the motor duration to be somewhat shorter than the criteria established for the motor design studies (down to 14-18 sec depending on the specific combination of I_{sp}, density and mechanical properties selected). Recent laboratory tests of this propellant with burning rate suppressants have resulted in burning rates of .094 in/sec which if incorporated into the motor would result in a duration of 22 seconds.

The preliminary design based on the most pessimistic characteristic burning rate of 0.14 in/sec indicates that 15% more total impulse could be loaded into a motor with 9 lb less inert weight and 9 inches shorter than the baseline design. A trajectory calculation showed that this motor would attain 619,285 ft; however, high acceleration loadings (≈60 g's) obtained with this configuration could be overcome by using a lower burning rate formulation of .12in/sec or .094 in/sec which have been demonstrated in laboratory tests. Another perhaps more attractive possibility is to use a relatively cheap throat which would experience considerable erosion and reduce the terminal pressure and thrust.

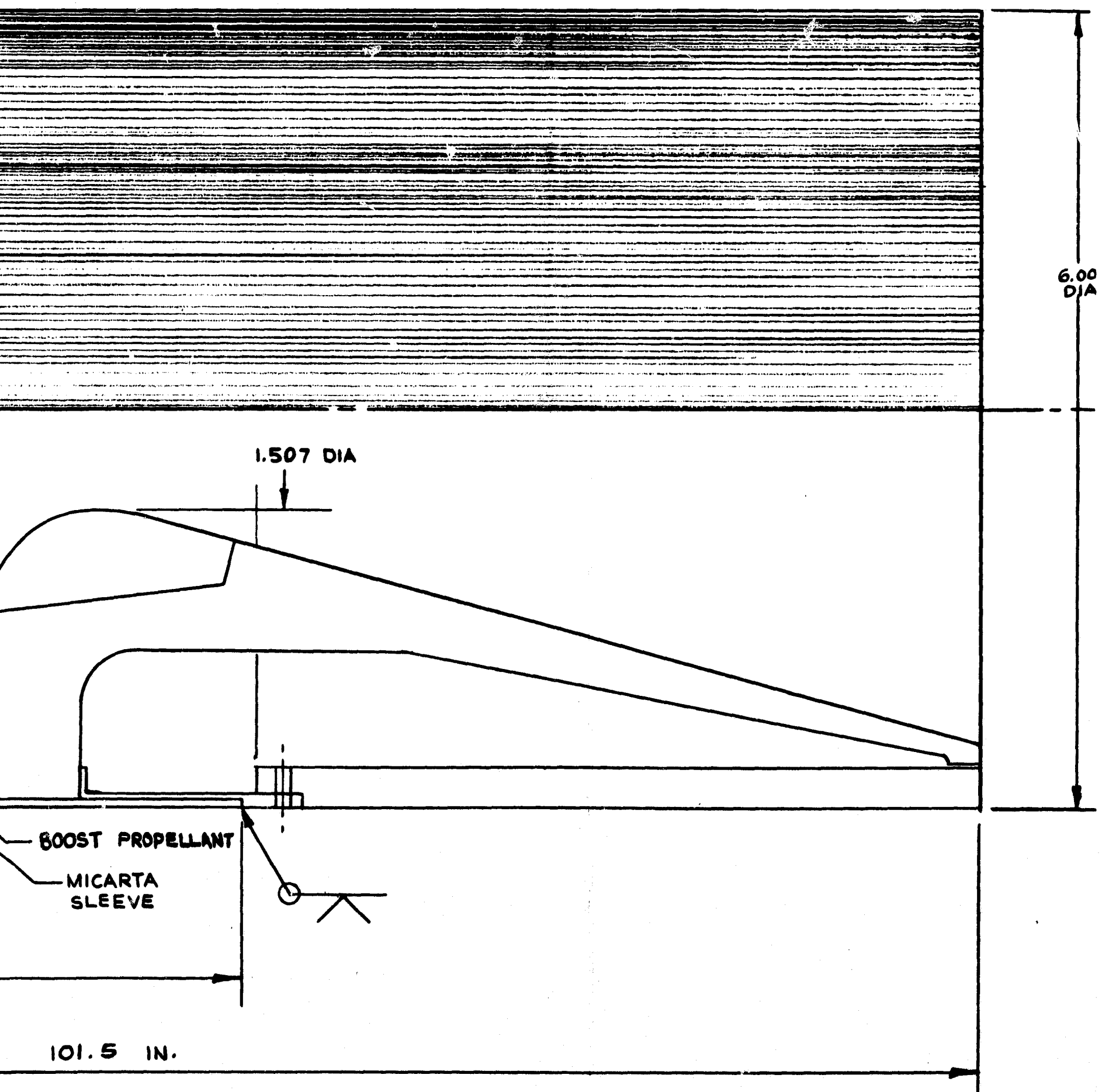
The preliminary design configuration of the case bonded Small Probe Solid Rocket Motor is shown in Figure 3.2.2.15. The motor is 101.5 in. long with a maximum diameter of 6.0 in. The propellant sustainer grain of high strain capability, low-burning rate propellant is cast into the chamber and bonded to the case for its full length. A single grain of high burning rate propellant is to provide a boost thrust capability of 4700 lb and is cartridge loaded aft of the sustainer propellant. The total nominal duration is 13.7 sec of which boost would account for the first second of operation. Total motor weight is 174.5 lb of which 142.1 lb is propellant and 32.4 lb is inert components. A summary of performance and weight characteristics is shown in

SMALL PROBE ROCKET MOTOR

CAST IN CASE DESIGN



OR



FOLDOUT FRAME 2

Table 3.22.20. An estimated nominal pressure and thrust versus time curve at 70°F is shown in Figure 3.2.2.16. Subsequent to this design effort propellant burning rate reductions sufficient to increase the duration to 22 sec have been demonstrated.

To provide a motor with wide operating and storage temperature range and minimum costs a cast in case design utilizing a new high strength propellant was evaluated. The sustain propellant is cast directly into the chamber and bonded for the full length. A cylindrical center core was used to minimize insulation requirements, simplify tooling, and provide a burning curve wherein a greater majority of the impulse would be delivered at higher altitude for more efficient usage.

Dual thrust capability is provided by a single boost grain cast into a Micarta sleeve and cartridge loaded in the aft end of the motor. Feasibility studies to evaluate the possibility of casting the boost grain directly into the chamber or of using a highly configurated boost using the sustain propellant would be conducted prior to selecting a final design.

Use of the cast in case design reduces the insulation requirements to a minimum. A rubber insulator would be required for the forward head and for the aft end behind the boost grain. The only sidewall insulation would be the liner used to bond the propellant to the chamber. External heating has not been evaluated for the case bonded concept, however, the internal heating which was foreseen as a potential problem with the cartridge loaded design would be minimized consequently greater external heating could be tolerated.

Initial chamber selection was the same as for the cartridge loaded design, however with the improved performance of the case bonded design it would probably be feasible to use an aluminum chamber and still meet the design requirements. The nozzle and igniter design for this motor would be essentially identical to those designed for the cartridge loaded design.

Selection of propellants for use in the booster and sustainer grains of the proposed motor was based on fulfillment of the following general

Predicted Motor Performance at 70°F and Vacuum	Boost	Sustain
Firing Duration, sec	1.0	12.7
Total Impulse, lb-sec	4700.	33000.
Average Thrust, lbf	4700.	2600.
Delivered Specific Impulse, lbf-sec/lbm	268.6	264.8
Chamber Pressure, avg. psia	1600.	770.0
Chamber Pressure, max nominal, psia	1800.	1170.
Propellant Ballistic Properties		
Std I_s at 1000/14.7 psia, opt exp lbf-sec/lbm	245.5	243.0
Burning Rate, in./sec	1.8 @ 2000 psi	0.14 @ 500 psi
Burning Rate Exponent	0.47	0.30
Propellant Density, lb/in.3	0.0663	0.063
Design Data		
Throat Area Initial, $in.^2$	1.784	1.784
Expansion Ratio, avg.	10.9	9.6
*Web Thickness, in.	1.60	2.02
Motor Length, in. (face of fwd boss to exit plane)	101.5	101.5
Motor Dia. in.	6.0	6.0
Exit Dia, in.	5.0	5.0

Weight Summary, lb	
Chamber	24.7
Nozzle Assembly	3.5
Insulation and Liner	3.4
External Paint	0.8
Total Inerts	32.4
**Propellant	142.1
TOTAL MOTOR	174.5
Mass Fraction	0.814

NOTE:

* Initial sustain propellant web is 2.22 in thick, however 0.20 in. are consumed during the boost phase.

** Propellant weight consists of 12.1 lb of boost propellant at I_s = 269.8 and 5.4 lb of sustain propellant at I_s = 267.0 consumed during boost. Sustain propellant wt - 124.6 lb at I_s = 264.8

Table 3.2.2.20

Performance and Weight Summary

6-in.-dia SOUNDING ROCKET
Case Bonded Sustainer HTPB

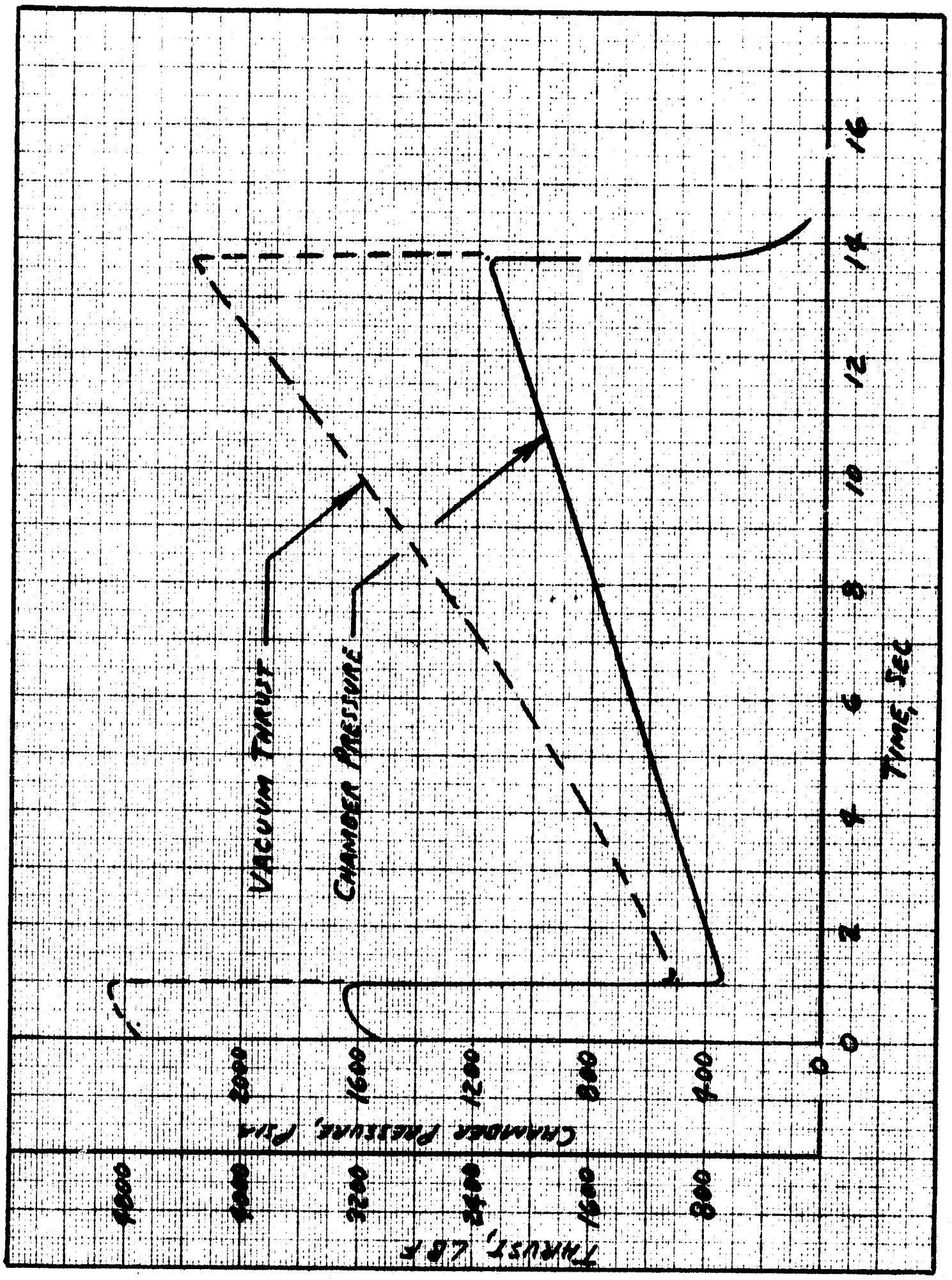


Figure 3.2.2.16

requirements plus the ballistic performance requirements obtained from grain design studies:

Low raw material costs.

Low processing costs.

A storage and operational temperature range capability of $-65^{\circ}F$ to $+165^{\circ}F$.

A class 2 explosive hazard rating.

Booster grain design was based on the use of propellant ANB-3254, a low-cost PBAN formulation used in 260-inch diameter motor 260-SL-3. The propellant is formulated with 69 wt% ammonium perchlorate, 15 wt% aluminum, and 1.3 wt% ballistic additive. This propellant, which will be used without modification, has the following ballistic properties:

Delivered Standard Specific Impulse, lbf-sec/lbm*	244.6
Burning Rate, in/sec at 600 psia	0.68
Burning Rate Pressure Exponent	0.38
Temperature Sensitivity, π_K, $\%/^{\circ}F$ from $-40^{\circ}F$ to $+150^{\circ}F$	0.16
Density, lb/cu in.	0.0633

The propellant has been well characterized with respect to ballistic, mechanical, bonding, safety, and processing properties. Over 1,800,000-lb of ANB-3254 have been processed and 319 5500-lb batches (1,754,500-lb) of propellant were cast into motor 260-SL-3 alone. Propellant properties are summarized in Table 3.2.2.21. On the basis of the mechanical and bonding properties of ANB-3254 and experience with other PBAN formulations, integrity of the cartridge-loaded booster grain can be assured on storage and operation over the specified $-65^{\circ}F$ to $+165^{\circ}F$ range. The storage stability of propellants of the ANB-3254

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* Specific impulse measured in 10KS-2500 size (100-lb grain) motors fired on a precision Aeroscience test stand at P_c/P_e = 1000/14.7 psia; 15° nozzle half-angle; optimum nozzle expansion.

Table 3.2.2.21

COMPOSITION AND PROPERTIES OF ANB-3254 PROPELLANT

Ballistic and Thermodynamic Properties	
Specific Impulse, lbf-sec/lbm at P_c/P_e = 1000/14.7 psia	
Theoretical	260.1
Measured in 10KS-2500 size (100-lb grain) motors	244.6
Flame Temperature, ^{o}F at P_c/P_e = 1000/14.7 psia	
Chamber	5780
Exhaust	3434
Characteristic Exhaust Velocity, C*, ft/sec	5167
Burning Rate, in/sec at 600 psia	0.68
Burning Rate Pressure Exponent, n	0.38
Temperature Sensitivity, π_k, %/^{o}F from -40^{o}F to +150^{o}F	0.16
Molecular Weight of Gases, g/mole at P_c/P_e = 1000/14.7 psia	
Chamber	28.922
Exhaust	29.837
Heat of Formation, Kcal/100g	-48.734
Physical Properties	
Density, lb/cu in.	0.0633
Autoignition Temperature, ^{o}F	495
I.C.C. Shipping Classification	Class B

Table 3.2.2.21 - contd.

Mechanical Properties at +77°F

	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
Tensile Properties*	90	27	29	429
Constant Strain Tolerance for 168 hr at 30% R.H.		20%		

Bond Strength to SD-850-2 Liner

Tensile, psi	Shear, psi
211	137

Propellant and Propellant/Liner System Allowables at +77°F

Interfacial Shear, psi	
Storage (1 yr)	9.8
Firing (0.04 sec)	385
Interfacial Tensile, psi	
Storage (1 yr)	16
Tensile Strain, %	
Storage (1 yr)	18
Firing (0.04 sec)	22

* Average Values for 56 of the 319 5500-lb batches cast into motor 260-SL-3.

type is illustrated in Figures 3.2.2.17 and 3.2.2.18 by +180°F and +150°F accelerated aging data for propellant ANB-3241-2, a PBAN formulation containing 87 wt% ballistic solids. Stability of the type exhibited at high temperatures is indicative of a long useful life.

Sustainer grain design studies indicated that a propellant with the following properties was required:

Delivered Standard Specific Impulse, lbf-sec/lbm in 10KS-2500 motors	
Minimum	235
Desired	237
Burning Rate, in/sec at 500 psia	0.12 *
Density, lb/cu in.	≥ 0.061
Strain Bearing Capability at -65°F, %	≥ 40

The strain capability requirement of 40% or greater at -65°F dictated the selection of a propellant based on a hydroxyl-terminated polybutadiene (HTPB) binder. A total ballistic solids content of 82 wt% was dictated by the very low burning rate requirement of 0.12 in/sec at 500 psia. The choice of this solids loading was based on the results of an extensive evaluation of carboxy-terminated polybutadiene propellants containing from 84 wt% to 90 wt% ballistic solids and various oxidizer/aluminum ratios which defined the lower burning rate limits for the polybutadiene propellant system. Extrapolation of the experimental data (Figure 3.2.2.19) to the 82 wt% ballistic solids level indicated (1) that the 0.12 in/sec requirement was at, or near, the lower limit of the 82 wt% system and (2) that formulations containing oxidizer/aluminum ratios between 60/22 and 64/18 wt% were potential candidates. On the basis of laboratory formulation studies and Crawford bomb burning rate tests with solid strands, the following formulation has been selected as the sustainer propellant:

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* .094 in/sec laboratory demonstrated potential.

EFFECT OF 180°F STORAGE ON MECHANICAL PROPERTIES OF ANB-3241-2 PROPELLANT

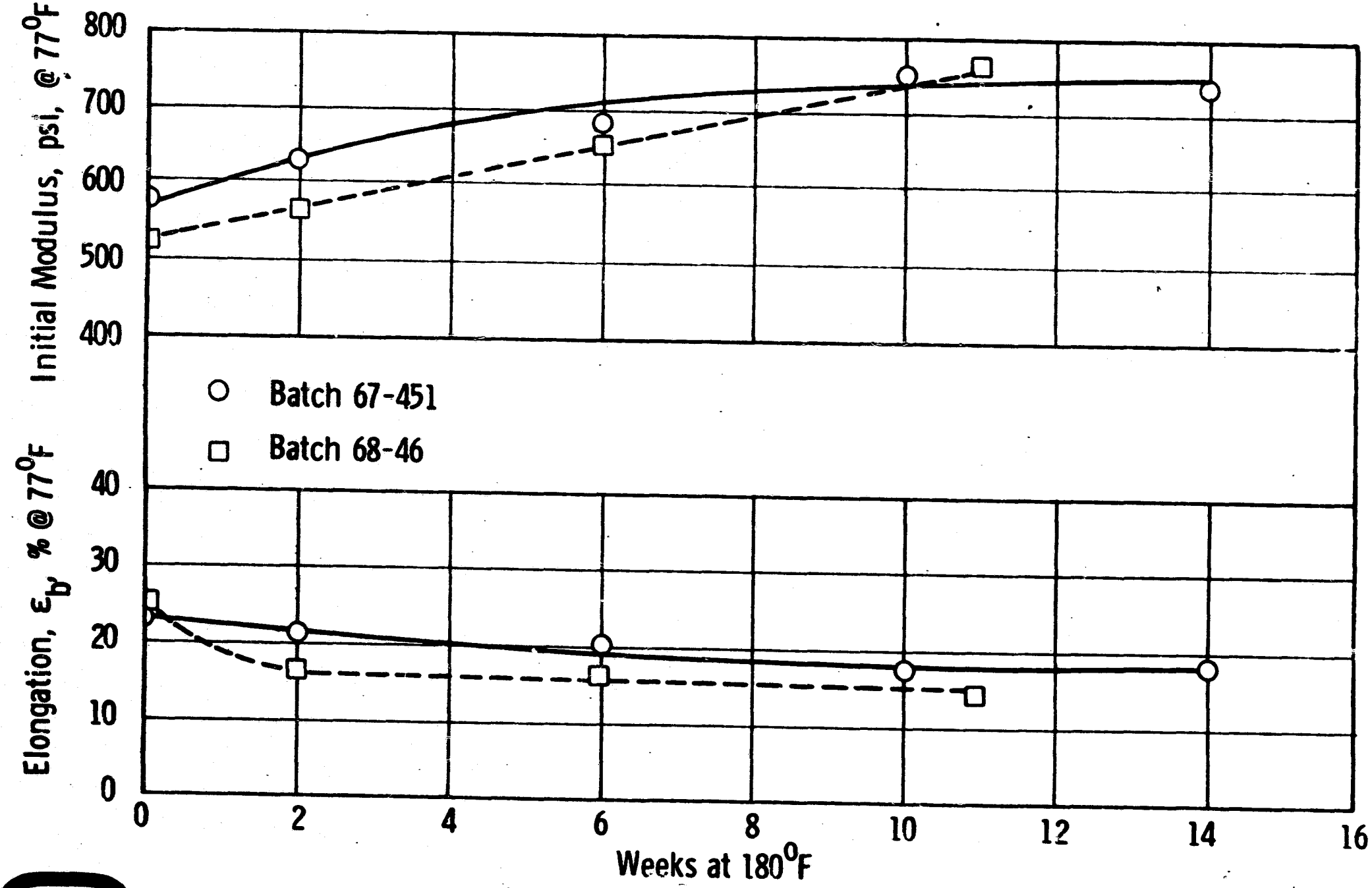


FIGURE 3.2.2.17

AEROJET GENERAL®

PROPULSION DIVISION

CJR 6-4-68 (5)

EFFECT OF 150°F STORAGE ON MECHANICAL PROPERTIES OF ANB-3241-2 PROPELLANT (77°F Properties)

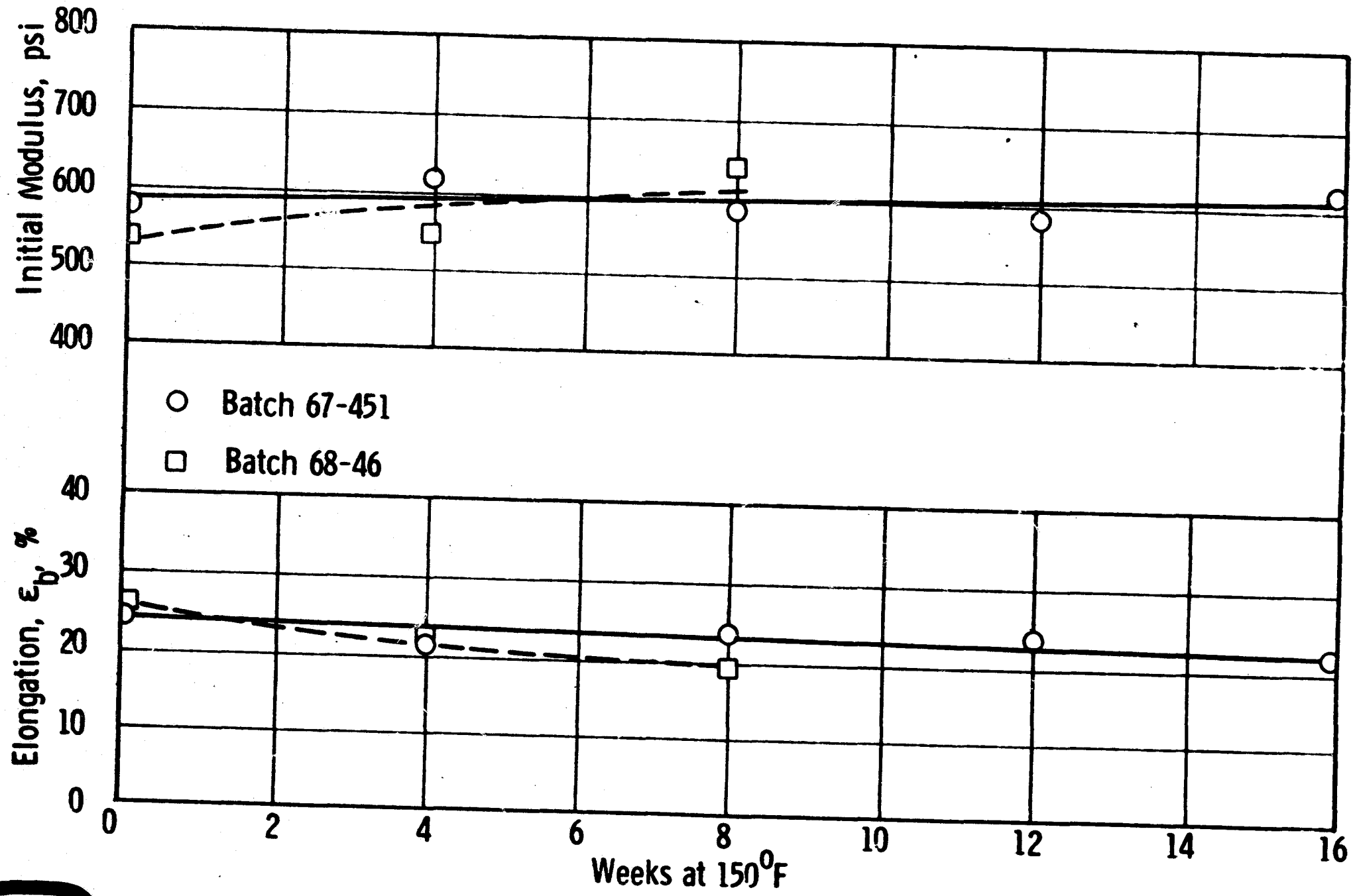
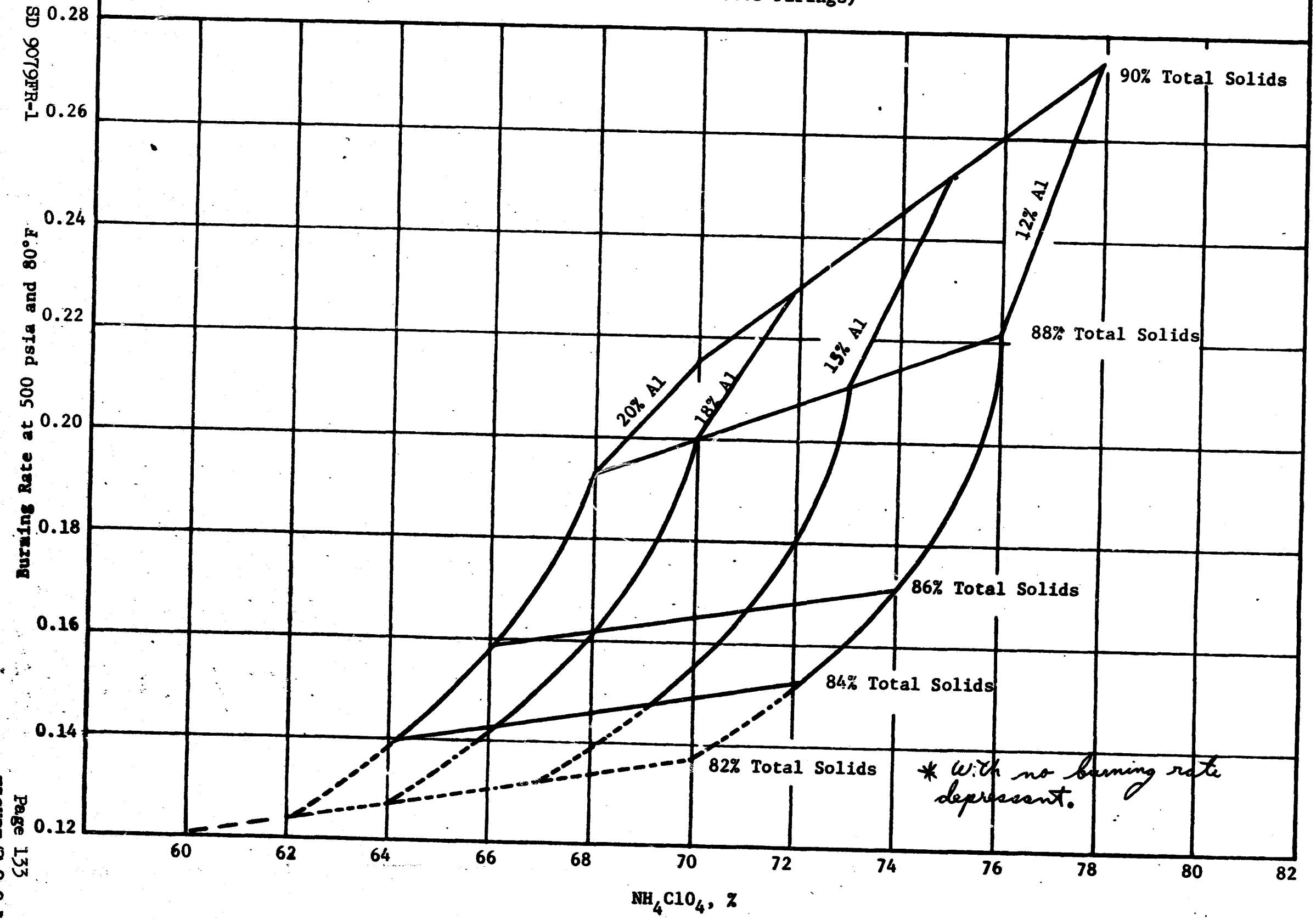


FIGURE 3.2.2-8

MINIMUM BURNING RATES FOR POLYBUTADIENE PROPELLANTS AT VARIOUS SOLIDS LEVELS*
(3KS-500 Size Motor Firings)



Page 133
FIGURE 3.2.2.19

	Weight %
Ammonium perchlorate	62
Aluminum	20
HTPB binder	18
	100

On the basis of solid strand burning rate results for propellant from a 1-lb laboratory batch and extrapolation of thermodynamic data for HTPB propellants with 84 wt% to 90 wt% solids, this propellant will be characterized by a delivered specific impulse of 237 lbf-sec/lbm, a density of 0.062 lb/cu in., and a motor burning rate of 0.12 in/sec at 500 psia. Although the solid strand burning rate at 500 psia was 0.124 in/sec, the propellant must be fired in 3KS-500 size (6-lb grain) motors to firmly establish both rate and pressure exponent. Additional formulation tailoring, including adjustment of the oxidizer/aluminum ratio, may be required after such tests are completed. The use of additives to further reduce burning rate may result in reductions to .09 in/sec.

Mechanical property data, presented in Table 3.2.2.22 for a very low burning rate propellant formulated with 64 wt% ammonium perchlorate and 18 wt% aluminum, illustrates the outstanding properties obtained with HTPB systems at the 82 wt% solids level. The very high elongations characteristic of HTPB propellants at low temperatures assure grain stability on storage and cycling to low temperatures. The results of cycling tests with the 64/18 wt% formulation in small strain evaluation motors are shown in Figure 3.2.2.20. In these tests, two grains having inner-bore hoop strains of 21.2% at -75°F and two grains with strains in excess of 40%* were held at +110°F for 12 days, cycled three times between +110°F and -75°F, and then three times from +180°F to -75°F. All grains completed this rigorous test program successfully and are now in storage for four weeks at +180°F prior

* 40% is the approximate motor strain requirement at -40°F.

Table 3.2.2.22

UNIAXIAL TENSILE PROPERTIES OF PROPOSED PROPELLANT FOR THE[1] SUSTAINED PHASE OF THE NINE-INCH SOLID PROPELLANT SOUNDING ROCKET MOTOR

	Tensile Properties			
Test Temp., ^{o}F	σ_m, psi	ϵ_m, %	ϵ_b, %	E_o, psi
180	31	16	25	295
150	37	19	36	305
110	49	25	39	350
77	65	32	46	440
40	84	41	54	535
0	128	43	60	1000
-40	225	38	70 *	2390
-75	417	29	45	4890

(1) Batch No. 10MM-6753

* Motor strain requirement is approximately 40% at this temperature.

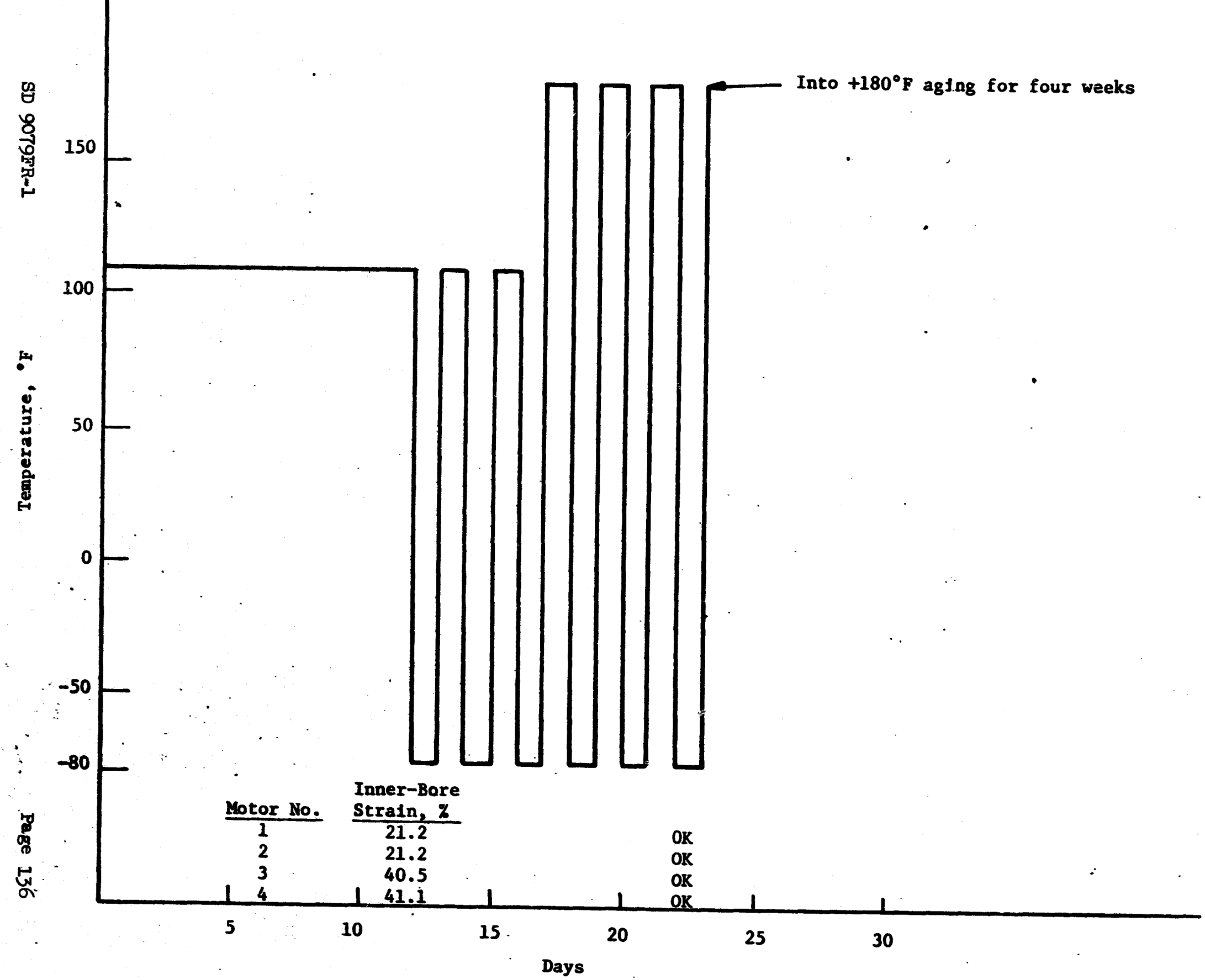


FIGURE 3.2.2.20

to additional cycling tests. Accelerated aging studies have shown HTPB propellants to be stable on storage at temperatures up to +180°F, as illustrated in Table 3.2.2.23 by data for a propellant containing 87 wt% ballistic solids. HTPB propellants and liners form strong and stable bonds, as shown in Tables 3.2.2.24 and 3.2.2.25.

To provide an HTPB formulation which fulfills the burning rate requirements of the sustainer grain, it will be necessary to (1) process the present formulation in a pilot-plant batch, cast 3KS-500 size motors, and establish burning rate, (2) adjust the present formulation, if necessary, to provide the required burning rate in 3KS-500 size motors, and (3) characterize the mechanical properties and safety of the final formulation. Thermodynamics calculations will also be carried out with the final formulation. These tasks can be completed on a low-cost, low level-of-effort program of short duration.

The scope of development effort will be the same as for the cartridge loaded design described in Section 3.2.2.7 of this report. The preliminary motor design based on the above discussion is illustrated in Figure 3.2.2.15. The chamber pressure and vacuum thrust time histories are illustrated in Figure 3.2.2.16. Overall motor physical and propulsion characteristics are contained in Table 3.2.2.20. A very preliminary reliability analysis, similar to that discussed in relation to the basic motor design was run and is summarized in Table 3.2.2.26.

Table 3.2.2.23

STORAGE CHARACTERISTICS OF HTPB PROPELLANT CONTAINING 87% SOLIDS

300-lb, Batch 67-511)

Storage Time, Weeks	Mechanical Properties at +77°F After Storage at the Indicated Temperatures											
	180°F				160°F				110°F			
	σ_m psi	ϵ_m %	ϵ_b %	E_o psi	σ_m psi	ϵ_m %	ϵ_b %	E_o psi	σ_m psi	ϵ_m %	ϵ_b %	E_o psi
0	125	57	61	662								
1	106	56	59	520	106	58	60	538	101	52	54	535
2	130	58	59	684	106	62	64	621				
7	173	46	48	841	103	49	50	540	107	60	62	463
12	188	45	46	910	125	53	53	682	101	48	52	579

Table 3.2.2.24

BONDING PROPERTIES OF HTPB LINER*

(88% Total Solid Propellants)

Formulation	Bonding Strength at 77°F					
	Holding Time Prior to Propellant Casting, Days					
	0			3		
	Tensile	Shear	Peel	Tensile	Shear	Peel
1	99	76	38	64	51	24
2	101	77	33	93	61	16
3	133	101	26	--	--	--
4	88	85	24	83	67	17

* Filler System: 10.3% asbestos, 1.75% Thixcin E

Table 3.2.2.25

STORAGE CHARACTERISTICS OF HTPB LINER-PROPELLANT BOND
(88% Total Solids Propellant, SD-891 Liner)

Storage Time, Weeks	Storage Temperature, °F			
	180°F		160°F	
	Tensile	Shear	Tensile	Shear
0	88	77	88	77
2	82	84	--	--
3	--	--	89	79
5	80	88	83	92

Table 3.2.2.26

SMALL PROBE ROCKET MOTOR RELIABILITY

Component/Event	Reliability Estimate for Design	Proposed Design Support Tests	Reliability at Production Start when Program Safety Margins are Demonstrated (90% Confidence Level)(*)
1. Ignition System a. Squib Actuation b. Igniter Actuation c. Igniter Chamber	0.9957	63	0.9987
2. Booster Grain a. Base Material Strength b. Ballistic Parameters	0.9778	30	0.9979
3. Sustainer Grain (1) a. Base Material Strength b. Ballistic Parameters	0.9778	30	0.9951
4. Case a. Base Material Strength b. Welds (2) c. (Not Applicable) d. External Paint	0.9986	69	0.9984
5. Internal Insulation a. Booster b. Sustainer (1) c. Forward Head d. Spacers (1) e. (Not Applicable)	0.9830	60	0.9977
6. Nozzle a. Entrance & Throat b. Exit & Shell c. Seal d. Fin Structure (4)	0.9984	105	0.9987
$R_m = R_1 R_2 R_3 R_4 R_5 R_6$	0.9329	-	0.9865

(*) "Best Estimate" Reliability = 0.9975 for reduced program and both designs, IF original program safety margins are demonstrated at start of production.

3.2.2.4 LOGISTIC CHARACTERISTICS

The Small Probe Rocket Motor is a very simple design and imposes few restrictions in employment. The motor design is safe and mobile, requires a minimum of maintenance, practically no support equipment, and has a long shelf life under most storage conditions.

SAFETY

The motor propellant is designated as a Class 2 explosive, which is not a severe restriction as the motor is insensitive to all but very extreme shock forces. The motor must not be exposed to flame or temperatures in excess of 519°F which is the autoignition temperature of the booster propellant. Very high charges of static electricity could conceivably ignite the motor. After the igniter is installed, direct or induced electrical currents in the igniter circuit up to 1 amp for 5 min can be tolerated, beyond that level the danger of motor ignition exists. No toxic or corrosive materials are used in the motor.

MAINTENANCE AND INSPECTION

The maintenance capability most probably required would be touch up of scratches to prevent rust and ensure even aerodynamic heating during flight. Replacement of the weather seal might be required if it is removed for inspection or accidently during transportation and handling.

No special inspections are required for the motor. Visual checks must be made to ensure that the motor has not been damaged, and during storage, inspection of the case for scratches will avoid the possibility of rust damage. An inspection to ensure that the weather seal is in place will avoid the possibility of exposing the propellant to unnecessarily extreme conditions. Chips and cracks in the nozzle exit cone or dents in the case should be reported and evaluated at that time to determine if the motor is satisfactory for use. The recommendation is made that an inspection log accompany each motor so that a history of environmental conditions and motor performance during operational use can be maintained for reliability analyses.

STORAGE AND AGING

The recommended storage temperature for the motor is 60 to 70°F although the storage capability ranges from -65 to +140°F. Higher temperatures tend to increase the motor aging process, and cycling to low temperatures induces stresses in the propellant that can reduce the propellant capability over a period of time. The minimum shelf life for this motor is 5 years under normal conditions.

SUPPORT EQUIPMENT

The only special tool required for support of this motor in the field would be a long rod type device that would be used to push the igniter through the motor bore from the aft end of the motor to the forward end of the motor. Just prior to installation and firing, an electrical check of the igniter circuit to ensure continuity through the squib would be recommended although not absolutely necessary. A power supply capable of supplying 5-8 amps to the igniter would be required for each firing.

In extremely cold environments, the use of an electrical heating blanket to keep the motor warm prior to firing would enhance performance reliability. Although desirable, this capability would probably not be required except after prolonged conditioning below -65°F; a lower temperature capability for the motor may be demonstrated during development.

3.2.2.5 RELIABILITY

The initial reliability of the Small Probe Rocket Motor design is estimated to be at least 84.12% at the 90% confidence level. The proposed reliability goal to be demonstrated by the start of the production program is 99.0% minimum, at the 90% confidence level. Reliability growth during the development and demonstration test phases will be assessed from measurement of actual equipment performance data, by comparison with data for rocket motors of similar configuration, and by detailed analyses of design improvements and corrective actions.

The first step in evaluating the probability of success was to classify the design configuration into a series of components and events that occur during the mission. The component/event arrangement (Table 3.2.2.27) which is termed a reliability model, is represented for this motor as an independent serial system; the assumptions are made that operation of an individual component/event does not affect operation of the others and that the failure of any one of these to meet requirements constitutes failure for the entire motor to complete its mission. Failures can be catastrophic, resulting in hazard to personnel or destruction of the system, but also can concern failure to attain a specified design goal, e.g., total thrust. The product rule for combining reliabilities was applied, and overall reliability at each phase was determined from the relationship:

$$Rm = R_1 R_2 \cdot \cdot \cdot \cdot \cdot Rn$$

where: Rm - overall motor reliability, and $R_1 R_2 \cdot \cdot \cdot \cdot$ Rn - reliability of the first, second, and nth component/event multiplied together. The product rule was applied also for the reliability calculation of each of the major component/events from subcomponent/event reliabilities.

The subcomponent/event reliabilities were estimated for the initial design from a detailed failure modes-and-effects analysis in conjunction with a review of historical data for tests of related types of components materials and events. The analysis also considered that quality control surveillance would be limited to the extent required to avert hazard-type failures (danger to personnel) and to minimize critical-defect type failures (abort mission).

The analysis of the design pinpointed two potential problem areas, both of which were related to the sustain grains (Table 3.2.2.27). It was considered that the stackup of ten individual sustain grains might require close control and possibly selective matching of individual units for the resulting sustain grain properties and ballistics to remain within specified limits. It is anticipated, however, that the allowable limits will be wide enough to preclude this requirement. The estimated reliability for the

Table 3.2.2.27

SMALL PROBE ROCKET MOTOR RELIABILITY

Component/Event	Reliability Estimate for Design	Proposed Design Support Tests	Reliability at Production Start when Program Safety Margins are Demonstrated (90% Confidence Level)(*)
1. Ignition System a. Squib Actuation b. Igniter Actuation c. Igniter Chamber	0.9957	63	0.9987
2. Booster Grain a. Base Material Strength b. Ballistic Parameters	0.9778	30	0.9979
3. Sustainer Grains (10) a. Base Material Strength b. Ballistic Parameters	0.9057	300	0.9988
4. Case a. Base Material Strength b. Welds (2) c. Compression Spring d. External Paint	0.9984	69	0.9984
5. Internal Insulation a. Booster b. Sustainer (10) c. Forward Head d. Spacers (9) e. Internal Paint	0.9570	330	0.9988
6. Nozzle a. Entrance & Throat b. Exit & Shell c. Seal d. Fin Structure (4)	0.9984	105	0.9987
$R_m = R_1R_2R_3R_4R_5R_6$	0.8412	-	0.9913

(*) "Best Estimate" Reliability = 0.9975 for reduced program and both designs, IF original program safety margins are demonstrated at start of production.

sustainer grain design is 90.57% because individual grain reliability estimates were raised to the 10th power.

The multi-grain design also influences the design estimate for the internal insulation. The junctures and spacers between grains may require modification, depending on test results. This uncertainty resulted in an estimate of 95.7% reliability for the insulation design since the 10th power relationship applied here also. The reliability estimate for the motor design is 84.12% leaving a remainder of about 16%. However, 14% of this 16% can be attributed to the sustain grain and its insulation/spacer system. Early tests will show whether these component/events will, in fact, be problem areas, and, if so, design changes for corrective action will be quickly incorporated.

A reliability goal of at least 99.0% at the 90% confidence level is proposed for the production motors. The goal is to be attained by the end of the development and demonstration test phases, or 42 weeks. When considered with minimum quality control surveillance, except to avert hazard-type failures and to minimize critical-type-failures, this reliability goal compares well with the observed reliability of similar Aerojet-produced solid rocket motors of 99.8%.

To demonstrate that the proposed reliability goal has been met or exceeded and is being maintained, use will be made of all applicable data coming from the development and demonstration test programs and from any additional tests and operational firings made during the production phase. The data includes not only direct information from test performance, but also comparison with rocket motors of similar configuration and a detailed assessment of the effects of design improvements and corrective actions.

3.2.2.6 POTENTIAL PROBLEMS

The basic technology incorporated in the design of the Small Probe Rocket Motor is well defined and would not be expected to cause any serious development problems. There are several design concepts utilized, however, that require demonstration and, potentially, some development effort.

The very long grain with a small port area creates the possibility of unstable or erosive burning during the initial phases of operation. Unstable burning would result in pressure and thrust oscillations, which, depending on their magnitude, may or may not be detrimental to the motor function.

The use of baffles between the grains or a modification to the propellant formulation would undoubtedly solve this potential problem. Erosive burning, if it occurs, would probably create a higher-than-predicted pressure during the initial portion of the firing. Reducing the initial burning surface area and increasing the port to throat ratio are two methods of combatting this problem. The magnitude of the over-pressure would determine the extent of the problem and appropriate adjustments in the grain design would then be made.

To predict the presence of these two phenomena has been difficult in the past, and there is no basis to state whether these conditions will or will not exist. During previous motor development programs, it has been possible to overcome these problems. The important factor in this situation is to have development motors available to demonstrate the design fixes. The number of motors available in the proposed development program is adequate to demonstrate a satisfactory grain design.

Historically, bonding of propellant to the pressure vessel or insulating container has been one of the most difficult problems to solve in the development of solid rocket motors. In the Small Probe Rocket Motor this aspect is not considered a major technical problem, however, the processing steps involved contribute significantly to the cost of the motor. These steps consist of coating the phenolic sleeves with FM-47 primer and then lining the sleeves with a rubberized liner prior to propellant casting. Ideally, it would be desirable to cast directly into the sleeve and avoid the two processing operations of priming and lining the chamber. This is a cost-oriented problem and effort is included in the development plan to demonstrate a simpler method of bonding the propellant to the insulation sleeve either by eliminating the primer or liner or by selecting a substitute material for the paper-phenolic sleeves that is more adaptable to bonding the propellant.

3.2.2.7 DEVELOPMENT PLAN

SCOPE

This plan describes the effort proposed for developing a low-cost, 6-in. dia., 30,000 lbf-sec impulse, dual-thrust solid rocket motor for use in meteorological applications. Included in this effort are preliminary ballistic demonstration tests, motor detail design and analysis, flight weight motor development tests, and flight motor demonstration tests. Propellant support effort and motor environmental testing is also included. Brief discussion is also provided of an alternative development plan.

OBJECTIVE

The objective of the development program is to develop and demonstrate a reliable, low-cost, solid-propellant propulsion motor for use in meteorological rockets. A maximum diameter of 6 in. is to be maintained, and the total impulse is to be 33,000 lbf-sec delivered during a 3500-lbf thrust boost phase and a 1100-lbf thrust sustain phase over a 27 to 30 second duration.

PROGRAM DISCUSSION

The development program will be conducted in accordance with the key milestone schedule shown in Figure 3.2.2.21 and the detailed milestones shown in Figure 3.2.2.22. For purposes of discussion, the program has been broken into four major tasks: (1) motor design, (2) component design, (3) component testing, and (4) motor testing.

Motor Design - A motor design has been selected that incorporates a case of AISI 4130 steel with a tensile strength of 180,000 psi. The forward closure and the aft closure-nozzle assembly are attached by welding. Provisions will be made for attaching fins on the aft end and the payload on the forward end.

The propellant grain will consist of ten sustain grains each approximately 9-in. long with a cylindrical bore plus a similar boost grain approximately

The use of the paper-phenolic sleeves as an insulator must be demonstrated to ensure that adequate case insulation is maintained throughout the firing. Exposure of the end portion of the insulator sleeves to the gas stream for the full firing duration plus the presence of gases behind the sleeves makes this area the most critical in the design. Calculations indicate that a considerable temperature rise will occur in the case by the end of the firing, consequently higher-than-anticipated heat transfer would cause a potential problem. Early tests will demonstrate the actual temperature rise in the case, and appropriate adjustments can be made as required. The additional temperature rise from aerodynamic heating will be evaluated by early flight tests. There are an adequate number of motors in the development program to demonstrate any design adjustments required because of heat transfer.

Welding of the nozzle on the loaded motor is a unique concept, and an adequate number of tests have been inserted in the program to demonstrate the techniques to be utilized. If some difficulty is encountered in developing the weld procedures, an alternative retention system could be incorporated for initial development tests. It is anticipated that this effort could be completed within the scope of the proposed program.

The proposed development program was formulated with these potential development problems in mind and an adequate number of tests were scheduled to resolve and demonstrate a satisfactory solution to any of these potential problems. In addition, a definition of motor performance over the operating range will be obtained. If no problems materialize, motor performance can be demonstrated in more detail and alternative materials and designs can be evaluated to a greater extent than would be otherwise possible, resulting in an optimum design. Recommendation is made that the development program to provide a higher performance propellant as discussed in Section 3.2.2.3 be incorporated into the program so that if weight penalties are incurred during development, the desired motor performance can still be achieved by virtue of the improved propellant characteristics.

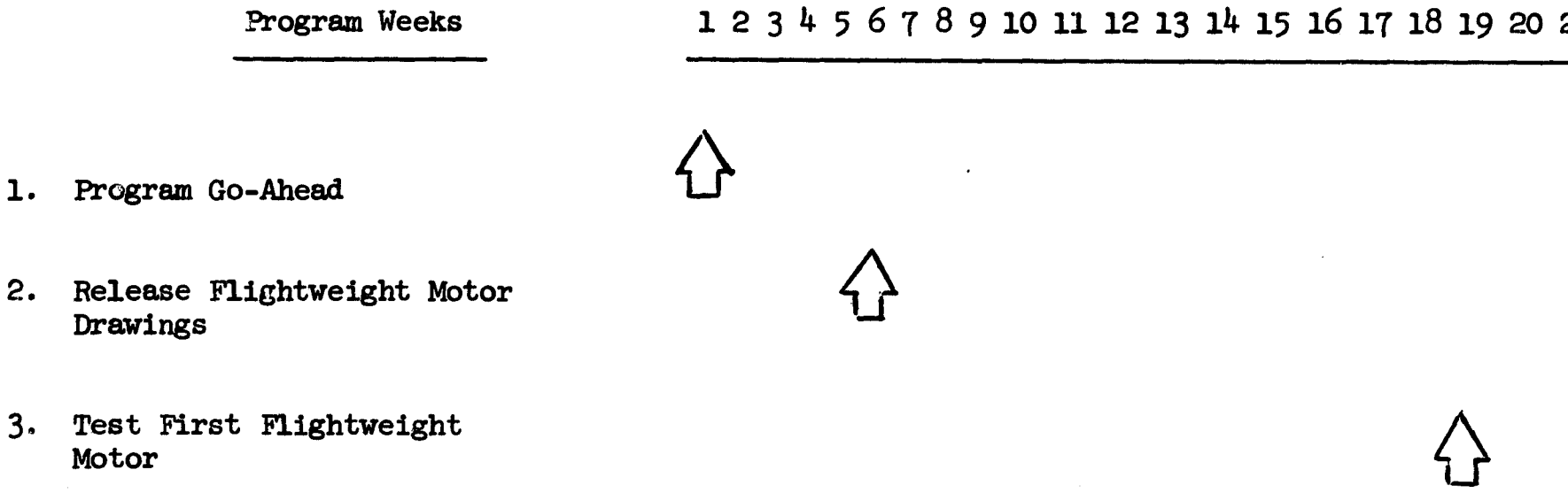


Figure 3.2.2.21

SMALL PROBE PROPULSION SOLID KEY MILESTONES

FOLDOUT FRAME 1

19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40 41 42 43 44 45 46 47 48 49 50

.2.2.21

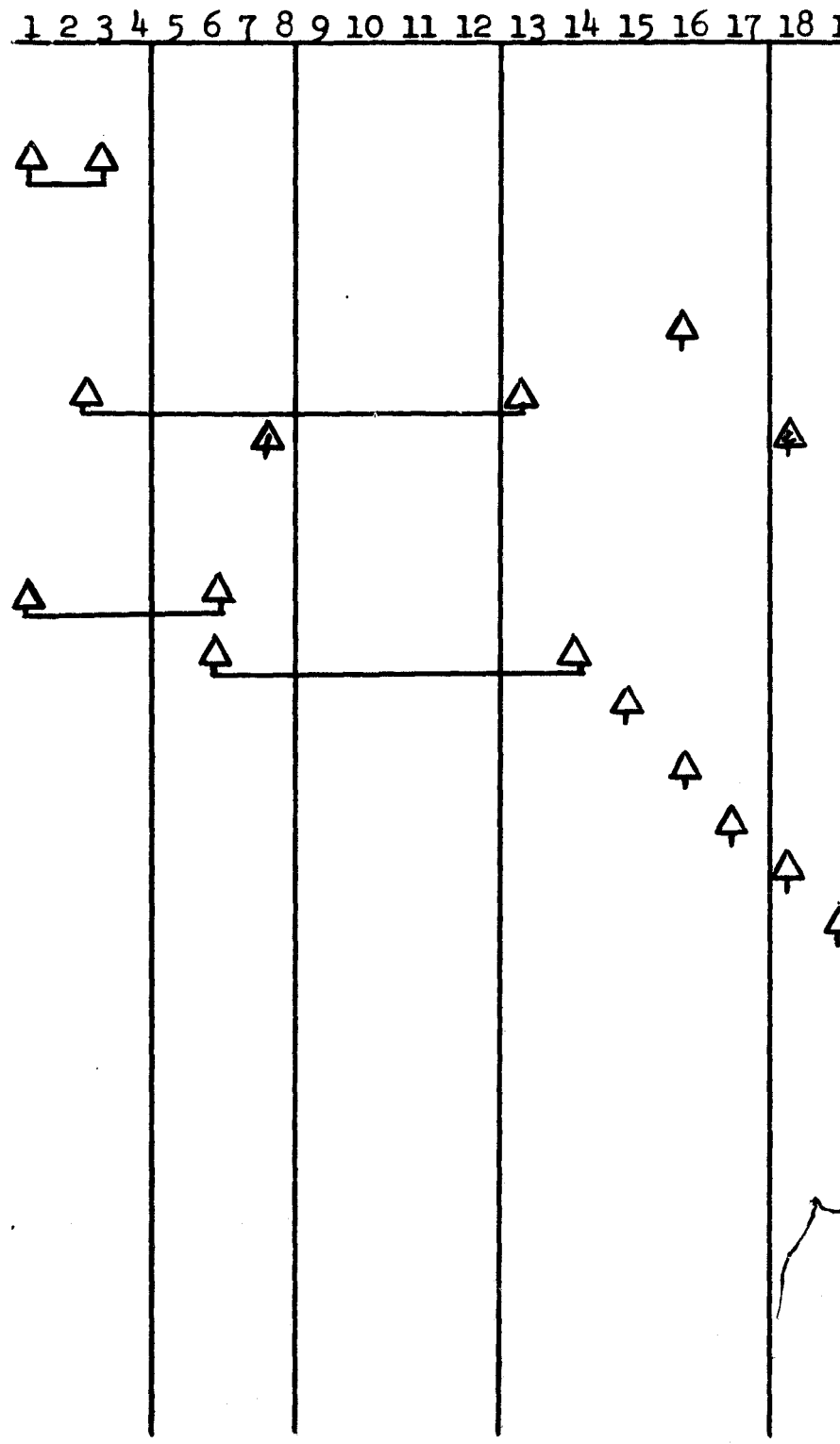
N SOLID ROCKET MOTOR

ESTONES

FOLDOUT FRAME 2

1

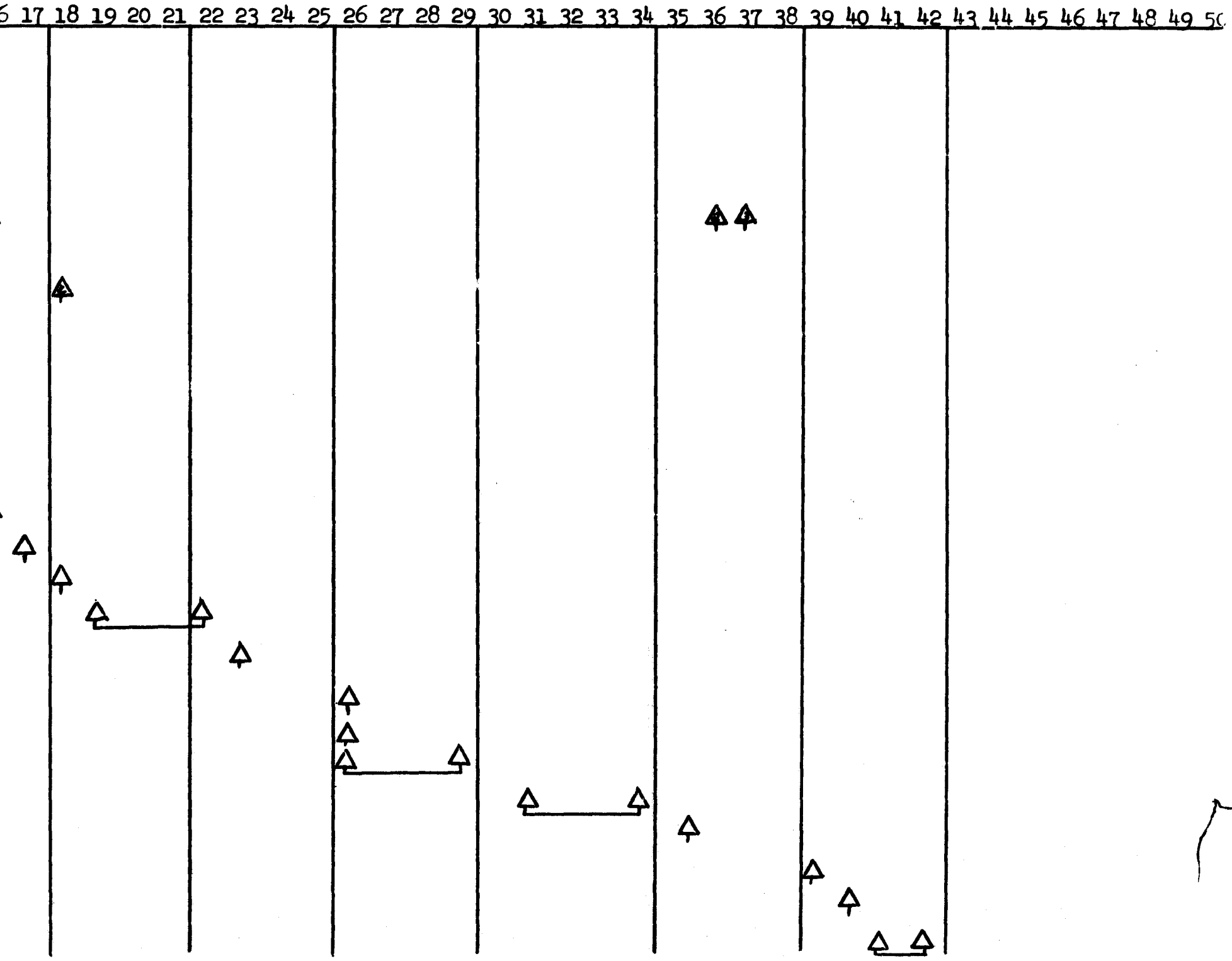
	Program Weeks
I. PROGRAM GO-AHEAD	
Prepare Program Plan	
II. COMPONENT DEVELOPMENT TESTS	
Case Burst Tests	
Propellant Bond Tests	
Igniter Lot Acceptance	
III. FLIGHTWEIGHT MOTOR DEVELOPMENT	
Analyses and Design Release	
Procure Hardware	
Case Grains (3 ea. Lot 1)	
Cut and Trim Grains	
Assemble Motors	
Test Fire (3 ea. -65, +60, +130°F)	
Analysis	
Cast Second Lot of Grains (6)	
Ballistic Test 2nd Lot of Mtr. (3)	
Flight Test Deliveries (3)	
Flight Tests	
Analysis	
Cast Third Lot of Grains (6)	
Fire Ballistic Tests (3)	
Flight Test Deliveries (3)	
Flight Tests (3)	



SMALL PROBE PROPULSIO

MILESTONE

FOLDOUT FRAME 1



PROPULSION SOLID ROCKET MOTOR

MILESTONE SCHEDULE

Figure 3.2.2.22

FOLDOUT FRAME 2

6 in. long. The grains will be cast in phenolic sleeves and loaded into the motor after curing. Spacers will be placed between the grains to permit end burning and to allow pressure to equalize between the case and grain. A spring in the forward end of the motor will apply compression to the grain and keep the grains and spacers in contact during temperature cycling.

Internal insulation of the steel chamber will be provided by the phenolic sleeves that contain the propellant. In addition, a thin coat of a rubberized material or a temperature-resistant paint will be applied to the chamber interior to provide additional thermal protection. The external surfaces of the chamber will be painted to inhibit corrosion and to partially overcome effects of aerodynamic heating.

Motor ignition will be provided by an aft-end inserted igniter containing boron potassium nitrate (BPN) pellets initiated by a single squib.

The nozzle will be a simple, low-cost design and will utilize an eroding throat and a minimum of component parts.

The overall motor design will be defined and controlled by a design-assembly technique that will utilize component details for the case, insulation, grain, nozzle, and igniter, all integrated by a Master Control Layout (MCL). The MCL will be used to (1) check installation fits, (2) verify the compatibility of mating surfaces with respect to corrosion prevention, and (3) check the adequacy of locking provisions as well as providing a means of reviewing over-all assembly problems. Detailed information on the MCL will include key dimensions, material and specification callouts, and weight and performance summaries.

Component Design

a. Propellant Grain Design

The motor grain design will consist of 10 small sustain grains of low-burning-rate propellant cast into plastic sleeves and cartridge loaded in the motor. Motor boost thrust will be provided by a single, additional grain cast with a high burning rate propellant. Final grain sizing will be

based on a detailed ballistic analysis for optimization of performance with respect to the selected configuration and criteria. Variations in nozzle expansion ratio will be evaluated to obtain the best total performance during the boost and sustain phases. Detailed stress analyses will be conducted to confirm that the propellant grain will withstand the environmental and firing conditions imposed by the design criteria.

Mechanical properties of the propellants selected for the boost and sustain grains will be reviewed in detail to ensure compatibility with the selected grain configuration. The propellant to be used in the sustainer will require a burning rate of 0.08 in/sec at 500 psi and the booster grain will burn at 1.8 in/sec at 2000 psi.

b. Insulation

A detailed thermal analysis will be conducted to establish the type and thickness of case insulation materials. The type of material and thickness of the phenolic sleeves will be established, and a material and thickness will be selected for the additional protective coating on the chamber interior surface. Similarly, the protection from aerodynamic heating obtained from external paint will be evaluated. A final combination of insulators will be selected such that total internal plus external heat transfer to the case will not raise the temperature of the case to an unsafe level prior to the end of motor action time.

c. Case

A design will be established ror a 180,000 psi tensile strength AISI 4130 steel case with a maximum OD of 6 in. An inverted forward dome will be welded in place prior to heat-treatment. To maintain the maximum propellant loading capacity and still maintain the 6-in. OD, it is planned to eliminate attachment bosses and to weld the nozzle to the chamber. Alternative nozzle attachment methods will be evaluated. A stress analysis will be conducted of the final design with special emphasis applied to the weld joints to ensure design reliability and minimum cost consistent with production procedures.

d. Nozzle

The nozzle design will be established with primary emphasis on simplicity and low cost. An eroding throat insert will be utilized, and low-cost phenolic materials will be evaluated for use in the entrance section and exit cone. Investigations will be made to determine the feasibility of using a single-piece plastic nozzle. Stress and thermal analyses will be conducted to ensure that the design is adequate.

e. Igniter

The igniter will be designed for installation at the motor aft end just prior to firing. A single squib will be used to initiate the igniter to save costs relative to a redundant squib design. The igniter will be designed to withst d the same environmental criteria as the motor although it normally will not be exposed to the same extremes since it will be packaged and stored separately.

Component Development Testing

a. Case

Three burst tests will be conducted, as outlined below, to demonstrate the case design margins and production techniques. One test will be conducted early in the program to demonstrate weld techniques and to gain additional design data. The remaining two tests will be conducted later in the program to demonstrate any design or fabrication changes that might be required or to obtain a better indication of the actual design capability. The structural portion of the nozzle housing will be included in these tests.

Case No.	Development Design	Production Design	Proof Test	Burst Test
1	X		X	X
2		X	X	X
3		X	X	X

b. Propellant-Liner

A low-level-of-effort, low-cost laboratory program is proposed to; (1) establish the bond strength of SD-886 and SD-793 liner materials to the Micarta sleeves into which the booster and sustainer propellants will be cast; (2) establish material allowables for the propellant/liner/Micarta systems; (3) determine if a single liner (SD-886, SD-793, or other can be used with both the booster and sustainer propellants; (4) determine if the Micarta sleeve must be primed prior to liner application, and; (5) determine if the propellants can be bonded directly to a primed or unprimed Micarta, thus eliminating need for a liner.

The first study must be conducted, since neither of the proposed liners has been extensively tested in Micarta sleeves and the Micarta-liner bond strengths have not been established. Effort under Items (3), (4), and (5) above, is proposed to provide the most economical grain processing technique commensurate with satisfactory bond strengths. The use of a single liner for both the booster and sustainer propellants or the elimination of a liner and the elimination of a primer on the Micarta would significantly reduce grain fabrication costs. Item (2), the establishment of bond system allowables, will be necessary regardless of the final system selected.

The program will be conducted, for the most part, with small laboratory test specimens. For screening purposes, composite propellant/liner/Micarta samples will be tested at a strain rate of 0.74 min^{-1} for tensile, shear, and peel strengths at -65, +77, and +160°F. The most promising approaches will then be refined, as necessary, and additional tests will be conducted under the same conditions both before and after accelerated aging at high temperatures. Test specimens prepared according to the final approach selected will be tested at five temperatures between -65 and +160°F and at from three to six strain rates, which will provide the low and high-rate data required for establishment of allowables. A second set of specimens will be stored at high temperatures and tested after four storage periods. Specifications and procedures for the selected approach will be prepared for incorporation

into the grain processing procedure. This effort should provide bonding systems that will ensure low cost processing while providing the bond strengths required to assure grain system integrity both before and after aging.

c. Igniter

Two lots of 18 igniters will be fabricated, and six igniters from each lot will be fired for acceptance testing. The remainder of the igniters will be used in motor development tests and spares.

Motor Development Testing

a. Flight Weight Motor Development

Immediately upon receipt of program "go-ahead", motor design analyses will be initiated and detailed motor and component drawings will be prepared and released. Fifteen sets of hardware will be procured to support the development tests. Initially, three motors will be cast to the design configuration. These motors will be fired for ballistic evaluation.

A second series of six motors will be cast pending success of the previous lot of ballistic tests. Three of these motors will be fired at three key temperature levels, and three will be flight tested.

A third series of six development motors will be tested to demonstrate selected nozzle materials, insulators, and potentially, a modified propellant configuration. Three motors will be tested for ballistic evaluation, and three will be flight tested.

During this phase of the program it is anticipated that certain design and material changes may be required. Consequently, provisions will be included for alternative insulator sleeves, nozzles, and propellant grain spacers. No backup requirements are anticipated for the case. Additionally, it is anticipated that some effort will be required to establish the procedures for welding the closure and for proof testing the loaded chambers. A study will be made to ensure that the case liner does not adhere to the propellant

sleeve during storage and then pull away from the case during subsequent handling.

Alternative Development Plan

a. Propellants Providing Improved Ballistic Performance Capability

Propellants similar in composition to the proposed sustain formulation, but with an expected delivered specific impulse of up to about 235 lbf-sec/lbm have been processed satisfactorily in laboratory batches. The compositions and ballistic properties of these formulations are compared below with the sustain formulation (Propellant A) proposed for use in the base motor design:

Propellant	NH_4ClO_4	NQ[1]	Al	PU Binder	Total Solids	Expected Delivered I_s, lbf-sec/lbm	Burning Rate at 500 psia, in/sec
A	61	21	-	18	82	217.6[2]	0.08
B	66	17	-	17	83	224.0	0.080
C	68	16	-	16	84	227.2	0.080-0.090
D	60	16	8	16	84	235.0	-

(1) NQ is nitroquanidine
(2) Measured

The burning rates shown for propellants B and C are the minimum, or near minimum, rates attainable with the formulation. A burning rate in excess of 0.09 in/sec at 500 psia can be expected for Propellant D, since aluminum is included in the formulation. Before this formulation could be seriously considered, it would be necessary to determine the minimum burning rate attainable with a formulation having acceptable mechanical properties.

To tailor any of these formulations (B, C, or D) as required to meet the requirements of a given grain design, a program of 20 weeks duration would be necessary. Such a program would permit (a) tailoring of burning rate and mechanical properties, (b) processing scale-up from 1-lb laboratory batches to 400-lb pilot-plant batches, (c) characterization of burning rate, pressure exponent, temperature sensitivity (π_K), mechanical properties, and bonding properties as a function of temperature, (d) measurement of temperature cycling capability in strain-evaluation cylinders, and (e) establishment of safety properties. Mechanical-property and bond tests would be conducted as required to establish preliminary material allowables. When a final formulation is selected and scaled-up to the 400-lb batch level, propellant and propellant-liner specimens would be stored at elevated temperatures to determine aging stability. To provide a sufficiently long storage period, this phase of the study would continue for at least 16 weeks beyond completion of the tailoring and characterization effort.

The improved propellant formulation would be available for incorporation in the third lot of development motor tests. Recommendation is made that this development effort be conducted to provide an additional margin of performance to overcome any potential weight increases that might occur during development.

3.2.2.8 DEVELOPMENT AND PRODUCTION COSTS

Two budgetary and planning cost estimates were prepared for the development plan proposed in Section 3.2.2.7 and for production lots of 100, 1000, 5000, and 10,000 units. The original production estimate was based on the use of present manufacturing methods and in an effort to reduce the estimated cost a subsequent estimate was prepared based on an automated manufacturing sequence. The second development program includes delivery of 100 production motors.

ESTIMATED COSTS BASED ON CURRENT MANUFACTURING SYSTEM (SRO 671265A)

	Design/ Devel.	Prod. Alt. I	Prod. Alt. II	Prod. Alt. III	Prod. Alt. IV.
Program Length	10 mos.	12 mos.	12 mos.	12 mos.	12 mos.
Deliveries	N/A	100	1,000	5,000	10,000
Deliv./Rate/ Mo.	N/A	10	100	500	1,000
Sub Total Labor Mat'l.	$538,783	*$390,792	*$2,351,773	*$6,361,008	*$10,167,847
Tooling	$110,830	Avail. from Des.& Dev.	$133,000	$200,000	$250,000
Unit Cost	N/A	*$3,908	*$2,408	*$1,312	*$1,042
Total Prog. Cost	$649,613	*$390,792	*$2,484,773	*$6,561,000	*$10,417,847

*Note: These production costs are based on receiving production go-ahead the fifth month of the Design and Development Program.

ESTIMATED COSTS BASED ON AN AUTOMATED MANUFACTURING SYSTEM (SRO 671265B)

	Design/ Devel.	Prod. Alt. I	Prod. Alt. II	Prod. Alt. III	Prod. Alt. IV
Program Length	12 mos.	N/A	12 mos.	12 mos.	12 mos.
Deliveries	100	N/A	1000	5000	10,000
Deliv./Rate/ Mo.	N/A		100	500	1,000
Sub Total Labor Mat'l.	$871,673		*$1,329,596	*$5,016,705	*$8,056,487
Tooling	$110,830		$133,000	$200,000	$250,000
Total Unit Cost	N/A	N/A	*$1,463	*$1,043	*$831
Total Prog. Cost	$982,503	N/A	*$1,462,596	*$5,216,705	*$8,306,487
Total Cost Alt. I less Design & Dev. Costs	$332,890	N/A			
Unit Cost (less Design and Dev.)	$3,333				

*Note: These production costs are based on receiving production go-ahead the fifth month of the Design and Development Program, with the exception of Alt. I which will run entirely concurrent with Design and Development.

PRELIMINARY CAST IN CASE ALTERNATE DESIGN

ESTIMATED COSTS BASED ON AN AUTOMATED MANUFACTURING SYSTEM

(SRO 671 265B)

	Design/ Dev.	Prod. Alt. I	Prod. Alt. II	Prod. Alt. III	Prod. Alt. IV
Program Length	10 mo.	12 mo.	12 mo.	12 mo.	12 mo.
Deliveries	N/A	100	1,000	5,000	10,000
Del./Rate/ Mo.	N/A	10	100	500	1,000
Sub. Total Labor Mat'l.	$317,944	$207,431	$1,034,598	$3,359,343	$5,809,320
Tooling	$ 50,877	$ 61,490	$ 132,746	$ 200,836	$ 253,319
Unit Cost	N/A	$2,689	$1,167	$712.	$606.
Total Prog. Cost	$368,821	$268,921	$1,167,344	$3,560,179	$6,062,639

BASIS FOR COST ESTIMATES

1. Motor hardware costs are print estimates at T-100 and extended on a curve for larger buys.

2. Standard propellant costs dated 14 December 1967 were used.

3. Manufacturing labor was estimated from a manufacturing process flow chart, other labor was estimated the same as on similar Development and Production Programs.

4. The SRO 671265B Design and Development was extended to 12 months and includes the production order of 100 units to be processed concurrent with the development effort.

5. Funding of approximately $500,000 for establishing the automated system would be required, however it should be noted that the associated costs would be saved prior to completion of the first 1000 production deliveries.

COST REDUCTION POTENTIAL

In conjunction with the Small Probe Propulsion Study several factors have been identified that could contribute to reduced costs of sounding rockets. Some of these cost factors arise from the rigid specifications and controls that are required on weapon systems programs and used as standards for solid rocket design but could be selectively relaxed for sounding rocket programs.

One factor where significant cost savings could be made is in the area of low cost design. Currently, specifications and controls are based primarily on military oriented specifications. Screening of specifications and procedures to eliminate unnecessary testing, certification, etc., could provide a significant cost advantage. Commercial grade materials can be substituted for AGC or military spec materials. Tolerance limits can be opened up on non-critical components because the weight and performance values are not necessarily controlling design factors. It would not be a simple task however to implement these recommendations. Each component and interface between components would require careful evaluation to ensure that the resultant design had a high reliability with a minimum of quality control effort. Close

coordination between the design engineers and Space Division personnel would be helpful in establishing the level of reliability and performance controls that are required for this type of vehicle. Coordination with potential vendors of component parts is necessary to establish which controls are required to obtain reliable parts. This task can be best implemented by funding a pilot study prior to initiation of detail design.

Elimination or reduction of pooled charges to a level consistent with the scope of work required would be appropriate. For example, Quality Control functions are ordinarily based on a straight percentage of manufacturing. For a particular application such as the sounding rockets, it might be appropriate to estimate costs based on the specific tasks to be performed. Processing costs for propellant are also based on a pool system where most of the propellant controls and acceptance procedures are somewhat more extensive than anticipated for sounding rockets. Here again, adjustment of the costing on a task basis might show significant cost savings. Support costs for line processing effort would also be significantly lower for a commercial type effort if work scope was identified and costed by task rather than as a percentage of manufacturing labor.

Automation of manufacturing procedures would offer significant advantages for high motor production rates. Only limited automation has been adapted to solid rocket manufacturing to date; however, automated manufacturing concepts have been established for small solid rocket motors. Assurance of relatively long periods of continued production would make the meteorological rocket an especially attractive area to establish the first automated production facility. A partially automated system might be useful in the production of other sounding rockets which are procured in smaller quantities. Consolidation of work areas to minimize transportation and engineering surveillance would be a first step in this area.

A somewhat more drastic method of reducing costs would be to establish a separate operation (similar to Mark 56 Mine Program) with separate overheads. Automation to the level optimum for the projected production rates should be employed.

3.2.3 VAPAK PROPULSION UNIT

3.2.3.1 SUMMARY AND INTRODUCTION

Liquid Rocket Operations, Aerojet General Corporation, Propulsion Division has completed a conceptual design of a new propulsion system for a meteorological rocket vehicle, designated MSR-2. This relatively low-cost vehicle can deliver a 10-lb payload to an altitude on the order of 140 km. The MSR-2 (Figures 3.2.3.1 and 3.2.3.2), composite propulsion system is 114-in. long, with a maximum diameter of 6 in. The composite concept provides a single combustion chamber with dual thrust capability. A short-duration, high-level boost thrust is provided by a fast-burning solid propellant grain loaded into the liquid propellant thrust chamber. Sustainer thrust is provided by relatively safe, reliable, inexpensive liquid propellants. A total impulse of approximately 26,000-lb/sec is delivered during a one second 2,000-lb thrust-boost phase followed by an average sustainer thrust of 488-lb for approximately 50 seconds.

A summary of the motor performance and weight properties are shown in Tables 3.2.3.1 and 3.2.3.2, respectively. Prefire and postfire propulsion system weights are 140-lb and 39-lb, respectively.

The design incorporates an advanced pressurization (VaPak)* concept whereby the propellant tanks are pressurized by the propellant's own vapors. This provides a liquid propulsion system with inherent design simplicity and low cost. The principle of vapor pressurization is similar to that of a self-discharging insecticide or paint can. The propellant, after conditioning (heating), is expelled by its own vapor pressure without a significant change in discharge pressure. This eliminates the heavy, expensive components required in a typical liquid rocket pressurization system. High pressure gas bottles, gas generators, or regulators are not required; in fact, no moving components or devices are required whatsoever, once liquid flow from the VaPak tank is initiated. As a direct result of VaPak pressurization, propellants injected into the combustion chamber vaporize readily and thereby simplify injector design with associated cost savings.

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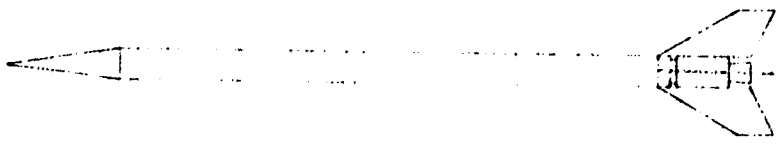
* Aerojet U.S. Patent No. 3,320,742

METEOROLOGICAL SOUNDING ROCKET

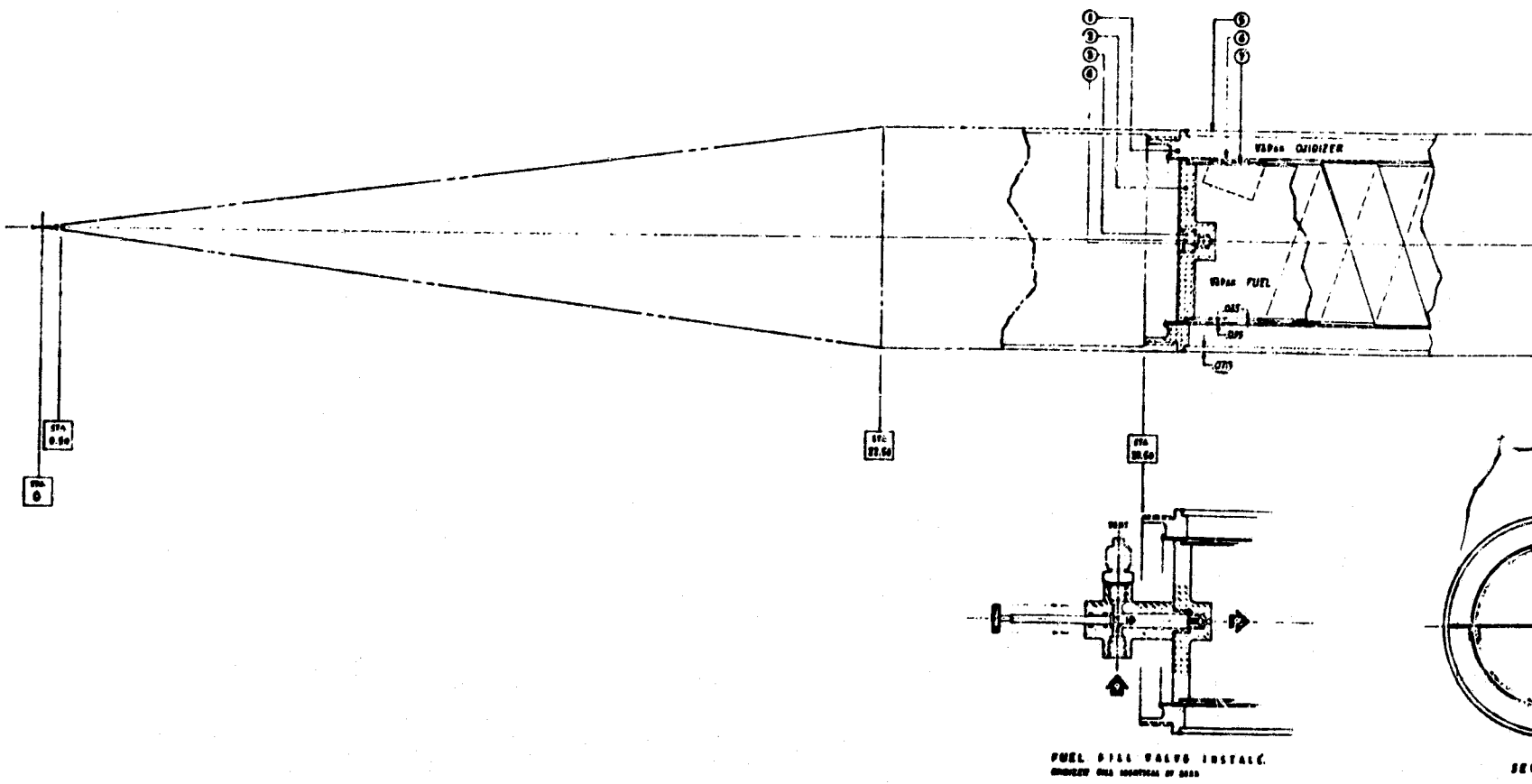


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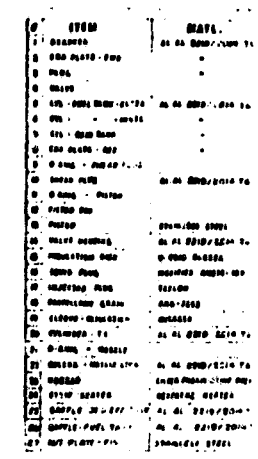
Figure 3.2.3.1

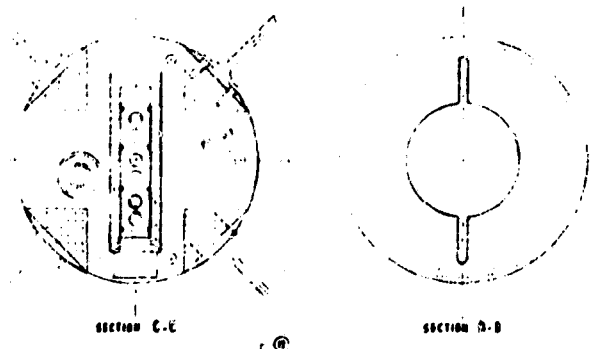


TOUCH SIDE



FOLDOUT FRAME 1





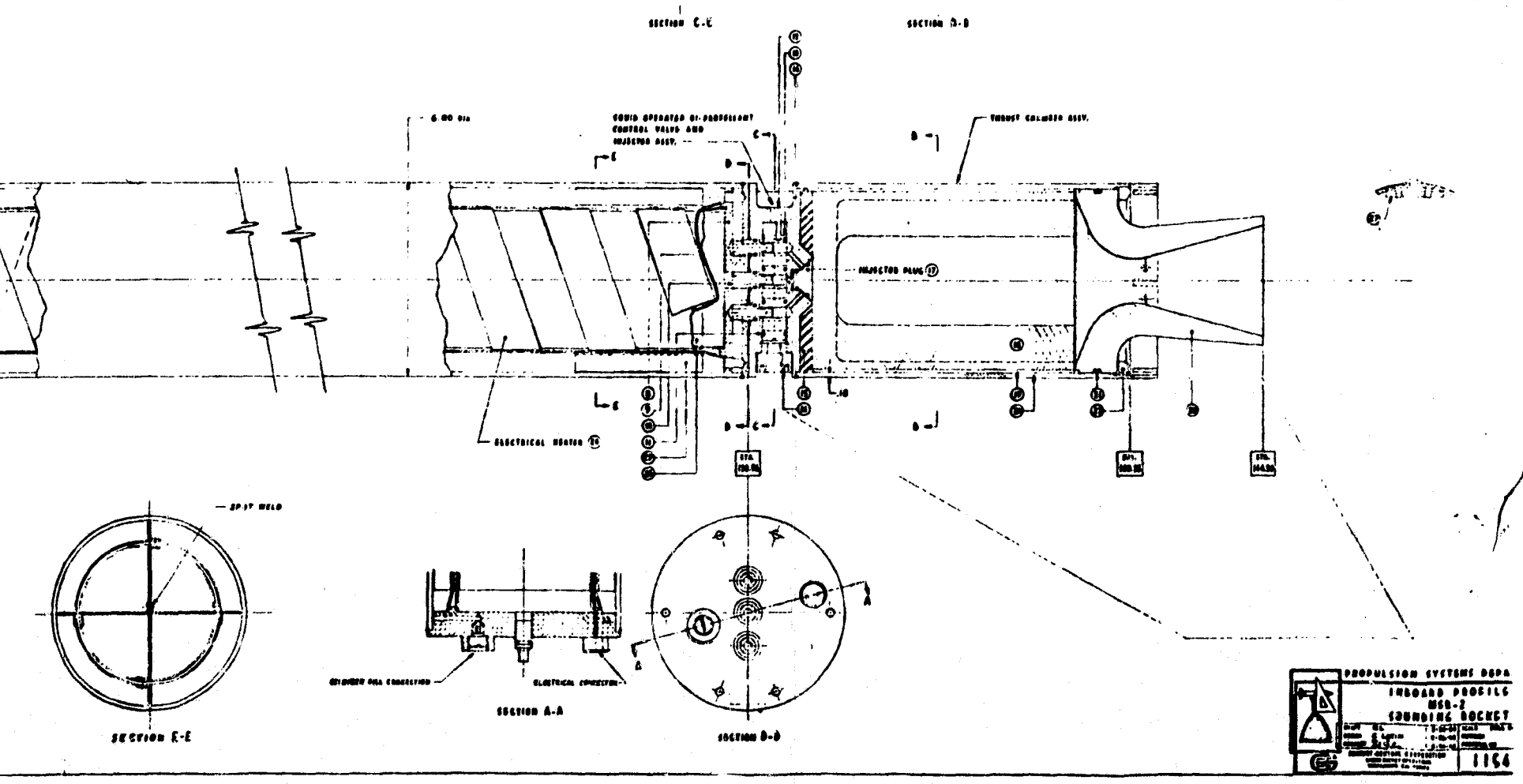


Figure 3.2.3.2

FOLDOUT FRAME 2

Table 3.2.3.1

ESTIMATED PERFORMANCE BREAKDOWN MSR-2
CONTINUOUS BURN DUAL THRUST METEOROLOGICAL SOUNDING ROCKET

(Net Payload 10 Pounds, 85° Launch Angle, 70°F)

BOOST PHASE

Solid Propellant (ANB-3248), WT. LB	8.6
Burn Duration, SEC	1
Specific Impulse, (Sea Level) SEC	235
Total Impulse Solid Grain, LB-SEC	2,000
Max Thrust, LB	2,750
Max Chamber Pressure, PSI	2,000

SUSTAINER PHASE

VAPAK Oxidizer	
.85 N_2O_4 + .15 NO, WT. BL	57.5
VAPAK FUEL	
.70 N_2H_4 + .30 NH_3, WT. LB	35
Mixture Ratio	1.64:1
Specific Impulse, (Avg Sea Level to 100,000 FT) SEC	260
Avg Thrust (LB)	488.0
Burn Duration, SEC	49.6
Total Impulse, Liquid, LB-SEC	24,204
VAPAK Propellant Temperature, °F	175
Tank Top Pressure, PSI	300
Max Chamber Pressure, PSI	225

Table 3.2.3.1 (Cont'd)

PROPULSION SYSTEM (without Payload, Fins and Nosecone)

Light-off Weight, LB	139.6
Burn-Out Weight, LB	38.5
Propellant Mass Fraction	.73
Total Burn Duration, SEC	50.6
Total Impulse, LB-SEC	26,204
Throat Diameter, INCH	1.37
Throat Area, Square INCH	1.48
Nozzle Area Ratio	6:1
Boost Thrust to Weight Ratio	10:1

Table 3.2.3.2

SUMMARY OF ENGINE COMPONENTS WEIGHT (LBS)

1.	Outside Fuel Tank	14.52	
2.	Inside Fuel Tank	4.86	
3.	Outside Oxidizer Tank	4.86	
4.	Forward Fuel Closure	.75	
5.	Forward Oxidizer Closure	.69	
6.	Aft Closure & Valve Assembly	4.80	
7.	TCA Shell	3.43	
	Total Weight =	33.31	
1.	Nozzle Weight	2.5	
2.	Insulation Weight	2.06	
3.	Heater	1.0	
	Weight =	39.0	Dry Tanks
	Solid Propellant	8.6	
	Fuel Weight	33.6	
	Oxidizer Weight	58.8	
	Total Weight =	140.0	Tanks Full

A similar composite rocket engine concept has been demonstrated by 27 test firings conducted at the Aerojet-Azusa Facility. The results of the test firings (rocket motor designated SSR-1) are summarized in Table 3.2.3.3. Ground test firings have demonstrated transition from booster-to-sustainer operation with the required combustion efficiencies in both modes.

In the following paragraphs details of the MSR-2 design features and performance characteristics are presented. A brief summary of the trade-offs which led to the selection of the current designs is included.

3.2.3.2 MOTOR CONCEPT

COMPOSITE ROCKET CONCEPT

In order to alleviate wind-sensitivity of meteorological sounding rockets, a unique concept which was previously conceived and tested was incorporated into the MSR-2 design. This concept provides the flexibility and performance offered by a two-stage design (high boost thrust and low sustain thrust) in a single stage. This is accomplished through the incorporation of a high-burning rate solid propellant grain cartridge-loaded into the liquid rocket chamber. The solid propellant is ignited first and provides the high initial thrust needed to minimize wind-sensitivity. At solid propellant burnout, the injector is exposed and the liquid hypergolic propellants are forced into the combustion chamber by their own vapor pressure. The flow of liquid propellant into the combustion chamber insures continuous combustion.

SELF-PRESSURIZING SUSTAINER PROPELLANTS (VaPak)

In order to minimize performance dependence on ambient conditions and to reduce system complexity and cost, self-pressurizing VaPak propellants were selected for the MSR-2. These propellants, 85% N_2O_4 + .15 NO and .70 N_2H_4 + .30 NH_3, are low-cost, well-known, and widely used chemicals with good performance. By automatically conditioning (heating) the propellants in the vehicle while on the launcher, using a simple internal strip-heater, the dependence of performance on ambient conditions is minimized. Regulators, valves and pressurizing units necessary in conventional liquid engines, are eliminated, resulting in higher vehicle reliability and lower cost.

Table 3.2.3.3

SSR-1 IR&D PROGRAM
TEST SUMMARY

Item	No. of Tests	No. of Successful Tests
Solid Booster Firings	8	3 (Successive satisfactory firings)
Liquid Sustainer Firings	8	6
Composite Firings	8	5
Heating	8	4
Nozzle	7	3
Valve	14	13
Igniters	11	11
Tanks (Burst)	4	1 (900 psi)
Gas Generator	4	1 (prototype)
Timers	4	3

Table 3.2.3.3 Con't.

SSR-1 IR&D PROGRAM
HARDWARE SUMMARY

Item	No. Fabricated	No. Tested	No. Delivered	No. on Hand
Thrust Chambers	16	9	6	9
Propellant Valve	16	6	6	9
Nozzles	20	7	6	7
Tanks	23	5	12	6
Sear Fittings	44	28	12	4
Pistons	18	9	6	12
Fins	60	-	24	36
Tank Heads	60	18	24	18
Fin Channels	60	-	24	36
Payload Transition Sections	12	-	3	9
Nozzle Retention Nut	12	3	6	6
Mid Sections	15	4	6	5
Aft Sections	15	-	6	9
Aft Launch Lug	15	-	6	9
Front Launch Lug	2	-	2	-

Table 3.2.3.3 Cont'd

SSR-1 IR&D PROGRAM
HARDWARE SUMMARY

Item	No. Fabricated	No. Tested	No. Delivered	No. on Hand
Igniters	18	11	7	-
Squibs	27	14	13	-
Gas Generators	10	3	7	-
Timers (Single)	6	3	3	-
Timers (Double)	4	1	3	-
Transmitters	2	-	2	-
Parachutes	3	-	3	-
P_c Transducers	11	-	9	2
Accelerometers	4	-	4	-
Propellant Heater	1	-	1	-
Solid Grains	15	9	6	-
Chamber Insulation	21	9	6	6
Propellant Mandrel	1	-	-	1
Workhorse Chamber (SS)	1	-	-	1
Test Stand - Horizontal	1	-	-	1
Test Stand - Vertical	1	-	-	1
Tubing Tank (2024-T3)	60	8	12	40
Templates and Shop Aids	13	-	-	13

THRUST CHAMBER ASSEMBLY-INJECTOR VALVE

The thrust chamber assembly is an integral unit containing a solid propellant grain, an injector-valve combination, and a nozzle. The solid propellant grain is precast into a micarta insulator sleeve and inspected before cartridge-type-assembly into the combustion chamber.

The one-piece injector-valve combination is a machined-aluminum casting which contains a triplet injector and a squib-operated propellant feed-valve. The triplet injector is one of the simplest types of injector designs possible and can be efficiently utilized in this VaPak pressurization application. VaPak propellants do not have to be mechanically broken down into small droplets, as accomplished in conventional multi-holed liquid injectors. This principle limits the MSR-2 injector to relatively simple and rapid machining operations, and is well suited to automatic production techniques.

The feed-valve features straight-through flow for both fuel and oxidizer, no contact of propellants with seals during storage, and positive lock in the "on" position after actuation. The only moving part in the MSR-2 propulsion system is the "piston" in the valve. Initiation of the squib actuates the piston, in turn cutting the propellant shear-nipples and instigating liquid propellant flow.

PROPELLANT TANKS AND HEATERS

The MSR-2 tankage is constructed of concentric tubes and provides an ideal location for internal strip-heaters between the self-contained oxidizer and fuel tanks. This simplified tank construction eliminates the piping and plumbing required by tandem tankage.

Internal strip-heaters are bonded to the exterior of the fuel tank and provide 3,000 watts of energy in the form of heat to both propellants when energized with 115 VAC. The electrical input can be regulated by thermostats located with ground support equipment.

PRE-PACKAGED LIQUID PROPELLANTS

The MSR-2 would be shipped loaded from the factory, with both the booster solid grain and the sustainer liquid propellants, in order to minimize

launch operations, reduce costs, and increase reliability. The propellant tanks ard shear-nipples are made of an all-aluminum welded construction. Only aluminum is exposed to the propellants during storage. By prepackaging, the highest standards for propellant purity and uniformity, as well as low-cost, can be maintained. The propellants tanks are designed to meet the rigid ICC regulations for interstate shipping.

The unique propellant fill-valve (Figure 3.2.3.2) provided for MSR-2 eliminates the need of costly and troublesome check valves normally required during the rocket fueling operation. The fill-valve is attached to the vehicle only during the propellant loading operation. Both propellants used in the MSR-2 are loaded under 50 psia pressure. Provision for venting the tanks is also incorporated into the fill-valve to eliminate any air entrapped during the filling operation and for emptying the tank should circumstances necessitate such an operation. Upon completion of fueling, an aluminum-plug is remotely seated, sealing the tank. After sealing the aluminum-plug, the fill-valve is removed and a secondary aluminum-plug is inserted and seal-welded.

LAUNCHING

As far as the motor is concerned, launch preparation is limited to supplying electrical power (115 VAC) to the heater, installation of explosive ordnance items (igniter and valve-squib), and electrical continuity check-outs. The launch operation is limited to turning on the thermostatically-controlled propellant-conditioning system, and, after the prescribed time, the ignition switch. The propellant valve piston, actuated by the squib, cuts the shear-nipples and allows the liquid propellants to flow to the injector-plug. The liquid propellant remains trapped behind the plug (separated by O-rings) for the 1-sec burn-duration of the solid booster. Upon booster burn-out, the pressure in the chamber drops below that of the liquid and the plug is expelled. Mixing of the hypergolic liquid propellant is sufficient to instigate combustion and sustainer operation.

As an operational safety feature, the valve squib would not operate from the ignition switch, but from a micro-switch installed in the launcher system. The micro-switch would be operated by vehicle motion. This is to

eliminate the situation where the squib might fire but not the booster igniter. In the later case the operational crew would then be faced with an activated sustainer and a potentially dangerous booster.

Incorporation of recent advancements in the state-of-the-art of solid and liquid propellants, together with the following system innovations - pre-packaged liquid VaPak propellant, a concentric fuel tank, an integral squib-valve and injector, an internal strip-heater, and a liquid thrust chamber that contains a solid propellant booster grain, results in a lightweight, high-performance, low-cost and high-reliability system that lends itself to mass production techniques.

3.2.3.3 MOTOR DESIGN

SUBSYSTEM SELECTION CRITERIA

The type and configuration of the MSR-2 propulsion subsystem were selected on the basis of best fulfilling the following set of requirements:

- Low wind sensitivity
- Performance relatively insensitive to ambient temperature
- Simple launch operations
- Low unit cost
- High margins of reliability and safety

Minimizing impact dispersion distance is, and always has been, a major objective of any sounding or meteorological rocket system. Because of cost active control systems are precluded. An alternative is a high acceleration, high velocity trajectory; e.g., artillery shell. However, high acceleration throughout the flight will result in high drag losses, excessive aerodynamic heating and severe structural problems. Thus, a compromise is indicated. Since the most critical effect of winds is in the initial phase of the flight, the high acceleration is most critical during this phase. A high-thrust/low-thrust sustain type of operation is therefore indicated. This is achieved in the Aerobee series by the use of separate stages. Use of a single stage offers simplicity and low cost, thus the approach used in the [illegible]m of a Composite Rocket which has been developed by Aerojet.

The Composite Rocket is a propulsion concept that provides boost/sustain performance utilizing single-stage hardware. Thrust ratios of 4:1 and higher are possible. Penalties in weight and complexity have been minimized by advanced design concepts.

Meteorological rockets can be expected to be launched under extreme ambient temperature conditions. The variation in performance with ambient temperature conditions should, however, be minimized. In the MSR-2, this is accomplished through the incorporation of a propellant conditioning system in the vehicle. For extreme conditions, supplemental capability may be provided by the launcher. The propellant conditioning system also enables the use of self-pressurizing propellants, eliminating the need of a complex pressurization system. A simple, internal strip heater incorporated in the vehicle conditions the propellants to a nominal 175°F prior to flight. Preliminary thermodynamic analysis indicates that for extreme minimum environmental conditions (-40 to -60°F) especially where 30 knot winds are encountered and an external shield of some type must be provided to achieve the nominal propellant design temperature. For these conditions a launcher of slotted-tubular construction with small external strip heater would provide the MSR-2 propellants a constant 175°F regardless of ambient conditions. Such a launcher concept is illustrated in Figure 3.4.3.6. The slotted tubular construction allows the fins to protrude from the close-fitting launch tube thereby eliminating the need for the vehicle support (Sabot) structure in conventional tube launchers.

Launch operations can be made to be very simple. The vehicle preparation for launch is limited to uncrating, visual inspection, installation on the launcher (manually by 2 or 3 people). Once the vehicle is on the launcher, the igniter, propellant valve squib, and the payload are installed and the heater connected. Continuity tests of the igniter to the firing circuit completes the launch preparation.

The unit cost is of obvious importance in an application where a large number of units are used. The high cost per pound of processed solid propellant of the type required for the meteorological rocket application,

as compared to the cost/lb of the liquid propellants of the same performance, and adaptability to mass production techniques, indicates possible cost advantages in liquid rockets for this application. However, such problems as pressurization, regulation, plumbing and tankage have to be considered.

The need for high margins of reliability and safety cannot be overemphasized. These can be achieved by system simplicity, conservative design, proven production techniques, use of subsystems and components of proven design, and design of safety features into the system.

The VaPak system is simple in concept in that only one moving part (valve piston) is required, valve and injector combined are in one unit, thrust chamber construction is standard with a pre-inspected cartridge-loaded grain.

Very high factors of safety were used in the design of all rocket components. Propellant tank designs meet ICC pressure vessel requirements. The thrust chamber design is based on the high boost pressures and incorporates a considerable safety factor.

All metal parts of the MSR-2 are of 2219, or 2014 aluminum alloy. The tanks are made of commercially available tubing to which tank heads are welded. All fittings are welded to the tank heads. The tanks are proofed to approximately 2 times the operating pressure.

Propellants used are well established and characterized. A fast-burning polybutadiene propellant, ANB-3248, is used on the booster. N_2O_4 + NO/N_2H_4 + NH_3 is used in the sustainer (well known characteristics and performance).

Company and Air Force funded programs (see References 3.2.3.1 - 3.2.3.12 Section 5.0) have established the groundwork for consideration of the composite rocket approach in a meteorological system. The basic design of the MSR-2 thrust chamber has been tested by Aerojet.

Many safety features have been designed into the MSR-2. The liquid propellant system can be made operative only after the solid booster has provided enough thrust to move the rocket along the launch frame. This is accomplished through the firing of the propellant valve squib by a micro-switch

activated by motion along the frame. No propellant is in contact with any seal until after valve activation. Thus, storage safety is assured. Additionally there are double welds and walls between propellants. The booster grain is precast and inspected thoroughly prior to installation. The rail launcher could be provided with a vehicle cooldown system (compressed CO_2) that will enable rapid depressurization of the fuel tanks in case of launch abort.

SYSTEM TRADE-OFFS

The requirement for a very high, short-duration, initial acceleration, coupled with the relatively low-sustain thrust pose severe limitations on the booster propellants. Either a high burning rate propellant or a very large burning area are required. A polybutadiene propellant, ANB 3248, was found to meet the high burning rate required (1.8 in/sec at 1000 psia) resulting in a minimum length chamber. To provide for the dual-thrust capability, the boost grain is cast into a micarta sleeve and cartridge loaded in the aft end of the combustion chamber.

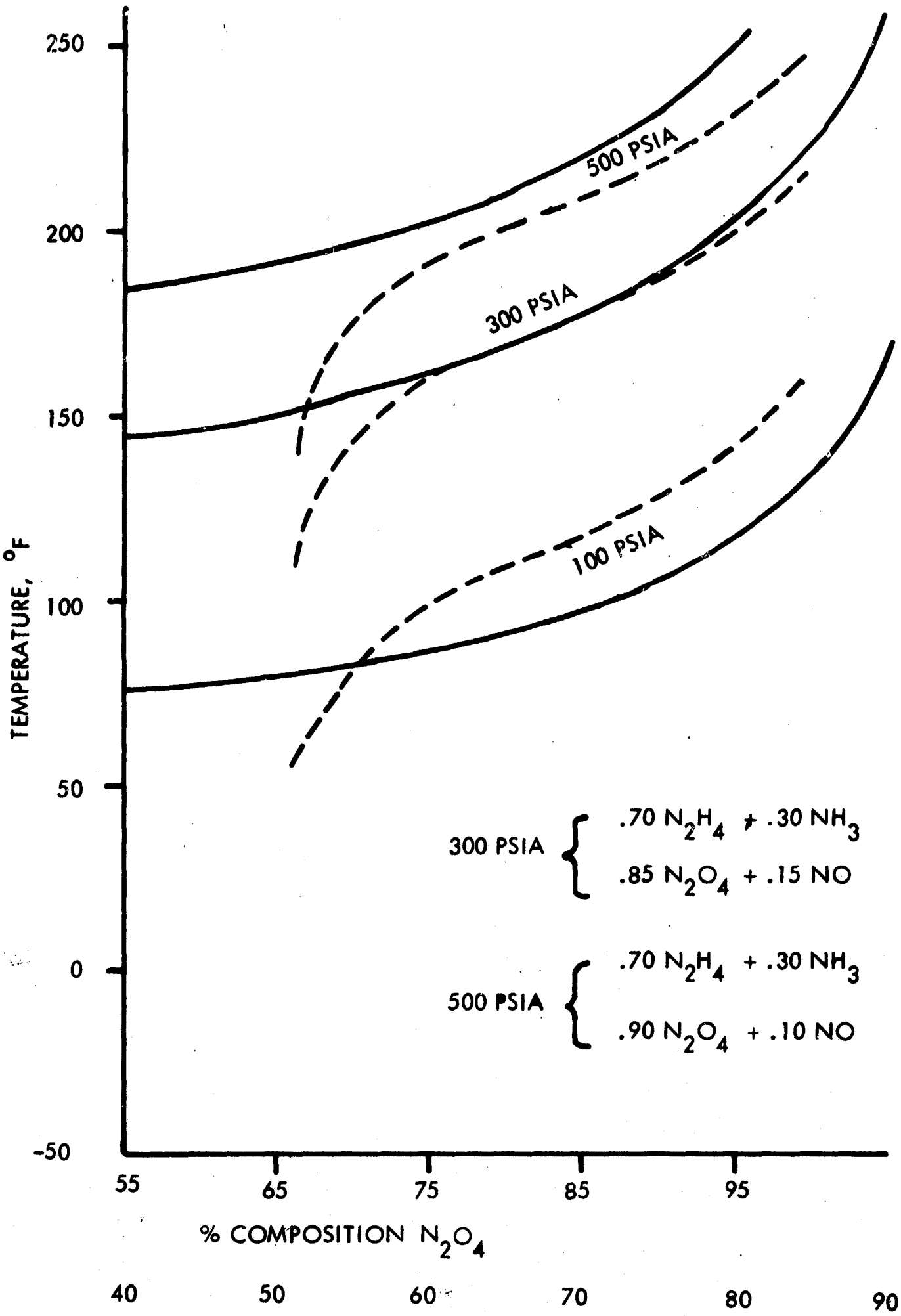
The selection of the sustainer propellants was limited to self-pressurizing propellants by the desire to simplify the system (eliminate pressurization system, regulators, etc.). The major considerations in the selection among the self-pressurizing combinations were previous experience with these propellants, cost per pound, and performance. On the basis of these considerations, the selection of a 85% N_2O_4 + .15 NO/ .70 N_2H_4 .30 NH_3 was made. This composition (Figure 3.2.3.3) of the propellants was chosen to provide 300 psia tank top vapor pressure in both tanks when an equilibrium temperature of 175°F was achieved. Experience with these propellants at Aerojet, Sacramento, in both company and Air Force funded programs has produced the data required for effective design. The cost per pound of these propellants (average approximately 40 cents) is among the lowest when compared with others of comparable performance.

Trade-off studies were performed in order to determine the optimum sustainer design utilizing the unique vapor-pressure (VaPak) pressurization system. These studies were based on maximizing apogee altitude with a given

TEMPERATURE VS VAPOR PRESSURE

FOR MIXTURES OF

$N_2H_4 + NH_3$ & $N_2O_4 + NO$



total weight and payload weight. An example of the results of the analysis is shown in Figures 3.2.3.4 through 3.2.3.7, Variation of Apogee Altitude with Expansion Ratio. The results presented are based on computer simulation of the MSR-2 vehicle flight.

The thrust chamber configuration was based on a trade-off between booster and sustainer requirements, the major parameter being the nozzle throat area and expansion ratio. The variation of I_{sp} with altitude for various area ratios is shown in Figures 3.2.3.8 through 3.2.3.10. The length of the chamber is dictated by the booster requirements. The construction of the chamber was selected on the basis of lowest weight, ease of manufacture and unit cost. The length of the MSR-2 conical nozzle was determined from Figure 3.2.3.11 for an area ratio of 6:1.

Tank Configuration and Construction

The criticality of the tank diameter to the performance of the vehicle (drag) and the vehicle length on its dynamic and structural behavior resulted in a design of a 6-in. diameter vehicle. An evaluation of different materials for tank construction resulted in a demonstrable advantage in the use of 2219 or 2014 aluminum alloy tubing with welded tank heads. The main considerations in selection of concentric tank construction were redundant welds between propellants, ideal location of internal strip heater, elimination of otherwise required piping (tandem tanks) and reduced number of external mechanical joints with potentially adverse drag and structural characteristics.

SELECTED VEHICLE CONFIGURATION CONSTRUCTION AND OPERATION

The MSR-2 vehicle configuration features all-aluminum construction with the exception of the thrust chamber valve piston which is fabricated of stainless steel. The vehicle has two main sections; the concentric liquid propellant tanks and the thrust chamber assembly. Figure 3.2.3.2 is the assembly drawing for the MSR-2.

Concentric tanks for both propellants are made by welding machined aluminum end plates into sections of seamless aluminum tubing. A strip heater is bonded to the exterior of the inner tank during the welding assembly

ALTITUDE, VELOCITY AND ACCELERATION VS TIME

EP = 6.0

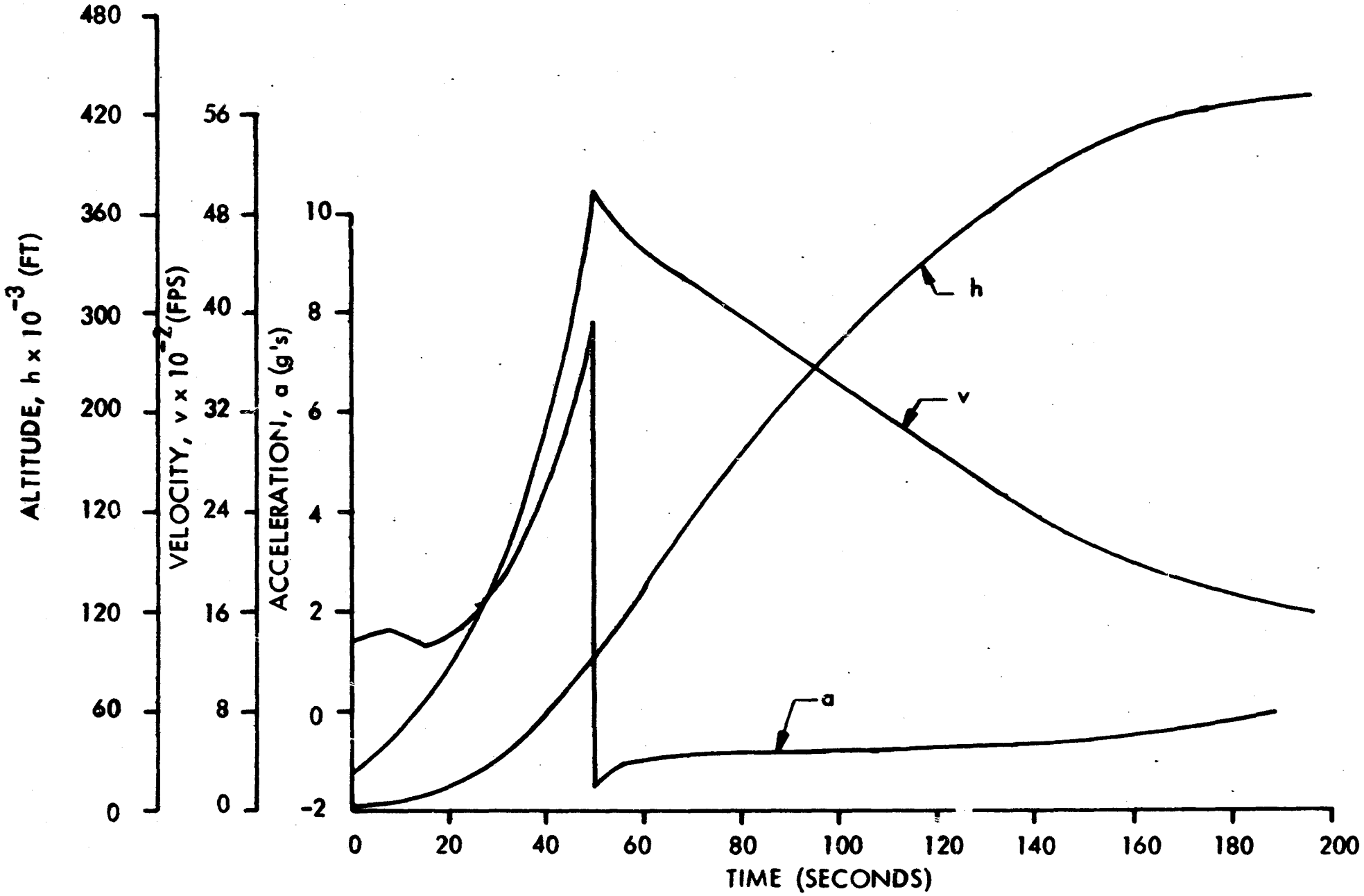


Figure 3.2.3.4

MACH NUMBER, THRUST, SPECIFIC IMPULSE VS TIME

ER = 6.0

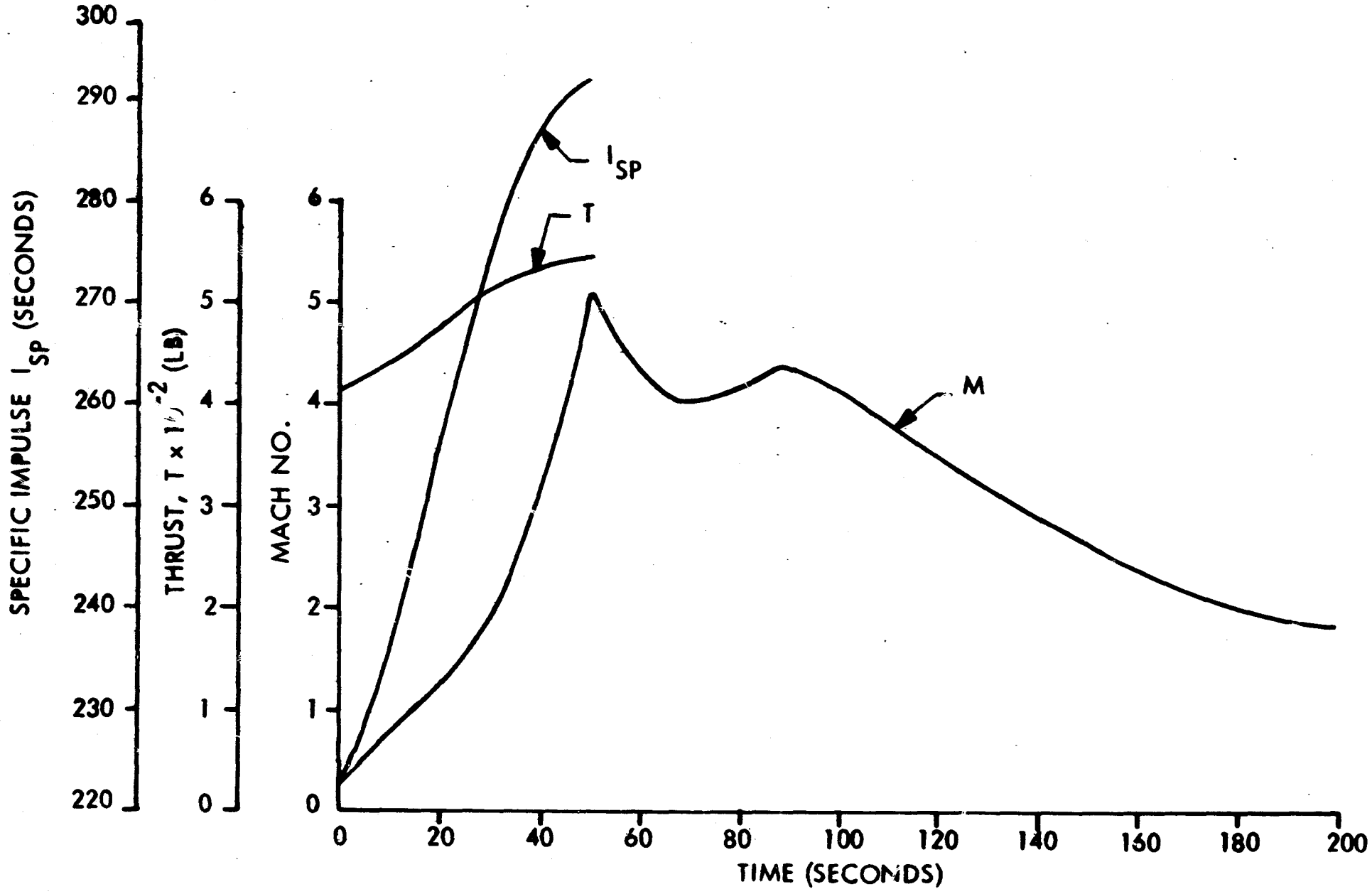
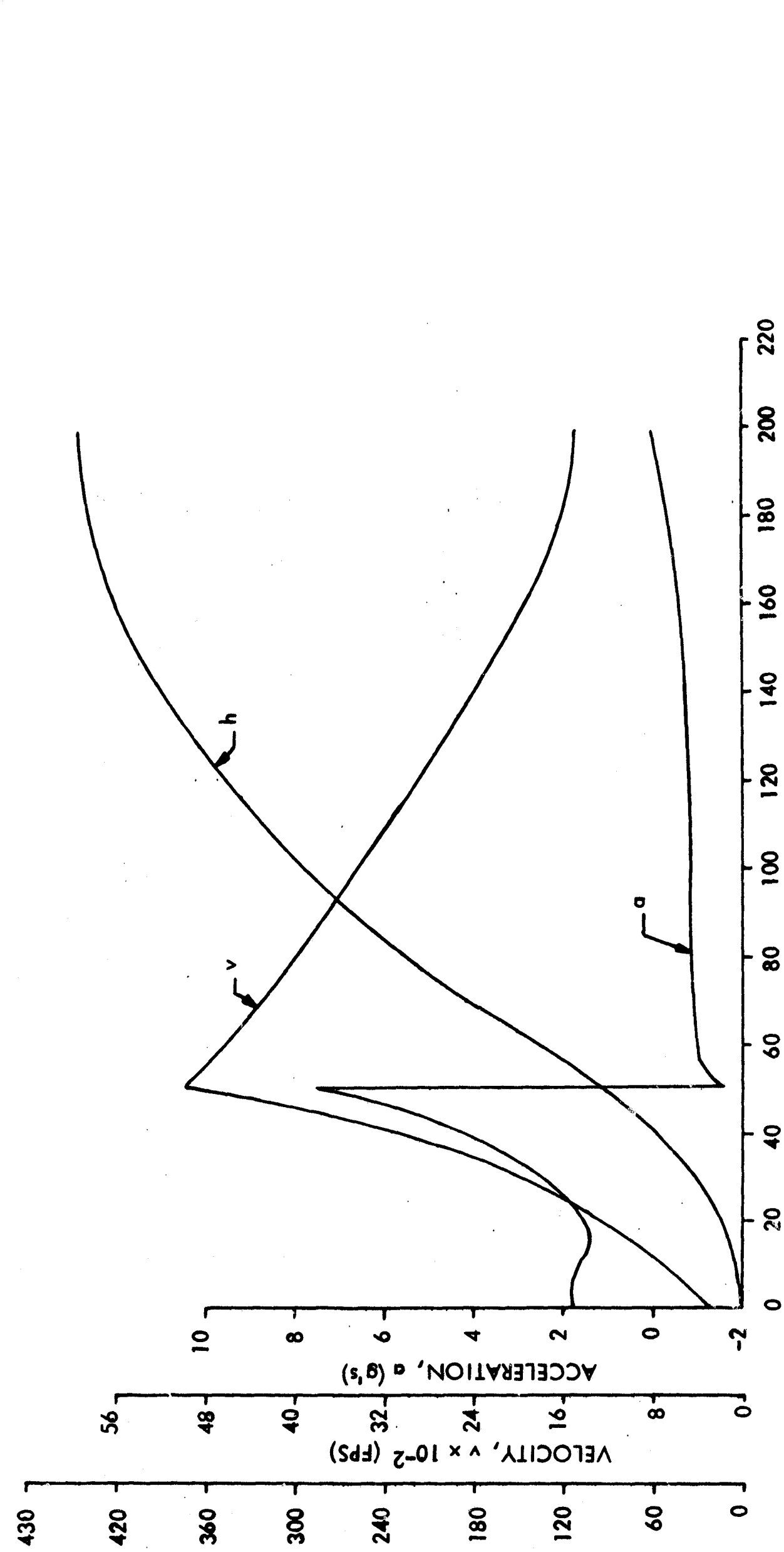


Figure 3.2.3.5

ALTITUDE, VELOCITY AND ACCELERATION VS TIME

EF = 4.0



ALTITUDE, VELOCITY, AND ACCELERATION VS TIME

EP = 8.0

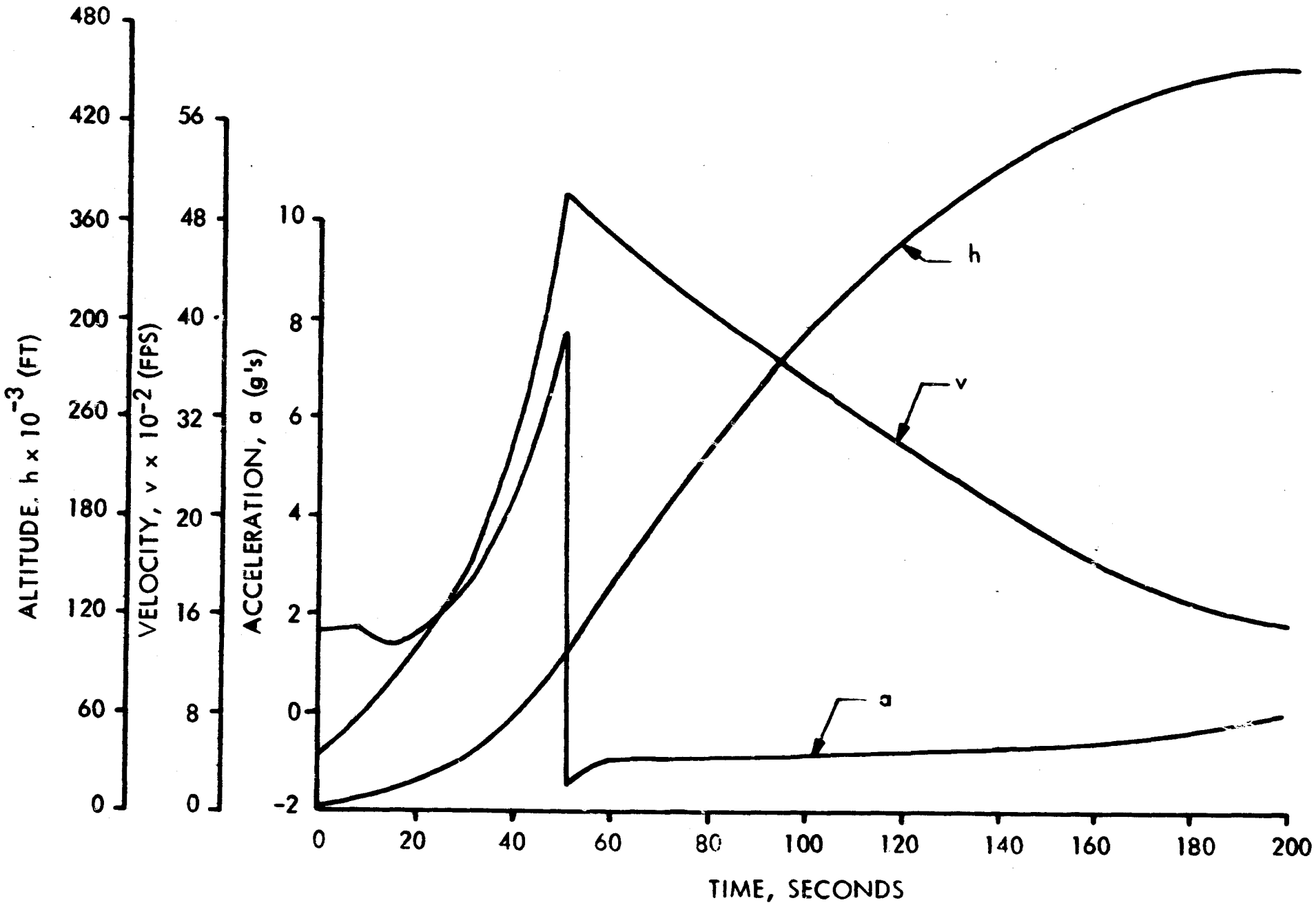
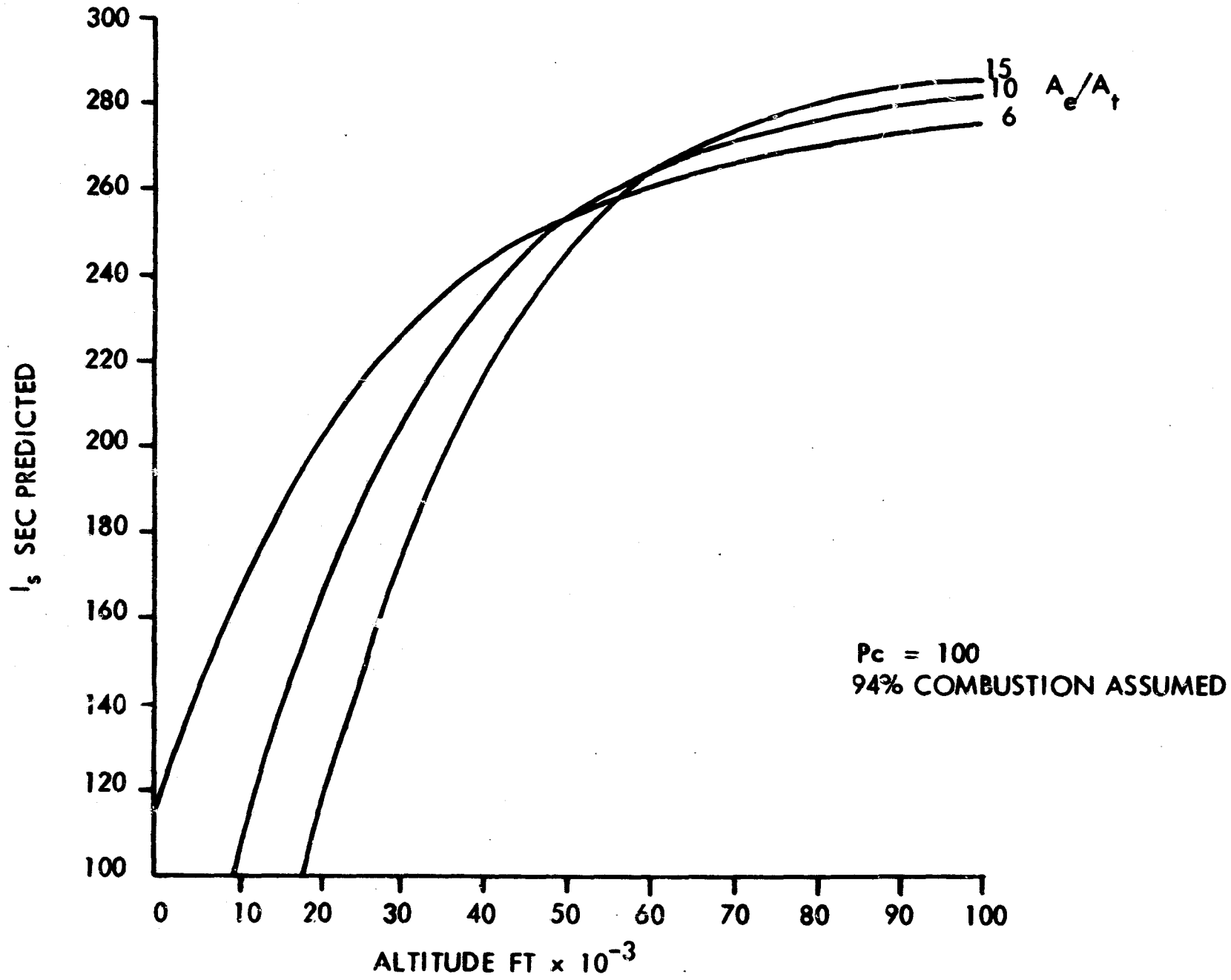


Figure 3.2.3.7

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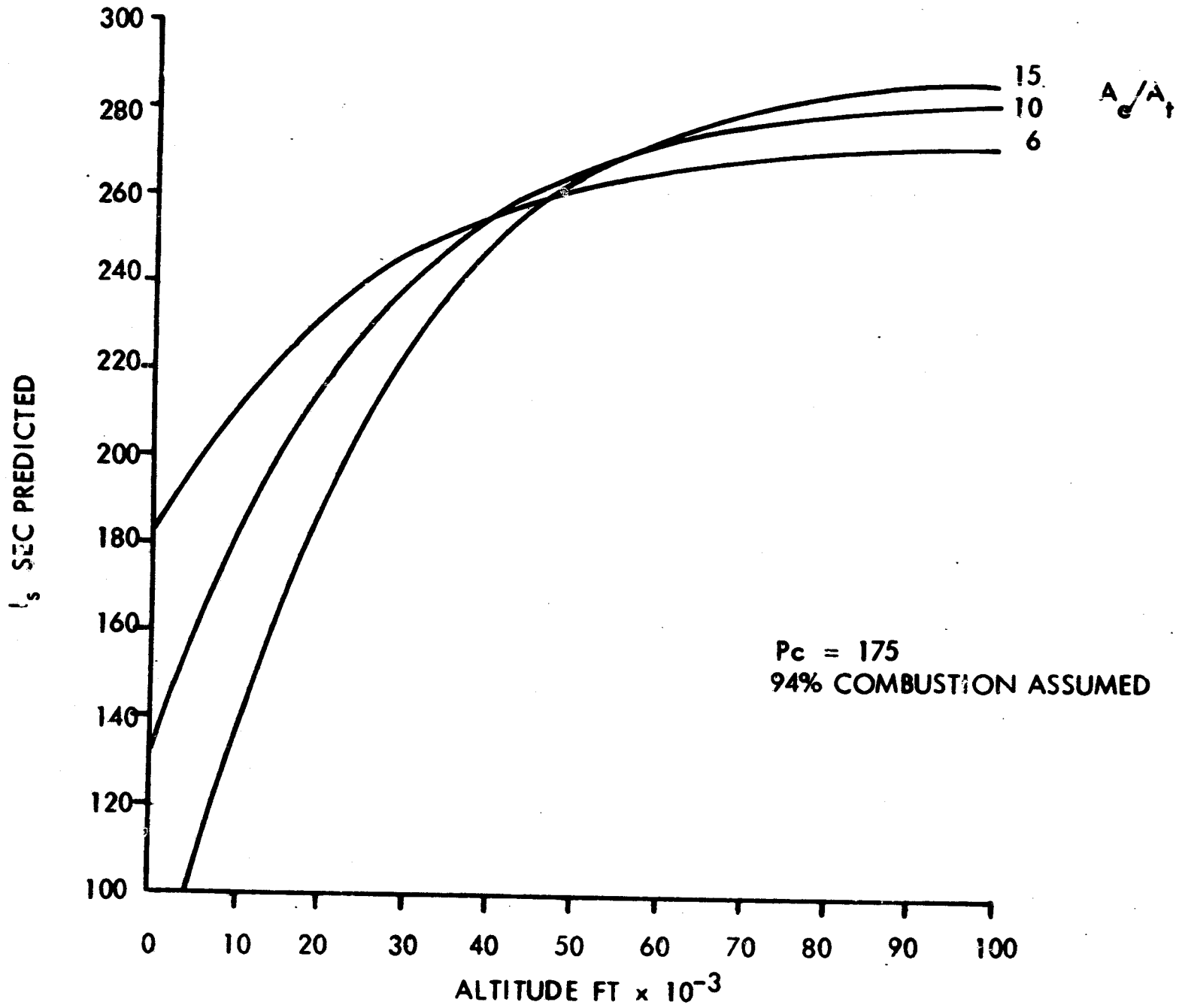
I_{sp} PREDICTED VS ALTITUDE



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Figure 3.2.3.8

I_{sp} PREDICTED VS ALTITUDE



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I_{sp} PREDICTED VS ALTITUDE

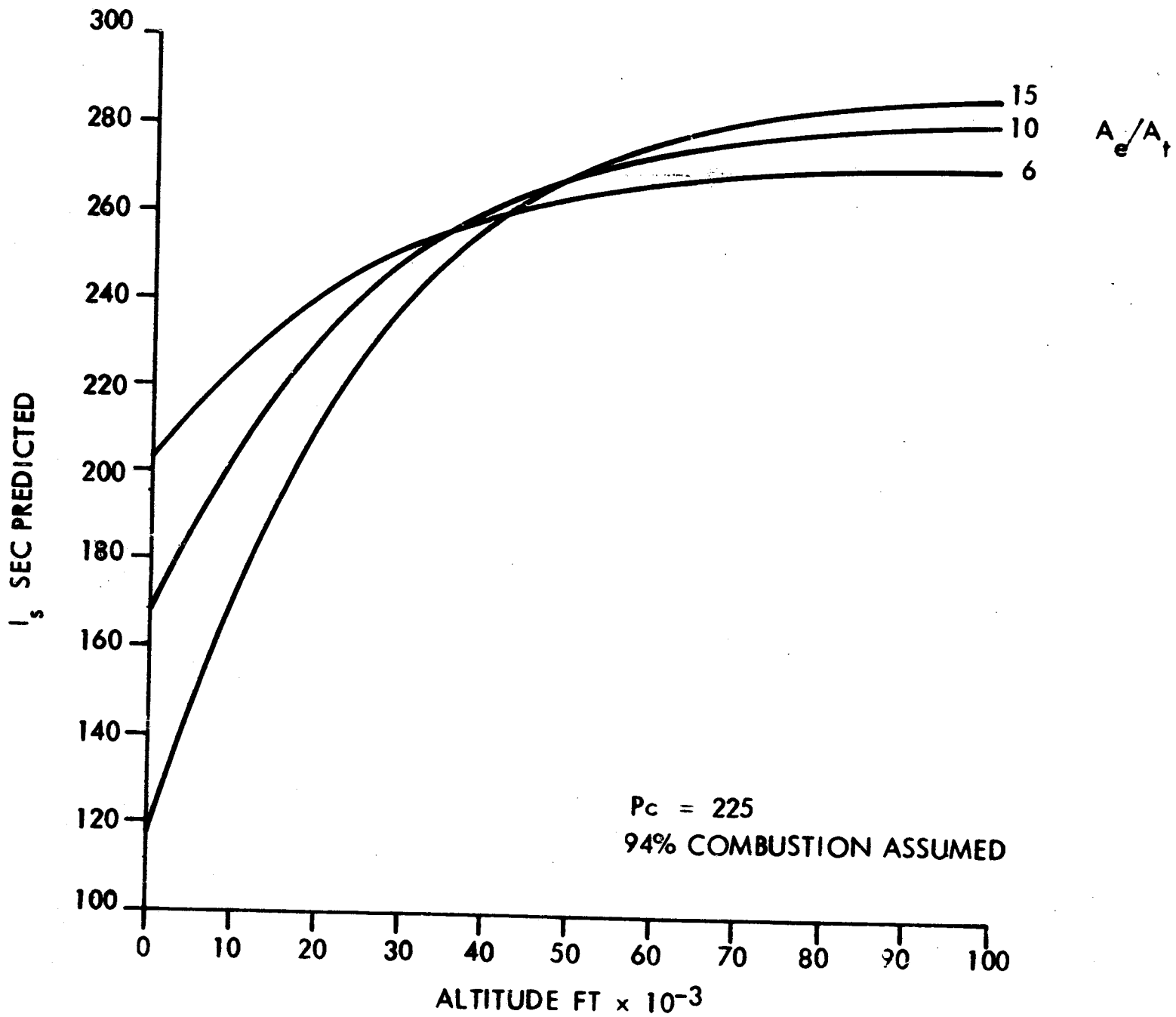


Figure 3.2.3.10

NOZZLE CONTOURS

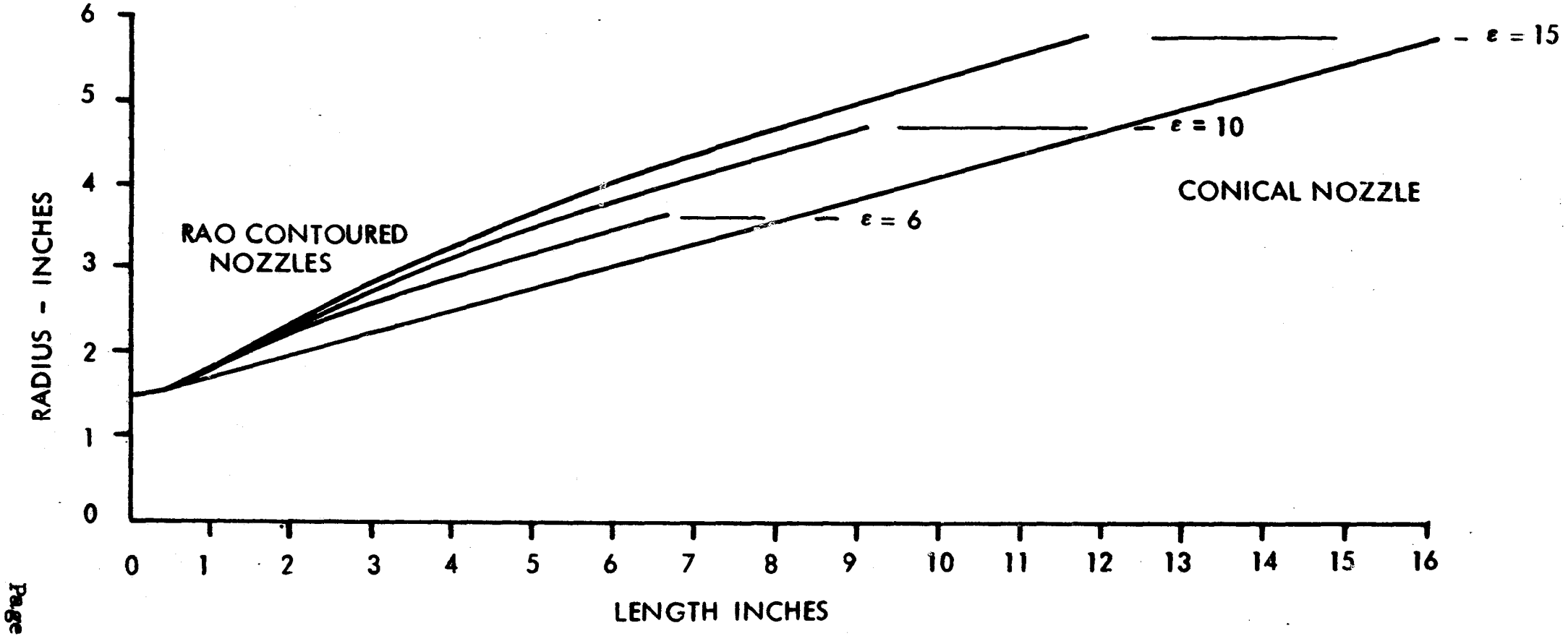


Figure 3.2.3.11

cycle of the tanks. The assemblies of 2219 will be aged to the T-87 and T-8511 conditions after welding. Prior to welding, the material will be in the T-37 condition (solution heat-treated and cold-worked with an 8% reduction in area). The aging cycle is 325°F for 24 hours and is compatible with the neoprene in the strip heaters that will accompany the parts through aging. Both GTAW and electron beam welding processes will be used. 2319 weld wire will be utilized in the former process. Aluminum alloy 2219 was selected for large production quantities over other high-strength aluminum alloys due to its superior weldability.

Chamber and Tankage - 2219 aluminum extrusions
T8511 condition

End Plates - 2219 aluminum plate
T-87 condition

The minimum mechanical properties of 2219 are as follows:

	T-87	T-8511	Welds
UTS (ksi)	63	58	45
.2% Offset yield strength (ksi)	50	42	35
T Elong.	5	6	6

Aluminum alloy 2219 is available in practically any condition and form desired; however, as a typical example, one vendor requires a minimum of 2000 to 4000 lb depending on the form requested with a minimum delivery time of six weeks. Further refinement of the design may indicate that it is possible to weld the material in the T-87 condition and eliminate post-weld aging with only a slight increase in weight. Should this technique prove feasible, additional vehicle cost savings would be realized.

As a backup, and for prototype vehicles, 2014 aluminum alloy is recommended because of its properties and availability. The corrosion rates of 2219 in N_2O_4 is comparable to that of 2014 aluminum, used for Titan tankage.

The material selected for the thrust chamber for limited production runs is aluminum alloy 2014-T6, having an ultimate tensile strength of 60,000 psi. The liquid propellant valve injector combination is welded to the heavy wall (.18) aluminum tubing forming the integral thrust chamber. After the propellant grain has been cartridge-loaded into the chamber and the nozzle inserted, an aluminum retaining ring is then screwed into place at the aft end of the chamber and locks the nozzle and grain in place.

The primary insulation material is the micarta (phenolic) sleeves in which the propellant is cast. The micarta insulation is .150-in. thick and is designed to insulate the chamber during both the solid and liquid combustion phase. The cartride-loaded grains have grooves cut into the aft face to allow chamber pressure between the aluminum chamber and the micarta sleeves. This prevents excessively high grain strains upon pressurization. The injector is protected during liquid engine firing by a V-4010 rubber insulator bonded to the injector head.

The nozzle consists of a glass phenolic housing and exit cone with possibly an ATJ graphite throat insert for erosion resistance. Inclusion of a graphite throat will depend on results of development tests. The throat diameter is 1.37 and the nozzle has an expansion ratio of 6:1.

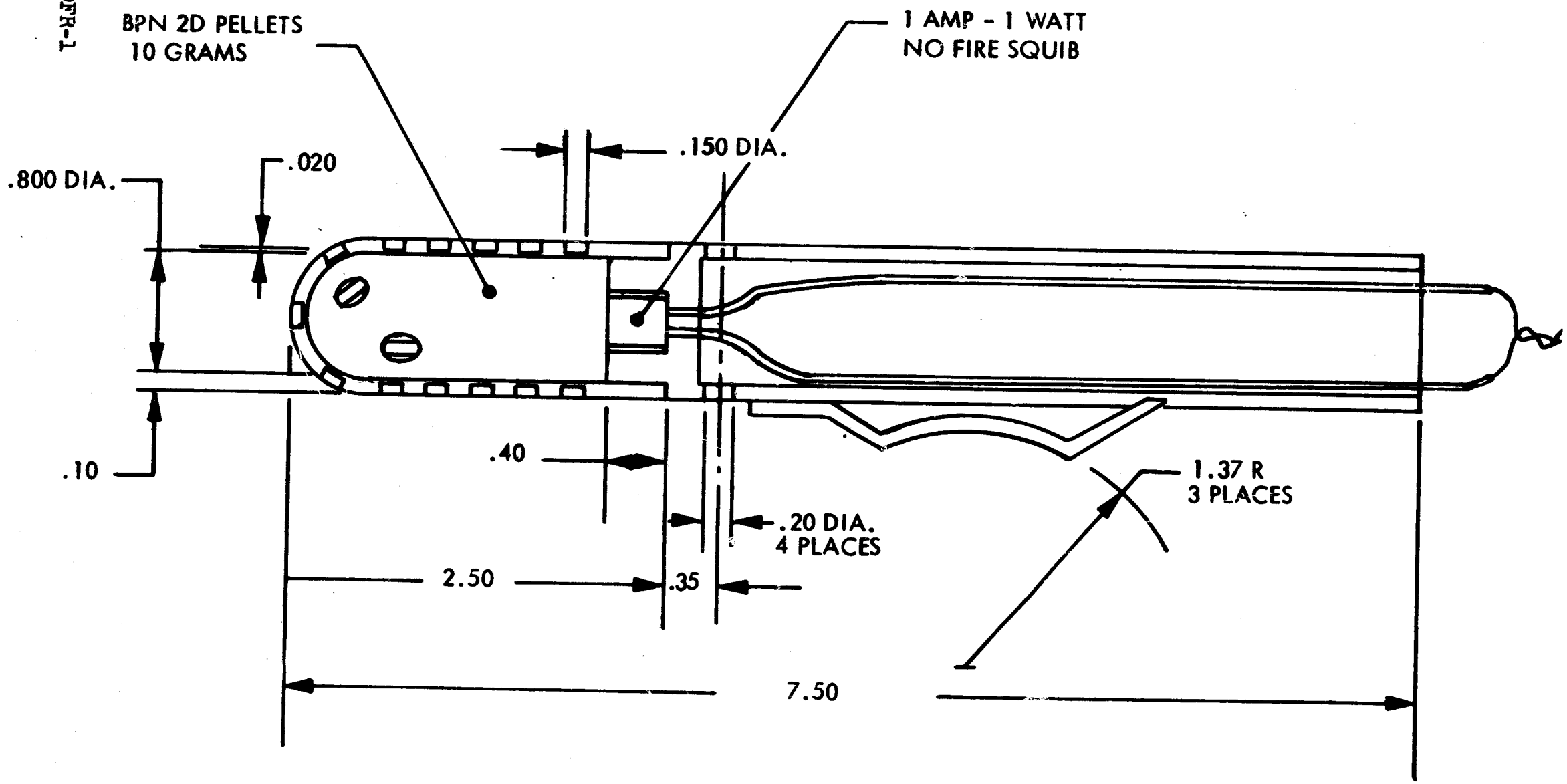
Fin attachment could be accomplished as shown in Figure 3.2.3.2. Provisions for attaching four fins can be located on the valve/injector casting and thrust chamber tubing. Thrust chamber and fin alignment could be designed in as a fixed valve and thus independent of field operations.

The accurate alignment of the propellant tanks and the thrust chamber and fin assembly would be controlled by using the propellant shear nipples as guide pins when bolting the two sections together. Machining tolerances of $\pm$ .010-in. for the mating sections of the vehicle guarantee that thrust and fin misalignment will be respectively $\pm$ 0.1 degree and $\pm$ 0.01 degree at their maximum values.

The type of igniter chosen is a nozzle-insertable, all-plastic, perforated-tube type, as shown in Figure 3.2.3.12. This type was selected

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SOLID GRAIN IGNITER



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Figure 3.2.3.12

after also considering a pad (rail) mounted, flame-jet type igniter. The latter type was rejected because it generally produces longer delays and less complete ignition for the same amount of igniter materials, it operates at higher pressures, and thus, would be considerably heavier, and it would require auxiliary mounting hardware.

With the exception of the lead-wires and the thin metallic case of the squib, the igniter assembly is constructed entirely of thermoplastic materials. These materials were selected in order to reduce the adverse effects of igniter debris as it passes through the nozzle after ignition. The igniter case is only sufficiently durable to retain the pyrotechnic pellets during the initiation phase. The case has a multiplicity of perforations through which the igniter flame issues.

The igniter consists of 10 grams of BPN pellets contained within a molded plastic housing. This housing hermetically seals the igniter prior to firing. Upon initiation of the 1 watt, 1 amp no-fire squib, initial pressurization causes the radial-oriented ports to burst, pressurizing the motor bore and providing sufficient heat flux to ignite the propellant grains. The remainder of the plastic housing will melt and disintegrate prior to being expelled from the motor. The igniter is held in place by a "four-fingered" plastic retainer which applies pressure against the grain bore providing a friction fit. The predicted ignition transient is shown in Figure 3.2.3.13. The ignition delay, time to reach 90% of initial steady-state pressure, is estimated to be 0.050 seconds.

The launch system can be either a standard rail or tube launcher, such as illustrated in Section 3.4.

In operation, once the vehicle has been placed on the launcher, an internal strip-heater is activated by 110 VAC electric supply that thermostatically conditions the VaPak propellants to 175^{o}F. At this time (approximately 30 min.), the vehicle is ready for launch. Launch is accomplished by ignition of the solid grain providing boost thrust, followed by squib valve actuation and efficient sustainer thrust. The launcher is equipped with a

micro-switch which closes after vehicle motion, activating the squib to open the liquid propellant valve. This is the means of providing assurance that the sustainer engine becomes operational only after successful booster initiation and remains inoperative in case of booster ignition failure. If the booster should fail to ignite, possibly because of a faulty igniter, the flight is not necessarily aborted because another igniter could be rapidly installed in the rocket nozzle with the firing sequence being repeated.

The motor would be shipped completely assembled (i.e., factor filled, sealed, inspected, etc.). Because the motor weighs only 140 lb, special handling equipment is not required; e.g., it would be 2-3 man portable.

The launch procedures have been simplified to the point where crew training is minimized. Launch preparations call for first, installation of the fin assembly, second, mounting the vehicle on the launcher rail; third, install the payload, forth, install propellant valve squib and booster igniter; fifth, connect the electrical strip heater; sixth, run electrical continuity checks. The vehicle would then be ready to operate.

3.2.3.4 MOTOR PERFORMANCE

DESIGN CONSIDERATIONS

Combustion Efficiency - A combustion efficiency analysis was conducted for the sustainer phase configuration using VaPak oxidizer (.85 N_2O_4 + .15 NO) and VaPak fuel (.70 N_2H_4 + .30 NH_3). This analysis considered propellant injection at 175°F and resulted in a calculated energy release efficiency, ERE, of 97.3%. However, no effort was expended to determine a mixture ratio distribution loss (MRD) so that the total combustion efficiency loss was estimated to be approximately 6%.

The engine efficiencies determined on the basis of static firings are: Sustainer - 94% of theoretical C^*, Booster - 88% of theoretical total impulse. The projected booster thrust curve is shown in Figure 3.2.3.14.

Nozzle Design - The contours of six suitable nozzle designs are shown in Figure 3.2.3.11. Three nozzles are conical and terminate at A_e/A_t of

SOLID BOOSTER

PREDICTED IGNITION TRANSIENT

T = 50°F

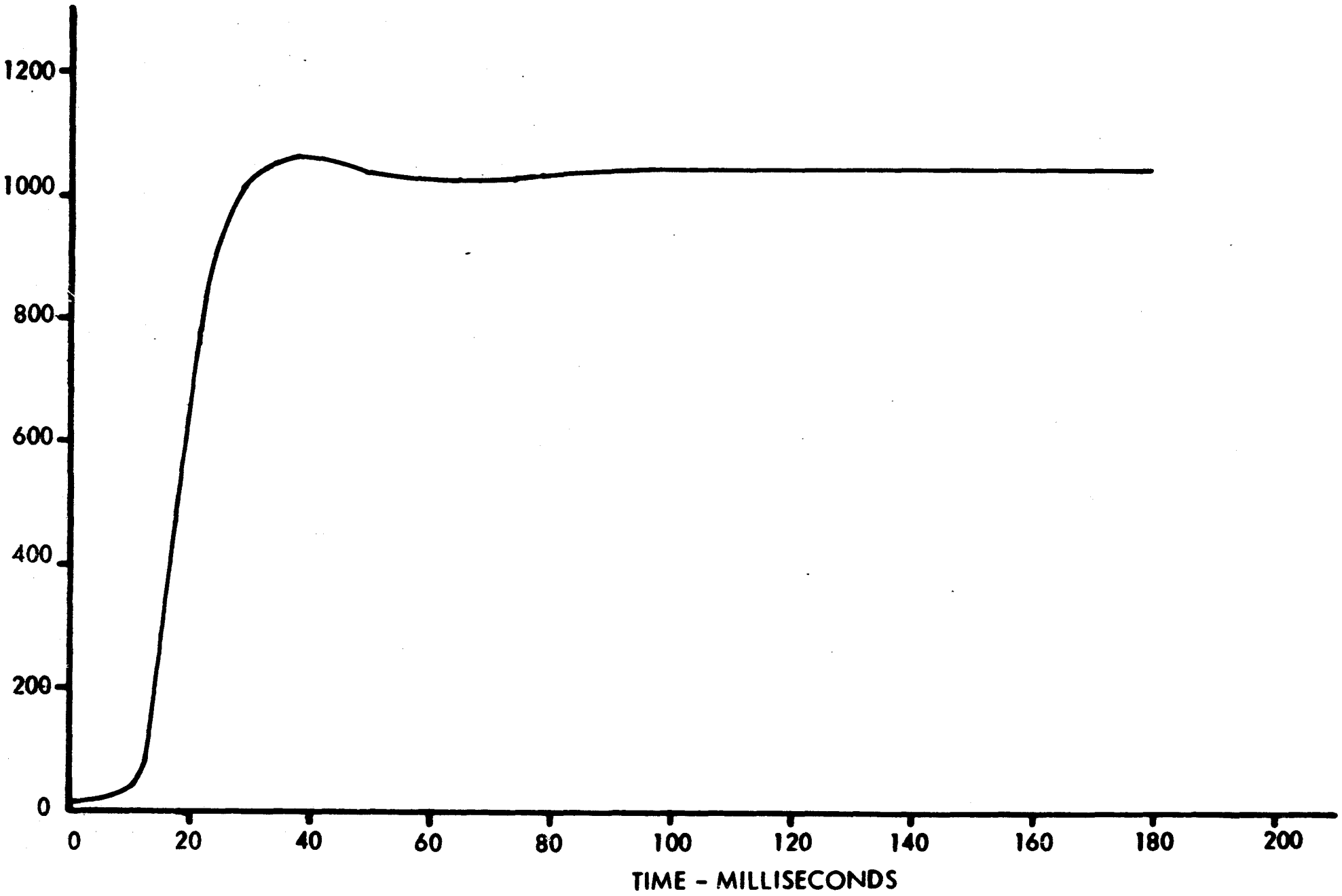


Figure 3.2.3.13

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SOLID BOOSTER

PREDICTED THRUST

$T = 50^{\circ}F$

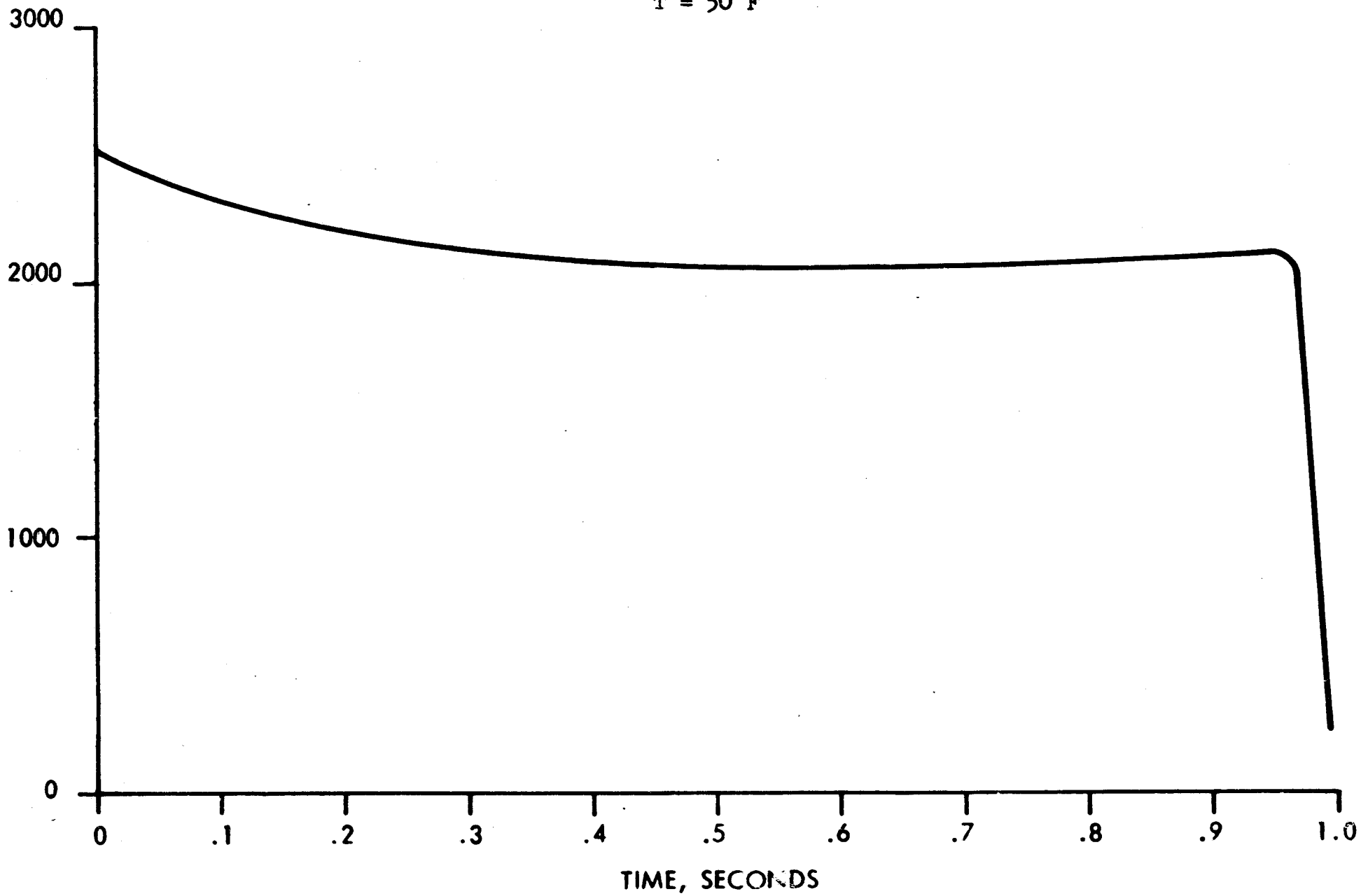


Figure 3.2.3.1.4
Page 139

of 6, 10, and 15-to-1, while the other three are Rao optimum designs terminating at the same A_e/A_t's. The Rao optimum nozzle design results in a nozzle which is shorter than a conical nozzle for the same performance and A_e/A_t. The Rao optimum nozzle design is described in detail in Engineering Systems Report No. E-3, Aerojet Computing Sciences Division.

Each nozzle was evaluated for divergence and shear-drag-loss and summed to derive the nozzle efficiency.

Figures 3.2.3.8 - 3.2.3.10 describe I_s as a function of altitude and A_e/A_t. The curves were derived from the equation:

$$I_{s_{\substack{\text{predicted}\\\text{altitude}}}} = \left[\eta_1 \, \eta_2 \left(\frac{F_{\text{vac theo}}}{\dot{W}_t} \right) \right] - \frac{P_a \, A_e}{\dot{W}_t}$$

Area Ratio - To derive the optimum area ratio for the MSR-2 design, the vehicle flight was simulated on a computer. The results of these runs are presented in Figures 3.2.3.4 - 3.2.3.7. Based on the computer results, a 6:1 area ratio was chosen.

ENVIRONMENTAL EFFECTS

A preliminary investigation was made of the transient propellant heating required to thermally condition the VaPak propellants to a nominal level of 175°F. The propellant combinations of 0.85 N_2O_4 + 0.15 NO and 0.70 N_2H_4 + 0.30 NH_3 have been selected such that at an equilibrium temperature of 175°F, the vapor pressure of both the oxidizer and the fuel will be 300 psia. The propellant thermal conditioning is accomplished by means of a strip-resistance electrical heater placed such that it becomes the interface between the oxidizer and fuel, in a concentric tank configuration (see Figure 3.2.3.3). This situation results in the inner tank being thermally isolated to heat-loss during the heating period. The outer tank does experience thermal energy interchange with its environment, the magnitude and direction, depending upon the prevailing ambient boundary conditions.

Fuel Located External to Oxidizer -
15 Minute Heating Period

The first propellant configuration investigated was for the fuel located external to the oxidizer, with the strip-heater compartment forming the interface between the two tanks. Figures 3.2.3.15 and 3.2.3.16 show the thermal response for a 5550 watt heater operating for a period of 15 minutes. Two ambient conditions were investigated: $+70^{o}F$ and $-40^{o}F$, both under no-wind conditions. The internally located oxidizer increased in temperature from $+70^{o}F$ to the desired $+175^{o}F$ in the 15 minute period. The fuel, located external to the oxidizer, rose from an initial value of $+70^{o}F$ to $+141^{o}F$ for the $+70^{o}F$ ambient conditions, whereas the $-40^{o}F$ ambient condition allowed the oxidizer to increase to $+138^{o}F$ in the same 15 minute period. This arrangement of propellants proved to be less effective from a thermal conditioning standpoint, due to the relative values of the propellant mass - C_p products.

Oxidizer Located External to Fuel -
30 Minute Heating Period

The locations of the propellants were reversed in an effort to make the thermal conditioning period more effective. Four conditions were investigated: $+70^{o}F$ and $-40^{o}F$ no-wind conditions, and $+70^{o}F$ and $-40^{o}F$ 30-knot-wind conditions. Figures 3.2.3.17 - 3.2.3.20 show the thermal responses for these external boundary conditions for a 3295 watt heater operating for 30 minutes. Indicated on the curves is the external energy required to compensate for the convective energy loss, which would occur if the oxidizer were exposed to the appropriate ambient conditions, at the bulk temperature indicated. The second oxidizer curve shows the bulk temperature response with the heat loss appropriate to the external boundary conditions.

The no-wind ambient-side heat transfer coefficient is 0.016 $btu/ft^2\ min^{o}F$ at $+70^{o}F$ and 0.019 $btu/ft^2\ min^{o}F$ at $-40^{o}F$. The oxidizer liquid-side natural convection heat transfer coefficient ranges from 0.41 to 0.49 $btu/ft^2\ min^{o}F$. Conditions of a 30-knot-wind yield ambient external heat transfer coefficients of 0.17 and 0.24 $btu/ft^2\ min^{o}F$ at $+70^{o}F$ and $-40^{o}F$, respectively.

VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 5550 Watts
EXTERNAL ENVIRONMENT +70°F/NO WIND
INITIAL SYSTEM TEMPERATURE: +70°F
FUEL EXTERNAL TO OXIDIZER

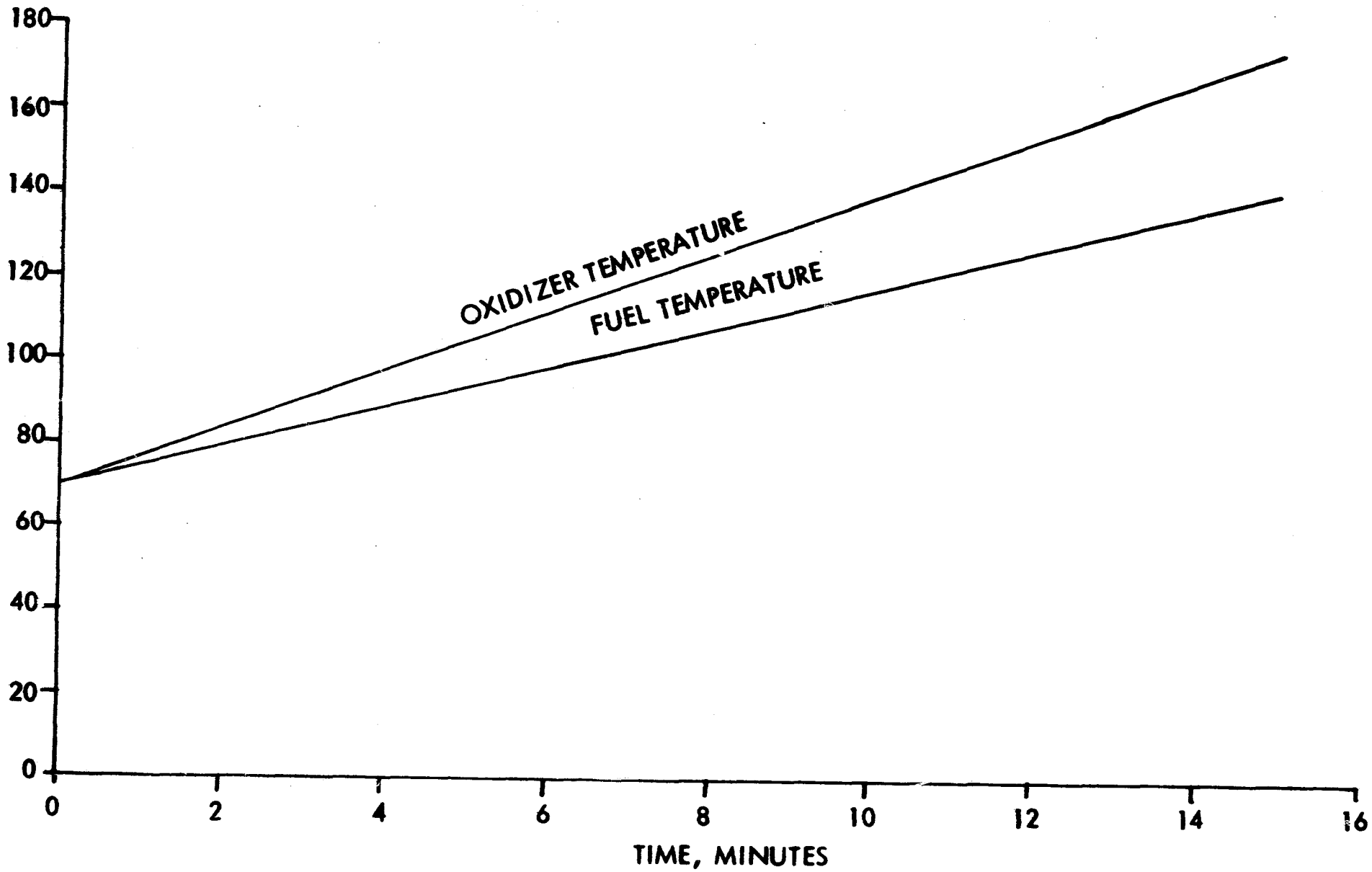


Figure 3.2.3.15

VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 5550 Watts
EXTERNAL ENVIRONMENT -40°F/NO WIND
INITIAL SYSTEM TEMPERATURE: +70°F
FUEL EXTERNAL TO OXIDIZER

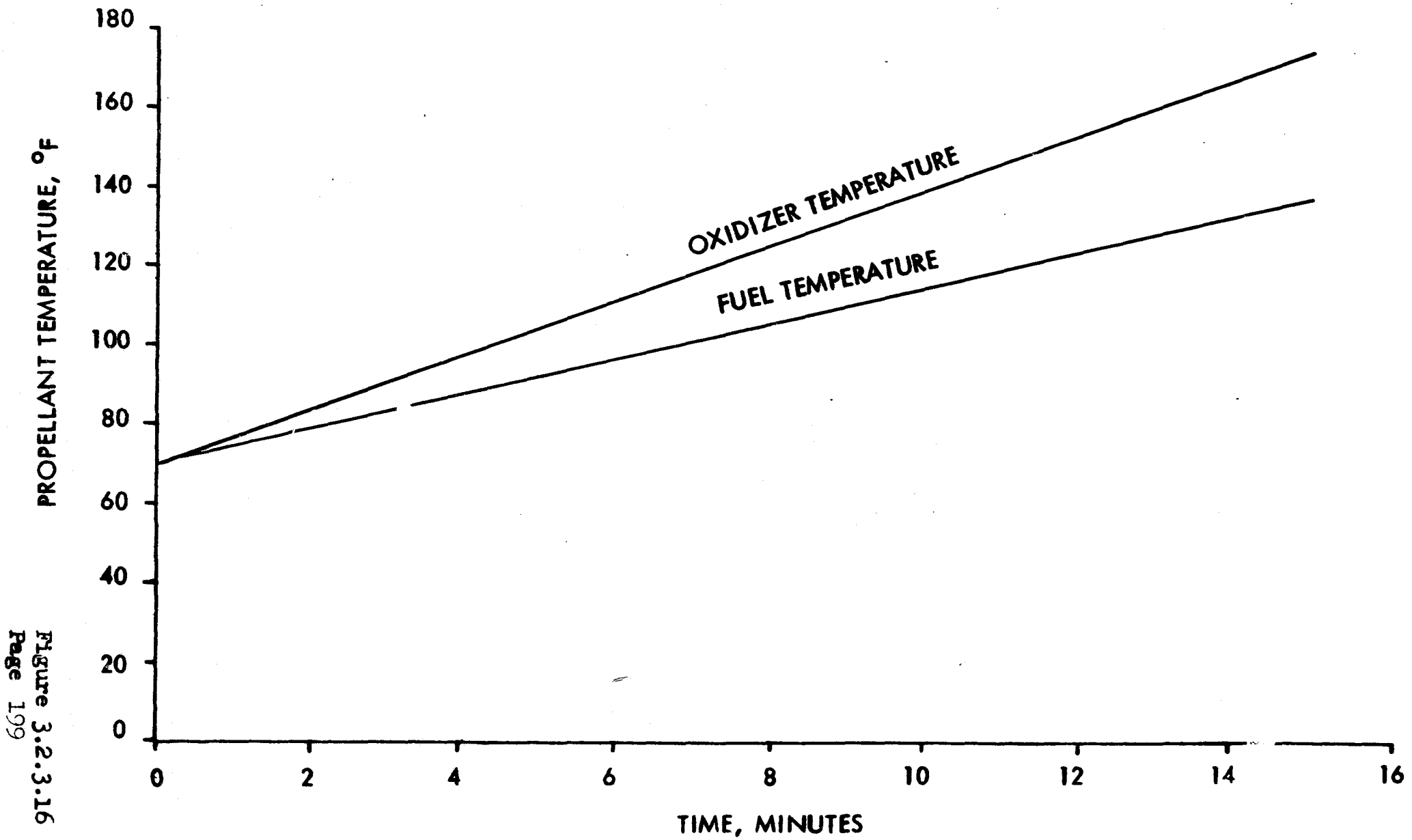


Figure 3.2.3.16

VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 3295 Watts
EXTERNAL ENVIRONMENT: +70°F/NO WIND
INITIAL SYSTEM TEMPERATURE: +70°F
OXIDIZER EXTERNAL TO FUEL

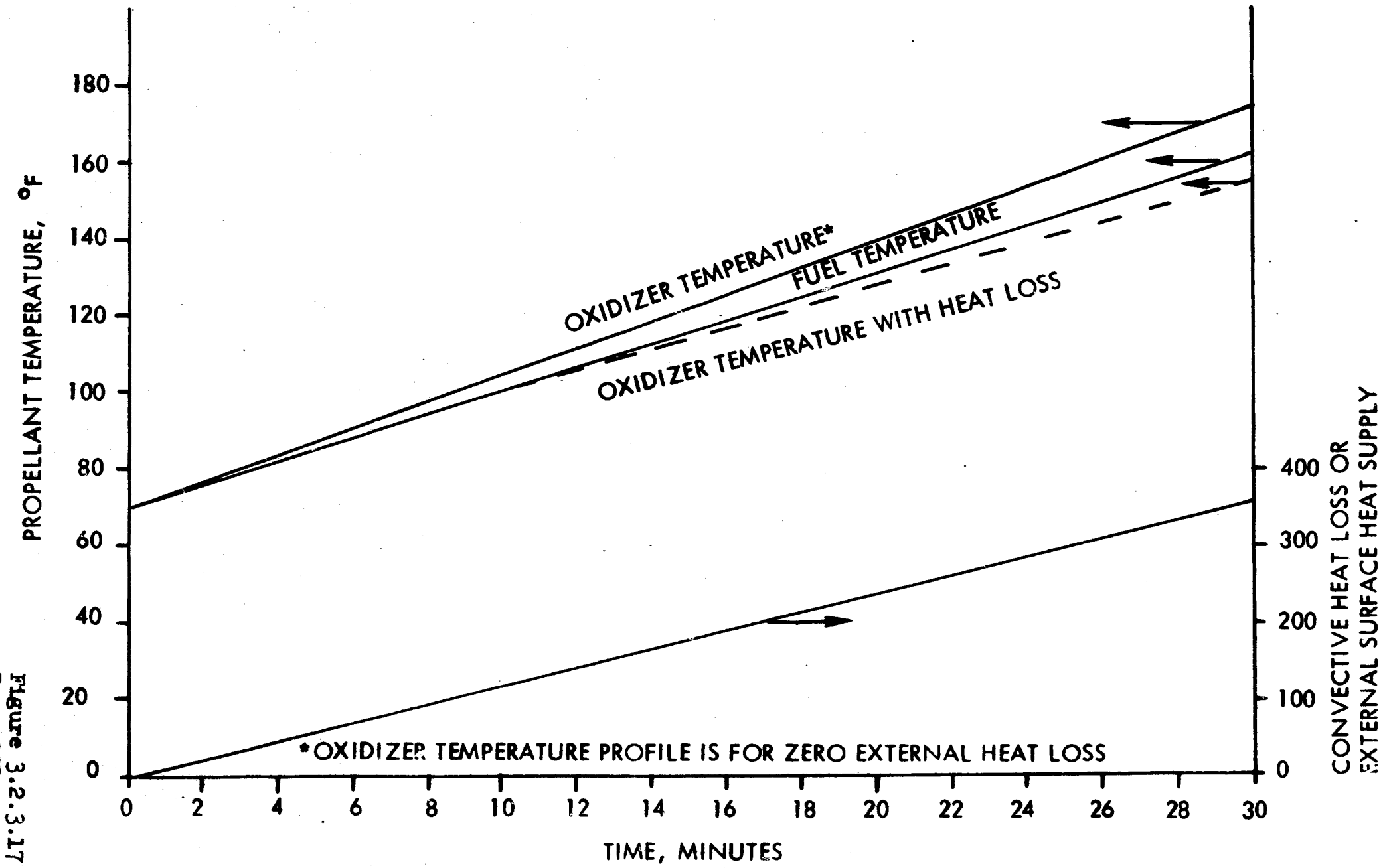


Figure 3.2.3.1.7
Page 200

VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 3295 Watts
EXTERNAL ENVIRONMENT -40°F/NO WIND
INITIAL SYSTEM TEMPERATURE: +70°F
OXIDIZER EXTERNAL TO FUEL

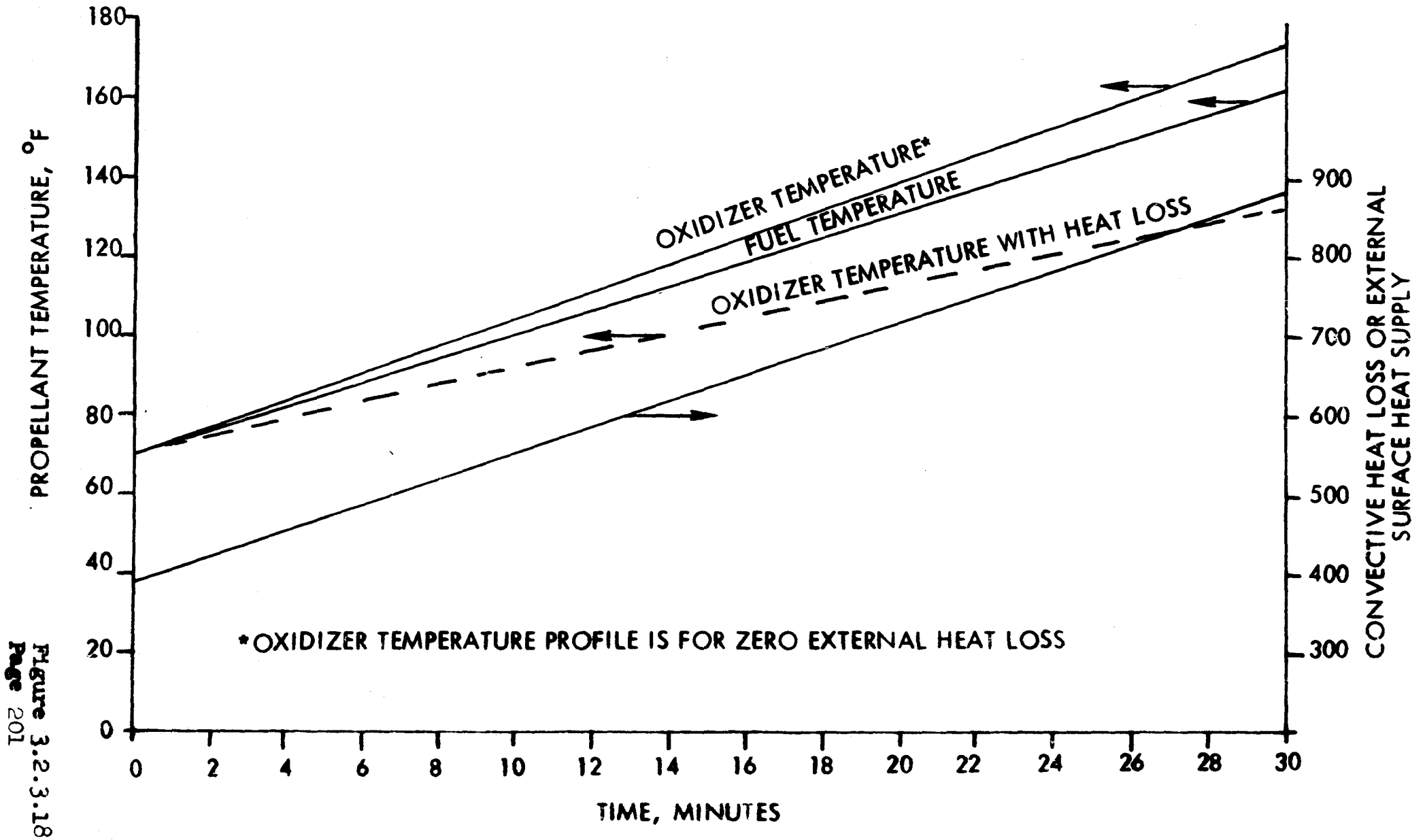


Figure 3.2.3.18
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VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 3295 Watts
EXTERNAL ENVIRONMENT +70°F/30 KNOT WIND
INITIAL SYSTEM TEMPERATURE: +70°F
OXIDIZER EXTERNAL TO FUEL

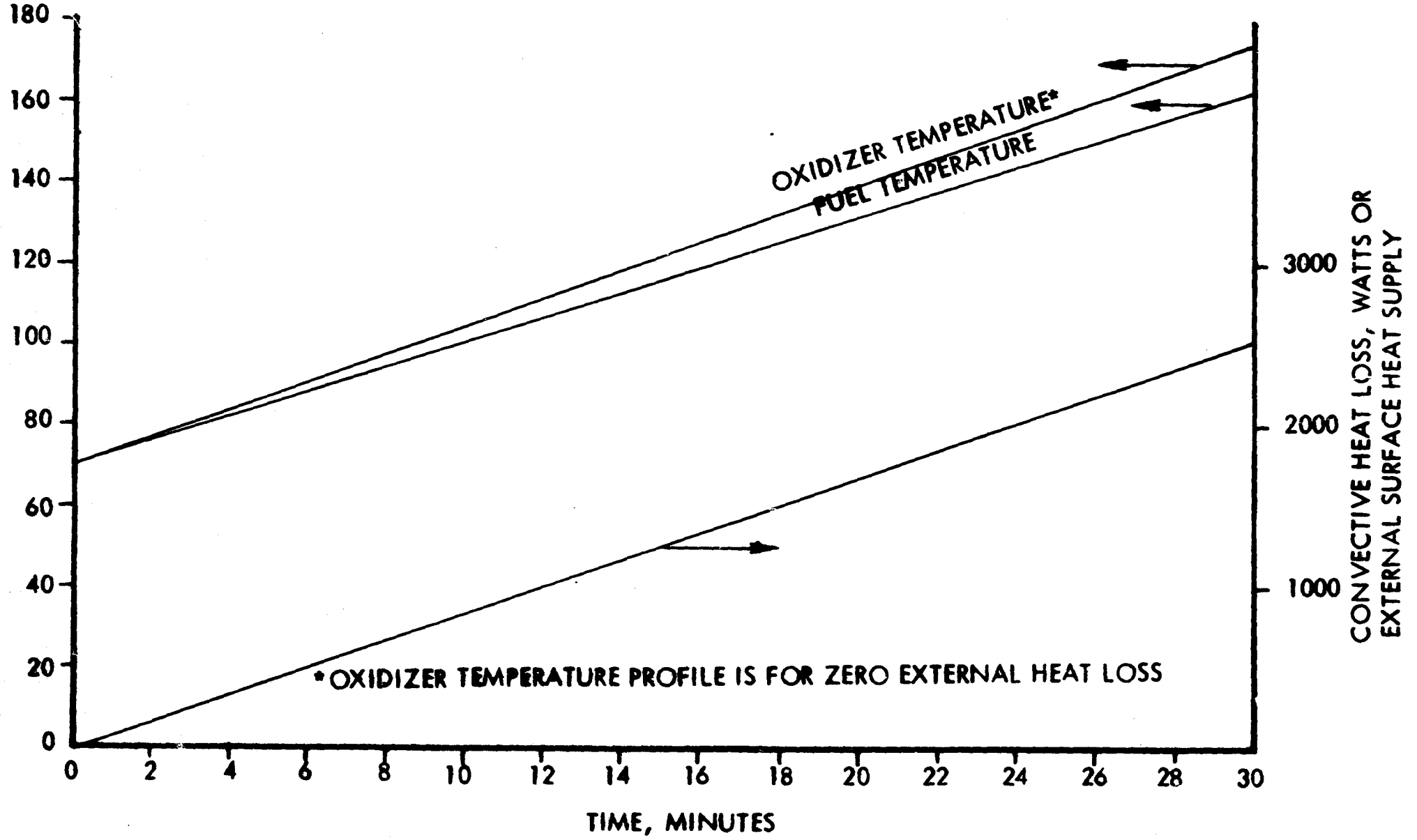


Figure 3.2.3.19
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VAPAK SOUNDING ROCKET
PROPELLANT HEATING REQUIREMENTS
INTERNAL HEATER: 3295 Watts
EXTERNAL ENVIRONMENT -40°F/30 KNOT WIND
INITIAL SYSTEM TEMPERATURE: +70°F
OXIDIZER EXTERNAL TO FUEL

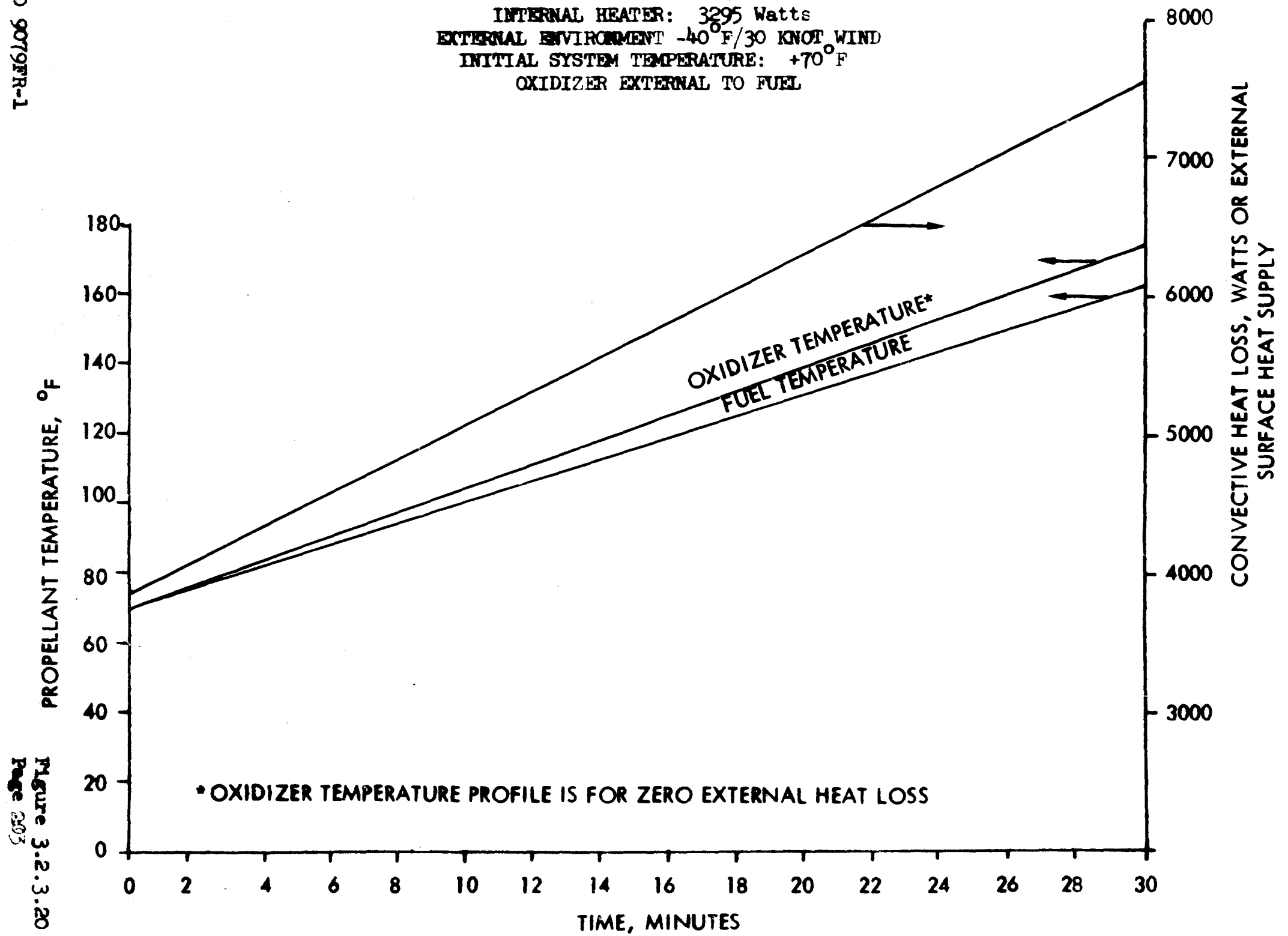


Figure 3.2.3.20
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3.2.3.5 CURRENT MOTOR CONCEPT STATUS

DEVELOPMENT STATUS

The composite rocket has undergone some development testing. The testing program (Table 3.2.3.3) included a number of static firings of the composite motor concept, designated SSR-1. Transition from solid to liquid VaPak operation was accomplished. Aerojet experience to date is as follows.

Storage - A long duration storage test was conducted with the VaPak bipropellants $NH_3 + N_2H_4/NO + N_2O_4$, maintained in a ready to expel (constantly pressurized) condition for over six months. This test was conducted to demonstrate the feasibility of maintaining these storable vapor-pressurized bipropellants in an instantly ready-to-fire condition. The storage tankage used for the test program was fabricated from Titanium A-110 and included a common bulkhead design. The VaPak propellants were maintained at $100^{o}F$ by 200 w electrical heaters, controlled by the propellant vapor pressures.

The VaPak storage program was terminated by a 60-sec small-scale engine firing in which the propellants were VaPak-expelled into the combustion chamber.

Expulsion - More than 38 VaPak expulsion tests have been conducted to investigate the expulsion characteristics of earth cryogenic and storable propellants. These tests have demonstrated the propellant tank pressure decay and correlation to predicted VaPak decay characteristics.

VaPak expulsion tests evaluated the following propellants: N_2O_4, $N_2O_4 + NO$, LO_2, LN_2, LH_2, and $NH_3 + N_2H_4$. The tests were conducted with tank vapor pressures ranging from 354 psia to 87 psia. Tests also demonstrated the pressure repeatability in tank pressure during the liquid expulsion.

Rocket Engine Tests - More than 120 VaPak rocket engine firing tests have been conducted to evaluate the bipropellant expulsion, injector design, ignition and combustion characteristics, and variable thrust control of the saturated propellants, $NO + N_2O_4/NH_3 + N_2H_4$ and $N_2O_4/NH_3 + N_2H_4$. Seventeen small-scale engine tests were conducted with a Lark rocket engine. These tests evaluated the VaPak engine combustion characteristics over chamber pressure range of 60 to 330 psia.

A flight propulsion system was fabricated from surplus aircraft oxygen tank, 5-3/4 in. dia., Lark valves and engine, and associate plumbing. A transtainer-launch cylinder with a 6-in. ID surrounded by a heater and insulation was used to transport the propulsion system and condition the propellants. The propulsion system was transported to Salton Sea, California, and loaded with VaPak propellants. The propulsion system was conditioned and then breech launched from the cylinder on schedule 15 April 1961 without any field checkout firings.

Thirty-six VaPak injector design tests were conducted utilizing a two-dimensional box rocket motor which allows the visual study of the combustion process. These tests supplied combustion data for saturated propellants using 12 various injector designs.

It was hypothesized that because the pressurant of a VaPak system is propellant vapor, a VaPak propulsion system could be successfully started in space without first stratifying the propellants. Therefore, 14 tests were conducted in the box motor to simulate the various propellant injection conditions that could be encountered in a zero gravity restart. The test apparatus was plumbed to supply vapor from the top of the tanks or liquid from the bottom of the VaPak propellant tanks. Liquid-liquid was injected into the chamber approximately 1-sec after FS_1 to simulate propellant stratification. The tests demonstrated that there were no hard starts or hardware damage even when liquid fuel and gaseous oxidizer were discharged into the chamber.

Box motor injector test results were used to fabricate injectors for a second-stage Titan thrust chamber. Forty-three engine tests were conducted at a chamber pressure of approximately 50 psia (A_t = 69.25 $in.^2$), thus developing an altitude thrust of nearly 6000 lb. These tests demonstrated the VaPak performance and system operating characteristics, using large-scale engine hardware.

Ten engine tests were conducted to investigate the saturated liquid variable thrust characteristics utilizing Titan hardware. The only hardware modifications required were the installation of positioning control for the thrust chamber valve (TCV) and a gas source for the VIP system.

The tests conducted with VaPak and VIP pressurized propellants successfully demonstrated that the chamber pressure (P_c) could be reduced from over 50 psig to 2 psig without combustion instability by simply changing the position of the TCV.

3.2.3.6 ADDITIONAL SYSTEM CONSIDERATIONS

MAINTAINABILITY

The simplicity of the MSR-2 design minimizes the need for maintainability. The prepackaged and sealed propellant tanks and the fact that the MSR-2 has only one moving part (valve piston) which is exposed to the propellants only after ignition, reduce maintenance requirements to a minimum, if any.

RELIABILITY

A high level of reliability for the MSR-2 is assured by virtue of its simplicity and highly conservative design. For example, the MSR-2 has eliminated the need for regulators, pressurizing systems, and elaborate valves. The propellant valve is the only part of the system that contains a moving part, the piston, which provides straight-through flow to a simple triplet injector.

The MSR-2 has been designed for failsafe operation wherever possible. Without exception, failsafe malfunctions, such as igniter failure, result in aborted launching with proper repair enabling restoration of the vehicle to operation.

The propulsion system reliability design goal is .995 including solid and liquid systems which indicates five failures or less in one-thousand system firings.

SHELF AND SERVICE LIFE

The shelf and service life of the MSR-2 are expected to be determined by the shelf and storage life of the booster grain. At the present time, the estimate of the shelf and service life of the grain is on the order of 5 years.

SAFETY AND HANDLING CONSIDERATIONS

The chemicals contained in the rocket are flammable at ambient temperatures but the concentric tank construction provides redundant-welds and separate tanks as a safety feature. The tankage will withstand pressure up to 600 psi at temperatures of 180°F. If leakage should occur, the liquid propellant fumes are toxic and must be avoided. However, the total contents of the tankage will vaporize in the event of leakage, leaving no residuals in 30 minutes at ambient temperatures. If both liquid tanks should burst, or be punctured, at temperatures of 180 degrees, spontaneous combustion will result. It is not advisable to use water to dilute either the fuel or the oxidizer in the event leakage is detected during shipping or handling of the MSR-2.

The solid propellant contained in the thrust chamber is polybutadiene. This is a fast-burning formulation that provides the high-thrust level for initial velocity. Handling of the solid, an integral part of the rocket, is in accordance with standard operating procedures. A grounding-strap must be attached to the rocket when removed from the shipping crate, and personnel warning-signs placed in the assembly area.

The MSR-2, although primarily a liquid rocket, would be handled in the same manner as a solid. Personnel safety regulations established for conduct during launch operations of solid propellant sounding rockets are adequate, with the exception of precautions in the event of possible leakage due to a punctured tank.

There are two squibs used to initiate explosive ordnance during the launch sequence. These squibs meet the one watt/one ampere/five minutes no-fire regulations.

The MSR-2 would be classed as a Class B explosive and shipped with permits under ICC regulations.

ANTICIPATED PROBLEMS AND SPECIAL TESTS

At the present stage of development, no major problems are anticipated. However, the normal development problems associated with rocket

component development such as injector compatibility with the chamber, valve squib sizing, nozzle configuration (with or without graphite insert) are anticipated. Long-term storage tests of the prepackaged liquid propellants are required; Aerojet data on results of such storage tests with MSR-2 materials and propellants are available for a six-month storage period.

3.2.3.7 DEVELOPMENT PLAN

This plan describes the effort required to complete the development of a low cost composite meteorological sounding rocket currently under study by Aerojet. Included in this effort are completion of the composite rocket detail design and analysis, individual workhorse testing of both VaPak liquid engines and solid rocket motors, followed by combined engine tests, and finally, flightweight vehicle demonstration and flight tests. The objective of the effort described in this plan is to complete development testing and demonstrate a reliable low cost composite engine for use in meteorological rockets.

The development program described below will be conducted in accordance with the key milestone schedule shown in Figure 3.2.3.21, and the detailed milestones, Figure 3.2.3.22.

PROGRAM DISCUSSION

Utilization of Existing Data - The MSR-2 development program can be reduced in scope, costs and duration by utilization of test data from a similar motor program conducted by Aerojet. Direct application of all prior test data is not possible, due to design differences; however, sufficient gain is anticipated to make this approach attractive. For instance, transition from solid grain to liquid combustion phase and VaPak pressurization has been demonstrated as well as valve-injector operation.

Component Design and Testi.

Injector - The injector design specified for the MSR-2 engine has successfully tested and demonstrated high combustion efficiency. Hardware

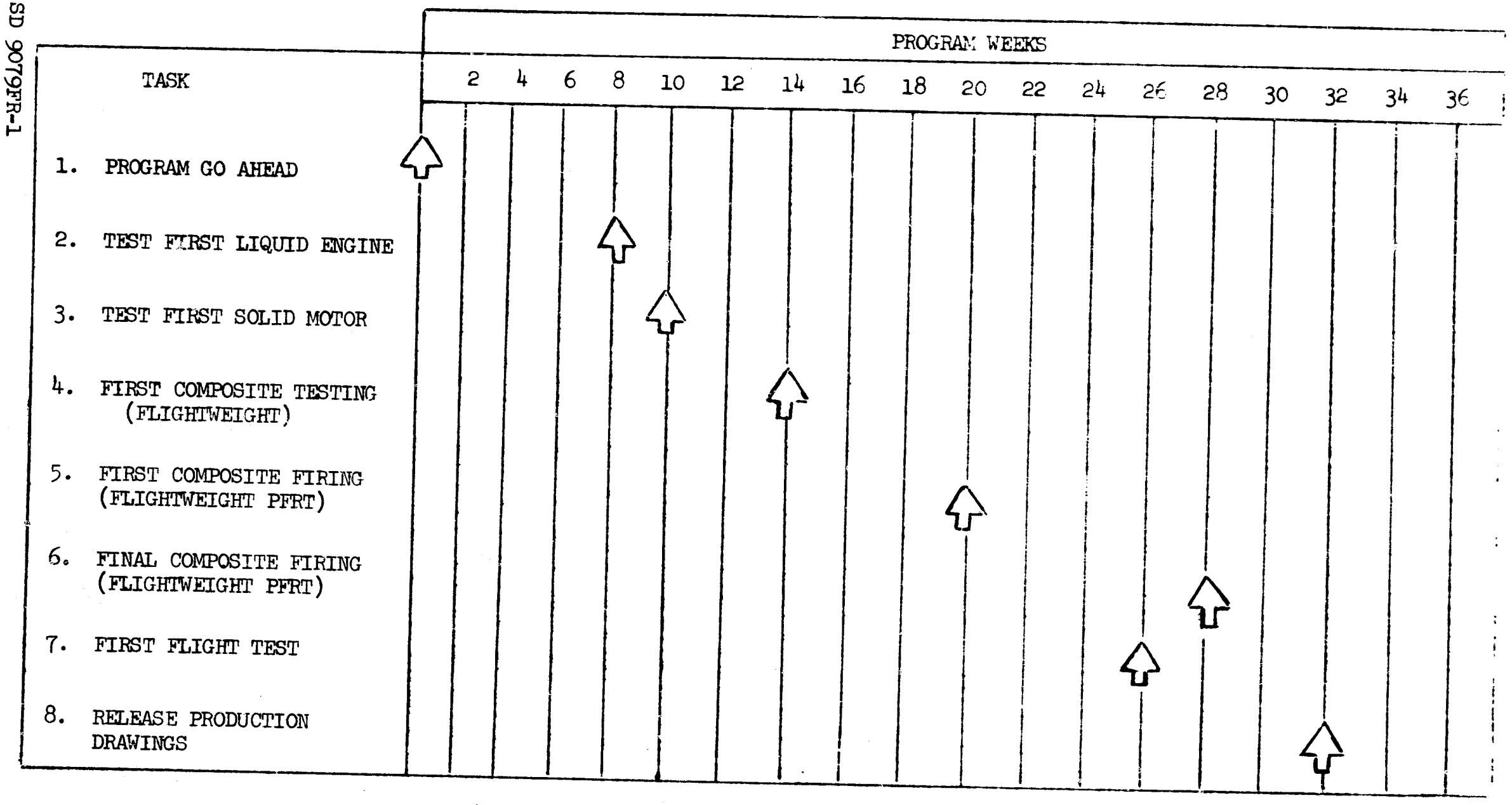


Figure 3.2.3.21

MSR-2 DEVELOPMENT PLAN

KEY MILESTONES

Figure 3.2.3.22

MSR-2 MILESTONE SCHEDULE

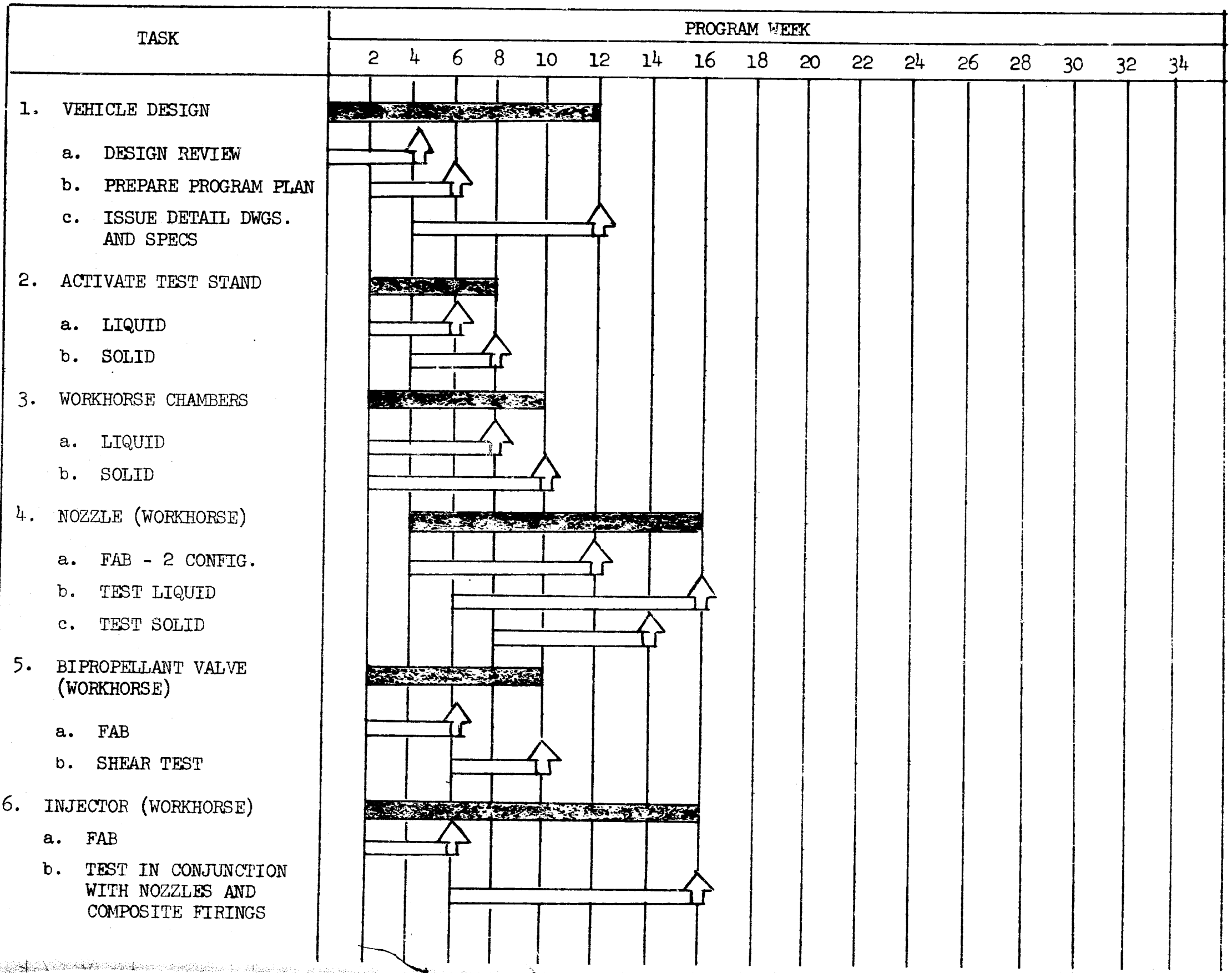
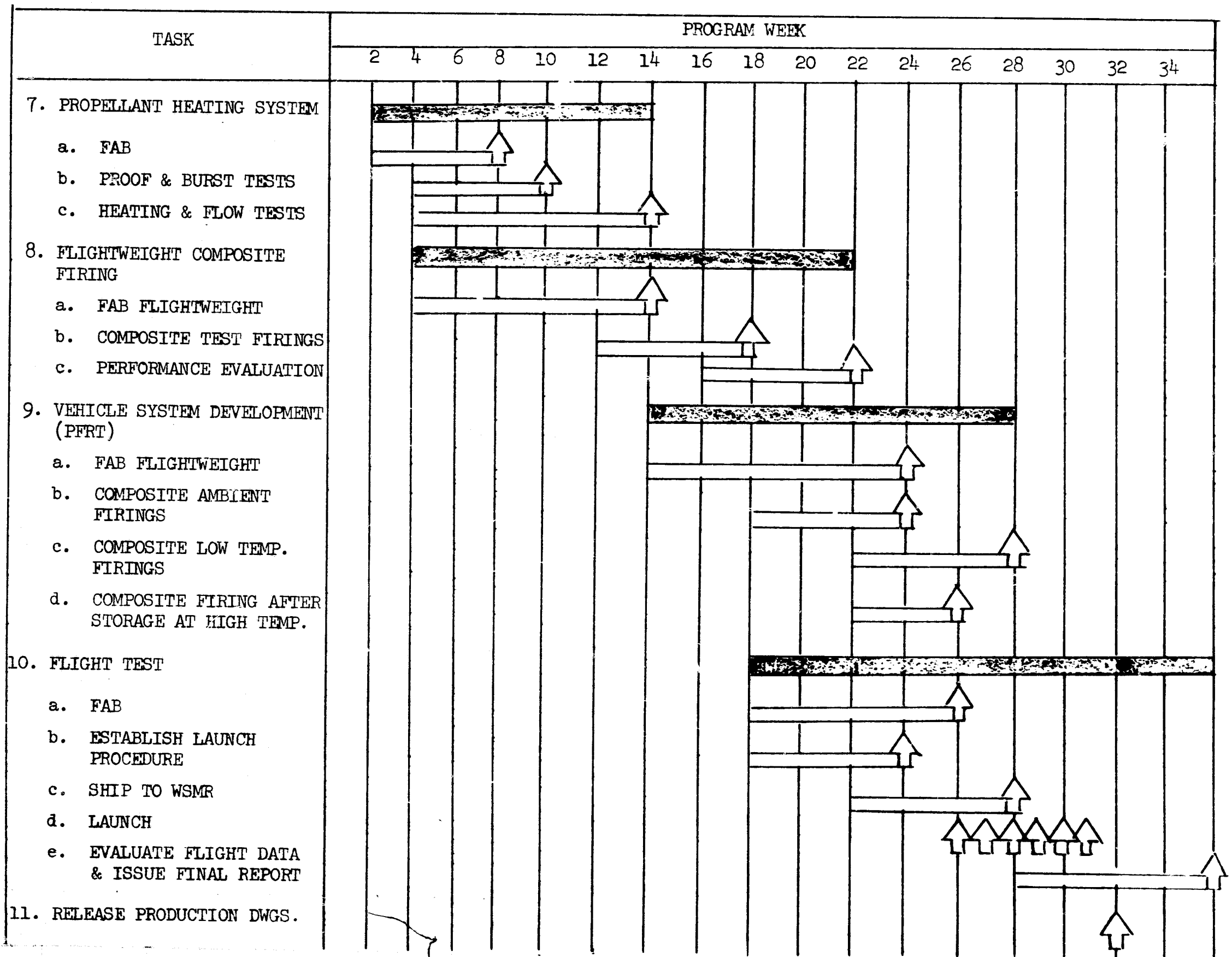


FIGURE 3.2.3.22

MSR-2 MILESTONE SCHEDULE



of this configuration injector is currently available and could be tested immediately upon activation of a test stand.

Testing of the injector will verify past performance data and investigate chamber and nozzle compatibility for this injector configuration.

Booster Propellant Grain - The solid motor that provides the vehicle with its initial boost thrust will be cast into a plastic sleeve and cartridge loaded into the liquid thrust chamber. Final grain sizing will be based on a detailed ballistic analysis to optimize boost performance.

Shortly after program go-ahead, five aluminum workhorse chambers will be fabricated for use in solid motor ballistic evaluation tests. The intent of these tests will be to obtain an early demonstration of the boost grain performance. Subsequent liquid testing on spent solid chambers will demonstrate overall design feasibility, and obtain insulation and nozzle erosion data.

Insulation - A detailed thermal analysis will be conducted to establish the insulation materials and thicknesses for the combustion chamber insulation. The material and thickness of the phenolic sleeves will be established as the insulator for both liquid and solid combustion phases. The insulator testing will be conducted in conjunction with injector, nozzle and solid grain evaluation and performance demonstration firings. This test effort is intended to uncover any significant problems that might occur and provide a confidence in the insulation design.

Nozzle - Nozzle designs, incorporating molded glass/phenolic (fiberite) with a G-90 graphite insert and a molded silica/phenolic, will be fabricated and evaluated.

Erosion rates after full duration test firings of the candidate nozzle materials will be compared to the allowable design criteria. Final material selection will be based on evaluation of the test results.

Propellant Valve - Workhorse propellant valves will be fabricated and tested to obtain the optimum squib charge versus the critical shear

nipple wall thickness. The object of this testing is to obtain repeatable squib performance and once achieved generate the necessary specifications to insure squib charge reliability.

Tankage - The concentric tankage is designed with the primary emphasis on safety, simplicity and low cost. Individual propellant tanks and redundant welds limit the possibility of fire in the event one tank were to develop leakage. Leak tests will be performed on all propellant tanks during the fabrication assembly welding operation.

Workhorse tanks will be fabricated and used during initial engine firing tests. Two flightweight tanks will be heated to 175^{o}F and tested to destruction (burst) to verify stress analysis and insure adequate safety factors for tankage even when in the heated condition. For actual flight vehicles, this situation will only be encountered on the remote launch pad, and will offer an additional factor of safety for tankage under normal storage conditions.

Igniter - The igniter will be designed to be installed in the motor from the aft end, just prior to firing. A single squib will be used to initiate the igniter to save costs relative to a redundant squib design. Failure of the igniter results in temporary flight abort until the igniter can be replaced. The igniter will be designed to withstand the same environmental criteria as the solid motor although ordinarily it will not be exposed to the same extremes because it will be packaged and stored separately.

Twelve igniters will be fabricated and tested to demonstrate the design (Fig.3.2.3.12). Six of these igniters will be fired to establish a ballistic reference and the remaining six will be subjected to environmental testing, consisting of temperature cycling, transportation and handling type vibrations and humidity exposure.

Propulsion System Testing

Flightweight Propulsion System - Upon successful completion of the individual component tests described above, 6 complete flightweight composite systems will be fabricated, assembled and test fired to verify system

integrity. Evaluation of the test data will pinpoint system discrepancies or abnormalities, if any. During this phase of the program, it is anticipated that certain design and material changes may be required, consequently provisions for alternate insulator sleeves, nozzles, and propellant grain are included in the plan. Upon successful demonstration of the composite flightweight system, fabrication of 12 flightweight propulsion systems will begin.

Preliminary Flight Rating Test (PFRT) - Three flightweight composite propulsion systems will be test-fired to establish nominal performance data and of the remaining nine, three will be subjected to environmental testing. This testing will consist of temperature cycling, humidity exposure and simulated transportation and handling vibrations, then fired to determine if these environments have a detrimental effect on engine performance.

Flight Test - The remaining six vehicles will be shipped to a test range for flight testing. Launch and handling procedures will be coordinated with the test range personnel prior to delivery of the first flight version MSR-2. Aerojet will supply flight crews for the initial flight tests. Test range personnel will track the vehicles and provide Aerojet reduced flight data and motion picture coverage of all flights.

PROGRAM PLAN

Vehicle Design

The MSR-2 basic design, theoretical performance calculations, and actual test data will be reviewed and revised as required. A complete review of detail drawings to establish proper detail design, materials and finish requirements, process andassembly procedures, will be made before approving all detail and assembly drawings. The necessary manufacturing, processing and assembly documents such as welding procedures, assembly procedures, bonding specifications, etc., will be prepared.

Activate Test Stands

Test stands in both Liquid and Solid Test areas with sufficient instrumentation and load cells to record engine/motor thrust, temperature, pressure and flow rates will be set up.

Workhorse Chambers

Four (4) reusable liquid workhorse chambers for liquid tests will be fabricated. Two of these chambers will be of heavy wall, and two thin wall, aluminum construction. All liquid workhorse chambers will contain provisions for removing test nozzles and injectors.

Ten (10) solid workhorse chambers for solid grain tests will be fabricated. These chambers will contain provisions for removing test nozzles.

Nozzle

Acceptable candidate nozzle materials and manufacturing procedures will be selected before fabricating 5 samples for liquid and 5 samples for solid test firing of each material.

Ten (10) five-second liquid and 5 one-second solid test firing on each candidate material will be conducted, the results evaluated and a final design selection made. Five liquid and three solid successful tests, without major redesign between tests should be conducted to establish the basis for the final selection. The following approaches should be investigated:

1. Molded glass/phenolic (fiberite) nozzle with G-90 graphite insert.

2. Molded silica/phenolic (U.S. Polymeric) nozzle.

Bi-Propellant Valve

Using workhorse valves, limits of shear fitting critical cross-section will be established to insure compatibility with squib charge. A minimum of five successful squib valve firings will be conducted with maximum shear section and at design pressure. The final design selection will be based upon 5 consecutive successful tests with no major design changes between tests.

Injector

Five (5) workhorse injectors that can be attached (by threads) to the liquid workhorse chambers for test firings will be fabricated. The

workhorse injectors will be tested in conjunction with nozzle evaluation and composite (workhorse components) test firings.

Propellant Heating System and Tankage

Two (2) workhorse and two (2) flightweight propellant tanks will be fabricated, including internal heating system. Leak, proof and burst pressure tests will be conducted to evaluate the material and design of the two flightweight tanks. Minimum acceptable proof pressure will be 1.5 times maximum working pressure. Minimum acceptable burst pressure will be 2.0 times the allowable proof pressure. Propellant temperature and pressure for given heaters and heating cycles of various durations will be monitored. Ten (10) propellant flow tests will be conducted. The system will be considered acceptable if temperature, pressure, and flow rates correspond to calculated values.

Flightweight Composite Firings

Six (6) flightweight propulsion systems will be fabricated to conduct six (6) composite vehicle firings. They will be used to evaluate system performance during composite vehicle firings. The system will be considered acceptable if performance during the vehicle firing test is satisfactory; and system does not require major redesign between tests.

Vehicle System Development (PFRT)

Six (6) complete flightweight vehicles, and two spare thrust chamber assemblies will be fabricated.

Three (3) composite firings at ambient conditions with full instrumentation of propulsion system will be made. Two (2) composite firings at minimum environmental temperature conditions of solid grain and full instrumentation of propulsion system will be included, as will one composite firing after maximum environment storage of vehicle, followed by ambient temperature firing.

Environmental testing will be conducted only on the successful completion of the composite ambient firings. The system will be considered successful if performance during the vehicle firing tests is satisfactory.

Flight Test

Six (6) flightweight vehicles will be fabricated. Procedures will be established for the test site.

The vehicles with factory prepackage liquid and solid propellant, will be shipped to the test range. The results of this test program will then be evaluated before moving to the next program phase.

3.2.3.8 PROGRAM COSTS

INTRODUCTION

The following paragraphs contain the preliminary cost planning figures covering the Aerojet proposed design for the Meteorological Sounding Rocket -2 (MSR-2).

The preliminary cost planning figures are presented for two separate phases (Development and Production) and are shown in Table 3.2.3.4 and Table 3.2.3.5, respectively. The total price for the Development Phase is $357,992, including profit. The prices for the Production Phase of varying quantities in lots per year of 100, 1000, 5,000, and 10,000 units are $171,258, $1,051,386, $3,812,017 and $6,512,535, respectively, including profit.

DEVELOPMENT PHASE

Estimated costs for the proposed nine-month development program are shown on the development cost summary Table 3.2.3.4 for engineering (management and design), manufacturing operations, test operations, and solid operations (solid rocket motor design and development). Quantities of development

Table 3.2.3.4

MSR-2

DEVELOPMENT PHASE

	DEVELOPMENT TASK		
1.	Program Management	$26,200	
2.	Program Engineering/design	$52,400	
	Sub Total		$78,600
3.	Manufacturing		
	Thrust Chamber (16 ea.)	$ 2,180	
	Nozzle (30 ea.)	9,159	
	Valve (5 ea.)	1,890	
	Tanks (4 ea.)	3,120	
	Composite System	22,970	
	Sub Total		$39,319
4.	Test Operations		
	Testing	$83,652	
	Propellant Cost	13,221	
	Administration	10,656	
	Sub Total		$107,529
5.	Solid Operations		100,000
6.	Total Development Cost		$325,448
7.	Profit at 10%		$32,544
8.	Total Price		$357,992

Table 3.2.3.5

PRODUCTION PHASE:

FUNCTIONAL AREA/TASK	100			1,000	
	UNIT COST ($)	SUBTASK COST ($)	TASK COST ($)	UNIT COST ($)	SUBTASK COST ($)
1. ENGINEERING					
Program Management		17,050			35,100
Program Engineering		15,100			70,200
Sub Total	321		32,150	105	
2. MANUFACTURING					
System Cost		59,700			501,000
Tooling		NONE			39,000
Sub Total	597		59,700	540	
3. TEST OPERATIONS					
Production Fill Costs		18,618			124,120
Propellant Mixing Costs		17,917			18,618
Propellant Cost *		2,104			21,040
Sub Total	386		38,639	164	
4. SOLID OPERATIONS					
Sub Total	252	25,200	25,200	174	174,000
5. TOTAL COST			155,689		
6. PROFIT (10%)			15,569		
7. TOTAL PRICE	1,713		171,258	1,081	
* PROPELLANT QUANTITY (lb)					
Oxidizer			7,000		
Fuel			4,235		

able 3.2.3.5

TION PHASE: MSR-2

FRAME 2

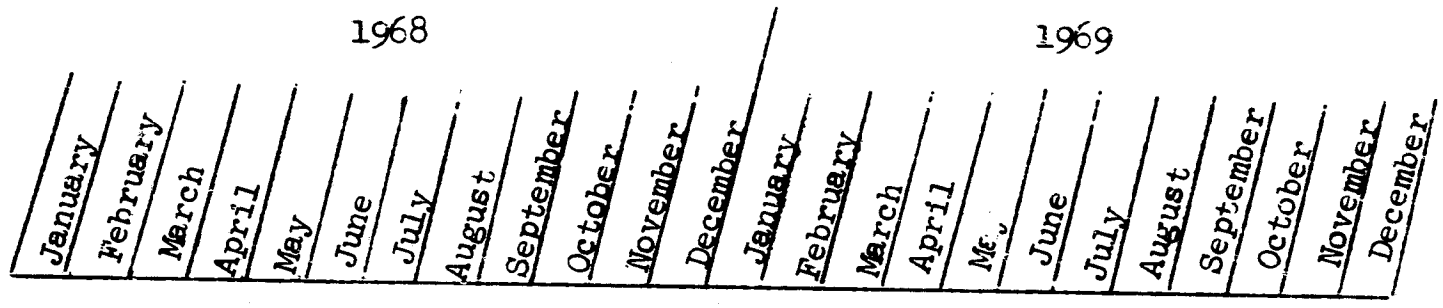
1,000		5,000			10,000		
SUBTASK COST ($)	TASK COST ($)	UNIT COST ($)	SUBTASK COST ($)	TASK COST ($)	UNIT COST ($)	SUBTASK COST ($)	TASK COST ($)
35,100			35,100			35,100	
70,200			175,500			245,700	
	105,300	42		210,600	28		280,800
501,000			2,120,000			3,910,000	
39,000			61,651			72,683	
	540,000	436		2,181,651	398		3,982,638
124,120			387,877			620,600	
18,618			93,091			186,181	
21,040			97,251			190,268	
	163,778	156		578,219	100		997,049
174,000	174,000	99	495,000	495,000	66	660,000	660,000
	983,078			3,465,470			5,920,487
	98,308			346,547			592,048
	1,081,386	762		3,812,017	651		6,512,535
	70,000			324,000			634,000
	42,350			197,530			385,900

hardware, as shown on the cost summary, are planned to be delivered and tested as shown on the development test schedule, Table 3.2.3.6. Propellant requirements for development testing are shown on Table 3.2.3.7.

TEST OPERATIONS - Test costs are based upon the following test plan.

1. There will be 10 expulsion tests to be run in Building D-5.
   a. All tests will be run with one propellant at a time, 5 with fuel and 5 with oxidizer.
   b. Water may be substituted as the second propellant if required.
   c. Propellant temperature and pressures will be measured and tests will be run at both ambient and heated conditions. Maximum propellant temperature and pressure are 175°F and 300 psi.
   d. VaPak propellants will be heated by a strip heater wound around the separation wall of the concentric VaPak tanks. Electrical requirement to the strip heater and controls in 110 VAC, 3 Amps.
   e. Since overspin of flow meters has been experienced at Azusa during flow tests of heated VaPak propellants, oversized flow meters will be modified and then calibrated during expulsion tests at VaPak propellant temperatures for subsequent flow measurement during TCA testing. Existing flow meters will be modified for VaPak propellant by replacing the teflon bearings with ball bearings. Calibration will be accomplished by expelling the hot VaPak propellant through the modified oversized flow meter into a water-cooled heat exchanger and then passing the cooled propellant through a calibrated second flow meter for comparison.

DEVELOPMENT TEST SCHEDULE



TEST STAND	PROGRAM	1	2	3	4	5	6	7	8	9
D-5	MSR-2 DEVELOPMENT TESTING									
	Design Criteria & Test Stand Modification									
	Propellant Heating and Expulsion			3	4	3				
	VAPAK Sys-WH Tanks, Chambers and Nozzle				6	7	7			
	Composite Sys. - WH Tanks							2	3	
	Composite Sys. - Flt. Wt. Tanks								4	2
	Flight Test Fills									6

Table 3.2.3.6

PROPELLANT REQUIREMENTS

DEVELOPMENT

PROGRAM MONTH	VAPAK OXIDIZER 85% N_2O_4 lbs	15% NO lbs	TOTAL lbs	VAPAK FUEL 70% N_2H_4 lbs	30% NH_3 lbs	TOTAL lbs	LN_2 tons
1							
2							25
3	93	17	110	49	21	70	25
4	416	74	490	210	90	300	25
5	467	83	550	238	102	340	25
6	382	68	450	189	81	270	25
7	110	20	130	56	24	80	25
8	382	68	450	189	81	270	25
9	442	78	520	224	96	320	25
TOTAL	2292	408	2700	1155	495	1650	200

NOTE: (1) VAPAK oxidizer can be purchased in 1500 lb cylinders at $.25/lb. Assume procure 2 cylinders, total 3000 lb x $.25/lb = $750.

(2) VAPAK fuel is proposed to be mixed in F Area in 300 lb lots. Assume procure constituents at contractor price and mix 6 lots total 1800 lb.

1260 lb N_2H_4 x $3.80/lb	=	$4790
540 lb NH_3 x .23/lb	=	130
TOTAL FUEL MATL. COST		$4920

Table 3.2.3.7

2. There will be 20 nozzle and chamber tests at Test Stand D-5 using workhorse VaPak tanks and thrust chamber valves supplied by the projects.

   a. Testing is planned in the vertical down firing attitude using the existing DICORAP stand modified for the MSR-2 and with a cooled flame deflector.

   b. A nitrogen tank pressurization system will be provided for initial development tests at ambient conditions. An electrical source of 110 VAC, 3 Amp will be supplied to the strip heaters for VaPak self-pressurization in final development tests. Projects will supply the heaters and controls.

   c. VaPak propellant temperature, pressures, and flow rates, chamber pressure and thrust are to be measured. Flow rates and thrust measurement accuracy within 5% are acceptable. Maximum propellant temperature and pressures are 175°F and 300 psi and chamber pressure and thrust are 225 psi and 500 lb. Since propellant flow rates will vary during the 49.6 seconds duration firing, propellant flow meters will be sized for maximum flows and calibrated during previous tank expulsion tests.

3. There will be 11 composite system tests in Test Stand D-5 with flightweight tanks, solid boost rocket and squib-operated bipropellant valve and injector assembly.

   a. The firing will be conducted essentially the same as nozzle and chamber tests. The solid boost rocket will contain 8.5 lb of solid propellant (ANB-3248) and will fire for one second duration at a maximum 2000 P_c and 2750 lb thrust. Thrust measurement accuracy within 5% is acceptable. The solid boost igniters will be supplied by the projects. The test area will provide 28 VDC to the squib.

b. A thrust cone threaded adapter will be fabricated by the test area to thread onto the forward flightweight tank bulkhead similar to the payload nose cone for thrust take out. Axial thrust only is required and will be measured through this adapter with a 5K load cell within accuracy of 5% at the 2750 lb thrust boost rocket and 500 lb thrust sustainer, dual burn test.

c. Thrust side loads will be taken out through flexures and adapters connected to the rocket fin attach points.

d. The flightweight development tanks are assumed to be provided with separate fill and vent ports for each propellant and that these ports will be located in the tank forward bulkhead. In this manner the tanks can be filled remotely with the rocket assembled and installed in the test stand ready for test.

e. Propellant flow rates cannot be measured during sustainer burn due to rocket geometry. Tanked propellants will be weighed prior to test during the filling operation.

f. Environmental conditioning is not required.

4. There will be 6 development flight test VaPak tanks to be filled in Test Stand D-5.

   a. The test area will receive 6 tank assemblies to be filled in the 9th program month.

   b. Propellant fill ports will not require a seal weld for closure.

   c. The rockets will not be assembled in the test area.

5. Data requirements are assumed similar to the DICORAP Program and are summarized below:

DESCRIPTION	PLOTS/ CHARTS	PARAMETERS/ CHANNELS
Start Transient	1	8
FS-1 to FS-2 Run	4	8
Millisadic		
Oscillograph		
Direct Writing (Browns)		
High Frequency	Not Required	

This program assumes that propellants will be furnished by the contractor for the development testing program. Nitrogen is assumed to be Government-furnished at no cost to AGC. It is assumed that VaPak oxidizer (.85 N_2O_4 + .15 NO, by Wt.) can be supplied premixed commercially. (For example Hercules Corporation can supply under specification MON-15, in 1500 lb, or 2000 lb net weight, N_2O_4 mono cylinders for shipment, storage and dispensing vessels.) For development testing, costs are approximately $.25/lb including $.03/lb shipping and handling costs. It is further assumed VaPak fuel (.70 N_2H_4 + .30 NH_3, by Wt.) will be mixed by AGC for development testing and estimated mixing labor costs are included. Mixing would be accomplished at F Zone in 300 lb lots using an existing 45 gal. 300 psi s.s. sphere (now used in Physics Lab). At these quantities, fuel constituent will be procured commercially at $3.80/lb for N_2H_4 and $.23/lb for NH_3 (if the N_2H_4 can be obtained as GFP the current stock fund price is $.58/lb).

SOLID GRAIN EVALUATION AND FABRICATION - The Solid Rocket Motor Design and Development is based on the following plan which assumes an "All-Goes-Well" Program:

1. Solid Grain Evaluation (Sacramento-SRO)

   a. Establish propulsion requirements for a 6 inch diameter by approximately 8 inch long grain developing approximately 2000 pounds thrust for one second and weighing 8 to 10 pounds.

(1) Thrust and P_c versus time (including tolerances)

(a) Conduct necessary propellant chemistry.

(b) Design grain.

(c) Conduct stress analysis on grain and chamber.

(d) Supply LRO with 5 grains cast in Micarta sleeve of candidate solid propellant grains for test firings. Existing surplus LRO chambers and nozzles will be used for this purpose and LRO will supply workhorse chambers.

(e) Fabricate necessary equipment and complete the casting of 15 propellant grains into prototype thrust chambers: Nine for static firings and six for flight test vehicles.

2. Test Firing of Solid Grain

a. Conduct static test firings of five workhorse chambers and five prototype thrust chambers loaded with solid propellant grains cast by SRO.

b. Establish propulsion performance from test results and advise LRO of any performance changes, if required.

(1) Thrust versus time performance to be determined.

PRODUCTION PHASE

Costs for the production phase are arranged in the following order: Engineering, Manufacturing, Test, and Solid Operations (see Table 3.2.3.5).

A. Engineering support during the 12-month production phase provides engineering management, technical coverage, as well as fiscal control/documentation services. This support is static for the first three increments, but increases for quantities of 5,000, and 10,000 units per year.

B. Manufacturing operations contain fabrication costs for the production of the selected quantities of hardware to be delivered during a 12-month period. These fabrication costs

are based upon using 2014-T6 aluminum because vendor quotes on 2219-T6 aluminum were not received in time for use in this program. It was impractical to secure material quotes from vendors; therefore, the manufacturing costs herein are engineering estimates which are accurate to within 10% of a firm cost proposal.

C. Test operations cost includes the estimates for production filling, special test equipment, propellant mixing, and propellant cost. While, for the production phase, it is assumed that all pressurants will be Government-furnished, quantities of the propellants required are shown on the production cost summary. The estimated costs are shown and are based upon procuring the mixed oxidizer at commercial prices; the hydrazine for the fuel at Government-furnished propellant prices; and the ammonia for the fuel at commercial prices. It is assumed that VaPak oxidizer (.85 N_2O_4 + .15 NO, by Wt.) can be supplied pre-mixed commercially. (For example Hercules Corporation can supply under Specification MON-15, in 1500 lb or 2000 lb net wt., N_2O_4 mono cylinders for shipment, storage and dispensing vessels). For small production quantities under 1000 per year oxidizer costs are approximately $.25/lb, including $.03/lb shipping and handling costs. For production quantities of 100 per year and over oxidizer costs may be discounted about $.05/lb for orders of 20,000 lb or more. It is assumed VaPak fuel (.70 N_2H_4 + .30 NH_3, by wt.) will be mixed by AGC and estimated mixing labor costs are included. Fuel constituent costs are estimated at $.23/lb for commercially procured NH_3 and $.58/lb current stock fund price for GFP procured N_2H_4. For small production quantities under 1000 per year mixing would be accomplished at F Zone in 300 lb lots using an existing 45 gal. 300 psi s.s. sphere (now used in Physics Lab). For production quantities of 1000 per year and over mixing

would be accomplished using a 500 gal. AF trailer mounted tank or 1500 gal. AGC trailer previously used for alumizine. Mixing costs at these quantities would be significantly lower since one setup would produce ten to thirty times the fuel blend as with the 45 gal. small lot mix.

Final closure of the propellant tank ports may be effected by a seal weld on production units, and costs to perform this operation are included herein.

No costs are included for welding equipment or the welding operation in the propellant fill costs for production units of 1000/year or less. However, an estimate of $30,000 for two remote welders (is included in STE costs for production units up to 10,000/year). (This estimate is included with the STE costs for the quantity of 5,000 deliveries.)

No costs are included for Quality Analysis of blended propellants in storage after initial batch mix qualification or for production fill samples, since sample requirements are undefined. Production batch testing costs are not included in this program, but if Batch Testing of production units is required, the estimated costs are available. These proposed test costs are dependent upon authority to use existing facilities and STE on a no-cost, non-interference basis. A listing of the appropriate contracts is also available.

D. Production costs for the solid motor deliveries are purely budgetary and are for planning purposes only. Detailed cost estimates may be formulated upon receipt of a request for a firm cost proposal.

3.3 PROPULSION SYSTEM COMPARISONS

Each of the propulsion systems designed during the Phase II effort were described in detail individually, in the proceeding section. In the following paragraphs, the major attributes of the three designs will be compared.

3.3.1 PHYSICAL

Weight and size constraints placed on the propulsion system were discussed at the outset, as were other system requirements and constraints. As pointed out weight and size constraints had as their basis, not only performance and potential cost considerations, but also operational considerations; i.e., portability and handling under adverse circumstances as required or desirable for the missions being considered. The physical characteristics of the three units designed during the Phase II effort are summarized in Table 3.3.1.1. While there are differences, both inherent to type of unit and due to designers' approach, overall the physical characteristics are quite similar. This is not particularly surprising, since the same goals and/or constraints were applied to each.

The most noticeable difference is between the operating mode of the VaPak unit and the other two. Whereas the Hybrid and Long-Burning Solid motors both have similar operating modes in terms of boost thrust level and duration, and total burning time, the VaPak unit has a lower boost thrust level (2000 lb compared to 4000) and a longer total burning time (50.6 seconds compared to 30.0 for the hybrid and 27.0 for the long-burning solid). Note, however, that the total impulse content is only slightly lower for the VaPak unit, while its propellant weight is significantly lower due to its relatively high specific impulse level. The VaPak data included in this report is based on this particular mode since this is the one for which the most data were available (i.e., old SSR-1 data discussed in Section 3.2.3). A preliminary examination indicated that the operational mode could easily be modified to more nearly approach those of the other units. The desirability of doing so will be discussed further with regard to performance and also under systems aspects.

One other prominent feature, as noted above, is the significantly lower weight of the VaPak motor.

MOTOR PHYSICAL CHARACTERISTICS

Characteristic \ Unit	UTC HYBRID	AGC VaPak Liquid	AGC Long-Burning Solid	Dimension
Initial Weight	171.0	139.6	178.0	Pounds
Final Weight	39.4	38.5	41.3	Pounds
Length/Diameter	124.2/6.0 = 20.7	11[illegible].9/6.0 = 19.2	110.13/6.0 = 18.4	Inches/inches
Boost Thrust/duration(1)	4336/1.0	2130/1.0	3530/1.0	lbs/sec
Sustain Thrust/Duration(1)	990/29.0(2)	545/49.6	1113/26.0	lbs/sec
Boost Total Impulse (1)	4336	2130	4226	lb-sec
Sustain Total Impulse(1)	28,712	27,035	28,938	lb-sec
Boost Specific Impulse(1)	262.5	250.5	270.0	sec
Sustain Specific Impulse(1)	249.0	293.5	248.8	sec
Throat Area	1.475	1.48	1.344	in^2
Exit Area	19.78	8.88	16.12	in^2
Expansion Ratio	13.4	6.0	12.0	-
Chamber Pressure	Boost = 1500/ Sustain = 330	Boost = 2000/ Sustain = 225	Boost = 1600/ Sustain = 500	psia
Case Material	2024T-4 Alum.	2219T-87 Alum.	4130 S.S. to 120 ksiv	-

(1) Vacuum conditions

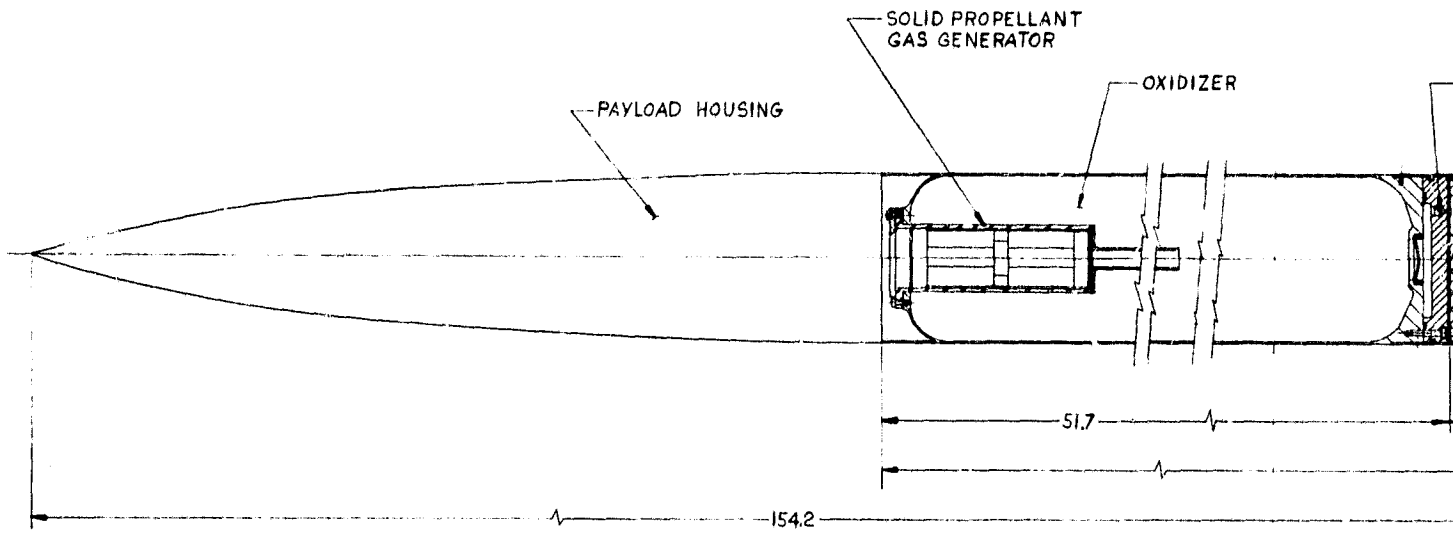
(2) Due to the expansion ratio and chamber pressure selected, nozzle separation will occur up to an altitude of about 10,000 feet; however, checks indicate that the performance calculations, based on vacuum conditions, yield slightly conservative results.

TABLE 3.3.1.1

3.3.2 PERFORMANCE

Typical vehicle configurations, based on the various design motor configurations, are illustrated in Figure 3.3.2.1. In all versions, the payload is housed in a nose section which is a 5.0 caliber (length-to-base diameter ratio) tangent ogive. This particular shape is a relatively low drag configuration with good internal volume. Although weight and balance data were not available at the time (i.e., during Phase II motor design, which is when potential performance evaluations had to be made), gross (and conservative) assumptions were used to estimate aerodynamic stability requirements, hence, stabilizing tail fin size requirements. Fin drag was estimated, at this point, by assuming the fin to be a 0.20 inch thick flat plate having an exposed semispan of 12 inches, and a hemicylindrical leading edge swept back 60°. This is somewhat conservative, however, the results were applied equally to all designs. The resultant drag curves used in performance calculations are shown in Figure 3.3.2.2. The variation in power-on drag simply reflects the effects of variations in nozzle exit area, between the designs, on body base pressure drag (i.e., body base pressure applied over an area composed of total base area, less nozzle exit area). As in Phase I, the vehicle weight was assumed to be the motor weight plus 20 pounds of gross payload weight, where the 20 pounds are assumed to represent 10 pounds of net payload weight plus 10 pounds of ancillary hardware (e.g., nose, fins, nozzle shroud, etc.).

As in Phase I, a two-degree-of-freedom trajectory simulation, programmed for the digital computer, was used to establish the performance capabilities of the vehicles based on each of the motor designs. The vehicle representations were based on the drag and gross payload weights as discussed above, and the propulsion system data supplied by each motor design group. Actually, an iterative procedure was followed in that motor data would be put into the simulation, the trajectory computed, the results analyzed and fed back to the motor designer, and the motor design revised accordingly. Resultant vehicle performance and trajectory data are presented in Table 3.3.2.1, and represents the final output of the iterative process. The computer printout



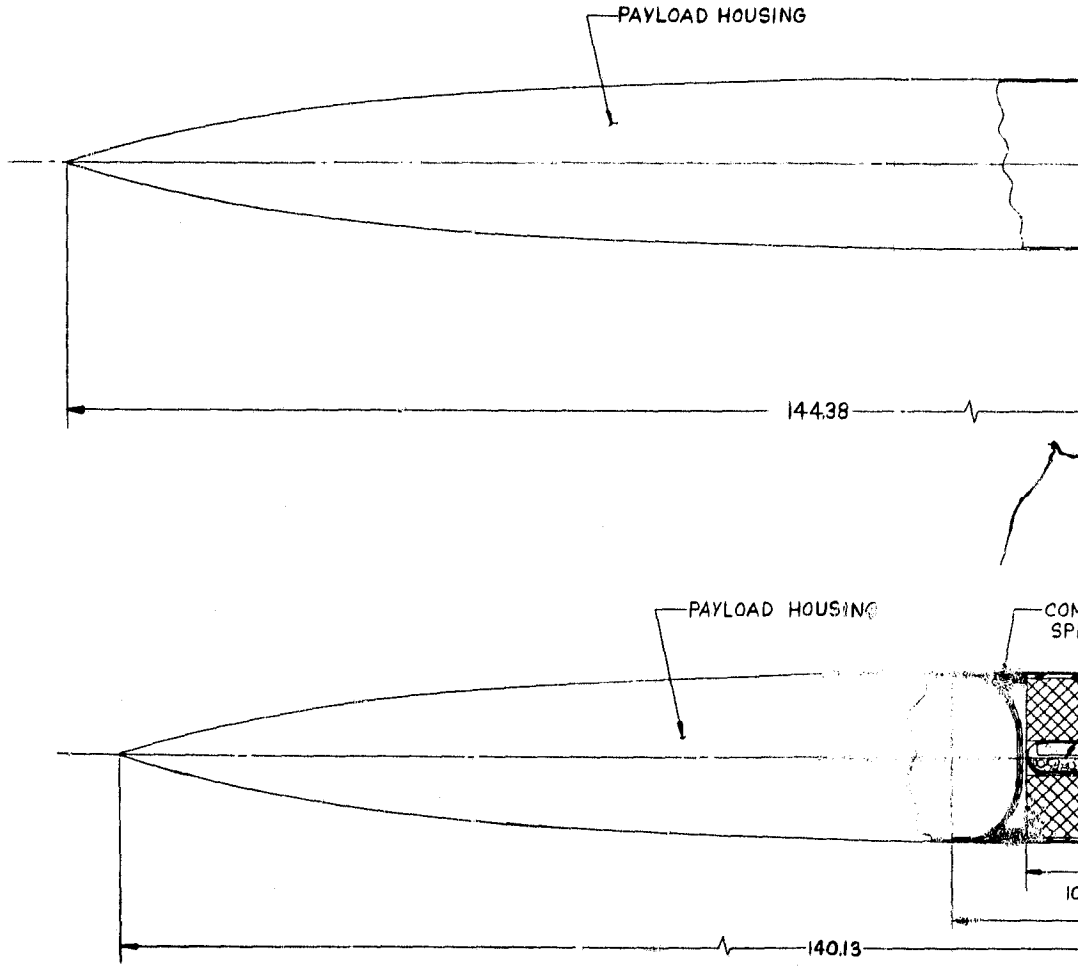
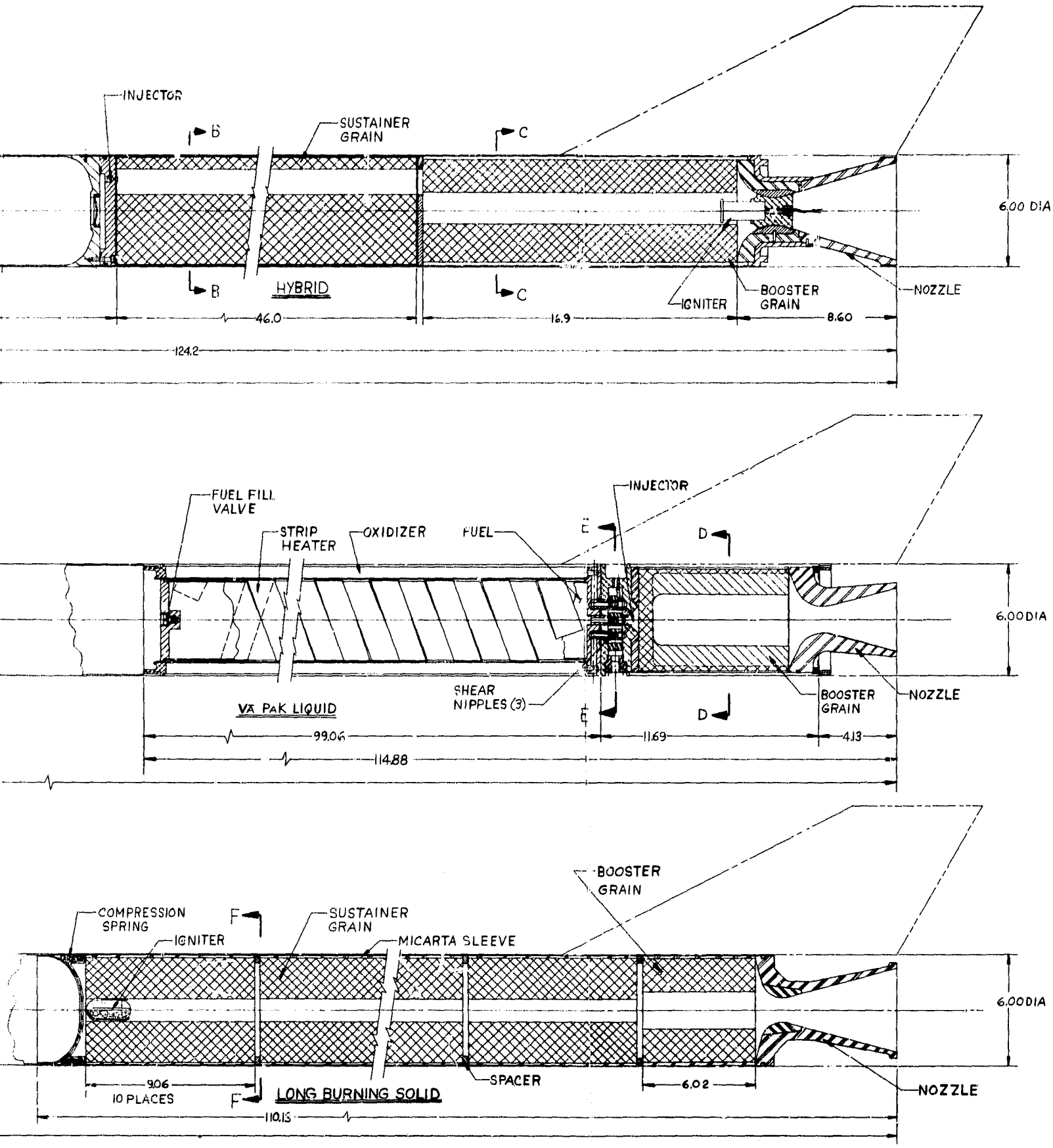


Figure 3.3.2.

FOLDOUT FRAME 1



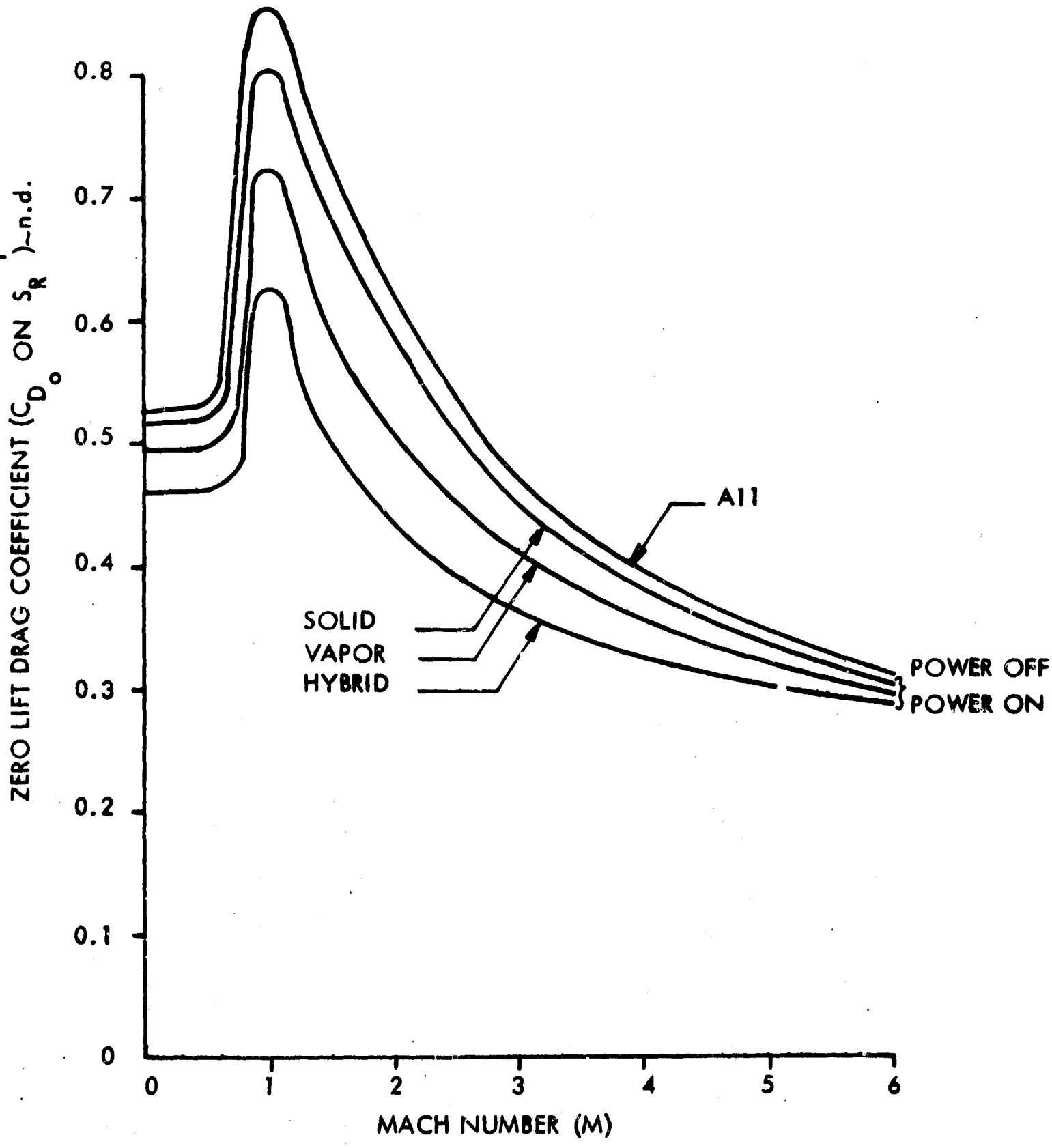
ure 3.3.2.1. Phase II Design Configurations

FOLDOUT FRAME 2

SMALL PROBE

ESTIMATED ZERO-LIFT DRAG CHARACTERISTICS

Note: (1) $S_R = \frac{\pi}{4} d^2$ motor



PERFORMANCE CHARACTERISTICS

Characteristic	UTC HYBRID	AGC VaPak Liquid(3)	AGC Long-Burning Solid	Unit Dimension
Apogee Altitude	469,808	450,816	483,952	ft
Apogee Range	138,252	258,875	131,970	ft
Apogee Velocity	803 (@ t = 191.0 sec)	1550 (@t = 202.2 sec)	750 (@t = 191.6 sec)	ft/sec
Sustainer Burnout Altitude	67,401	95,304	64,002	ft
Sustainer Burnout Range	9,868	26,147	8,752	ft
Sustainer Burnout Velocity	5377 (@t = 30.0 sec)	5056 (@t = 50.6 sec)	5525 (@t = 27.0 sec)	ft/sec
Sustainer Burnout Flt. Path Angle	9.29	18.51	8.55	deg from vertical
Booster Burnout Altitude	329.3	189.9	319.5	ft
Booster Burnout Range	30.4	17.8	29.5	ft
Booster Burnout Velocity	660.3 (@t = 1.0 sec)	381.1 (@t = 1.0 sec)	640.5 (@t = 1.0 sec)	ft/sec
Booster Burnout Flt. Path Angle	5.4	5.57	5.40	deg from vertical
End of Launcher Velocity	138.1(@t = 0.216 sec)	105.9(@t = 0.283 sec)	136.2(@t = 0.219 sec)	ft/sec
Max. Dynamic Pressure	3472 (@ t = 22 sec) (and M = 3.16)	1572 (@ t = 30 sec) (and M = 2.08)	4008 (@ t = 19 sec) (and M = 3.10)	lbs/ft^2
Max. Mach number	5.554	5.087 (@ SBO)	5.707 (@ SBO)	n.d.
Max. Axial Acceleration	21.35	12.13	20.62	g's
Launch Weight	193.6 (4)	159.6	198.0	lbs
Final Weight	62.0 (4)	58.8	62.2	lbs

Notes. (1) Effective launcher length = 15 ft.

(2) Effective launcher angle = 5^{o} from the vertical.

(3) Operational mode as given in Table 3.3.1.1.

(4) Note final data in Table 3.3.1.1 indicates inert motor weight 2.6 lb lower than when "final" trajectories were run.

TABLE 3.3.2.1

tapes are included as Appendix B. Note, that while there is some variation in apogee altitude, all three systems are very close to the target of 140 kilometers ($\approx$ 459,000 feet). By continuing the iterative process, each could have been refined further; however, refining beyond the present point would be of dubious accuracy and of no significance.

Several observations concerning the data in Table 3.3.2.1 are of interest. First, note the close comparison between the Hybrid and Solid motors. This is to be expected since both are quite similar in size, weight, impulse content, etc. Now compare the VaPak results with the other two. Recall from Section 3.3.1 discussion that the operational mode of the VaPak unit was different from the other two, mainly in having a much longer sustainer burning time. Also, although total impulse content of the VaPak unit was about the same as the others, its propellant weight (hence initial vehicle weight) was less due to the higher specific impulse of the liquid propellants. This is reflected in the performance and trajectory data of Table 3.3.2.1. Although the impulse delivery mode is quite different from the optimum (see Phase I discussion) the VaPak powered vehicle does achieve satisfactory altitude, but it does so at almost twice the distance downrange. Although the relationship is not a direct one, generally, the further downrange apogee is attained, the more wind sensitive the vehicle becomes. In opposition to this, three items should be borne in mind: 1) the dual thrusting capability would be expected to relieve this tendency to a large degree; 2) although the data contained herein are for a particular operating mode, as discussed previously, preliminary indications are that the burning time of both phases can be varied considerably even back to match the mode of the other two motors. In fact, a trajectory calculation indicates that if burning time of both phases is halved while total impulse remains constant (near optimum as indicated in Phase I) the apogee altitude would increase to 509,000 feet. This in itself is probably not too significant in terms of cost/size reduction potential, but does indicate a flexible capability to adjust final operation mode to the most desirable configuration when exact mission and payload constraints become known; 3) reduction or elimination of the so-called falling mass hazard may become mandatory, in

which case, wind sensitivity may become a lesser consideration. More will be said on this point later.

It should be noted that preliminary investigation indicates that the operational mode of the Hybrid unit can also be varied considerably (e.g., double the sustainer burning time), but at a somewhat reduced specific impulse. The long burning solid unit is probably at its maximum burning time for a motor restricted to 6.0 inches in diameter, so its mode can be varied only in shortening its burning time.

Based on the data given in Table 3.3.2.1, it is concluded that all three systems have equal capability in terms of placing a given payload weight (i.e., 20 pounds gross for 10 pounds net) at the target altitude ($\approx$ 459,000 ft). The question then resolves down to system features and/or constraints and economics which will be considered in subsequent paragraphs.

3.3.3 COSTS

Since all of the Phase II prime designs are about equal in performance capability or can easily be made to be, the question resolves itself down to two considerations; 1) special attributes or drawbacks of a given design, when considered in the complete system context; 2) motor economics as they affect the overall system cost-effectiveness. Actually, as an overall system design evolves, the above items will fracture into a number of components, including mutual interactions resulting in something in the nature of a design influence coefficient matrix. A simple example would be the determination of the effect of RDT&E dollars spent to achieve a given motor reliability level which, in turn, affects the overall system cost-effectiveness. As noted previously, a well defined operational system specification is not now in hand (although efforts are now under way to produce such a specification); therefore, any comments concerning overall system aspects must be of a general nature, and will be touched upon in subsequent sections. What can be done now is to examine direct propulsion system costs which will be required for the ultimate overall system cost analysis.

At the beginning of the Phase II effort motor designers were asked to give a breakdown and commentary concerning their cost data. As seen in the individual discussions (Section 3.2) this was done in varying degrees. Inquiries resulted in comments varying from "all time would allow", to "given in standard form". The data from the individual discussions are summarized in Table 3.3.3.1. Cost estimates are given for both the RDT&E, and production phases. The production costs are further broken down into unit costs, at varying production rates, and tooling cost where given.

Looking at the data in Table 3.3.3.1 it is seen that the unit cost estimates for the Hybrid and VaPak liquid unit changed little from the Phase I estimates as a result of the Phase II design refinements. The solid data, however, shows a marked decrease in estimated unit costs between Phase I and Phase II results. There are several points of interest in the solid data.

PHASE II COST DATA SUMMARY

Production Rate	Unit Item		UTC Hybrid	AGC VaPak Liquid	AGC Solid Strd. Prod. Tech. ①	AGC Solid Autom. Prod. Tech. ②	AGC Alt. ③ Solid Autom. Prod. Tech. ①
--	RDT&E	$	380,000	357,992	374,000 ③	374,000 ③	368,821 ③
100/yr	Unit Cost	$	1800	1713	3908	3333	2074
	Tooling Costs	$	None Indicated	From RDT&E	110,830	110,831	61,490
1000/yr	Unit Cost	$	940	1042	2408	1330	1035
	Tooling Costs	$	None Indicated	39,000	133,000	133,000	132,746
5000/yr	Unit Cost	$	770	750	1272	1003	672
	Tooling Costs	$	None Indicated	61,651	200,000	200,000	200,836
10,000/yr	Unit Cost	$	550	644	1017	806	581
	Tooling Costs	$	None Indicated	72,683	250,000	250,000	253,319

① Standard and Automated Production Techniques are discussed in Section 3.2.2

② Although the alternate design is very preliminary, it is included because of indicated attractive cost potential for a solid design (see Section 3.2.2 for further discussion).

③ Requires RDT&E and production (at 100 unit level) phases to dovetail for mutual cost sharing, principally tooling.

Table 3.3.3.1

The first point is the apparent unit cost reduction from Phase I to the Phase II design. It was to investigate this anticipated cost reduction, that the solid unit was included in the Phase II effort (see beginning of Section 3.2). The second item of interest is the fact that costs are given for two manufacturing processes for the main solid design, one based on standard solid rocket motor manufacturing techniques, and one based on an automated line technique (the latter now coming into use on some military programs). The benefits of automation are as obvious in this instance as they have been in other fields.

A third, and very significant point as far as solid motor design is concerned, is the cost data for the alternate motor design. Recall that an initial premise of this study was that the unit cost would be strongly coupled to impulse content. Note that the relationship suggested is "strongly coupled", not directly proportional. That is, it was realized that other factors certainly would enter the picture, but it was anticipated that the impulse content would be the more important factor. Comparison of the three solid cost estimates with each other indicate that while impulse content may be a major consideration, other considerations are potentially of equal importance. That is, significant reductions are evident at all production rates, when going from present day standard production techniques to an automated technique. Also, significant reductions are indicated in going from the cartridge loaded design (basic) to the cast in case design (alternate). Thus, production technique, design philosophy, and operational requirements/constraints may all be as significant as the impulse content (i.e., material plus handling costs).

A further point regarding cost concerns the subject of reliability as it influences actual operating costs. Only the solid motor section contains more than a passing mention of reliability considerations even though a discussion of the subject was requested for each design. At this stage of the designs, however, it seems reasonable to assume that all designs can be engineered to have a preproduction level on the order of 0.99 or higher, and to project beyond this level would require further design definition and refinements.

Finally, recognition should be given to the previous work in this general line of inquiry, particularly with regard to cost projections. This work is well summarized by a Stanford Research Institute study in solid propulsion economics reported in Reference 3.3.3.1. The results, detailed in the SRI report, may be briefly summarized as follows; 1) using as a figure-of-merit the ratio of vehicle cost-to-capability in terms of net payload weight times apogee attitude (i.e., dollars/lb-mi), available systems were examined and found to have values between $1.0/lb-mi and $2.0/lb-mi; 2) based on the data given in Reference 3.01, the SRI study uses the results of its correlation work to project a figure-of-merit range of $0.40/lb-mi to $1.7/lb-mi for advanced systems based on hybrid and/or advanced solid systems. Using a similar figure-of-merit the data contained herein give values between $0.66/lb-mi to $1.7/lb-mi. Clearly, this figure-of-merit is of interest only as a rough assessment tool. In a final analysis considerations other than just cost and payload weight time apogee altitude are required to assess the systems' effectiveness (e.g., reliability, capability, etc.).

3.3.4 PHASE II SUMMARY

In general, the Phase II effort confirms the general implications of previous work (Advanced Motor Study of Reference 3.01), and the tentative conclusions of the Phase I work (feasibility and preliminary cost projection). That is, a more detailed design and cost analysis confirms that advanced propulsion technology is now capable of delivering a small probe propulsion system at a relative low cost at high production rates.

Further, rather than being confined to either the solid or hybrid category, it appears that there is across the board capability, including the liquid system, at least in the small probe size system.

Now, several items are of particular interest and deserve further comment. First, the work contained herein is based largely on certain assumptions as to likely requirements and/or constraints imposed on the propulsion

system by overall system considerations. This is as it should be, for the motor is, after all, only one component of an overall system. Secondly, there is now under way a NASA study to define the overall system for a world-wide meteorological rocket network (MRN). Thus, the data generated in this study should be quite useful, in the overall system study. In this regard it will be necessary to reconcile system requirements/constraints assumed herein with those generated in the overall system study. In this regard, the motor designs generated herein are rather sophisticated for this class of motor, but are also relatively cheap at high production rates. In addition the designers indicate a wide range of adaptability. Thus, reconciliation between assumptions of this study, and actual requirements as will be determined in the current system study should have little affect on the cost data contained herein.

Additionally, purely from the propulsion technology standpoint, the units designed indicate a wide spectrum in capability. That is, even though a particular impulse delivery mode was selected at the outset, these units have a wide range capability, the least being the solid unit(s). Specifically, although the hybrid unit designed for this study has a burning time of 30.0 sec, its time could be doubled according to the designer, with little reduction in specific impulse. Actually, the shortest or longest operating capability was not determined. The same is true for the VaPak liquid system. These two systems then have an extremely wide range in operating time capability, something that has not been generally available in this small size motor. The solid units do not have as wide a range as the other units, the maximum being on the order of 27.0 to 30.0 seconds, at least for the small diameter internal burning design, at moderate to good specific impulse levels. For an end burner, which was not considered in this study, this could perhaps be extended somewhat. Although not specifically required for this study, the cartridge loaded solid holds potential for an interesting concept. That is, thrust-time curve shaping by loading grain segments of varying burning rate. If there were a sufficient number of

missions in which a variable thrust curve was highly desirable, and all could be satisfied by a common motor size, it would not be impossible to envision a common set of motor hardware and a standardized set of solid charges (all standardized for manufacturing economy) such that individual motors could be loaded (either at the factory or from a stockpile at a field station) to approximate specified thrust curve shapes. Quite often scientific experimenters mention the desirability of trajectory control either from the standpoint of acceleration limitations, or trajectory times within certain altitude bands. Also being of a scientific or experimental nature the experiments differ and so do the trajectory requirements. Thus, although there are not now stated categories of missions waiting or pressing for such a development, the possibilities should be pointed out. Something on the same order could also be applied to both the hybrid and VaPak liquid system in that a time operating throttling device could be installed forward of the injector (in fact, a current hybrid unit which is much more complex in capabilities is currently under development).

3.4 SYSTEMS CONSIDERATIONS

It has been noted several times that, although the work reported herein is concerned with propulsion requirements, the motors designed will be no more than a component or subsystem to a complete system; however, it was beyond the scope and intent of this study to perform a complete systems analysis. However, while most of the effort expended during this program was concentrated on the motor proper, some consideration was given to its place in a conceptual operational system with particular emphasis given to those areas of direct interfacing with other subsystems. It may be expected that, when the final Meteorological Rocket Network (MEN) requirements are established, there may be some differences between that system and its requirements and those used in this study. As noted in Section 3.3, however, the designs generated herein have a great deal of flexibility in adapting to various operational modes. The important points of this study are the demonstration of design capability, favorable economics, and design flexibility with regard to the propulsion system. In fact, these items should feedback to the complete system study, and perhaps could influence the overall system definition.

It is now recognized that, while there are instrument problems to be overcome, there is a great need to make meteorological measurements at altitudes significantly above those attainable by present equipment, but below the satellite domain. Specifically, a previous study (Reference 3.0.1) which was of a general survey nature, indicates the lack of a relatively economical sounding vehicle for the 100-150 kilometer region. Further, that study indicated that a vehicle capable of performing a meteorological function, by carrying a payload on the order of 10 pounds to approximately 120 kilometers, could also "double in harness" as a D-region scientific carrier (requirements estimated as 30-40 pounds of payload to approximately 60 kilometers). The study also indicated that the later function was unfulfilled.

Thus, a rather fortunate set of circumstances appeared to exist, in that two active missions could be fulfilled by a single carrier vehicle,

thereby mutually sharing potential manufacturing economies. Additionally, that study indicated a good economic potential for a hybrid propulsion system if applied to the missions indicated. It was to evaluate the missions in general and the hybrid system cost-effectiveness, as applied to the Met/D-region mission, in particular, that this phase of the study was initiated. In addition, it was intended that other rocket propulsion systems/ technique would be given at least a cursory examination from the standpoint application and cost-effectiveness.

With regard to cost-effectiveness, other propulsion systems studies previously completed (Reference 3.0.2) indicated that the term cost-effectiveness had to be approached with some care. That is, a propulsion system is not an endin itself but is a component or subsystem to a complete system which has as its aim an assigned mission. Thus, any cost-effectiveness criterion should be based on the overall system and not a particular component or subsystem. This is the philosophy followed in this study, and, in fact, some discussion of overall system consideration and subsystem interfacing is given in this section.

3.4.1 GENERAL

Up to this point all references to a complete system have been limited to stating mission objectives and some rather broad guidelines based on overall system consideration. Before proceeding, a general description of likely system features, as presently envisioned, will probably be helpful, not only in understanding some of the guidelines used, but also in explaining some of the options available as the overall system development progresses.

Simply stated, the system proposed is an extensive and remote means of "instrumenting" the atmosphere, and getting the results to an appropriate analysis center on a timely basis. The obvious question is, if it is so simply

a concept why has it not been implemented? The answer is two part. The first, it has, to the extent the U.S. or other national weather bureaus operate. The second part is, the technology allowing more extensive aspirations are now coming on line: i.e., largely a fall out of the very extensive technology developed for the national space program. Principal contributors to this new found capability are the advanced generations of high speed, high capacity computers (coupled with advanced mathematical techniques for data handling, analysis, and modeling); the advanced state of instrumentation techniques (coupled with the still rapidly advancing capabilities in electronic instrument design and fabrication); and the relatively new rapid, timely communications systems, i.e., the communication satellites. All are, to a very great extent, traceable to the national space program. Here then, is an excellent opportunity to cash in, in a very real, practical, and highly demonstrable way, on the national space program investment.

First, a restatement of the envisioned missions:

1) The prime mission is that of synoptic meteorology, and requires the placing of a standard 10 pound payload at an altitude of 140 kilometers (probably over-performing, but easier to scale down than up - also if this is over-performance it will not be by much). This will be a repetitive function; i.e., same payload on same trajectory for almost every flight from all stations.

2) The secondary mission is that of a D-region scientific probe; i.e., 20-40 lb atmospheric physics experiments to approximately 60 kilometers. Performance calculations indicate that a vehicle capable of satisfying the met mission could easily meet the secondary mission.

Emphasis is placed on the synoptic met mission since it is an order of magnitude larger in scope than the D-region probe mission.

The general make-up of the met rocket system envisioned is as follows:

1) Manufacturing and Logistics System
2) Operational Sites or Complexes
3) Launching System

4) Carrier Vehicle System
5) Payload System
6) Data Acquisition System
7) Communications System
8) Data Analysis and Dissemination System

The objective of this system is to make in situ measurements of atmospheric state variables, using rocket borne payloads, launched on a synoptic schedule from a widely dispersed network of operational sites or complexes. These data are to be used in weather prediction using advanced systems and techniques now under development. The ultimate objective is, of course, accurate long-range weather forecasting. In summary, the met rocket system is intended as a data acquisition subsystem for an advanced weather system.

Certainly all functions shown were not analyzed during this study, however, their existence and influence should be noted, if for nothing more than to be aware of the possibility of propulsion system-functional interactions. There is, in fact, a great deal of interplay between the various system components, particularly in terms of economics. Additionally, if the system is used as extensively as anticipated, particularly if the mission does become world wide in scope, the magnitude of the program investment would be very large. This leads immediately to the obvious requirement that the whole program be approached from the systems analyses standpoint such that cost-effectiveness techniques can be brought to bear from the inception.

In the following paragraphs consideration is given to the functions of the various system components, their likely operating mode, and requirements and/or restrictions placed on the components by the system or on the system by the components. Particular attention is given to the effects of mutual interactions as they affect both economic and technical functions. With regard to the scope and purpose of this study, emphasis is placed on the propulsion system proper, and its direct interfacing components. In fact, some preliminary engineering work directly related to the propulsion system and/or resultant flight vehicle is also contained herein.

3.4.2 CONCEPTUAL SYSTEM OPERATIONAL MODE

It is anticipated that the operational mode of a met rocket system will be some variation about the following general theme.

1) A number of stations will be established, either in U.S. territory, or world wide, in a pattern based on meteorological considerations (e.g., established storm track, etc.).

2) The station will launch rocket borne instrument packages on a coordinated (through communications system), synoptic basis (e.g., simultaneous launch at a given Greenwich mean time, or a sequencial launch based on a common local site time, etc.).

3) The payloads will measure the desired atmospheric state variables and communicate the data to a data acquisition system (e.g., advanced GMD ground station), which will probably be located at the launch site.

4) All the data acquisition systems will then feed the data (either raw or reduced) through the communications system to either local, regional, national, or a central station, and possibly all of them. A natural tie in with the developing communication satellite systems is obvious, particularly for a world wide system.

5) The data analysis and dissemination centers will receive and operate on the data, according to their various functional levels (e.g., local, regional, national forecasting).

This operational mode is of necessity, conceptual in nature, and will require several related systems studies to arrive at a more definitive state. This is of particular concern here since several, as yet to be determined requirements/constraints, related to operational, have a great effect on both design and economic consideration, directly related to the propulsion system.

A pressing need is a more concrete determination of the extent that a met rocket system can be effectively utilized. This is the subject now under study. It holds great implication for the work contained herein, and vice versa. That is, quite simply, the number of units required directly

affects the unit costs as was illustrated in Section 3.3. These results, in turn, feed back and affect the determination of the complete system cost, which must then be evaluated in terms of both cost-effectiveness and total mission worth. The study should also specify required operational reliability levels for each of the system components, or at least give an overall level required, along with a tentative allocation for each component. After examination of the effect of such allocations to each component, data, in the form of influence coefficients could then be fed back to the complete system analysis, thus arriving at the most cost-effective allocation. As a simple example, consider the motor data generated herein. Generalizing for this example, an RDT&E program allotment of $350,000 - $400,000 is projected to a propulsion unit reliability level on the order of 0.99 at the start of production. What really should be considered, in any overall program study, is the determination of RDT&E cost vs unit reliability vs total program cost. This should be done for each of the system components to arrive at the most effective allocation.

A parenthetic reliability consideration, that will be an absolute must during any development program, is a detailed analysis of disastrous failure modes. By the nature of the mission under consideration, there is a distinct possibility of requiring operations near populated areas. In addition, the envisioned operational mode considers routine operation by personnel having capabilities perhaps no higher than that of a semi-skilled technician (particularly if world wide deployment is achieved). Even one disastrous event would have very serious bearing on the successful operation of such a system; therefore, every effort must be made to identify and minimize, in the extreme, potentially disastrous failure modes. This may be accomplished by such means as mechanical design and engineering, operational mode specifications and/or restriction, operational crew training, etc. In any event, this problem too, should be examined in the context of total system analysis to arrive at the most cost-effective solution.

With regard to a potential operation mode, as it would affect both safety, and possibly economics, consideration should be given to automatic

station, particularly for remote site operations. Certainly the state-of-the-art in control system and communications make this a very real possibility. It is one approach to at least alleviating a portion of the disastrous failure potential discussed above (i.e., direct site exposure time of personnel limited to logistics/maintenance functions). Two concepts are illustrated in Figure 3.4.2.1 and Figure 3.4.2.2.

In the first, vehicles are stored in a movable "cartridge" barrel which would be located in a bunker for safety and weather protection. To fire a given vehicle the barrel would rotate until that particular chamber was on the carrier truck centerline. The truck would then translate to place the chamber centerline coincident with that of the vertical launch frame (tube launch shown in illustration). The vehicle is moved from its chamber to the launcher by the loading piston which moves up through the chamber, pushing the vehicle into the launch tube. With the vehicle inserted a given distance into the tube a retaining detent could hold it while the load piston withdraws back into its stored position and the loading doors, shown on the roof of the bunker, close. These doors would probably serve both as weather doors and to shield the bunker from the rocket blast at launch. They would probably be "geared" to the loading piston; i.e., open when it moves up to load a vehicle, and close as it withdraws upon completion of the loading cycle. The launcher proper may be tubular as shown, or a rail, or some other form. It really doesn't matter particularly, although the tubular concept may offer advantages in terms of weight and perhaps significantly weather protection, particularly in arctic use. Also the launcher could be trainable in both elevation and azimuth with no technical difficulties. Such a control system could be geared to a local automatic weather station. The launcher control system could have a built-in statistical wind weighting curve such that the launcher setting could be used to partially correct for ambient weather variability, at least as well as a relatively simple ground station could determine it.

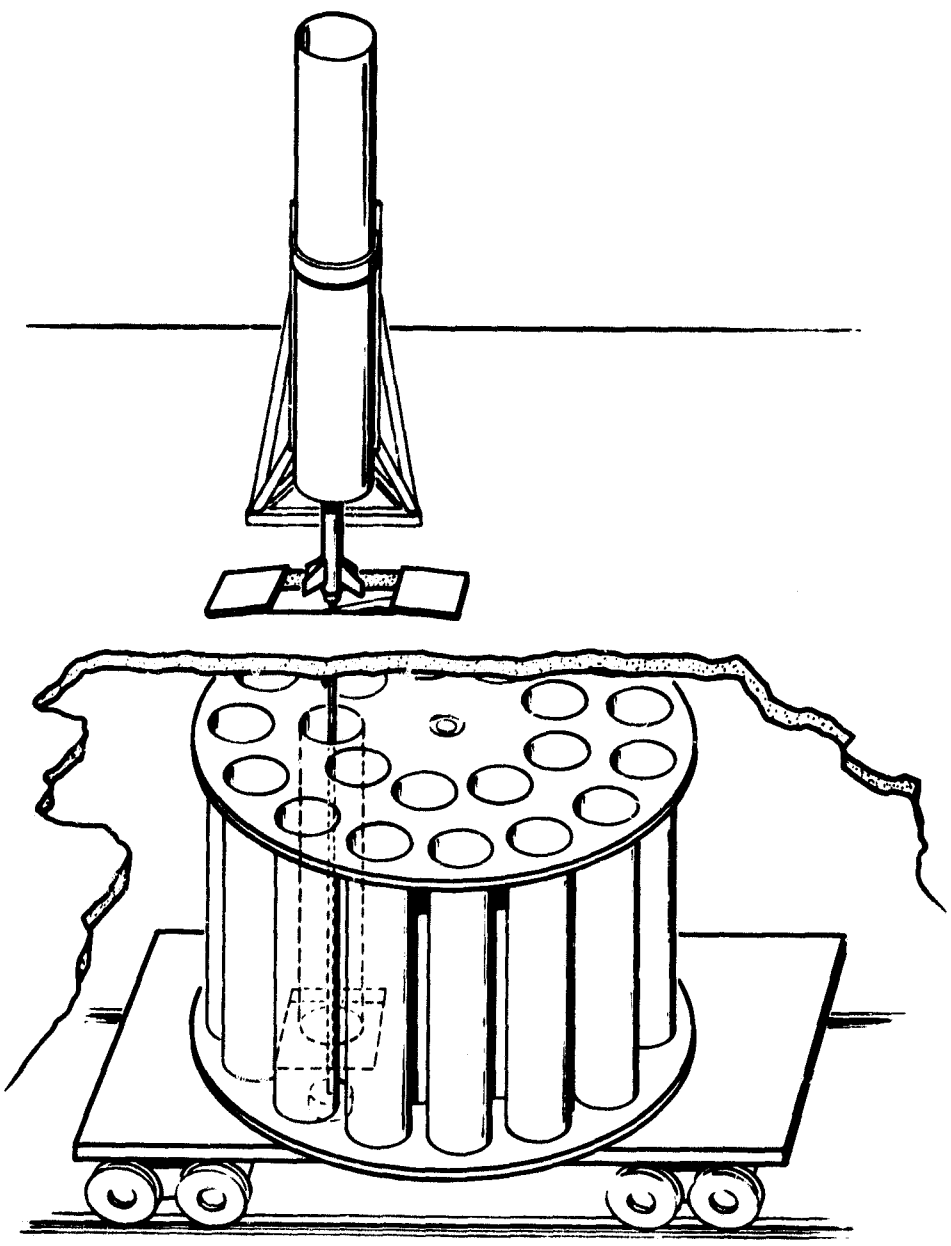


Figure 3.4.2.1. Cartridge Barrel Automatic Launcher Loading System Conceptual Illustration

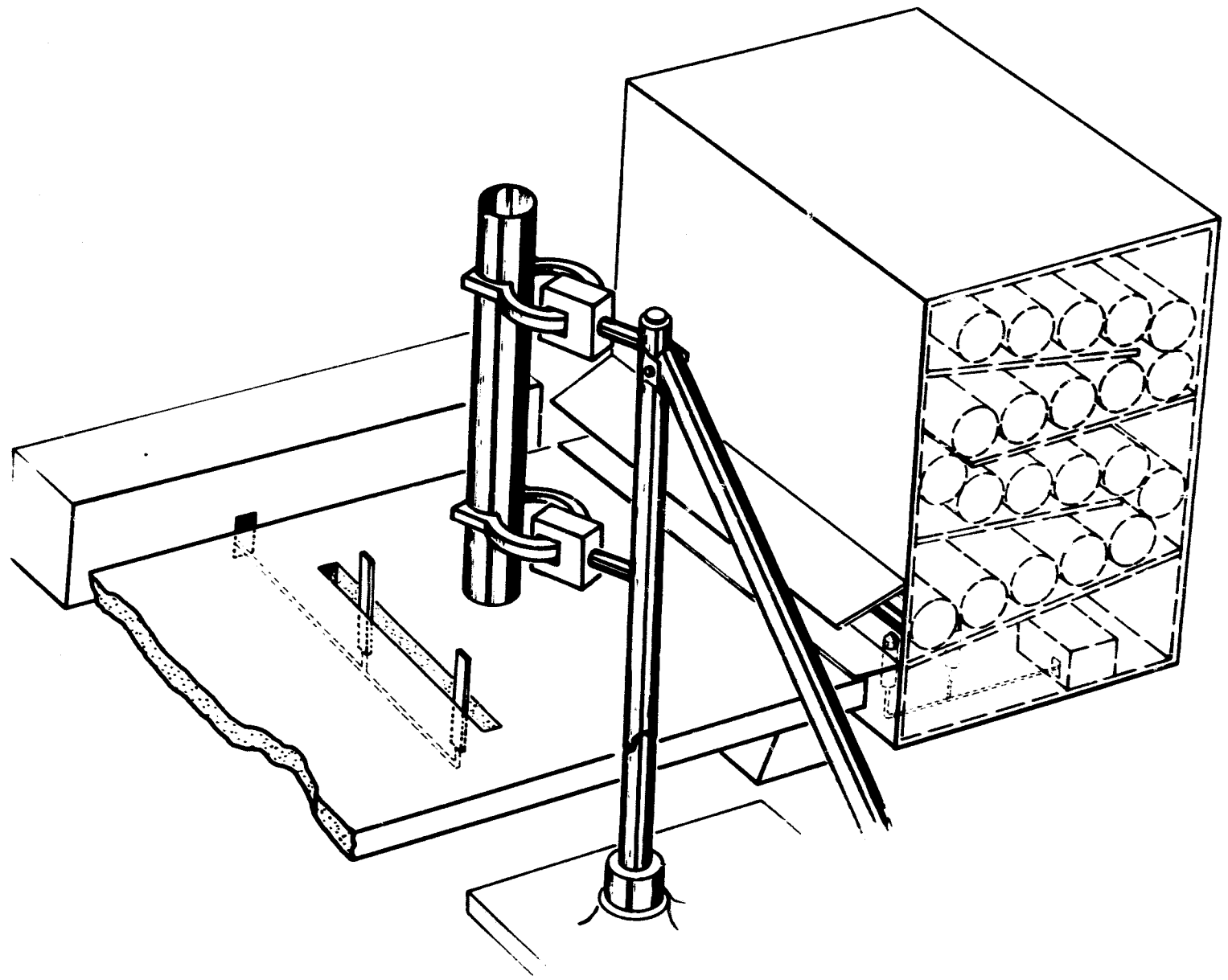


Figure 3.4.2.2. Hopper Type Automatic Launcher Loading System
Conceptual Illustration

A variation on the same theme is illustrated in the other conceptual illustration, Figure 3.4.2.2. Here, instead of the barrel, the carrier vehicles are contained in a hopper system, similar to that used to feed stock in a canning operation. The vehicles themselves are in a container, either special design or perhaps their shipping container. When firing time comes, a detent mechanism drops from in front of the ready round. A second detent holds the next and succeeding rounds unit, the front detent returns to its "stop" position. The second detent then lets all other rounds advance one space, and all is in readiness for the next cycle. Meanwhile the current round has rolled to the ready position, at another detent. Striking this detent, signals the loader that the round is ready. The loader bends over and its mechanical arms grab the case of the ready round. The carrier vehicle, contained inside the case, requires electrical power to function, and would require at least a system go-no-go check at this point. A band of electric contacts could be located around the periphery of the case in a set pattern, under a flexible or frangible covering. Corresponding contacts in the loader arms could make contact by penetrating the protective covering. If the readiness test is no-go, the arms would simple release the unit, the detent retract, and that round roll to a reject area for maintenance/logistic action. The system would then re-cycle to bring the next round into position. If, however, the readiness test was go, then the loader would erect the round and move it to the launch frame (not shown in the illustration) and load it. In fact, the loader may even be a part of the launcher.

There are any number of mechanical contrivances that could be used to accomplish the objectives just illustrated. The point is that current technology makes such a concept feasible. If there is reason for considering such an approach the only deciding factor should be economic (or perhaps political). As for reasons, in addition to reducing potential personnel hazard, there is also the potential for alleviating the requirement for highly skilled technicians, particularly in world wide application. As for economics, consider the following. Current estimates place manpower requirements at 4 per site counting both direct

operations and effective requirements for maintenance and logistic support. If station automation (including rate sensing and monitoring as well as control) could reduce the requirement by two, and the manpower replaced represented skilled technicial level with average salary levels around $7500.00/year, the resulting reduction in direct operating cost for a 10-year program would be of the order of $150,000 - $200,000. In fact, for remote site operation, the usual additional incentive pay could make it run higher. After a reasonable RDT&E program, a site should be easily automated for this amount or less, which is, after all, over and above the basic manned site cost. If these rather crude estimates are reasonable, it may even be desirable to automate all sites.

Another consideration, particularly in regard to remote site, is the possibility of mixed mode operation. This is the inclusion of scientific vehicles (e.g., D-region mission) in the storage system to be launched at pre-determined times or an opportune time (e.g., aurora occurences). In any event, any level of operation, from complete automation to single round control could be controlled completely from either a central or regional command post, with the potential for further reductions in direct operating manpower, hence costs.

Preliminary consideration was also given to another operational systems problem - that of the following mass hazard; i.e., spent motor cases, etc. This particular aspect of operating such a system has immediate impact on the overall probability of success. That is, for the system to be operationally effective, the capability of locating launch sites in a geographic pattern favorable to timely and meaningful atmospheric measurements is considered mandatory. Although the operations analysis necessary to confirm it has not been conducted yet, the assumption is made that the operational sites cannot be limited to coastal, remote, or regions undesirable for habitation. Under these circumstances, some provisions will have to be made to insure a safe impact of spent equipment. One approach is to make the motor case self-liquidating, i.e., either self-consuming or by explosive fragmentation into "non-lethal" pieces. It would seem, however, that to achieve a high level of capability and particularly a reasonable level of

reliability requires a design which would not be particularly suited to mass production and a relatively high RDT&E and unit cost. A second approach is essentially that now followed of providing sufficiently clear real estate for an impact zone; e.g., the ocean for ETR and WTR, and a desert for WSMR. However, it is very unlikely that such a limitation could be accepted if a truly effective system is to be developed. Additionally, it is doubtful that there are available sufficient "unused" land, particularly U.S. continental land, located in a satisfactory geographic distribution, which could satisfy the requirements for impact sites. This means that with this operating mode, some real estate purchases would be likely and, under present day circumstances, could represent a heavy program cost burden. An ROM idea of the potential magnitude of such an expense can be obtained by the following calculations. For this class of vehicle the dispersion of impact points of spent equipment (e.g., motor case) about a nominal (or unperturbed) point is approximately circular. This dispersion is caused by mechanical imperfections in manufacturing (e.g., misaligned fins), and/or variability of the flight environment (e.g., shifting winds). The need for the impact site is to provide a restricted access area into which the spent equipment can be impacted safely. Assume that an extreme dispersion situation requires an area having a radius R_{Req}. The area impact site would probably be a square circumscribing the dispersion circle, and thus cover an area of:

$$A = 2560\ R^2_{Req}$$

where R is in statute miles and A is in acres. The real estate cost for an impact site, as a function of real estate cost and the required dispersion radius (i.e., extreme requirement to safely impact a particular system, are shown in Figure 3.4.2.3. Now consider for example a cost level of $10.00/acre, which is probably a lower limit attainable in the U.S. today. A good boosted rocket system such as considered herein should be able to draw the system extreme dispersion radius down to the order of 10 to 15 miles (10 miles would, in fact, be excellent). Looking at Figure 3.4.2.3, the site real estate cost

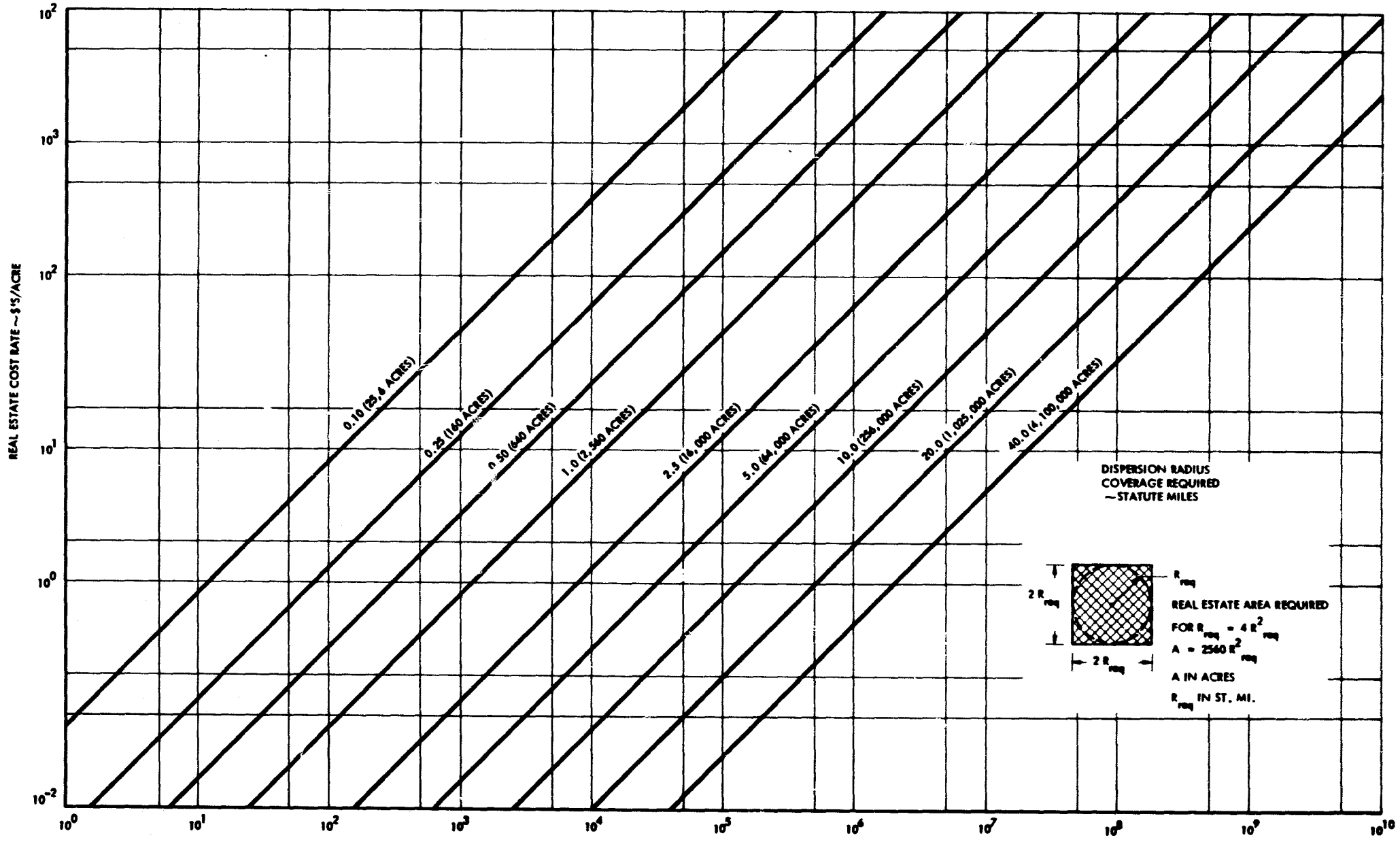


Figure 3.4.2.3. Real Estate Costs Versus Cost/Acre and Dispersion Radius

would be between $2,500,000 to $5,000,000/site. This does not include site engineering and maintenance. Even if all but a few sites could be located at available governmental land holding, those few requiring such real estate purchases would impose extremely heavy financial burdens on the program.

A particularly attractive alternative is to consider recovery of the complete carrier vehicle system. What makes this approach particularly attractive, is the indication that devices of the parawing concepts such as the Rogallo paraglider or the Jalbert parawing can be packaged very compactly (i.e., similar to a standard parachute), can develop good lift-drag ratios (i.e., ≈ 5) hence have good glide range, and can be controlled by relatively simple systems (e.g., Reference 3.4.2.1). Thus these devices hold the potential for greatly reducing the real estate required for impacting spent equipment, but also make the overall system more flexible and adaptable in operation. That is, a parawing, roughly controlled by a very crude control system, homing on a ground signal could substantially reduce (perhaps by an order of magnitude) the "effective" dispersion radius. Additionally there is the potential for at least partially offsetting the cost of including such a recovery package in that the payload, the recovery unit, and possibly other components could be reconditioned and reused. This certainly seems like a more attractive alternative to making a frangible or consumable motor. It could have other attractive benefits. For example, by greatly reducing wind sensitivity effects, a lower velocity trajectory could be used thus imposing the mildest of environment on the payload. Additionally, and for the same reasons, there is at least the potential for eliminating the need for setting the launcher to correct for winds, but to build permanent fixed installations.

The foregoing are some of the overall systems considerations, which have either direct or indirect bearings on a propulsion system design, which should receive considerable attention before proceeding to a final meteorological rocket system definition.

In the following paragraphs, consideration will be given to the propulsion system interactions with the other system components.

3.4.3 SYSTEM COMPONENTS

As noted in succeeding paragraphs, the propulsion system is simply a subsystem or component of an overall system or system group. Each of the component systems must perform its intended function in concert with the others in order that the complete system perform its function in an effective manner. The following paragraphs will deal, in a general way, with the envisioned meteorological rocket system components, as noted in paragraph 3.4.1.

3.4.3.1 ANCILLARY SYSTEMS

Ancillary systems is intended here to include those components which have only a general or implied relationship with the rocket propulsion system. For example, data acquisition, analysis, and dissemination system would not be directly affected by interactions with the propulsion system, except perhaps in the general sense. That is, the capability of producing a high speed, high capacity data and communications system allows contemplation of an effective synoptic meteorological rocket system.

Likewise, manufacturing and logistics systems, would not be expected to be greatly different from present military systems. The only item of special concern during logistic studies would be perhaps giving extra consideration to handling procedures, and emergency directions should the propulsion system be based on the VaPak or Hybrid concepts; i.e., safety engineering with regards to toxic N_2O_4 vapors or highly corrosive IRFNA.

3.4.3.2 PAYLOAD SYSTEM

There is not a great deal that can be said about the payload itself at this time, due to its lack of definition. That is, the overall systems study now being conducted elsewhere is to deal with the determination, in

detail, of the payload system to be used in any synoptic meteorological rocket system. It is to be expected that any new payload will be more extensive in capability and hence more complex in makeup. For example, the most complex systems now in use measure only wind vector and ambient temperature. It is anticipated that the advanced system will require the addition of such parameters as pressure and/or density, ozone concentration, water vapor, etc. Furthermore, the instrumentation will be required to operate at a higher altitude, hence a wider range of fluid dynamic and/or kinematic states.

Thus, even though the payload is not well defined right now, some commentary concerning interactions between the payload system and the propulsion system is possible. In fact, recall from the introductory and background remarks, the reasoning behind choosing a single stage, dual thrusting concept for this study. That is, to provide as yet undefined instruments and electronics with among other things a reasonable "soft" ride (i.e., relatively low axial acceleration level), and adequate payload volume. In other words, payload-propulsion interactions are at the very heart of all considerations included herein. This is also one area where several options are available, and should be considered in the complete system context, before motor development proceeds further.

The first option has to do with the payload operational mode. Present systems operate only after the expulsion of the payload from the carrier vehicle nose cone, with data being taken by the payload as it descends by parachute. The process often exceeds thirty (30) minutes, and occasionally the winds carry the package-chute combination beyond the range of the current ground acquisition systems. A most desirable mode would be to take data from launch to impact, with the payload system remaining a part of the carrier vehicle. This would result in a more timely method, and would eliminate the occasional loss of data. Furthermore, the payload system could be reduced in complexity in that

two of its components, a physical severance and expulsion device, and a parachute deployment device could be eliminated. If such a payload should become available, its operation could require a specified trajectory in terms of, for example, a specified altitude-velocity-time history on the ascent leg. As noted previously the designer indicated a great deal of flexibility in varying the operational mode for those motor designs contained herein. In fact, the motor capability, in terms of both flexibility and economy, may even allow or encourage consideration of the "data on the fly" concept.

Investigation of methods and instrumentation for accomplishing the tasks required with this type payload are currently under investigation by several companies; however, its development is not eminent. Thus utilizing this very attractive option may finally depend upon timing, mainly that of payload development, as well as the results of overall system requirements.

Another option that should be noted is somewhat related, but may be useful whether the payload is of conventional or advanced concept. That is, the flexibility indicated for the designs contained herein, could be used to keep the flight almost as mild as desirable, if used in conjunction with the complete, vehicle recovery technique discussed earlier in this section. This concept might allow consideration of more sensitive, less durable, and hopefully less expensive instruments in the payload. This is simply one more example of the possibilities, related to the propulsion system, that should be explored in an overall system cost-effectiveness analysis.

3.4.3.3 OPERATIONAL SITES

Operational sites obviously have a direct interface with the propulsion system. There is simply no propulsion system capable of the mission that is not potentially explosive; therefore, safety, to the highest degree, must be engineered into the system, particularly the operational sites. One of the

units designed for this study work require particular attention during design and construction of operational sites be they regional storage areas or launch sites. The VaPak unit propellants are toxic, particularly the N_2O_4. As noted in Section 3.2.3, the design of the tankage of this motor provides safety checks in redundant welds, high safety margins, etc. This coupled with clear, concise handling instructions and procedure should minimize dangers from accidently rupturing tankage. With regard to site location, however, the possibility of wholesale release of N_2O_4 must be acknowledged. Thus if a site is to be located near a populated area, sufficient separation distance, based on number of units on site, to allow dissipation of vapors in case of a catastrophic event. To a lesser extent this foregoing is also true of the hybrid unit which contains highly corrosive, and toxic nitric acid (IRFNA). The later point concerning measured separation of site from adjacent populated areas, must also be applied to the solid unit. In fact, adequate separation must be maintained for all types of propulsion units which, as noted above, are potential explosives. The above discussion should not raise undue alarm for there are now Nike sites around most major U.S. cities. It is simply intended to flag additional areas of propulsion system interfacing which should be examined with care during any overall system evaluation and cost-effectiveness analysis, and must be incorporated in a final system design.

Other major considerations related to site-propulsion interfacing were noted on a more general basis in paragraph 3.4.2. These considerations had to do with the potential for automating sites, and with complete vehicle recovery. The discussion will not be repeated here, rather the relationship is noted and the reader is referred to paragraph 3.4.2.

3.4.3.4 LAUNCH SYSTEM

The launching system interfaces with the propulsion in several very important ways. Launch system is herein intended to mean inclusively, that

equipment required to put the vehicle into the air, beginning with removal from storage to physical launcher release. This means ground handling equipment, checkout and fire control equipment, and the launch frame.

Little can be said with regard to ground handling equipment at the stage of design except that one of the instructions given to the motor designers during this study was to keep the design simple and avoid, if possible, designing in any feature that would require any special tools or handling equipment. The resulting designs are indeed about as simple as could be expected, with all screw and/or bolts being standard, general use commercial stock. The only special tool required is a rod required to insert the igniter in the head end of the solid motor designs. Of course, the basic requirement of keeping the propulsion system under 180 lb, (i.e., vehicle equal to or under 200 lb) is specifically aimed at minimized ground handling equipment requirements.

The second piece of ground handling equipment is quite critical to the system operation from several aspects. That is the electrical checkout and fire control equipment. This piece of equipment is absolutely critical to both safe and reliable operation, particularly if the system is to be operated by semi-skilled technicians. On this basis some detailed thought was given to the design of a checkout fire control box. The equipment to be discussed can be used by any of the designs, and, although the discussion implies manual operation, it is easily adaptable to remote operation.

The electrical firing system especially should be oriented for simple operation. Figure 3.4.3.1 shows how a simplified checkout and launch box might be ordered.

Instrumentation on this box is kept to a minimum (1 meter) while the basic circuit logic is designed to preclude accidental firing. The unit can be plugged into the rocket with the switches thrown without igniting it. This is

accomplished by safety circuitry which permits arming only after a proper sequence is followed. Additionally the equipment then disarms, automatically, 5 sec after the fire switch is thrown.

This, or a similar type of unit, would allow routine operations without undue concern about accidental ignition. As all operations in the checkout routine are performed at one point, it would not be necessary to have any personnel in the vicinity of the launcher during these operations. In the case of a postponement or reschedule, the unit can be left engaged and safety assured by removal of the key.

A simple light system is used to indicate the success or failure of each step in the checkout sequence. The meter is provided for quantitative evaluations of the battery check and continuity check functions; however, this is also keyed to the light system. If at any point the "reject" light appears, the rocket can be cycled out of the system, to a storage area, until the logistics/maintenance crew can make an evaluation of or correction to that particular unit.

The logic diagram is given in Figure 3.4.3.2. A brief description of the box function and operation is given below.

1. Plug rocket into umbilical. Firing power is not available even if Key Switch is in ARM and Fire Switch in ON because ARM Permit needs results of both Battery Test and Continuity Test before being ready.

2. Normal sequence would be as follows:

   a. Umbilical plugged in the rocket.

   b. External power applied to Firing Box.

   c. Key Switch turned ON and battery charge is supplemented.

   d. Key Switch turned to Battery Test and batteries given 5 sec load test at normal firing current. Relay Meter will register GO and a Proceed Signal will be sent to the ARM PERMIT.

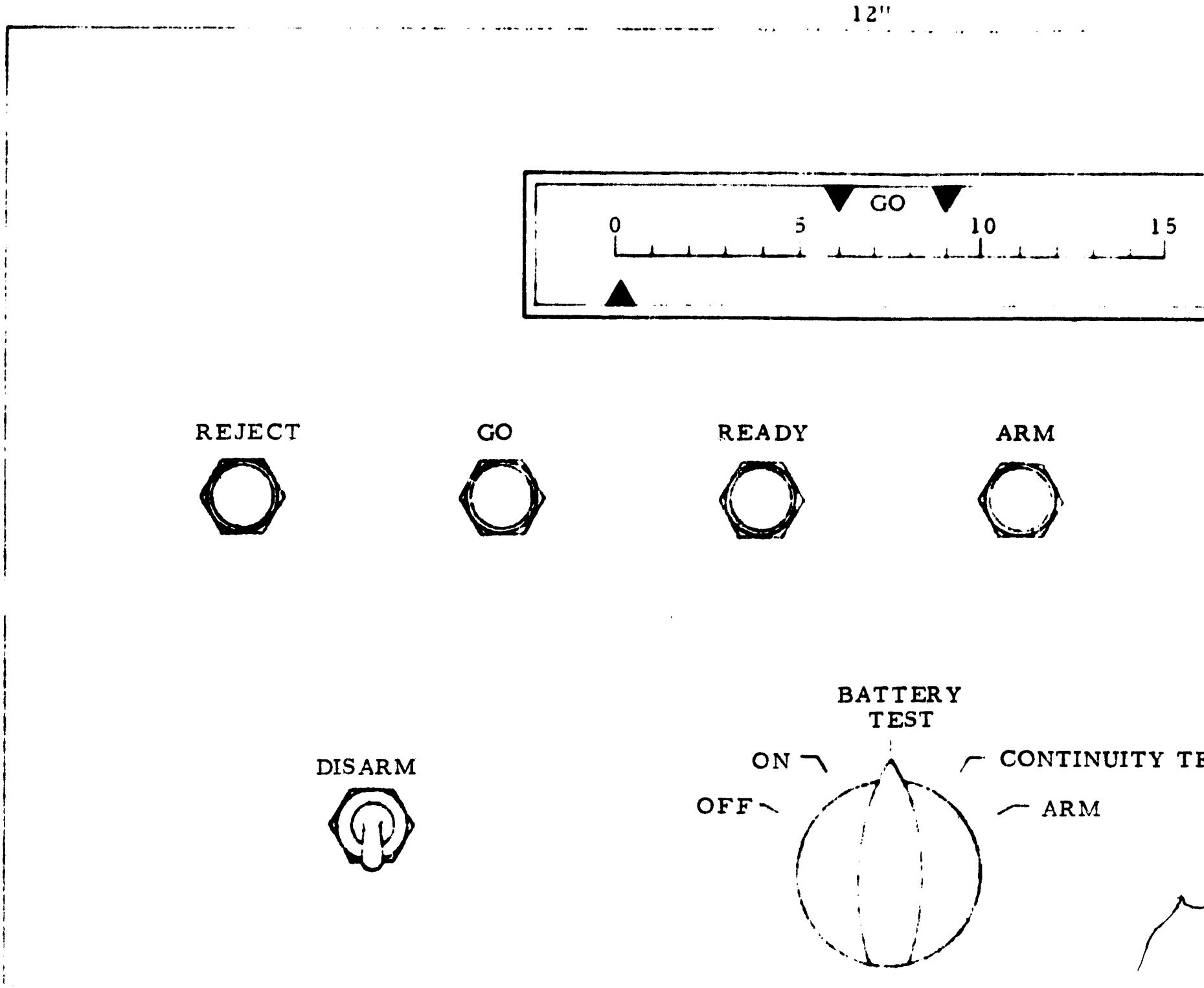


Figure 3.4.3.1. Simplified Electrical Checkout
Panel Conceptual Illustrati

FOLDOUT FRAME 1

15

DISARM

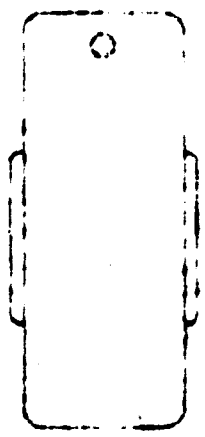


LIFT OFF



8"

FIRE



NUITY TEST

1

CL

. Checkout and Fire Control
Illustration

FOLDOUT FRAME 2

1

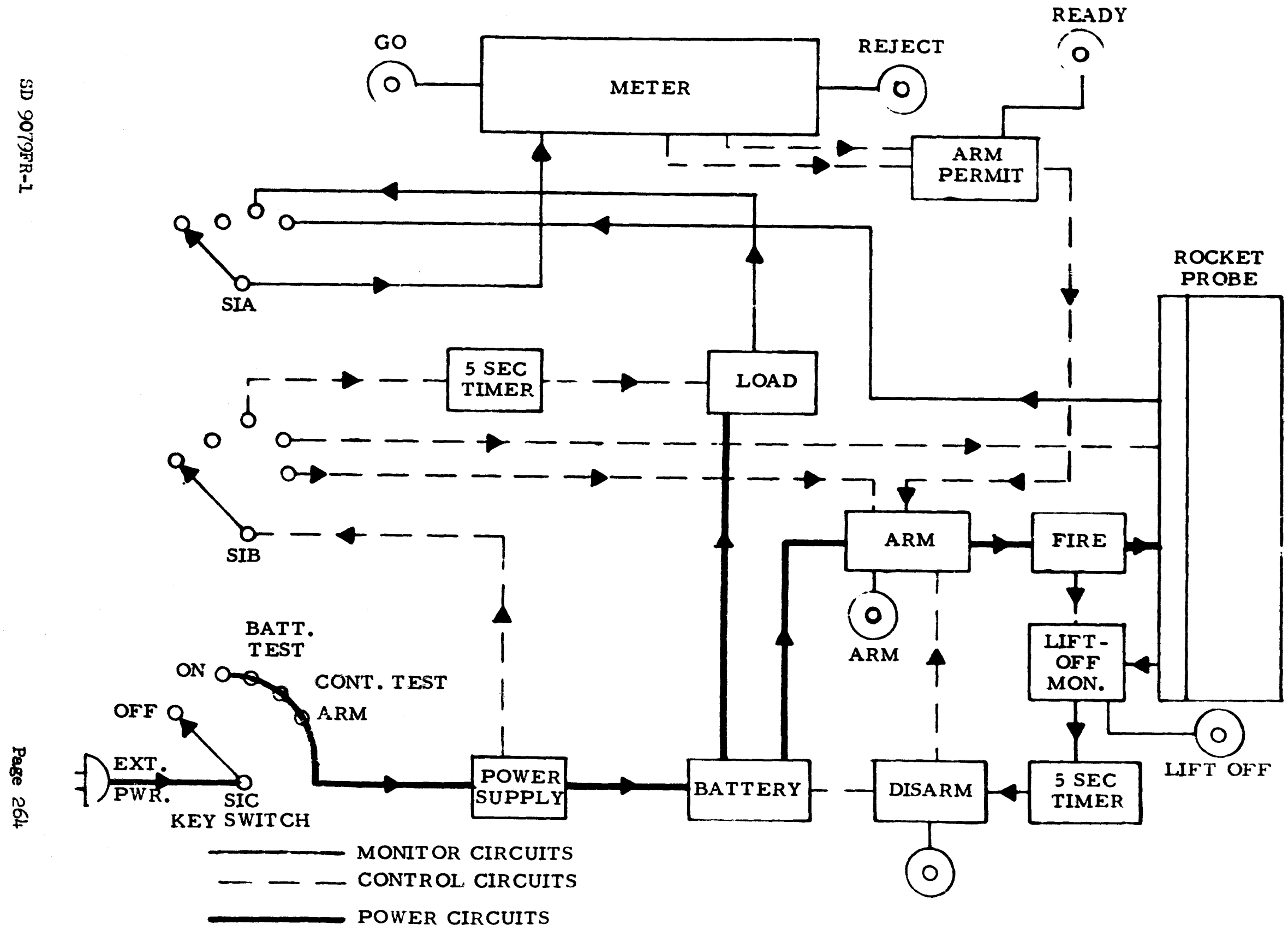


Figure 3.4.3.2. Fire Control Box Logix Diagram

e. Key Switch turned to Continuity Test and the Igniter Squibs are given a continuity test. The Relay Meter will register GO and the second Proceed Signal will be sent to the ARM PERMIT which will light the Ready Light.

f. The Key Switch is turned to ARM and providing the continuity check was still good immediately before, the ARM light comes ON and the GO and Ready lights go OFF.

g. The FIRE switch cover is lifted and the switch is turned ON which ignites the rocket motor.

h. The Lift Off Monitor is activated as well as a 5 second timer. Unless lift off is detected within these 5 seconds, the timer activates the DISARM which removes the FIRE signal.

i. The DISARM signal may be manually activated at any time and this will prevent the system from being armed.

This simplified check procedure could be performed (or remotely monitored) by a relatively low skilled operator.

The third member of this system is the launch frame. The purpose of this piece of equipment is to provide lateral restraints and guidance during the initial phase of flight. Beyond the end of the launch frame the vehicle must have sufficient inherent stability to hold perturbations to the planned flight path down to an acceptable level. The carrier vehicle envisioned for this system is stabilized aerodynamically by afterbody mounted fins. There are several points of interaction between the launcher and the propulsion system that should be noted here and fully accounted for during the overall system development program. The first is related to performance capability, and the second to the vehicle dispersion characteristics.

With regard to performance effects, the launch frame must guide the vehicle long enough for the velocity to build up to a level such that the so-called gravity rotational term in the equation of motion is minimized. The trajectory calculations made during this study assumed an effective launcher length (i.e., length over which the vehicle is restrained laterally) of 15 feet, which is a reasonable length for intended purpose, yet could also reasonably

be included in a portable system. A quick check indicated that for launch accelerations levels above about 10 g's, additional length had a diminish[illegible] effect on performance.

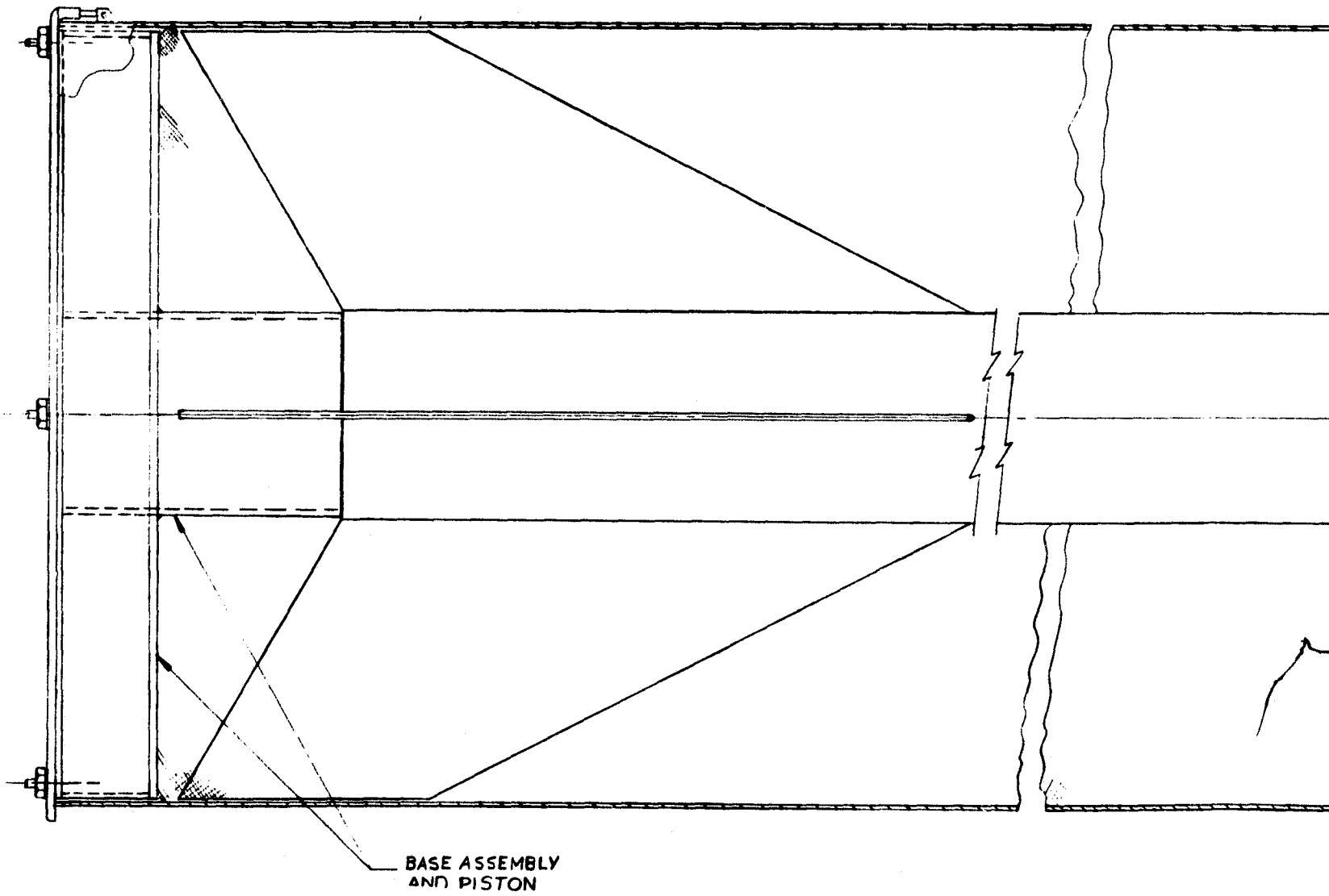
The dispersion problem is much more complex, depending upon an interaction between the launcher, the motor, and the inertial and aerodynamic characteristics of the particular vehicle being considered. It was, in fact, one of the reasons for including the dual thrust requirement in this study, and is a major consideration of including boosters in most sounding rocket systems. During this study, the variation in both boost and sustain thrust levels, and operating time were examined to optimize the delivery mode from the performance standpoint. During the development program a similar trade-off study should be made from the standpoint of dispersion minimization. Optimization trade-offs were not made during this study since the results are very much dependent on the characteristics of the individual vehicle, and the complexities involved are beyond the scope of this effort. Also an optimal thrust delivery mode (from the dispersion control standpoint) would not be grossly different from that arrived at from the performance standpoint, especially when considered within the framework of the study ground rules and practical limitations on a dual thrusting, single chamber motor. It should be mandatory, however, that the suggested relationship between the propulsion system and launch frame be fully investigated during the development program when the overall system characteristics are being firmed up, but prior to freezing the motor design. In this regard, the motor designs generated herein, are considered quite flexible and therefore very adaptable.

As for the physical aspects of the launch frame, several concepts are applicable to this particular class of vehicles and the mission envisioned. Recall from paragraph 3.4.2 the discussion related to automated site operation that a tube type launch frame was used in the example, however, a rail tube

could be used also. A simple conceptual sketch, typical of a tube launcher, is illustrated in Figure 3.4.3.3. Note in the drawing that, in addition to the sabot blocks, the vehicle base area is protected by a base plate and tube to vent the exhaust to the rear preventing recirculation into the fin area thus preventing structural damage to the fins as has sometime been experienced in the past. The desire to minimize the size of a tube launcher for its inclusion in a portable system, resulted in the concept illustrated in Figure 3.4.3.4. Here, the tube physical diameter is held down to only a little more than the motor diameter. The fin assembly is mounted on a carrier ring that has a short cone frustrum section. Either a similar section, or as illustrated, matching wedge section, are added to the motor aft end. At launch the vehicle passes through the ring assembly, with its base section picking up the ring and fins as the vehicle fully clears the launch tube. The concept was used for other reasons some years ago, by Aerojet, in a test program and is now, in fact, used in a foreign military system.

As for rail type launchers, standard techniques are certainly applicable. The top sketch in Figure 3.4.3.5 is a simple illustration of typical rail adaption by riding lugs. Note that the class of vehicle considered herein is somewhat drag sensitive; therefore, there is then good reason to reduce drag levels wherever possible. Note the streamlining of the lugs in the upper sketch (which are part of and remain with the vehicle), and the concept for removing the lugs completely as illustrated in the middle sketch. Still another possibility is to minimize the size (particularly the frontal area) of the attachment device. A possibility along the line is illustrated in the bottom sketch of Figure 3.4.3.5.

A configuration that may prove particularly useful for extreme climatic operations is illustrated in Figure 3.4.3.6. The vehicle may be protected from either cold winds or hot sun, if extended hold time is anticipated (e.g., awaiting aurora phenomena, etc.). The tubular section could be either the main launch



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FOLDOUT FRAME 1

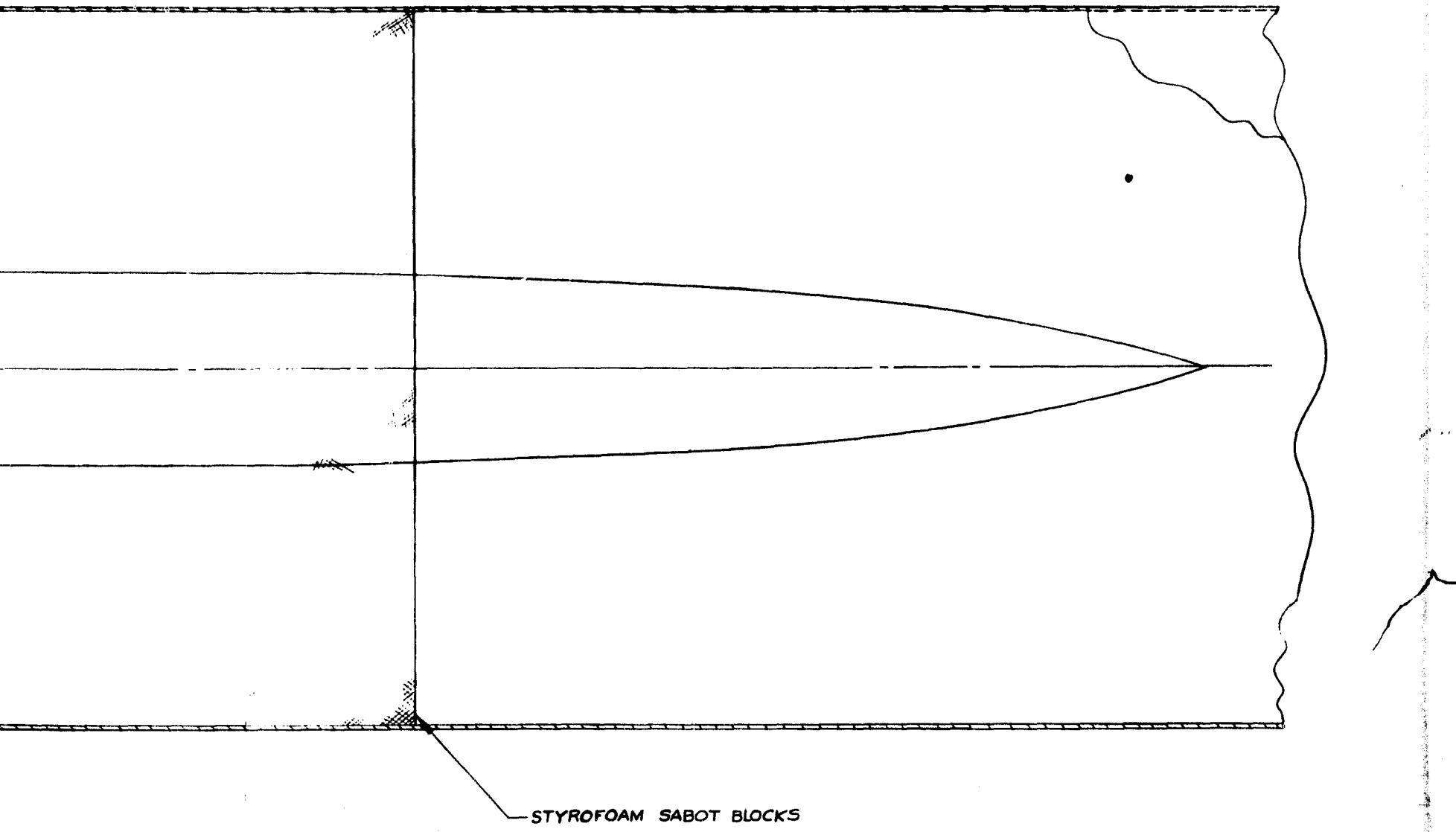
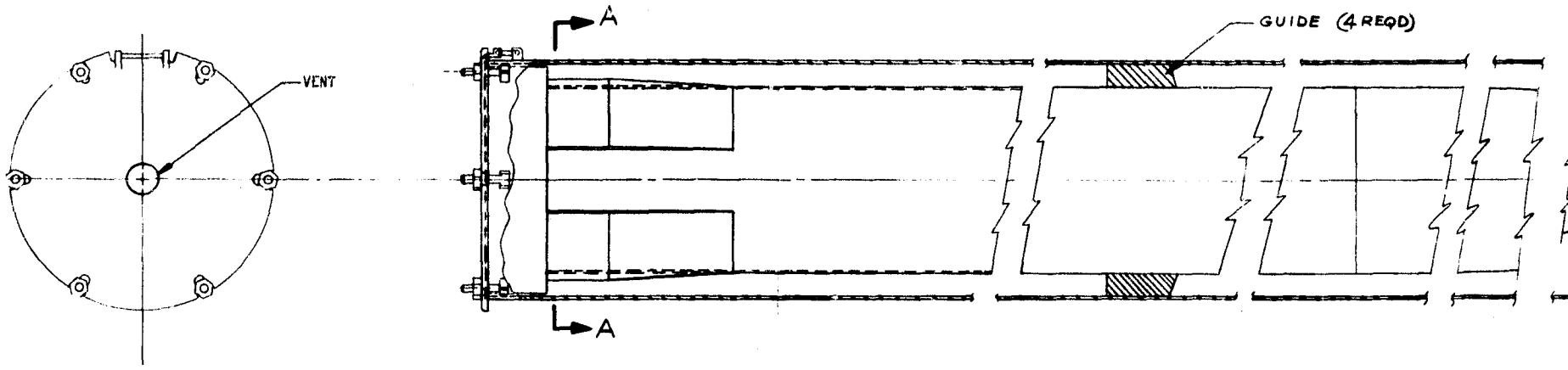
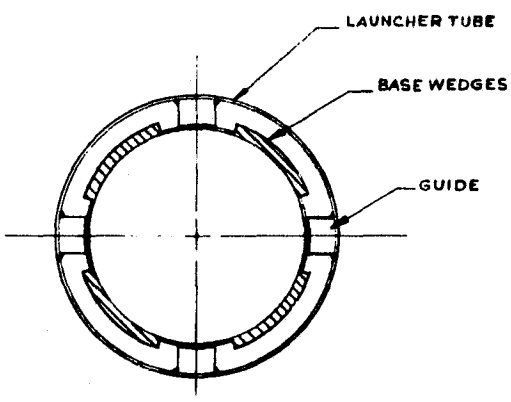


Figure 3.4.3.3. General Tube Launcher Illustration

FOLDOUT FRAME 2



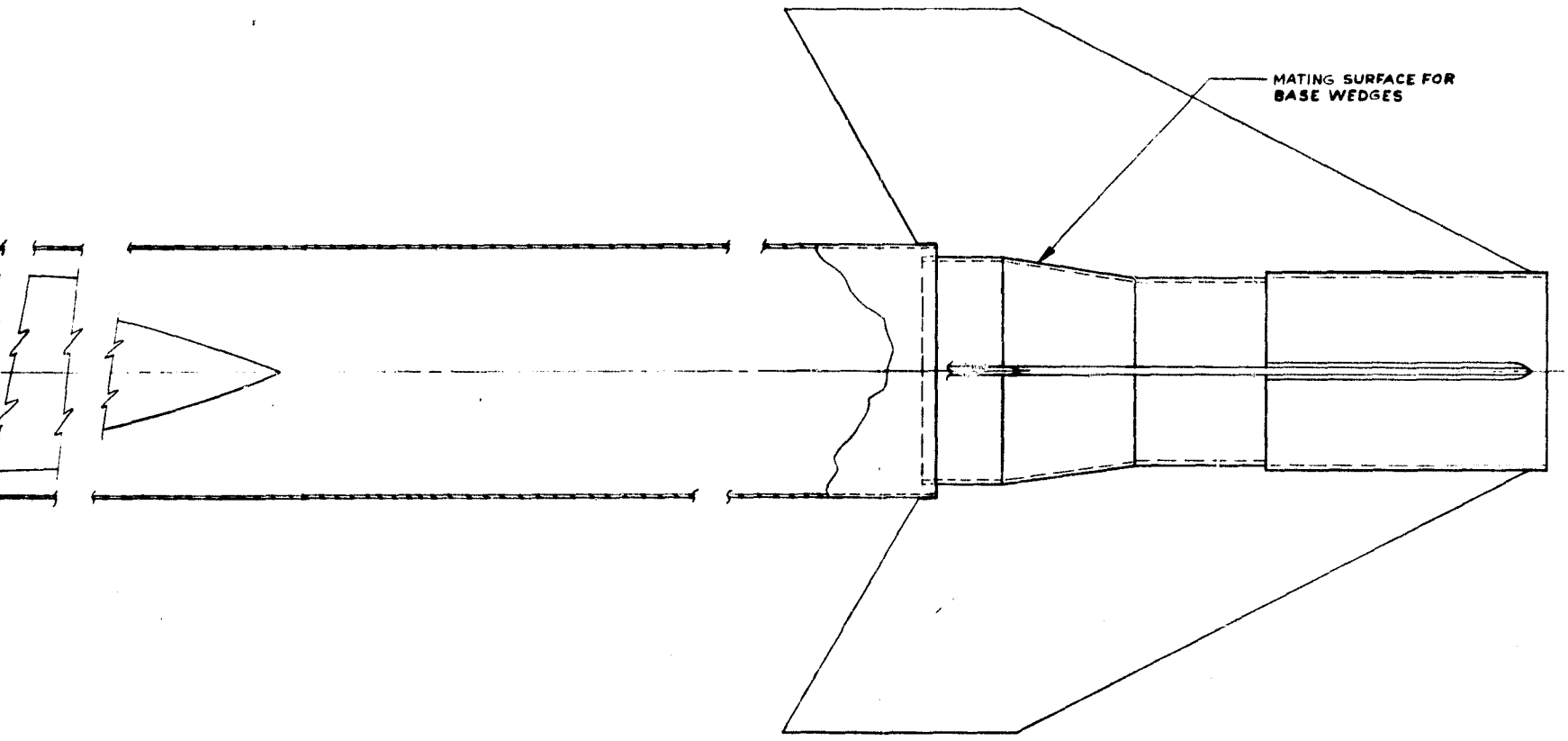


SECTION A-A

MINIMUM DIAMETER TUBE LAUNCHER

Figure 3.4.

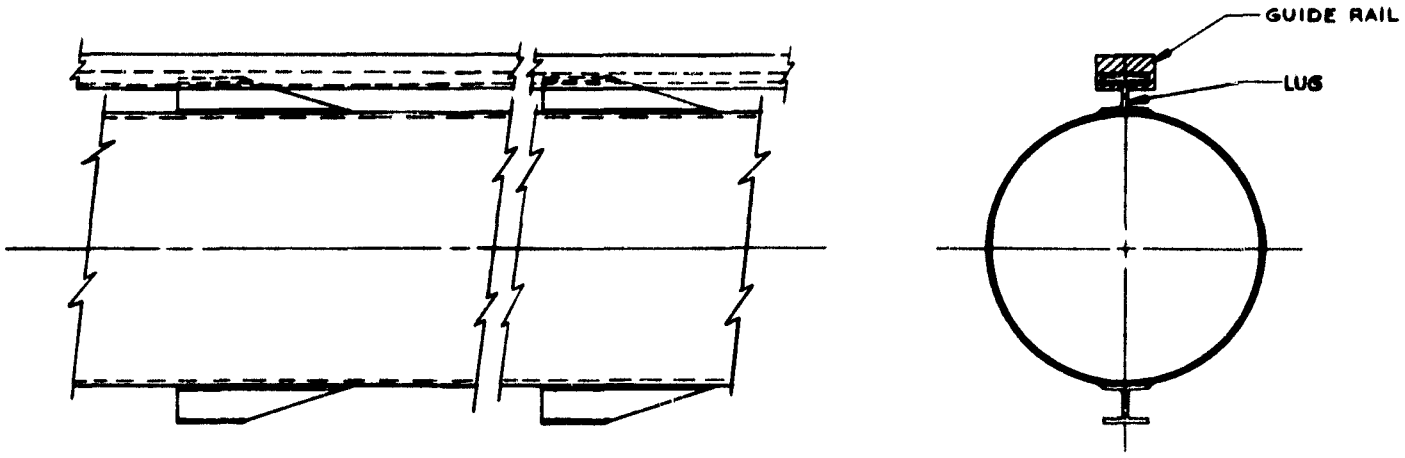
FOLDOUT FRAME 1



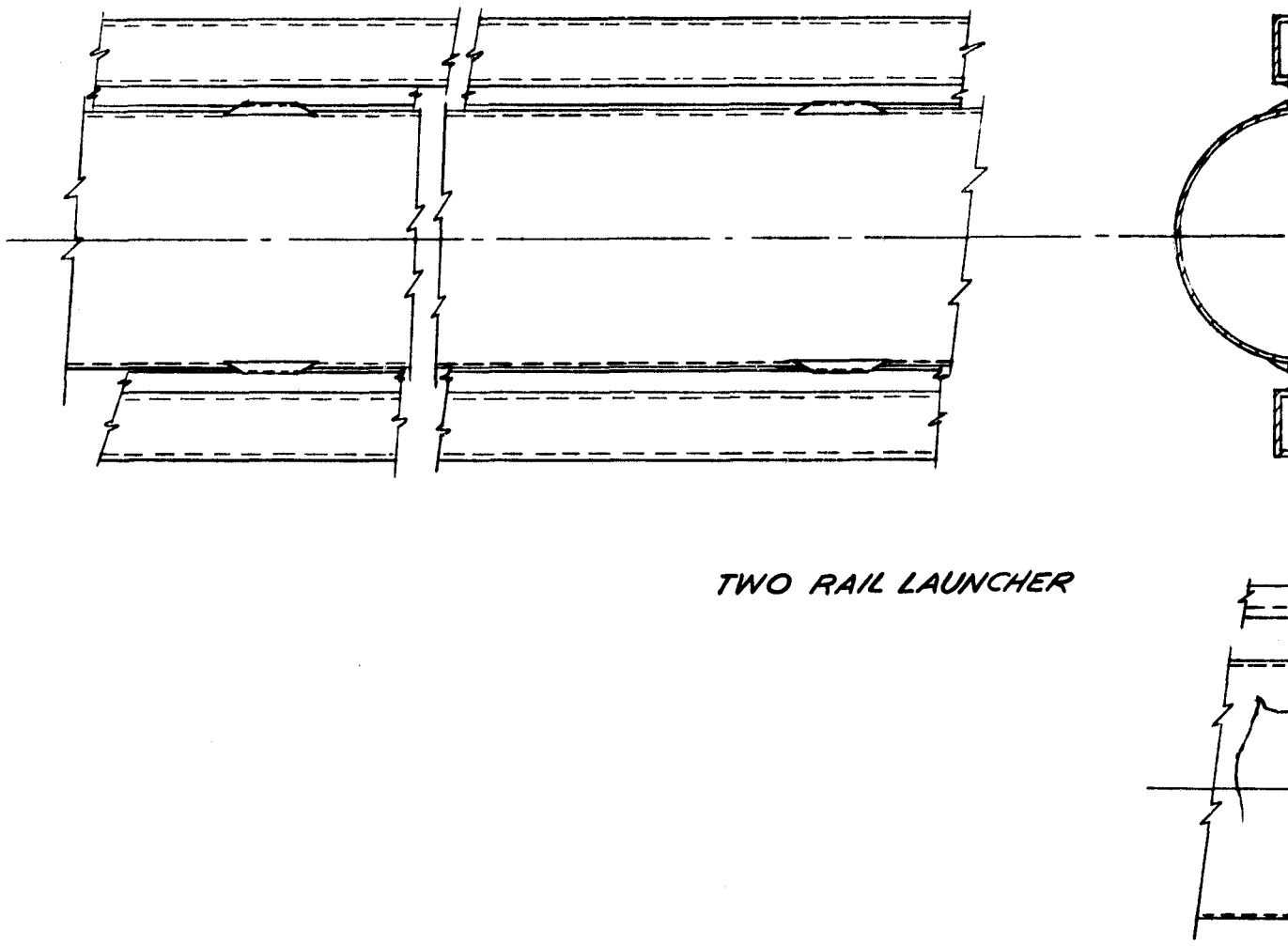
ure 3.4.3.4. Compact Tube Launcher Illustration

FOLDOUT FRAME 2

1



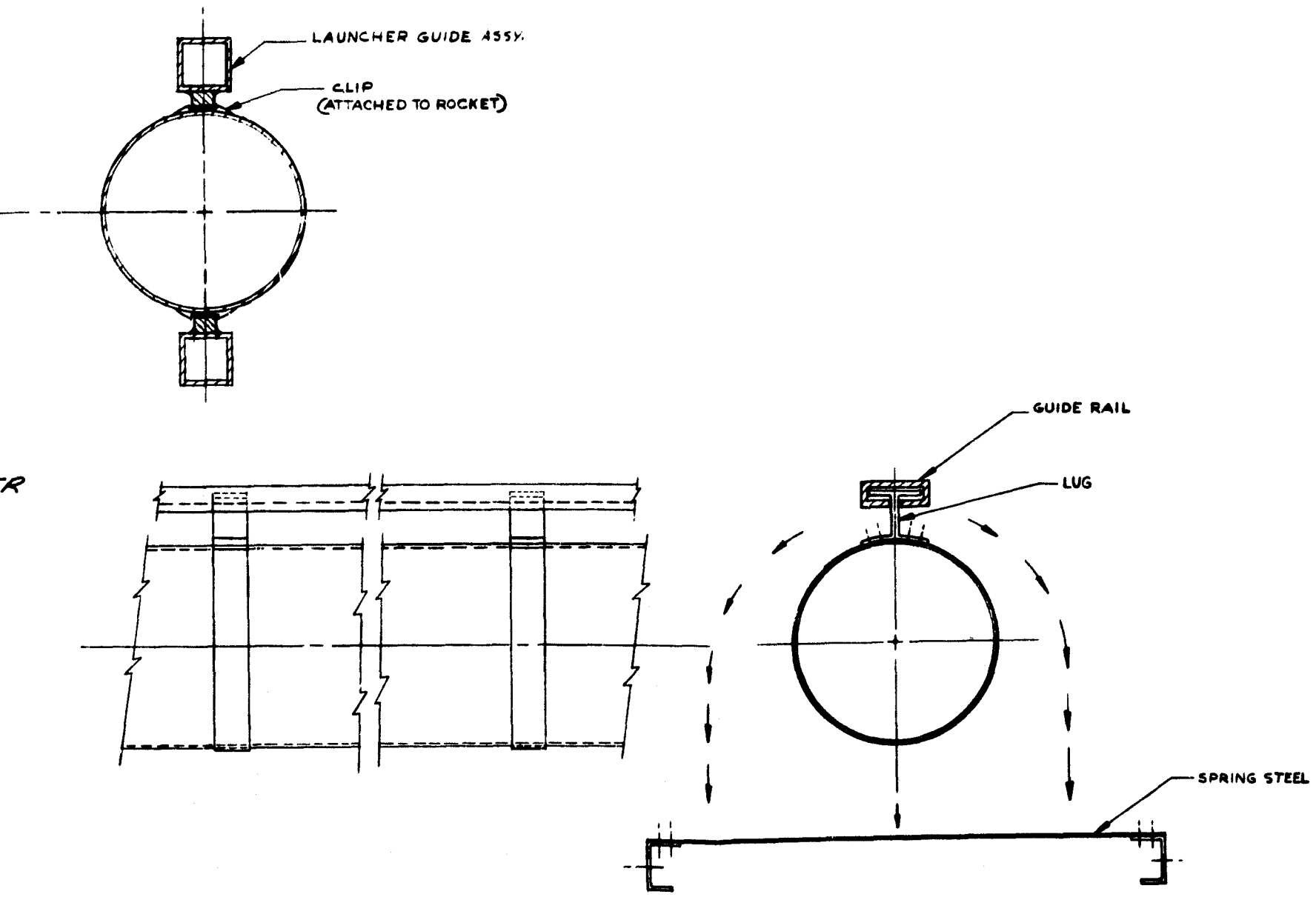
STREAMLINED LUGS
RAIL LAUNCHER



TWO RAIL LAUNCHER

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FOLDOUT FRAME 1



SEPERABLE LUGS
RAIL LAUNCHER

Figure 3.4.3.5. Rail Launcher Adaptors

FOLDOUT FRAME 2

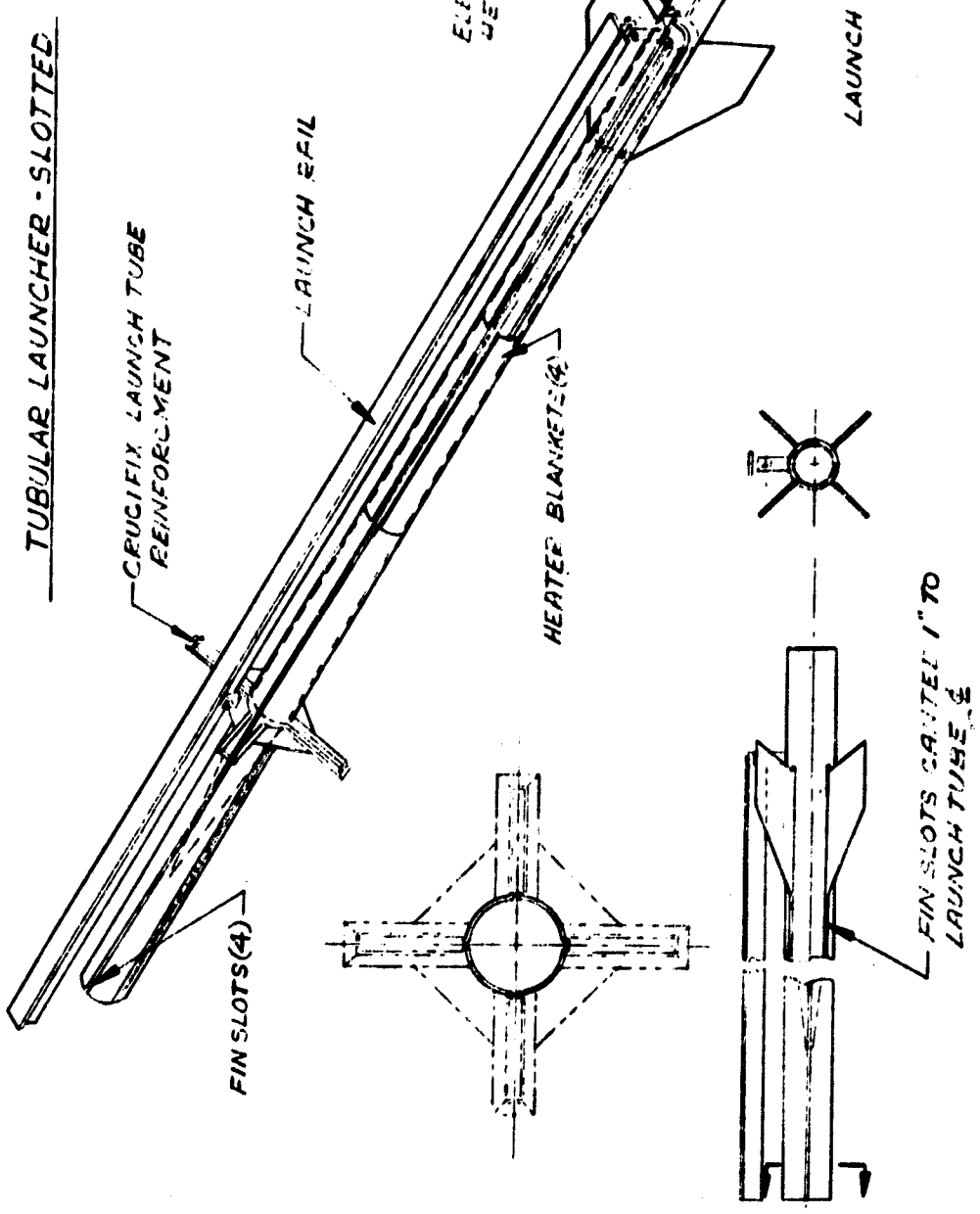


Figure 3.4.3.6. Weather Shielded Launch System

frame or simply a weather shield with the rail being the launch member. This concept is particularly adaptable to the VaPak motor concept, which is somewhat environmental sensitive in extreme cold when winds are present.

These are but a few illustrations of potential launch mechanisms. Actual selection and design of the launch frame, like all the other system components, must be made in the context of the overall system requirements, and cost projections.

3.4.3.5 CARRIER VEHICLE SYSTEM

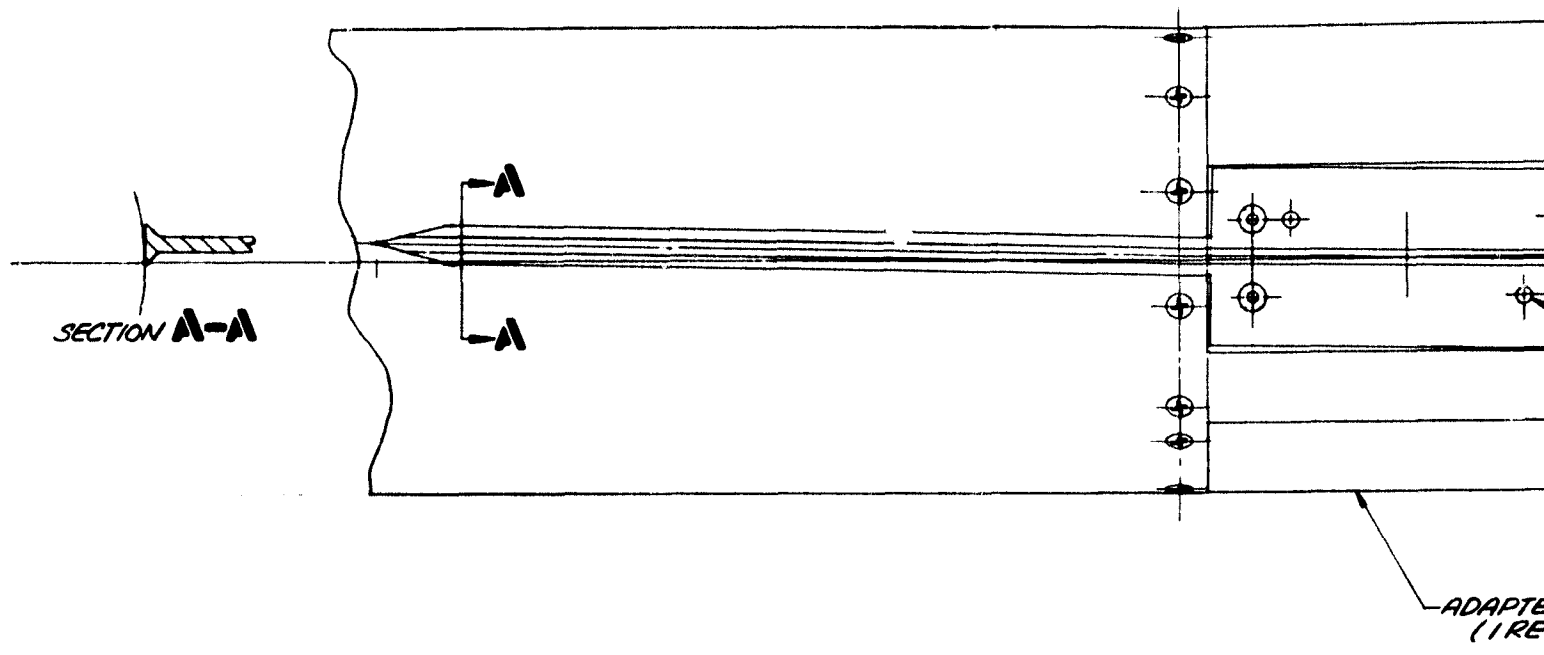
The whole purpose of the carrier is to place the payload along the desired trajectory through the atmosphere on a timely and reliable basis. The vehicle system is meant here to include the nose section, the motor, and the nozzle shroud-afterbody fin combination.

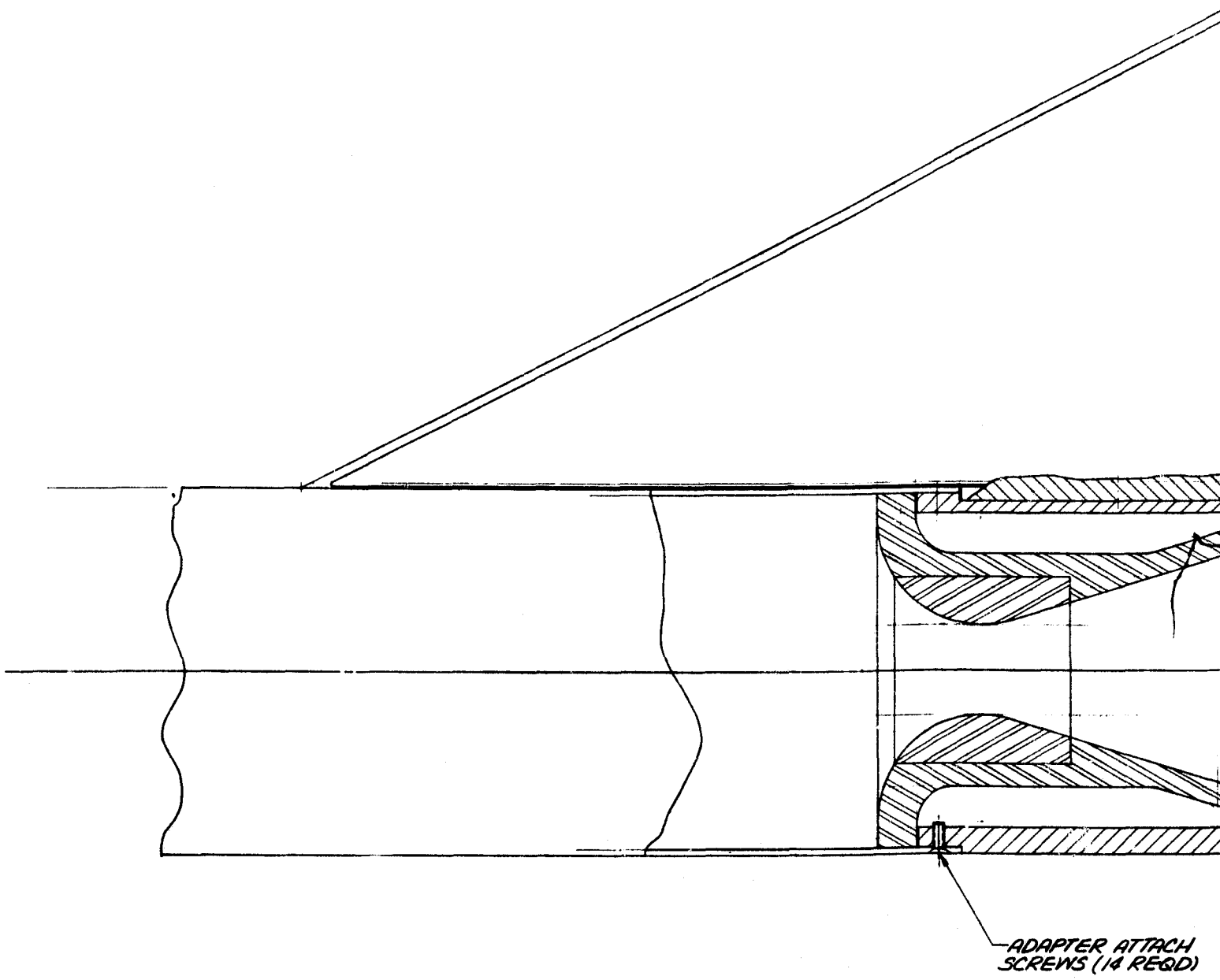
The major item is, of course, the motor, which is amply discussed and described in foregoing sections. A complete and final design was not the objective of this study, however, certain checks were required to insure that the motor designs would be realistically capable of performing its assignment. In this regard two checks were made. First, since it was impossible to determine in detail the flight dynamic characteristics of potential vehicles based on each motor design, a conservative estimate of the structural loading the motor case would be expected to encounter. Based on past experience a bending moment of 50,000 in-lb (in addition to the motor internal pressure loading) was imposed (i.e., given the designers as a requirement), at the aft end of the case. Secondly, a very cursory examination of aeroheating effects was made. The AGC solid design discusses its heat environment effect in detail in Section 3.2.2. Although not specifically covered by the other designs, these checks were made (particularly for the hybrid unit since the pressurization system represented an additional source of heat for the forward tank), and the designs found to be

adequate. This does not mean that structural design and assessment will not be required for a final design. Indeed, that will be a major task in any final development program.

The nose cone received scant attention during this investigation as far as details concerning structure or heat. Of course, the shape was specifically selected because of its low drag, good internal volume, and easy fabrication characteristics. As far as structural and thermal adequacy are concerned, there are ample methods and materials which can be used in relatively simple (therefore cheap) fabrication techniques.

The remaining item is the nozzle shroud-afterbody fin combination. Here again, although detailed design was not the objective or justified at this stage, some extra consideration was given to this component since it plays a relatively large part both in the vehicle flight success and in the unit cost. The performance trajectories (see Appendix B) were used to determine the Mach number-time history for the various designs, and the inertial data for each motor design was used to estimate the center of gravity-time history for each of the resultant vehicles. Using these estimates and arbitrarily requiring a minimum aerodynamic static stability margin equal to 10% of the body length (i.e., the distances between the center of gravity and the effective aerodynamic center of pressure), an estimate was made of the size of the stabilizing fins required. This was, in turn, used to make a preliminary design of the nozzle shroud-fin set. Here a rather unique approach, particularly applicable to low cost, high production objects was used. That is, using all common commercially available extrusions with a minimum of fabrication required. A conceptual design is illustrated in Figure 3.4.3.7. The nozzle shroud is simply a tubular extrusion similar to the motor case. Each fin is an extended T-section which is worked to the desired shape by a minimum of cuts and/or planning operations on flat surfaces (all can be accomplished by guides and jigs thus has the potential for automation). It is anticipated that the vehicle



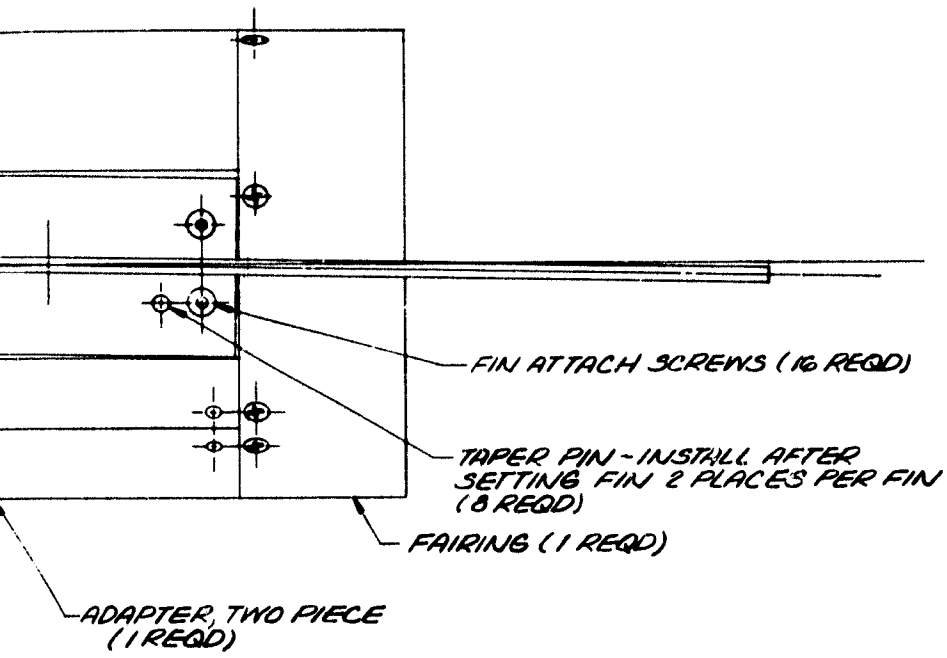


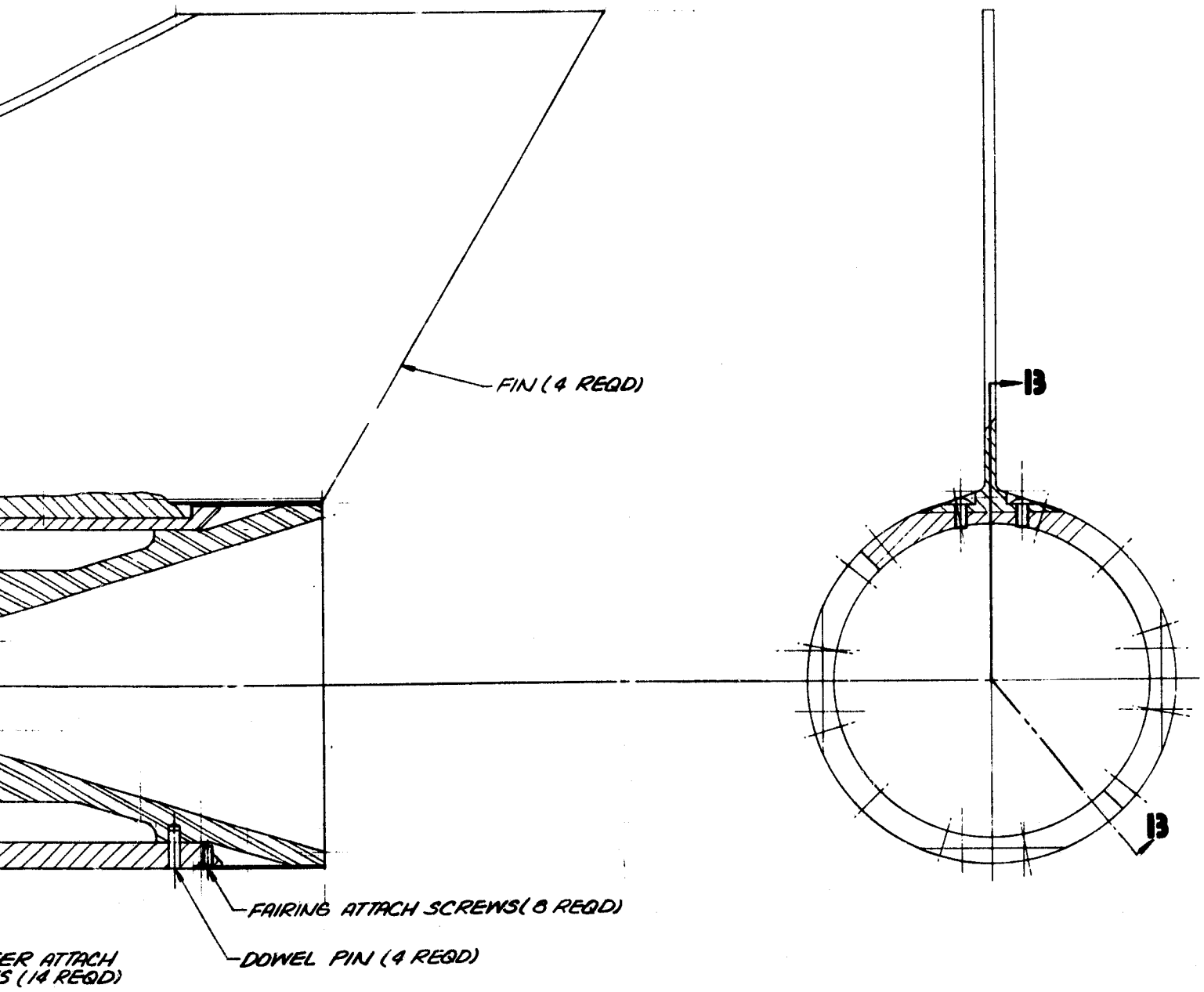
SECTION B-B

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Figure 3.4.3.7.

FOLDOUT FRAME 1





3.4.3.7. Small Probe Fin Conceptual Illustration

FOLDOUT FRAME 2

will be rolled during flight to minimize trajectory dispersion due to body fixed torques by differential cant on the fins (i.e., similar to an aircraft aileron roll; a standard procedure for sounding vehicles). The fins are attached to the shroud by screws, at the cant angle required to produce the desired roll rate. The shroud then slips over the nozzle where its forward end of the shroud is connected to the motor aft end skirt by screws.

4.0 PROJECTED DEVELOPMENT PROGRAM

4.1 TASK DEFINITION AND SCHEDULE

As requested, each of the Phase II designs have, in varying degrees, indicated their intended development programs as given in Section 3.2. What is presented here is a unified program based on the development of a flight vehicle system, not just the propulsion unit. This is absolutely necessary as far as the carrier vehicle is concerned, since the motor is not only the propulsive source, but an integral structural member of the carrier vehicle. Note that the scheduling indicated by the individual designers is somewhat different from the other, and may also be somewhat different from that projected herein; however, there is sufficient flexibility in all cases to allow fitting any of the designs into the integrated development program.

The development program outlined herein is related to the synoptic meteorology mission. The resulting vehicle would, with little or no effort, also be capable of performing the D-region physics experiment mission.

The development program is, for all practical purposes, summarized in Figure 4.1.1. Note that there are essentially three phases to this projected program, separated from each other by program reviews. Note also, this program is concerned only with the vehicle system. That is, development of the payload subsystem, and its separation, descent and recovery subsystems are assumed to be a separate and distinct program. This does not mean that this program is not cognizant of the payload development program. In fact, the two should be coordinated; however, the vehicle development program outlined herein is general in nature and therefore should be amenable to schedule/objective coordination.

The first two phases indicated essentially make up the RDT and E phase, while the third phase is essentially the production/operational phase. The first, or preliminary design phase, essentially confirms the work contained herein, and reconciles differences between current designs and concepts with a

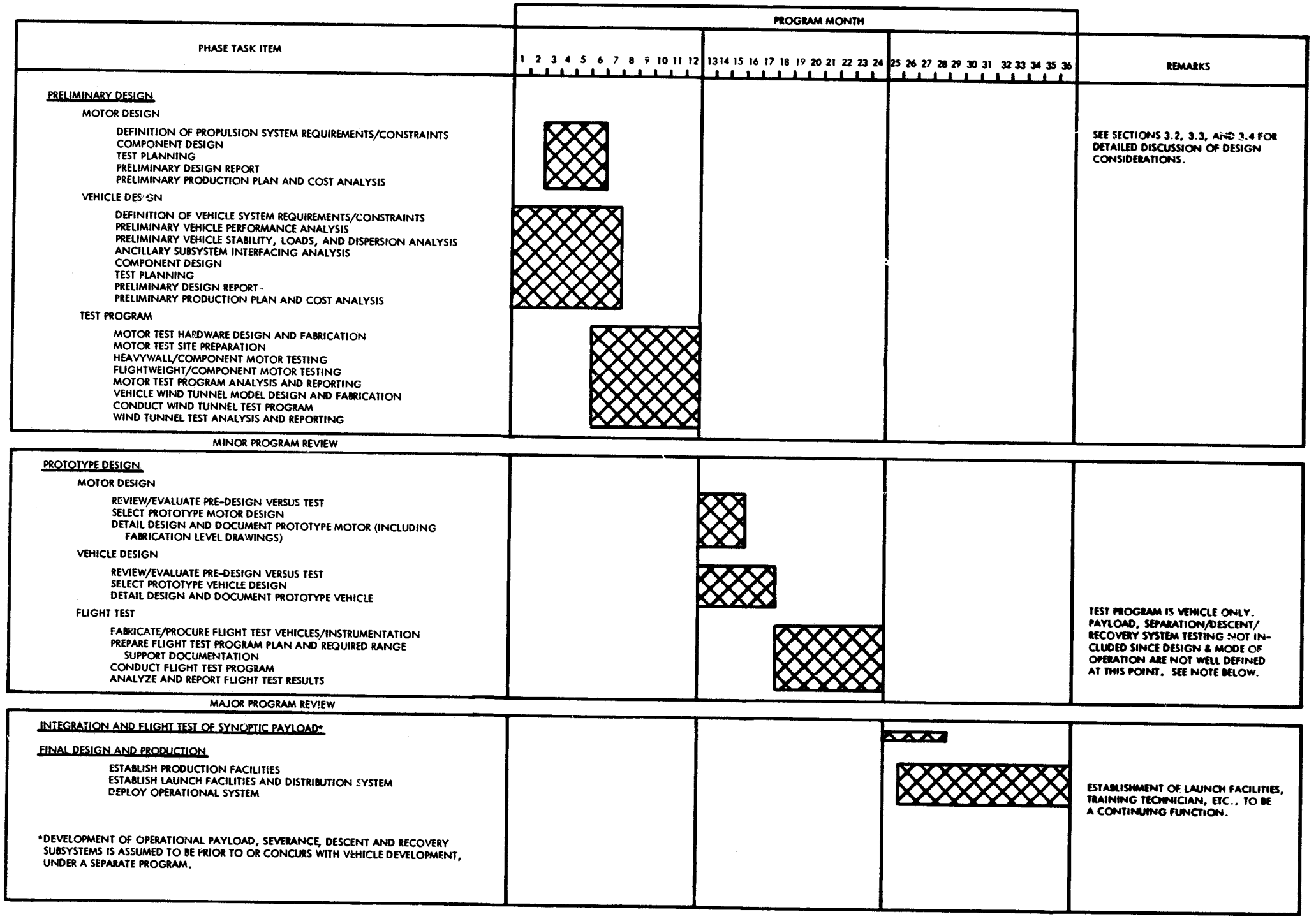


Figure 4.1.1. Development Task and Schedule Diagram

more precise definition of the overall system configuration (which is assumed to have taken place under a separate, but related program) and its specific requirements. It then moves to a testing program to provide the necessary experimental data for confirmation or revision of the preliminary motor and vehicle design concepts. At the end of this first phase a program review would be held to ascertain how well the preliminary objectives and design concept were meeting overall system requirements as indicated by the experimental results. The output of review would likely be a decision to either: 1) commit to the second phase, or; 2) redirect the program along other lines which may appear effective after the program review.

The second phase task is essentially that of proving the indicated potential by prototype fabrication and operation. The initial Phase II effort is to review the results of the Phase I work, and the design review findings and recommendation, and to evolve from these findings a prototype design. This includes the motor, the flight vehicle system, and the ground handling system, including electrical checkout/support and launching equipment. The output of the Phase II motor and vehicle design tasks are a complete vehicle system documentation and prototype fabrication level drawings (including at least prototype ground handling equipment). The final task, and really the objective, of this phase is the fabrication and flight testing of the prototype vehicles. At the end of this Phase II effort the major program review occurs. First item of review is, of course, did the Phase II effort adequately demonstrate the intended system capability? Secondly, are cost projections still attractive after prototype fabrication experience? Third, are all other factors (political and economical for world wide systems use) favorable to system objectives? After weighing these and related questions, the discussion facing this program review is to commit to operational deployment (either in U.S. territory or world wide) or redirect.

Assuming the decision is to deploy an operational system the program would move into the third phase. Recall from the discussion above that it is assumed that the payload and its ancillary systems would be developed under

another program. The first item of the next phase would be a short demonstration program, uniting the operational payload into the flight vehicle system. The next step would be the establishment of the fabrication facilities, development of the logistics system, establishment of operational launch sites, all leading to operational deployment of the system.

Note that the overall program, as outlined in Figure 4.1.1, is 3-years, with each phase lasting about one year. Actually, the last phase would be a continuing function, but would require a much higher level of activity during its first stages. Now, as pointed out above, this schedule is of a general nature, and could be either shortened or lengthened depending upon budget, extent of deployment planned, or adjustments to pacing item development schedule (i.e., payload, motor, etc.).

4.2 ESTIMATED COSTS

Costing such a program as envisioned herein depends very much on overall system considerations; however, ROM estimates were made for each phase of the proposed vehicle system development program outlined above simply to illustrate the magnitude of the undertaking.

Preliminary Design (Phase I)	-	250,000
Prototype Design (Phase II)	-	450,000
Total RDT&E Program	-	700,000

This includes $400,000 for the motor proper (see paragraph 3.3.3), therefore, the balance of 300,000 represents RDT&E costs for the complete vehicle system (including structural and wind tunnel testing), as well as launch system (including ground station components directly related to vehicle system, but excluding the payload and its ancillary subsystems and equipment, and the data acquisition/handling system). This assumes then that there are three (3) separate but related activities either under way or completed under separate programs. The first is that which defines the final system configuration,

operational mode and deployment. The second is that related to the development of the meteorological payload and its ancillary subsystems and/or equipment. The third is related to the data acquisition and handling systems. While it is fully realized that all of the above activities must be coordinated, the only activities considered in the program outlined in this section are those directly related to the flight vehicle.

It should be noted that the ground station work included in this program envisions a conventional manned station. As noted in paragraph 3.4.3, some consideration should be given to automated stations particularly for remote site operations. An ROM estimate of $1,500,000 additional RDT&E funds would be required to develop a completely automated station.

As far as the production/operational costs are concerned, cost estimates are included for the carrier vehicle only. Other program costs such as payload, data acquisition, site or station, etc., depend too heavily upon the overall system definition to yield to meaningful analysis at this time (see particularly paragraph 3.4.3). The estimate is based on an effective program duration of 10 years. The vehicle cost includes propulsion system (see paragraph 3.3.3), nozzle shroud and fin assembly, and nose cone.

Yearly Usage Rate	← UNIT COST →			Total Flight Cost for Program ③
	Propulsion ①	Ancillary Hardware ②	Flight Vehicle	
100	2000	275	2275	2,975,000
1,000	1350	200	1550	16,200,000
5,000	900	150	1050	53,200,000
10,000	700	135	835	84,200,000

① A conservative assumption based on $2000/unit @ 100 unit/year level and @ 90% learning curve. Actually selected to conservatively approximate motor data of Section 3.3.

② ROM estimate based on $150 @ 5000 unit/year level and 10% learning curve.

③ 10-year total, includes both flight vehicles cost based on unit cost shown plus the estimated RDT&E cost of $700,000.

5.0 REFERENCES

Section 3.0

3.0.1 "Application of Advanced Solid and Hybrid Motors to Sounding Rockets"; SGC 929FR-1; Space-General Corp., El Monte, California; July, 1966.

3.0.2 "Design Criterion for Spacecraft Propulsion Systems"; SGC 1069FR-1; C.N. Jennings, Aerojet-General Corp.- Space-General Plant; El Monte, California; Oct. 1967.

Section 3.1.1

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Section 3.1.2

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APPENDIX A

FLUID-CONTROLLED SOLID PROPULSION UNIT
DESIGN AND COST DATA

SD 9079FR-1

1.0 INTRODUCTION

Lockheed Propulsion Company (LPC) is pleased to submit information regarding a low-cost controllable motor for NASA sounding rocket applications. This includes ballistics and design characteristics, and unit production costs. Particular effort was expended to obtain reasonably accurate production costs.

The proposed controllable motor concept has evolved from nearly ten years' of hybrid and controllable solid motor R&D experience at LPC. Conventional hybrid (liquid oxidizer/solid fuel) programs have proceeded on a continuous basis at LPC since 1959 (Refs. 1-13). Research programs on the fluid-controlled solid, or RSVP motor (any liliquid/solid propellant) began in 1961 (Ref. 14) and have proceeded at a greater level of effort since 1964 (Refs. 15-18). RSVP design studies (Refs. 19-21) and exploratory R&D testing (Ref. 22) have been performed under NASA sponsorship since early 1966. The RSVP is considered to be a more general form of hybrid motor, the liquid oxidizer/solid fuel combination being a special case. The particular choice depends upon the mission requirements, and a judgement as to which combination of propellants will best meet these requirements, and not by semantics. For the subject application, low cost is a critical factor.

The design approach used in this proposed motor concept is strictly state-of-the-art. Only well proven principles were considered. The development effort required is that of (1) developing and proof testing the flight weight components and system, and (2) a complete characterization of the solid propellant. A complete motor design and cost analysis was made for the proposed motor. These data, complete with several potential vendor quotes on such components as pressure regulators, explosive valves, timing devices, etc., plus cost analysis of hardware fabrication, propellant loading and assembly, are available as back-up information as needed. Only the end results of this effort are supplied. The back-up information can be provided to the NASA upon NASA project personnel request.

It is cautioned that comparison of this proposed controllable motor with alternate systems should take into consideration technological similarities, any use of cost-saving technological or fabrication methods proposed by other systems which are not unique to them, and the significance of non-unique advanced concepts proposed.

2.0 CHOICE OF PROPULSION SYSTEM TYPE

LPC conducted exploratory testing of a low cost propellant combination in 1967 (Ref. 23). This was based upon the utilization of N_2O_4, an inexpensive high energy liquid oxidizer, as the control fluids. The solid propellant consisted of fertilizer-grade ammonium nitrate and butyl rubber (tire buffings)

mixed with a conventional binder, such that overall propellant raw material costs were approximately 20 cents per pound. For increased performance, a low-cost grade of aluminum powder could be added to the solid propellant at a small overall cost increase. The binder is by far the most costly ingredient, so that any conventional hybrid fuel would be far more expensive. The particular solid propellants studied possess ballistics properties very close to what is called for in the subject mission, thereby minimizing future propellant development work.

It has also been established (Ref. 24), as seems intuitively obvious, that propulsion hardware costs can be minimized by minimizing the size of the liquid component system in a liquid/solid propulsion system. Because the solid propellant contains appreciable ammonium nitrate solid oxidizer, comparatively little N_2O_4 need be injected to achieve a desired performance level for this application. In fact, the average mixture ratio is on the fuel-rich side to reduce temperature and minimize nozzle erosion.

It was found from the testing program that N_2O_4 is not hypergolic with solid propellant or hybrid fuel; the surface just blistered. For an application not requiring multiple stop-restart, this was successfully overcome by incorporating a small vial of UDMH in the N_2O_4 injector design. Requested throttling is accomplished by a dual injector to provide two definite liquid flow rates.

An alternate approach to throttling could be by solid propellant design; i.e. use two propellants of different burning rates, or an intricate variable surface area grain design, at a constant liquid flow rate. However, a preliminary evaluation indicated that the added processing and insulation costs which would be fixed would outweigh the cost of a dual injector versus a constant flow injector in large quantities. Indeed, these processing costs would exist for a dual-mode conventional solid (no liquid) in performing this mission. Moreover, the solid propellant that would be necessary to achieve requisite performance, without liquid injection, would have to be based upon ammonium perchlorate which would greatly increase

the raw material costs. Thus, the added raw material and processing costs could well outweigh the benefits of removing the liquid components.

In conclusion, the proposed liquid/solid RSVP motor should best meet the low-cost requirement of this dual-thrust application.

3.0 BALLISTICS AND DESIGN CHARACTERISTICS

A summary of the proposed motor characteristics is presented in Table 1. An outline drawing is shown in Fig. 1. Additional detail drawings were made for cost estimates (not included).

The solid propellant grain design is a partial slotted-tube to provide good neutrality and 80% solid volumetric loading. Good volumetric loading is required in order to best remain within the maximum allowable length, and the slotted tube best meets this requirement in high L/D chambers. Other internal-burning grain designs would either reduce volumetric loading, increase processing costs or detract from neutrality. An end-burner would require burning rates somewhat higher than current state-of-the-art for solid propellants. A slotted-tube has also been proposed for the LPC 2.75-inch rocket, a quantity production item of similar L/D. No mixing aids are required for good combustion efficiency.

The nozzle is a conventional conical nozzle. Contouring or submerging to conserve length is not proposed because of the added cost involved. The expansion ratio was optimized at 6.3 for dual-thrust at the indicated pressure levels and altitudes.

The low liquid/solid ratio permits a spherical nitrogen pressure vessel within the diameter envelope, at a reasonable 5000 psi initial pressure level. A simple integral pressure regulator-valve feeds the pressurant to the liquid. No other liquid expulsion devices are required. The injection interface is similarly constructed with integral valve and injector body. The valve serves to initiate full N_2O_4 flow for the ignition delay plus one second, and then shuts off the N_2O_4 flow to a portion of the dual injector for the remainder of the duration. The UDMH cup is mounted on the injector face and is ruptured by the N_2O_4 hydraulic pressure at start-up. Sophisticated plumbing configurations to reduce interfacial lengths are not proposed in order to maintain low cost. The hardware is manufactured from heat-treated 4130 steel.

Solid propellant burning rate and performance with N_2O_4 injection are presented in Figs. 2 and 3 . The ballistics (i.e. pressure and flow rate levels) were optimized with respect to performance and weight. The pressure and mixture ratio excursions in throttling are dependent upon the solid propellant burning rate slope, and are reduced with increased slope. However, for this propellant, a slope not greater than 0.7 was taken as realistic based upon the previous LPC work with it. Nevertheless, it was purposely designed to perform the major duration at low mixture ratio from hardware and mass fraction considerations, so that a reduced mixture ratio excursion would not have been taken advantage of for peak Isp in any case.

4.0 PRODUCTION COSTS

Although based on a highly preliminary motor design and specification, LPC is confident that the budgetary costs presented in this section are realistic and attainable.

These production costs would be achieved after completion of an eighteen month development program which could be conducted for costs in the range of $3,000.000.

No. Units Per Yr.	1st Year Unit Cost	2nd Year Unit Cost
1,000	$2,637	$ 2,320
5,000	$2,114	$ 1,911
10,000	$1,919	$ 1,761

The 1st year unit costs shown above include amortization of rate tooling, tool design would be completed under the basic development program.

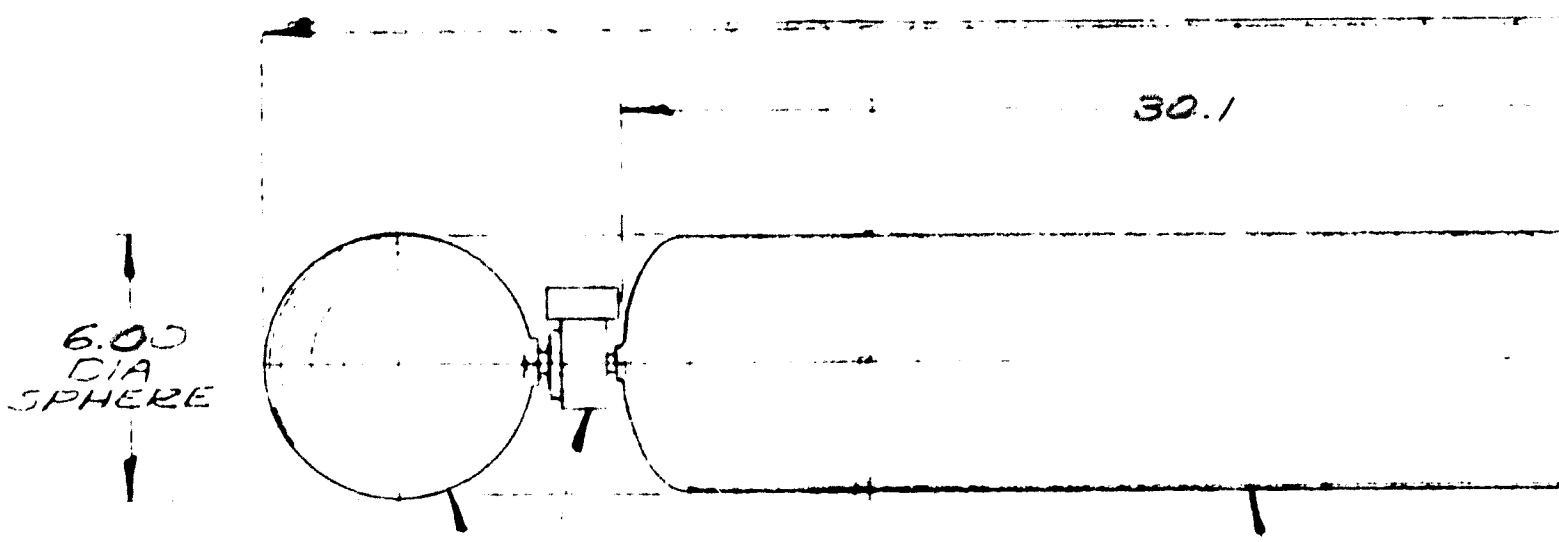
Cost estimates include the following groundrules and assumptions.

- No lot acceptance testing is required other than standard quality control verification of strand burn rates.
- First year production period of performance in 1970
- Estimates include overhead, G & A, profit, and appropriate escalation factors.
- Deliveries are F.O.B. Redlands, California
- Motors will be delivered as assembled units; however, the N_2 sphere will not be pressurized.
- Estimates do not include launch support effort.

Table A-1

CHARACTERISTICS OF LOW-COST, DUAL-THRUST RSVP SOUNDING ROCKET MOTOR

Control Liquid (0.050 lb/in^3)	N_2O_4
Solid Propellant (0.053 lb/in^3) 65% NH_4NO_3-10%Al-25%Rubber	
Theoretical Specific Impulse	Figure
High Thrust, ϵ =6.3, sea-level (lb)	4000
High Thrust Duration (sec)	1.00
High Thrust Fluid/Solid Ratio	1.11
High Thrust Combustion Pressure (psi)	1426
Low Thrust, ϵ =6.3, 15K ft. (lb.)	800
Low Thrust Duration (sec)	29.50
Low Thrust Fluid/Solid Ratio	0.40
Low Thrust Combustion Pressure (psi)	300
Total Impulse (lb-sec)	27,600
Propellant Weight Expended (lb)	118.8
Average Delivered Specific Impulse (sec)	232.3
Motor Diameter (in)	6.00
Total Length (in)	114.50
Liquid Components Inert Weight & Residuals (lb)	25.9
Solid Rocket Motor Inert Weight & Residuals (lb)	16.9
Skirts, Mounting provision weights (lb)	3.2
Total Inert Weight (lb)	46.0
Total Propulsion Weight (lb)	164.8
Mass Fraction	0.721



N_2 PRESSURE
CHAMBER

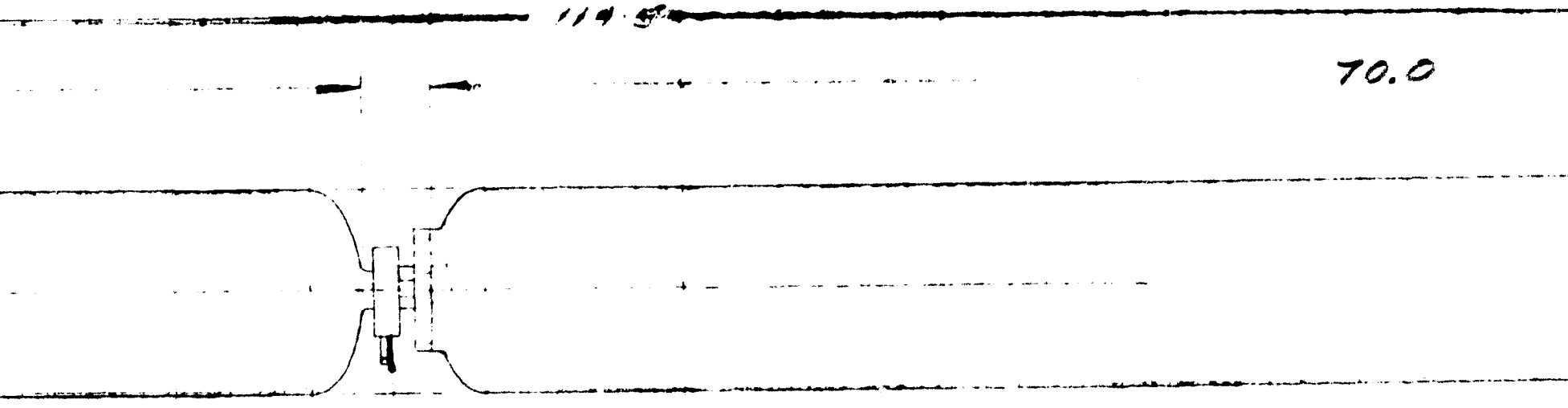
PRESSURE
REGULATOR & VALVE

LIQUID
CHAMB

FOLDOUT FRAME 1

FOLDOUT FRAME

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VALVE & DUAL
INJECTOR ASSY

LOW COST R3VP
SOUNDING ROCKET

SCALE: 1/4

1-10-68

FOLDOUT FRAME 2

70.0

SOLID PROPELLANT
CHAMBER

R3VP
ROCKET

1-10-68

FOLDOUT FRAME 3

FOLD

AME 2

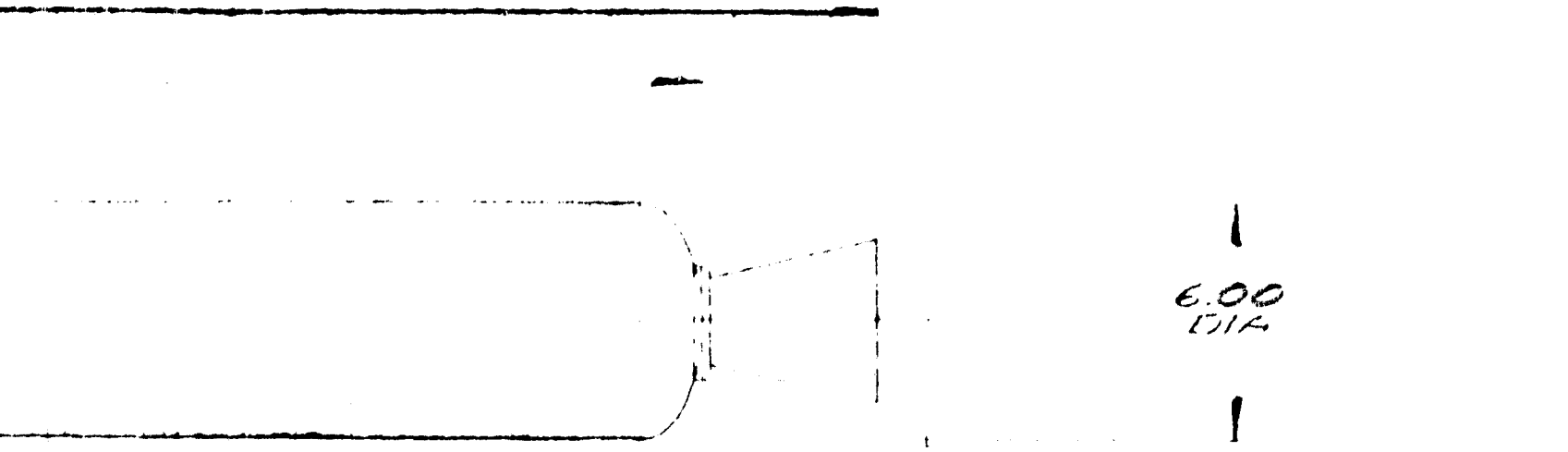


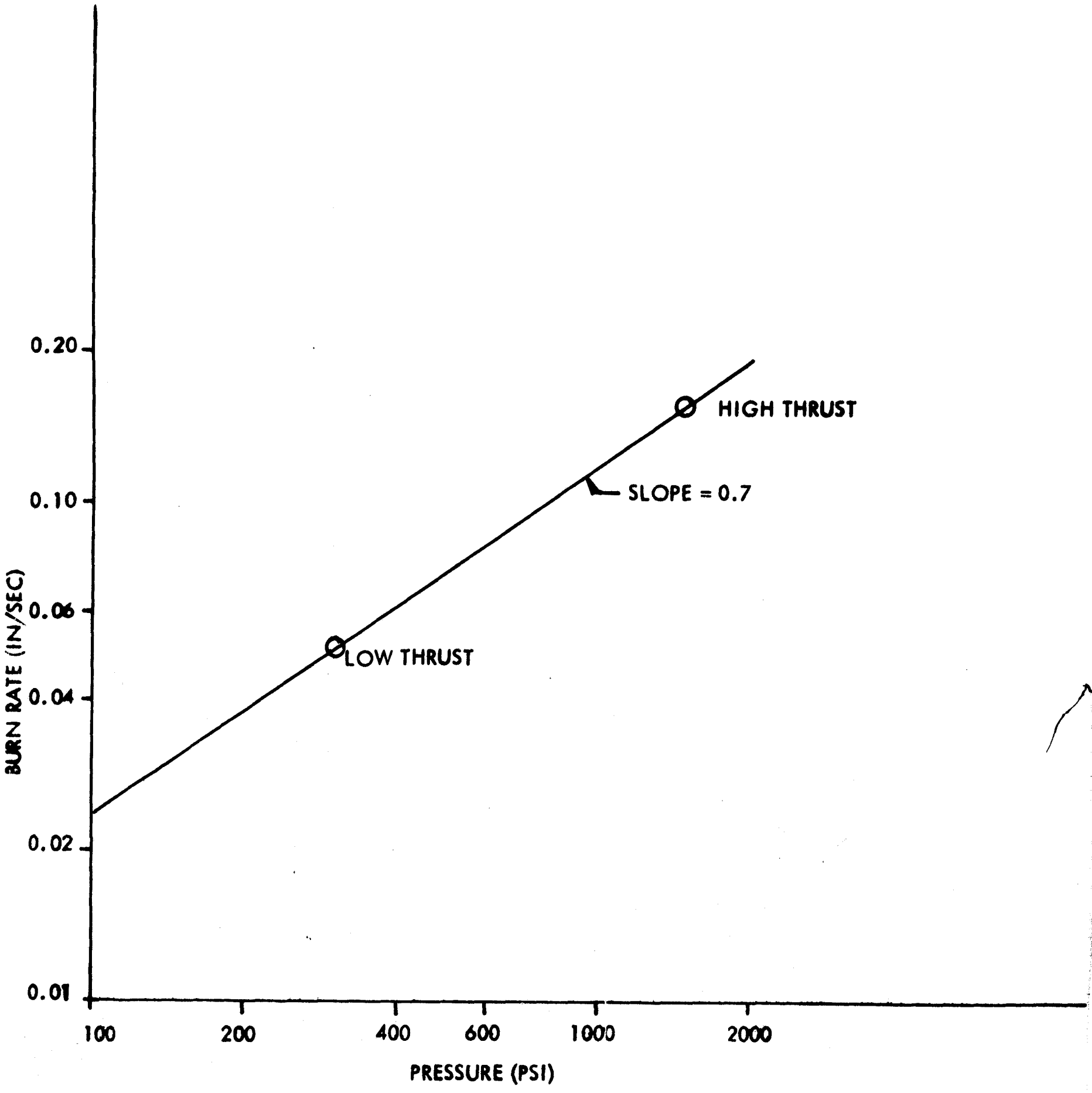
Figure A-1. RSVP Unit (LPC/FPC)

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BURN RATE VS PRESSURE (LPC/FPC)



SD 9079FR-1

Figure A-2
Page A-

THEORETICAL INSPECTION VS F/S RATIO (LBC/FBC)

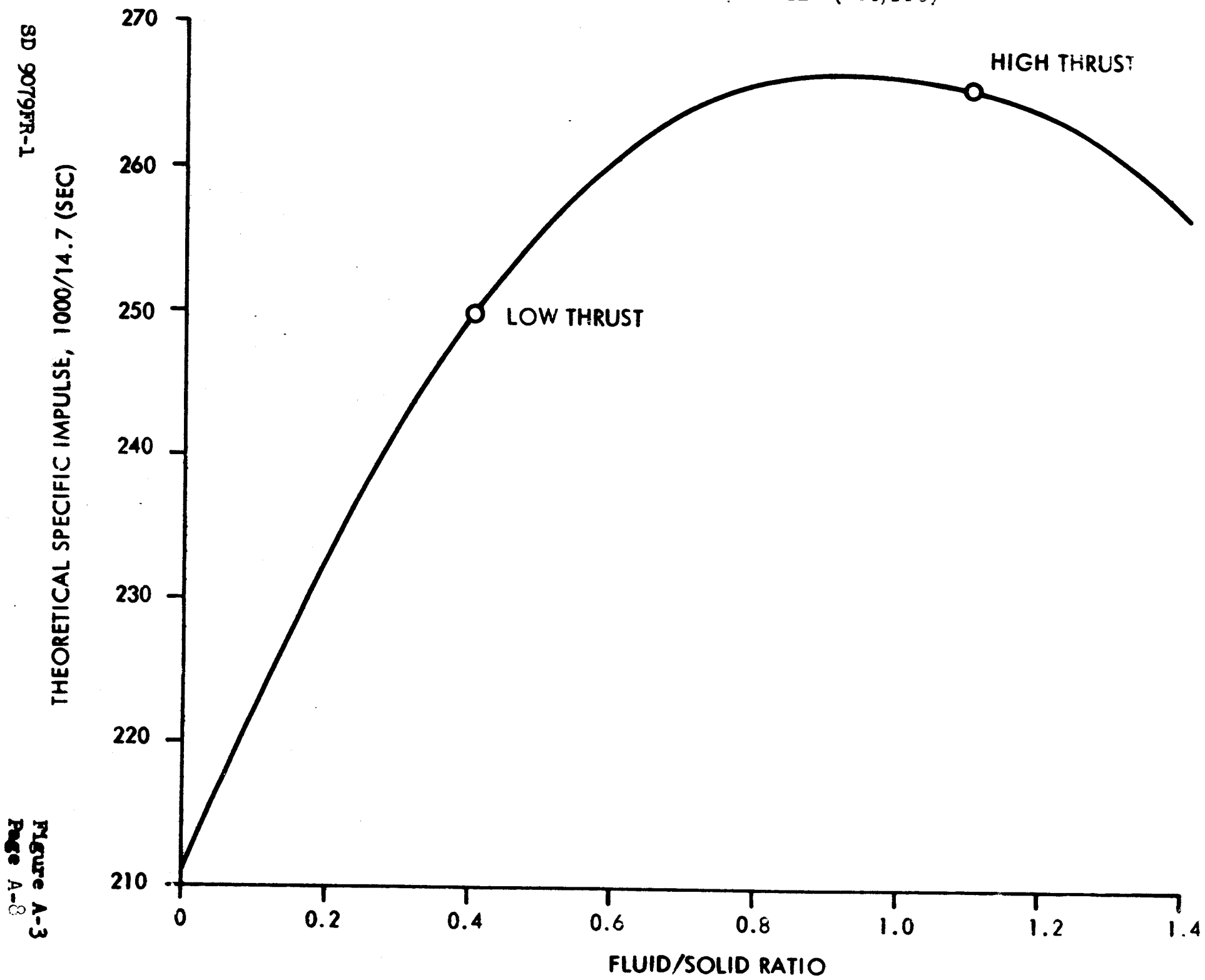


Figure A-3
Page A-8

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APPENDIX B

COMPUTED PRINTOUT OF FINAL TRAJECTORIES

SD 9079FR-1

APPENDIX B

The heading on the printout tapes are, for the most part, self-explanatory; however some clarifying definitions are given below:

1) Stage 1 - boost phase
   Stage 2 - sustain phase
   Stages 3 and 4 - postburnout coasting stages (Stage 4 is included only to change printout timing and is not a trajectory function; therefore, for all practical purposes Stages 3 and 4 may be considered as one.

2) Gamma - is the local flight path angle (γ); i.e., angle between the velocity vector and the local vertical.

3) Acc - axial acceleration

4) C_D - zero-lift drag coefficient (see Figure 3.3.2.3).

5) Q - dynamic pressure

6) S - reference area on which C_D is based

7) AE - nozzle exit area

8) The simulation is referred to a round, non-rotating earth, having the 1962 Standard Atmospheric model.

COMPUTER PRINTOUT - FINAL TRAJECTORY

UTC HYBRID MOTOR

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACCEL.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC D.,SLUG/FT	X,SLUG AE,SQFT
1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	242.4	0.196
	[illegible]	[illegible]	[illegible]	153.5	0.0	0.461	0.0	0.159
1	0.230	127.7	12.7	20.91	4000.38	1.793	242.4	0.196
	0.114	1.1	5.00	149.30	867.80	0.461	19.4	0.159
1	0.215	136.1	15.7	20.94	4000.11	2.076	242.4	0.196
	0.124	1.3	5.00	149.33	937.52	0.461	22.9	0.159
1	[illegible]	337.5	[illegible]	20.35	4000.60	7.127	242.5	0.196
	0.231	5.0	5.15	147.00	1734.90	0.461	78.7	0.159
1	[illegible]	389.7	120.5	21.7.	4001.36	10.261	242.5	0.196
	[illegible]	10.8	5.25	143.75	2501.90	0.461	179.9	0.159
1	[illegible]	523.9	211.4	21.04	4002.43	23.445	242.6	0.196
	0.472	19.3	5.44	140.49	3469.20	0.462	324.3	0.159
1	1.080	556.3	329.3	21.45	4003.42	46.854	242.7	0.196
	0.592	33.4	5.49	137.13	4335.50	0.465	513.3	0.159
2	1.305	590.3	329.3	2.45	357.42	46.853	155.5	0.196
	0.592	38.4	5.49	177.13	0.0	0.465	513.3	0.159
2	[illegible]	939.4	2051.5	2.52	699.42	93.310	172.4	0.196
	[illegible]	264.7	7.11	155.13	2969.96	0.525	970.1	0.159
2	7.155	1153.8	5729.2	2.43	717.43	162.380	180.8	0.196
	1.054	512.4	8.71	153.29	5933.92	0.620	1333.5	0.159
2	[illegible]	1429.2	9559.0	2.32	754.41	187.879	190.1	0.196
	[illegible]	1351.8	7.54	141.33	8999.40	0.529	1808.5	0.159
2	13.333	1737.4	14251.4	3.47	794.14	216.127	200.1	0.196
	1.633	1697.3	7.69	129.47	11379.58	0.476	2314.3	0.159

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CASE 110 JET MIXED

3-12-69

STAGE	TIME,SEC / MACH NO.	ALT,FT / VACUUM ALT,FT	WEIGHT,LB / FUEL WT,LB	CD,0 / THRUST,LB	TAMBST,R / TT(TAC),R	WB/W0 / CP	ISP,SEC / V,LB/SQFT	Q,SQFT / AE,SQFT
2	15.000	21 1.7	12 .9	.11	334.51	245.471	210.3	0.196
	3.025	2496.3	5.78	117.57	14349.57	0.428	2804.7	0.159
2	15.999	2537.+	28 65.8	4.77	374.37	245.469	220.3	0.196
	2.515	3382.3	5.43	105.25	17313.45	0.388	3221.8	0.159
2	21.999	3171.+	38 35.0	3.17	911.37	243.[illegible]	229.5	0.196
	3.153	4732.3	8.72	93.75	20743.35	0.352	3472.3	0.159
2	24.998	3753.+	45161.7	3.33	941.11	245.541	237.1	0.196
	3.874	6297.1	8.27	81.35	23759.21	0.324	3229.7	0.159
2	27.997	4335.3	57524.3	12.49	952.77	162.239	242.6	0.196
	4.788	8267.3	9.17	69.95	25729.10	0.391	2748.1	0.159
2	30.000	5375.7	67403.4	12.61	973.01	130.521	245.2	0.196
	5.554	9867.+	9.29	62.00	23712.36	0.288	2306.2	0.159
3	30.000	5375.7	67403.4	-3.33	0.0	145.332	0.0	0.196
	5.554	9867.8	9.29	62.00	0.0	0.322	2306.2	0.0
3	35.000	5512.7	92013.4	-1.60	0.0	39.414	0.0	0.196
	5.367	14085.+	9.57	62.00	0.0	0.340	574.7	0.0
3	40.000	4747.7	117034.1	-1.17	0.0	12.359	0.0	0.196
	4.631	17159.3	9.37	62.00	0.0	0.359	179.3	0.0
3	45.000	4522.7	140239.7	-1.04	0.0	4.202	0.0	0.196
	4.285	22272.7	10.19	62.00	0.0	0.377	56.8	0.0
3	50.000	4459.9	162567.9	-1.00	0.0	1.708	0.0	0.196
	4.033	26327.7	10.54	62.00	0.0	0.393	22.3	0.0
4	50.000	4459.9	162567.8	-1.00	0.0	1.708	0.0	0.196
	4.033	26327.7	10.54	62.00	0.0	0.393	22.3	0.0

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ASE 11 UTS HYBRID

STAGE								
4	93.333	4143.3	2479[illegible].7	-3.37	3.0	0.355	0.1	0.196
	2.343	3445[illegible].2	11.32	52.33	4.3	0.303	4.4	0.3
4	73.305	3834.7	243547.1	-3.35	0.3	0.051	0.0	0.196
	4.225	42441.7	12.22	52.33	0.3	0.363	0.3	0.1
4	83.333	3524.3	273754.7	-3.33	0.3	0.0	0.0	0.196
	4.163	39451.3	13.7[illegible]	52.33	0.3	0.1	0.1	0.3
4	93.333	3724.[illegible]	312437.1	-3.34	0.3	0.1	0.1	0.196
	3.813	38436.3	14.53	52.33	0.3	0.0	0.0	0.3
4	103.333	3924.[illegible]	3421[illegible].3	-3.33	0.3	0.0	0.0	0.196
	3.454	35307.3	15.94	52.33	0.3	0.0	0.0	0.3
4	113.333	4124.4	373734.7	-3.33	0.3	0.0	0.0	0.196
	3.133	32433.2	17.41	52.33	0.3	0.0	0.0	0.3
4	123.333	4323.[illegible]	333174.[illegible]	-3.33	0.3	0.0	0.0	0.196
	2.755	32255.7	19.21	52.33	0.3	0.0	0.0	0.3
4	133.333	4525.2	413[illegible].5	-3.33	0.3	0.0	0.0	0.196
	2.413	30141.1	23.14	52.33	0.3	0.0	0.0	0.3
4	143.333	4725.2	453[illegible].7	-3.33	0.3	0.0	0.0	0.196
	2.145	28[illegible].1	27.17	52.33	0.3	0.0	0.0	0.3
4	153.333	4927.3	493940.3	-3.33	0.3	0.0	0.0	0.196
	1.923	25[illegible].2	32.49	52.33	0.3	0.0	0.0	0.3
4	163.333	5127.[illegible]	535025.2	-3.33	0.3	0.0	0.0	0.196
	1.731	23[illegible].3	40.11	52.33	0.3	0.0	0.0	0.3
4	173.333	5331.5	583031.5	-3.33	0.3	0.0	0.0	0.196
	1.53[illegible]	21[illegible].2	51.15	52.33	0.3	0.0	0.0	0.3

REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR"

COMPUTER PRINTOUT - FINAL TRAJECTORY

ASE 103 AGC/SRO-

AGC LONG BURNING SOLID MOTOR

JOB 0022 PAGE 1

AE =85 DEG 1-29-68

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	AC...,S WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
1	0.0 0.0	0.0 0.0	0.0 5.00	19.15 198.00	3988.97 0.0	0.0 0.492	256.4 0.0	0.196 0.112
1	0.200 0.111	124.2 1.1	12.3 5.00	19.46 194.39	3989.08 345.20	1.773 0.492	256.4 18.3	0.196 0.112
1	0.219 0.122	136.2 1.3	15.0 5.00	19.49 194.39	3989.10 925.75	2.157 0.492	256.4 22.3	0.196 0.112
1	0.400 0.224	250.5 4.8	54.1 5.16	19.77 191.78	3989.43 1590.40	7.201 0.493	256.4 74.5	0.196 0.112
1	0.600 0.339	378.7 10.5	116.8 5.27	20.07 188.66	3989.96 2535.60	16.435 0.493	256.4 169.8	0.196 0.112
1	0.800 0.456	508.7 18.7	205.1 5.34	20.39 185.55	3990.70 3380.80	29.362 0.494	256.5 305.7	0.196 0.112
1	1.000 0.574	640.5 29.5	319.5 5.40	20.62 182.44	3991.67 4225.00	47.240 0.498	256.6 483.1	0.196 0.112
2	1.000 0.574	640.5 29.5	319.5 5.40	3.56 182.44	878.67 0.0	47.240 0.499	190.1 483.1	0.196 0.112
2	4.000 0.891	985.9 275.9	2748.3 6.10	3.48 168.57	898.55 3338.99	144.445 0.691	194.4 1065.2	0.196 0.112
2	7.000 1.207	1319.1 660.2	6183.1 6.63	3.53 154.70	924.33 5577.92	224.307 0.564	199.9 1719.3	0.196 0.112
2	10.000 1.557	1673.6 1195.9	10633.0 7.07	3.83 140.84	953.90 10016.82	274.462 0.580	206.3 2411.3	0.196 0.112
2	13.000 1.962	2065.3 1904.0	16184.6 7.45	4.32 125.97	935.42 13355.73	310.630 0.515	213.2 3070.2	0.196 0.112

SD 9079FR-1

"REPRODUCILIBITY OF THE ORIGINAL PAGE IS POOR."

ASE 103 AGC/SRU-

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
2	16.000 2.459	2519.0 2812.0	22980.7 7.77	5.14 113.10	1016.90 16694.64	323.975 0.454	220.0 3630.6	0.196 0.112
2	18.999 3.103	3070.1 3960.3	31254.8 8.04	6.37 99.23	1046.39 20033.54	315.930 0.401	226.4 4008.3	0.196 0.112
2	21.999 3.897	3772.6 5407.5	41367.4 8.27	8.36 85.37	1071.75 23372.45	273.820 0.355	231.8 3915.6	0.196 0.112
2	24.998 4.869	4713.6 7244.7	53891.0 8.45	11.31 71.50	1090.34 26711.36	211.044 0.320	235.9 3356.9	0.196 0.112
2	27.000 5.707	5525.1 8752.4	64001.9 8.55	14.00 62.25	1099.03 28940.26	166.218 0.298	237.7 2844.5	0.196 0.112
3	27.000 5.707	5525.1 8752.4	64001.9 8.55	-3.82 62.25	0.0 0.0	176.789 0.317	0.0 2844.5	0.196 0.0
3	32.000 5.202	5121.3 12732.0	90178.4 8.80	-1.71 62.25	0.0 0.0	45.518 0.335	0.0 591.4	0.196 0.0
3	37.000 4.840	4894.1 16598.9	114883.8 9.08	-1.19 62.25	0.0 0.0	13.503 0.350	0.0 196.5	0.196 0.0
3	42.000 4.479	4718.2 20426.1	138598.7 9.37	-1.05 62.25	0.0 0.0	4.511 0.366	0.0 62.7	0.196 0.0
3	47.000 4.209	4554.3 24234.4	161456.6 9.69	-1.00 62.25	0.0 0.0	1.718 0.381	0.0 23.0	0.196 0.0
3	48.000 4.179	4522.3 24994.5	165929.6 9.75	-0.99 62.25	0.0 0.0	1.449 0.382	0.0 19.3	0.196 0.0
4	48.000 4.179	4522.3 24994.5	165929.6 9.75	-0.99 62.25	0.0 0.0	1.449 0.382	0.0 19.3	0.196 0.0

REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR

ASE 103 AGC/SRD-

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB ITP(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
4	58.000 4.093	4207.6 32573.9	208895.0 10.46	-0.97 62.25	0.0 0.0	0.258 0.387	0.0 3.4	0.196 0.0
4	68.000 4.222	3897.7 40120.7	248593.0 11.28	-0.95 62.25	0.0 0.0	0.043 0.380	0.0 0.6	0.196 0.0
4	78.000 4.062	3590.5 47539.9	285348.7 12.23	-0.95 62.25	0.0 0.0	0.009 0.389	0.0 0.1	0.196 0.0
4	88.000 3.673	3285.6 55134.3	318874.6 13.37	-0.94 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	98.000 3.335	2983.2 62606.2	349281.9 14.73	-0.94 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	108.000 3.000	2683.7 70057.2	376579.1 16.40	-0.93 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	118.000 2.669	2387.9 77490.0	400774.1 18.45	-0.91 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	128.000 2.344	2095.4 84907.2	421874.1 21.14	-0.90 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	138.000 2.029	1812.0 92310.8	439885.0 24.54	-0.87 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	148.000 1.713	1537.0 99702.7	454812.5 29.42	-0.83 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	158.000 1.428	1277.7 107085.7	466850.7 36.19	-0.77 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0
4	168.000 1.163	1045.2 114461.4	475433.1 46.18	-0.65 62.25	0.0 0.0	0.0 0.0	0.0 0.0	0.196 0.0

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR"

CASE 103 AGC/SRO-

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB ITOTVAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
4	178.000	851.3	481132.0	-0.46	0.0	0.0	0.0	0.196
	0.963	121832.4	61.09	62.25	0.0	0.0	0.0	0.0
4	188.000	761.6	483759.4	-0.14	0.0	0.0	0.0	0.196
	0.852	129200.3	81.75	62.25	0.0	0.0	0.0	0.0
4	191.559	753.3	483952.0					
		131969.7	90.00					

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR"

COMPUTER PRINTOUT - FINAL TRAJECTORY
AGC VAPAK MOTOR

JOB E135 PAGE 1

CASE 105 AGC/LRO-

QE =85 DEG 3 12-68

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
1	0.0 0.0	0.0 0.0	0.0 5.00	11.53 159.60	1999.54 0.0	0.0 0.515	235.5 0.0	0.196 0.062
1	0.200 0.067	74.6 0.6	7.4 5.00	11.55 157.90	1999.68 426.00	0.671 0.515	235.5 5.6	0.196 0.062
1	0.283 0.095	105.9 1.3	15.0 5.00	11.72 157.20	1999.71 603.05	1.351 0.515	235.5 13.4	0.196 0.062
1	0.400 0.134	150.1 2.8	31.5 5.15	11.79 156.20	1999.79 852.00	2.713 0.515	235.5 26.8	0.196 0.062
1	0.600 0.203	226.4 6.2	69.1 5.33	11.91 154.50	1999.96 1278.00	6.155 0.517	235.5 59.8	0.196 0.062
1	0.800 0.272	303.3 11.2	121.8 5.46	12.02 152.81	2000.20 1704.00	11.060 0.517	235.5 109.0	0.196 0.062
1	1.000 0.342	381.1 17.8	189.9 5.57	12.13 151.11	2000.51 2130.00	17.424 0.517	235.6 171.7	0.196 0.062
2	1.000 0.342	381.1 17.8	189.9 5.57	1.64 151.11	415.36 0.0	17.424 0.517	223.3 171.7	0.196 0.062
2	4.000 0.487	548.9 186.9	1563.8 6.38	1.57 145.53	421.65 1634.99	33.907 0.520	226.7 332.3	0.196 0.062
2	7.000 0.637	703.1 412.3	3413.5 8.04	1.53 139.95	429.75 3269.99	55.565 0.532	231.0 531.3	0.196 0.062
2	10.000 0.788	862.4 763.3	5738.1 9.09	1.57 134.37	439.34 4904.93	95.224 0.551	236.2 745.0	0.196 0.062
2	13.000 0.928	1005.7 1231.2	8505.0 10.08	1.40 128.79	449.85 6539.81	143.373 0.784	241.9 930.7	0.196 0.062

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR."

CASE 106 AGC/LRO-

3 12-68

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQFT AE,SQFT
2	18.000	1135.1	11666.1	1.38	460.85	170.176	247.8	0.196
	1.063	1820.3	11.02	123.21	8174.69	0.803	1079.1	0.062
2	18.999	1276.0	15211.4	1.50	471.90	180.284	253.7	0.196
	1.208	2540.3	11.93	117.63	9809.57	0.758	1210.4	0.062
2	21.999	1429.5	19171.0	1.68	482.33	185.517	259.6	0.196
	1.374	3408.6	12.80	112.05	11444.45	0.709	1332.6	0.062
2	24.998	1602.0	23592.1	1.90	493.44	187.691	265.3	0.196
	1.568	4447.0	13.64	105.47	13079.34	0.664	1438.2	0.062
2	27.997	1798.9	28534.0	2.19	503.49	185.117	270.7	0.196
	1.797	5681.2	14.42	100.39	14714.22	0.620	1521.3	0.062
2	30.996	2027.7	34074.2	2.57	512.83	176.334	275.7	0.196
	2.075	7142.7	15.16	95.31	16349.11	0.571	1572.4	0.062
2	33.996	2299.3	40315.7	3.09	521.11	157.749	280.2	0.196
	2.375	8871.2	15.83	89.73	17933.99	0.525	1529.9	0.062
2	36.995	2627.2	47398.9	3.72	527.97	134.953	283.8	0.196
	2.7..	10918.6	16.45	84.15	19618.87	0.483	1423.8	0.062
2	39.994	3019.6	55493.3	4.43	533.42	110.568	286.8	0.196
	3.119	13347.5	17.01	78.57	21253.75	0.440	1278.8	0.062
2	42.993	3484.8	64791.2	5.23	537.58	86.306	289.0	0.196
	3.600	16229.5	17.51	72.99	22888.64	0.403	1090.9	0.062
2	45.993	4032.1	75507.4	6.13	540.55	63.551	290.6	0.196
	4.165	19645.7	17.94	67.41	24523.52	0.369	876.7	0.062
2	48.992	4671.5	87887.4	7.14	542.51	42.847	291.7	0.196
	4.771	23687.5	18.33	61.83	26158.40	0.340	642.3	0.062

CASE 106 AGC/LRO-

3 12-68

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQF AE,SQF
2	50.600	5056.0	95304.2	7.73	543.24	33.186	292.1	0.196
	5.087	26147.4	18.51	58.84	27034.93	0.327	516.6	0.062
3	50.600	5056.0	95304.2	-1.53	0.0	34.470	0.0	0.196
	5.087	26147.4	18.51	58.84	0.0	0.340	516.6	0.0
3	55.000	4870.9	115964.9	-1.15	0.0	12.916	0.0	0.196
	4.711	33128.7	19.01	58.84	0.0	0.356	184.9	0.0
3	60.000	4699.0	138533.9	-1.01	0.0	4.653	0.0	0.196
	4.369	40989.8	19.52	58.84	0.0	0.372	63.7	0.0
3	65.000	4541.6	160246.4	-0.96	0.0	1.889	0.0	0.196
	4.107	48811.3	20.27	58.84	0.0	0.386	24.9	0.0
3	70.000	4389.5	181143.2	-0.93	0.0	0.867	0.0	0.196
	4.026	56607.3	20.96	58.84	0.0	0.391	11.3	0.0
4	70.000	4389.8	181143.2	-0.93	0.0	0.867	0.0	0.196
	4.026	56607.3	20.96	58.84	0.0	0.391	11.3	0.0
4	79.000	4123.3	216748.2	-0.91	0.0	0.222	0.0	0.196
	4.163	70592.6	22.34	58.84	0.0	0.383	2.9	0.0
4	89.000	3833.5	253314.1	-0.89	0.0	0.040	0.0	0.196
	4.373	86073.0	24.08	58.84	0.0	0.372	0.5	0.0
4	99.000	3549.4	286747.9	-0.87	0.0	0.0	0.0	0.196
	4.193	101501.3	26.10	58.84	0.0	0.0	0.1	0.0
4	109.000	3271.5	317062.4	-0.85	0.0	0.0	0.0	0.196
	3.865	116863.1	28.47	58.84	0.0	0.0	0.0	0.0
4	119.000	3001.0	344266.4	-0.83	0.0	0.0	0.0	0.196
	3.545	132223.4	31.26	58.84	0.0	0.0	0.0	0.0

"REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR."

CASE 106 AGC/LRU-

3 12-68

STAGE	TIME,SEC MACH NO.	VEL.,FT/SEC RANGE,FT	ALTITUDE,FT GAMMA,DEG	ACC.,G WEIGHT,LB	THRUST,LB IT(VAC),LB-SEC	DRAG,LB CD	ISP,SEC Q,LB/SQFT	S,SQF AE,SQF
4	129.000	2739.6	368367.2	-0.79	0.0	0.0	0.0	0.196
	3.237	147527.1	34.60	58.84	0.0	0.0	0.0	0.0
4	139.000	2490.5	389372.2	-0.75	0.0	0.0	0.0	0.196
	2.942	162798.7	38.61	58.84	0.0	0.0	0.0	0.0
4	149.000	2256.6	407287.6	-0.70	0.0	0.0	0.0	0.196
	2.666	178042.1	42.48	58.84	0.0	0.0	0.0	0.0
4	159.000	2043.2	422118.9	-0.63	0.0	0.0	0.0	0.196
	2.414	193261.7	49.42	58.84	0.0	0.0	0.0	0.0
4	169.000	1857.0	433869.9	-0.53	0.0	0.0	0.0	0.196
	2.194	208462.6	56.63	58.84	0.0	0.0	0.0	0.0
4	179.000	1706.7	442544.3	-0.40	0.0	0.0	0.0	0.196
	2.016	223648.5	65.28	58.84	0.0	0.0	0.0	0.0
4	189.000	1602.3	448144.4	-0.24	0.0	0.0	0.0	0.196
	1.893	238824.3	75.30	58.84	0.0	0.0	0.0	0.0
4	199.000	1552.9	450672.1	-0.06	0.0	0.0	0.0	0.196
	1.834	253994.2	86.33	58.84	0.0	0.0	0.0	0.0
4	202.231	1549.7	450916.0					
		258674.6	90.00					

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