NASA N91-24247

HYBRID PROPULSION TECHNOLOGY PROGRAM[1]

G. E. Jensen and A. L. Holzman
UNITED TECHNOLOGIES/CHEMICAL SYSTEMS
P O Box 49028
San Jose, CA 95161-9028

14728

815

ABSTRACT

UW 292359

Future launch systems of the United States will require improvements in booster safety, reliability and cost. In order to increase payload capabilities, performance improvements are also desirable. The hybrid rocket motor (HRM) offers the potential for improvements in all of these areas.

This paper presents the designs for two sizes of hybrid boosters, a large 4.57-m (180-in) diameter booster duplicating the Advanced Solid Rocket Motor (ASRM) vacuum thrust-time profile and a smaller 2.44-m (96-in), one-quarter thrust level booster. The large booster would be used in tandem, while eight small boosters would be used to achieve the same total thrust. These preliminary designs have been generated as part of NASA contract No. NAS8-37778, Hybrid Propulsion Technology Program. This program is the first phase of an eventual three-phase program culminating in the demonstration of a large subscale engine.

The initial trade and sizing studies resulted in preferred motor diameters, operating pressures, nozzle geometry and fuel grain systems for both the large and small boosters. The data were then used for specific performance predictions in terms of payload and the definition and selection of the requirements for the major components: the oxidizer feed system, nozzle and thrust vector system. All of the parametric studies were performed using realistic fuel regression models based upon specific experimental data.

INTRODUCTION

HRMs offer the potential for improvements in rocket motor launch vehicles in the form of increased payload capabilities, booster safety, reliability and lower cost. By virtue of separation of its inert solid fuel and liquid oxidizer, a hybrid booster offers improved ground and flight safety. Even in the event of a major vehicle structural failure, a propellant explosion or major fire remains a highly improbable occurrence with the hybrid system. Moreover, the hybrid booster offers launch abort capability, throttleability to increase launch trajectory performance, and insensitivity to grain anomalies during operation, all of which are not available with solid propellant rocket motors. The safety aspects of the hybrid would allow for modifying the manufacture and launch operations, thereby resulting in a reduction of payload-to-orbit costs. Another important benefit associated with the development of large HRM technology is that it provides the means to test critical components, such as nozzles and insulation, under actual operating conditions with full capability to stop and restart the motor. This capability permits evaluating the components at various times throughout the motor operation.

The objectives of this program were to: (1) define preferred hybrid launch concepts and configurations, (2) identify the concepts and technologies required to enable development of an HRM booster, (3) plan for acquisition of this technology and plan for demonstration of a large subscale HRM in later phases. This report presents the designs for two sizes of hybrid boosters, one duplicating the Advanced Solid Rocket Motor (ASRM) vacuum thrust-time profile and the other a smaller one-quarter thrust-level booster. The large booster would be used in tandem, while eight small boosters would be used for the same total thrust.

[1]This work was performed under NASA contract No. NAS 8-37778 with the NASA Marshall Flight Center

As indicated in Table I, the primary requirement for the large hybrid booster was to meet the ASRM vacuum thrust-time profile depicted in Figure 1. The small hybrid booster meets the one-quarter thrust requirement. Booster designs and major subcomponent designs were completed for both sizes of hybrid booster. The designs were generated using the other design requirements summarized in Table I.

After completing the initial studies to select the oxidizer (liquid oxygen) and potential fuel systems, trade studies were performed with the CSD hybrid design/performance computer model. Fuel performance tables were calculated and representative fuel regression rate data were used to evaluate the effect of fuel composition, motor diameter, number of fuel ports, port geometry, oxidizer delivery system (pump versus pressure fed), oxidizer flow rate, and operating pressure upon the booster configuration and weight. These initial studies were nonspecific as to component design with only general size and weight models being given. In all cases the designs met the thrust-time profile and the total impulse requirements of Figure 1. Because the hybrid can be throttled to always meet the nomina[Bl thrust-time curve, the minimum and maximum thrust-time limits are not germane. The booster design study was therefore performed using the nominal thrust-time curve and a total nominal impulse of 1.44×10^9 N-sec ($324. \times 10^6$ lb-sec) which corresponds to an action time of 134 sec.

The initial trade and sizing studies resulted in preferred motor diameters, operating pressures, nozzle geometry and fuel grain systems for the large and small boosters. These data were then used in the performance predictions for the payload and for definition and selection of the requirements for the major components: oxidizer feed system, nozzle and[B thrust vector system. All of these parametric studies were performed using realistic fuel regression models based upon experimental tests measuring regression rate. The parametric and sizing studies resulted in the selection of a 4.57-m (180-in) diameter large booster and a 2.44-m (96-in) diameter small booster. An average operating pressure of 5.17-MPa (750-psi) was fixed for the pump-fed oxygen systems, while an average pressure of 3.45-MPa (500-psi) was selected for the pressure-fed systems. These values were selected on the basis of minimizing booster weight. Given more precise performance requirements, additional optimization studies would result in slightly different values but the differences would have minor effects upon the basic overall HRM designs.

A second design effort was performed using preliminary weight and size requirements for alternative oxygen feed systems, fuels, injector designs, case designs, and nozzle and thrust vector systems. These studies were also used to generate performance requirements for the subsystems, which could be used to refine their designs. This second effort generated two large booster designs: a baseline 4.57-m (180-in) diameter, inert-fuel-grain design and an alternative 3.96-m (156-in) diameter booster using a high-regression (oxidized) fuel grain. Both systems utilize a multi-pump GOX delivery feed system. Additionally, a large booster was designed using four parallel 2.44-m (96-in) diameter combustors (quad design) and a single LOX tank. A fundamental advantage of this system is that, with individual GOX pumps, the system has engine-out capability thereby significantly improving abort capability.

DESIGN SUMMARY

The sizing studies showed that hybrid boosters offer significant configurational flexibility. Typical sizing trends for the large liquid oxygen (LOX)/hydrocarbon fuel hybrid booster are presented in Figure 2. Payload capability is relatively insensitive to diameter over the range of 3.8 to 5.8-m (150 to 200-in), which corresponds to booster L/Ds of 8 to 18. This is the preferred booster diameter range for the Shuttle-compatible thrust and impulse values defined in the requirements. For the specified requirements, boosters smaller than 3.81-m (150-in) begin to get too long and lose payload. Boosters larger than 5.08-m (200-in) in diameter require an increasing number of ports and also lose payload. For the quarter-scale thrust booster requirements, boosters smaller than 1.78-m (70-in) diameter begin to get too long and those larger than 3.05-m (120-in) diameter require numerous ports. This preferred diameter range is driven by the performance requirements and would change in response to variations in the required performance. This trend also holds for the other hybrid fuel systems.

Figures 3, 4 and 5 summarize the selected configurations for the large and one-quarter-scale hybrid booster applications. All of the hybrid systems use oxygen as the oxidizer because of its commonality at the launch bases, cost, ease of use, and general safety in comparison to other oxidizers. In general, performance comparisons of the optimum pump-fed and pressure-fed systems favored selection of a pump-fed system. The preferred pump system actually produces gaseous oxygen (GOX), which is injected into the forward dome through a manifold injector system.

The booster selections were made partially on the basis of the largest improvement in possible payload capabilities. If lower payload requirements were specified, the motor weights and lengths could be reduced from the values presented. Also, if the advantages of hybrid throttleability were fully used, the engines could be down sized.

The booster designs are discussed in the following subsections. These designs have been generated for two types of fuels which are discussed in more detail, along with the other system characteristics, in the following sections.

LARGE MOTOR DESIGN(S)

For the large motor application, three designs have been selected: a baseline single 4.57-m (180-in) diameter booster a 3.96-m (156-in) diameter booster and a combination system that has four parallel 2.44-m (96-in) diameter grains with a common oxidizer tank. This latter combination offers definite advantages over the single large booster since it has engine-out capabilities and permits abort of the mission.

The results of the analysis for the large booster indicate that within the constraints of port L/D and oxidizer mass velocity on hybrid operation, there is a large variety of fuels and motor diameters that could be used. Therefore, the selection process was based upon factors other than performance alone; these other factors included reliability, life cycle costs (LCC), development risk, fabrication requirements, and transportation issues. These factors were used to select the baseline 4.57-m (180-in) diameter large booster and the 2.44-m (96-in) diameter small booster systems.

4.57-m (180-in) Diameter Booster. The preliminary design studies show that the minimum system weight and best packaging were achieved at a diameter of 4.57-m (180-in). At this diameter, launch pad modifications would be required if the ultimate application of the hybrid booster were to be a Shuttle SRM replacement. Without this constraint, the 4.57-m (180-in) diameter is optimum and was selected as the baseline design.

The preferred grain formulation for the 4.57-m (180-in) motor is based upon an inert, all-hydrocarbon fuel, (designated fuel No. 7) which provides the highest improvement in payload increment, as well as the highest system safety. The fuel consists of a hydroxyl-terminated polybutadiene (HTPB) binder and a polycyclopentadiene reinforcing agent (Escorez). To achieve the necessary fuel flow, over 30 fuel ports are required in the grain. These are arranged in two rows with a central port. While an open dome is used, an equivalent number of oxygen injectors would be required. To increase the system reliability, this booster would have multiple GOX pumps (3) with one-out capability. These feed into a common manifold which supplies the injectors.

Figure 3 shows the large HRM boosters relative to the Shuttle with RSRM boosters. Their performance is summarized in Table II, along with that for the alternative 3.96-m (156-in) diameter large booster and the 2.44-m (96-in) diameter small booster. As noted, performance is indicated to be higher than that for the solid propellant RSRM booster. However, it should be emphasized that these values are based on RSRM performance partials and the RSRM as a reference. Ultimate performance potential must be based on specific hybrid mission requirements.

Table III presents a breakdown of the weights of the 4.57-m (180-in) and 2.44-m (96-in) hydrocarbon-fueled pump-fed boosters. System trades favor a metal two-segment case with separate foward and aft domes. The oxidizer manifold is located above the forward dome with the oxygen being fed through solid cone injectors located above and in-line with the fuel ports. Because of the large number of ports (34), an open dome design has been chosen that permits the use of a common ignition system consisting of supplemental fuel injectors and redundant pyrogen initiators. Individual ports with fuel cast to the injector face would result in lower reliability because of the potential for non-ignition in one or more ports and a structural failure in the grain due to pressure differences between the fuel ports.

Clevis pin field joints similar to the RSRM design for the case segments have been selected on the basis of better performance, lower weight and lower cost. The fuel grains would be processed to eliminate exposure of the joints completely, thereby avoiding any of the similar problems encountered with the solid propellant solid rocket motors (SRMs).

The steel cases and aft nozzle dome are insulated with strip-wound Kevlar filled EPDM insulation. The forward dome is insulated with a trowelable EPDM insulation that covers the dome and the sides of the oxygen injectors. Because of termination grain stresses, the fuel grains are slotted at the upper end of the forward grain and the lower end of the aft grain. This is done by using release strips installed in the course of the continuous mix operations selected for casting the fuel grains. The two segments are bonded together to eliminate any leak paths to the middle field joint. This is done by installing a preformed fuel gasket between the segments as they are assembled together. The fuel gasket is bonded to the surfaces of both segments using a catalyzed liner system. The forward and aft field joints are insulated with additional thickness of EPDM insulation as well as the fuel grain which extends past the joints.

A flexseal nozzle with hydraulic actuators has been baselined for this application on the basis of proven technology and performance. The performance improvements due to LOX injection, however, were not fully considered in analyzing the alternative liquid injection TVC system. This is an area requiring further review utilizing more precise mission requirements and TVC requirements in particular.

At this time, the preferred oxidizer feed system consists of three pumps located in the interstage space between the fuel grain and a composite over-wrapped LOX tank. The feed system is based upon a version of the integral oxidizer-rich burner turbine and LOX pump proposed by Accurex Corporation. This system provides the highest performance with the burner products being added to the oxygen delivery. This is not a critical design issue as alternative pump systems could be used, but at the expense of about 544.3-kg (1200-lb) in delivered payload.

<u>Alternative 3.96-m (156-in) Diameter Booster.</u> To promote compatibility with the current processing and launch facilities, the preliminary sizing studies show the booster diameter could be reduced to minimize launcher impacts if required. Figure 3 shows a 3.96-m (156-in) diameter version of the large-thrust hybrid booster. For this version, three fuel systems were evaluated. The CSD selection for this size is designated fuel No. 8 and is a high-regression fuel (30% AP/HTPB) which provides a shorter booster length than the inert fuel formulation for this diameter booster. Overall safety is compromised, however, by the use of the oxidized fuel. To increase the system reliability, this booster would also have multiple LOX pumps with one-out capability.

<u>Alternative Full-Scale Booster/Quad Combustor.</u> The hybrid booster system studies have also led to a multiple chamber design option. This configuration (Figure 4) clusters four of the 1/4-scale fuel grains with a single oxidizer tank in order to perform the large motor mission. To minimize the size of the combustion chamber/solid fuel case and to provide increased system safety, an all hydrocarbon fuel was selected. Each chamber is self-contained with its own oxidizer feed pump and thrust vector control system TVC. Alternatively, for increased pump and system reliability, a common feed system consisting of three (3) pumps could be used. The central space between the chambers may be used for a

common propane tank to drive the four LOX pumps. The single oxidizer tank diameter can be selected to achieve the desired attachment length to the external tank (ET) in the case of the Shuttle or other core vehicle structure. For the large hybrid booster application, the multiple-chamber option offers several advantages, the most of which is enhanced manned flight safety achieved through continuous engine-out capability. Normally, command shutdown in response to detected failures or impending failures provides enhanced safety only for allowing the orbiter to safely land following booster shutdown or jettison.

The multiple-chamber design is sized to maintain an adequate thrust-to-weight ratio following single chamber shutdown on each booster. This shutdown can occur at any time from booster ignition to burnout. As the flight progresses, the number of engine failures that a booster can withstand and still permit a safe orbiter landing increases, due to the velocity and altitude imparted to the vehicle before any additional engines fail and/or are shutdown. This enables the hybrid booster to achieve a minimum of single engine (pair) failure capability throughout the flight, expanding to multiple failure capability as the flight progresses which is a significant enhancement in manned flight safety.

The multiple-chamber option also offers reduced development costs, design simplicity, and enhanced operational flexibility. Using the small motor chamber obviates the need to develop a single large chamber. The small motor can be used singly or in clusters of two, three, or five for other missions without additional chamber development. Each chamber has roughly 1/2 the number of fuel ports as the large motor, which enhances the design simplicity and reduces grain processing costs. The reduced size of the small motor simplifies recovery if so desired.

SMALL MOTOR DESIGN

Figure 5 shows how eight of the small 1/4-scale hybrid boosters would be clustered around an Advanced Launch System (ALS)-size payload. The inert fuel grain is the same as was selected for the multiple chamber design discussed in the previous subsection. Table III summarizes the weight breakdown of the 2.44-m (96-in) hydrocarbon quarter-scale booster. The system definition is less precise for the small booster. The low launch rate overall preferred design is pressure-fed with LITVC. This is primarily due to a lower life cycle cost. However, highest performance is achieved with a pump-fed system and a flexseal nozzle. At the high launch rate the preferred design has three integral GOX burner/turbine/pump units in the interstage. These units use the same basic design as that of the units used in the large motor. GOX is fed into a manifold that is an integral part of the dome.

On the basis of reliability and costs, an expendable composite case overwrapped on the pre-cast fuel grain has been selected for the low launch rate application. At the high launch rate a recoverable steel case is preferred. Also, the trade studies resulted in the selection of either liquid injection fixed nozzles or flexseal nozzles for the small booster, as it was specified that both size boosters have TVC capability. Life cycle cost considerations favor LITVC while higher performance is obtained with a flexseal nozzle.

A unique feature of the expendable small booster design is that the fuel grain is cast into a cartridge using consumable, non-removable mandrels. After cure, the cartridge is overwrapped with the forward dome and nozzle polar boss. This facilitates the overwrap process and eliminates mandrel withdrawal problems.

Better definition of the small booster requires further discussion of cost and performance. Highest performance was obtained with an expendable, composite case, pump-fed, flexseal nozzle configuration that resulted in a payload increase of 28,100 lb. The predicted low and high launch rate LCC for this system resulted in \$9,440 x 10^6 and \$30,150 x 10^6 respectively. The system with the lowest LCC (\$8,475 x 10^6 for the low rate and 26,200 x 10^6 for the high rate) is a recoverable, metal case, pump-fed, LITVC configuration; however, this system resulted in a payload increase of only 16,700 lb which is approximately 11,400 lb less than the highest performance configuration. Consequently, it becomes practical to discuss overall performance in terms of dollars per pound of paylaod to LEO.

In order to accomplish this objective, one must define the total launch cost which, in addition to the HRB's, also includes 1) flight operations and field services, 2) launch operations, 3) orbiter spares, 4) external tank and main propulsion system, etc. It was assumed that the total launch cost (@12 launches per year) of a current STS is $150 million, which inserts a 50 K lb payload into LEO, (this is equivalent to $3000/lb) and that 25% of this cost (i.e., $37.5 million) stems from the cost of the SRB's. Consequently, these other "fixed" costs were added to the predicted HRB costs per launch and divided by the predicted payload weights (i.e., 50 K lb + payload increment) to generate the dollars per pound figures.

Based on a dollar per-pound payload to LEO basis, the small expendable, composite case, pump-fed, flexseal nozzle configuration is the preferred system. The calculations show $2252 per pound at the low launch rate and $2082 per pound at the high launch rate. Because of the payload increase with the hybrid, these costs are significantly lower than for the existing SRB system.

BALLISTIC PERFORMANCE SUMMARY

Details of the large booster ballistic performance levels are shown in Figures 6 through 9. These are essentially the same as for the small booster except for the thrust and oxygen flow rate which are one-quarter the values of the large hybrid.

The pump-fed designs were optimized to a pre-selected average pressure of 5.17-MPa (750-psi). Consequently, the peak value of chamber pressure is approximately 40% higher than the average. Figure 6 shows the variation of chamber pressure as a function of burn time. The curve shape is the same for all designs, independent of average pressure level or booster size as it reflects the required thrust profile. Figure 7 plots the thrust profile which meets the Statment of Work nominal thrust time schedule. The action time integrated total impulse is 1.44×10^9 N-sec (324×10^6 lb-sec). This exceeds the specified minimum integrated value of 1.42×10^9 N-sec (320.15×10^6 lb-sec).

For the large 4.52-m (180-in) diameter booster using the all-hydrocarbon inert No. 7 fuel, the oxygen flow rate schedule is shown in Figure 8 as a function of time. Except for the tailoff after 120 sec, there is less than a 2-to-1 turndown in the oxidizer flow, which is easily achievable with turbine-driven pumping systems. This head-end oxidizer flow variation will result in a minor variation in the propellant mixture ratio passing through the nozzle. Figure 9 shows that there is only a $\pm 5\%$ variation in mixture ratio, which eliminates the need for any aft-end oxidizer injection or more complicated injection distribution scheme. This variation in mixture ratio does not seem to vary with motor size or total impulse. Note that the supplied total impulse schedule is the same as that to which the ASRMs were designed and does not reflect the optimum shape for the hybrid capabilities. The ASRM curve shape is limited by the ability to design a solid propellant motor with changing surface area for producing changes in thrust level. Since the hybrid can be throttled like a bi-propellant liquid rocket, the thrust-time curve can be prescribed with much greater freedom.

PERFORMANCE UPDATE

The designs shown in the previous sections do not take advantage of the hybrid throttleability. An item of particular importance to large size boosters is their thrust-energy time distribution. Large launch vehicles have a relatively low thrust-to-weight ratio, which results in lower overall accelerations requiring a steeper trajectory. Such trajectories are characterized by dominating gravity flight losses. A significant reduction of these losses can, therefore, have a correspondingly consequential effect on payload improvement. Gravity flight losses can be reduced by preparing a proper rocket motor thrust profile that reduces burn time and/or the average flight path angle, while complying with trajectory, aerodynamic and structural load limits. The Space

Transportation System (STS) solid rocket boosters (SRBs) have been designed to have a saddle-shaped thrust profile for this reason (see Figure 7). High up-front thrust decreases the average flight path angle; the saddle section is designed to reduce aerodynamic loading during the period of maximum dynamic pressure. The linear gradual thrust decreasing section (before tail-off) was designed to minimize the burn time while subjecting the structure to their load limits of no more than 3 gs acceleration throughout this period. However, this profile has been set by solid fuel grain ballistic limitations. These limitations impede an enhanced thrust profile that can result in further performance gains.

Since the hybrid rocket motor is throttleable, it doesn't have the thrust management limitations inherent to the solid fuel system. A trajectory study was conducted to quantify the payload lift advantage that can be realized from this hybrid feature. The mission trajectory consisted of a launch out of the Eastern Test Range (ETR) to a 28.5-deg inclination orbit with a 296 km (160 nmi) altitude. Trajectory constraints were obtained from Rockwell International and used to guide the shape of the hybrid motor thrust shape. Three booster motors were looked at: (1) a NASA strawman ASRM design that was used as a guideline during the ASRM phase B study contract No. NT018; (2) a hybrid booster that matched the aforementioned study contract's thrust shape; and (3) the same hybrid motor with an enhanced thrust profile.

The enhanced thrust profile was arrived at by a manual interactive procedure wherein performance was maximized while adhering to the trajectory limits. This profile is not necessarily the desired optimum, but serves to illustrate the performance advantage of the hybrid booster. Figure 10 shows the improved hybrid booster performance within the operational limits.

Table IV summarizes the weight and propulsive characteristics of these boosters and the calculated main engine cut-off (MECO) weight from the trajectory simulation. This weight represents the remaining weight of the system after the boost phase has been completed, and as such is an indication of the performance achieved from the boosting system (SSMEs and boosters). Since the core (SSMEs) propulsion is the same in each case, any changes in MECO weight are due to changes in the characteristics of the boosters. MECO weight changes representing changes in useful payload lift capability are also shown in Table IV. Based upon the booster characteristics used in the study, the throttleability feature enchances hybrid payload capability by approximately 3175 kg (7000 lb).

CONCLUSIONS

A hybrid booster can be designed to fit the desired thrust-time curve and improve the payload capability of a shuttle-type system in a higher performance lower weight unit than a solid propellant rocket booster. Because the hybrid can be throttled like a bi-propellant liquid system, the SRM thrust-time curve can be improved to gain additional payload. The HRM is larger than the SRM due to the lower propellant density but equal or smaller than the LRB.

Except for scale-up testing of the fuel grain, the technology for the HRB exists in liquid and solid systems.

The HRB has less critical failure modes than the SRB or LRB and contains an inert fuel that cannot detonate or burn catastrophically. Like the LRB, the HRB can be designed to have a clean exhaust containing no HCl or aluminum oxide. The pump-fed hybrid offers the lowest costs in delivering payload-to-orbit compared to other propulsion systems.

TABLE I. REQUIREMENTS AND TRADES

Parameter	Requirements/Trades
Thrust-time profile	Table 1 of SOW (March 31, 1989)
Impulse values	Figure 1 (March 31, 1989 SOW)
Motor size	• Two-booster: Shuttle • Eight-booster: Advanced Launch System (ALS)
Thrust vector control	Utilize TVC
Asbestos-containing materials	None allowed
Control systems	Active control: • Performance • Thrust imbalance • Propellant utilization • Transients
Environmentally degrading exhaust products	Minimize
Shelf life	Maximize
Extinguishability	• Goal: extinguish upon fluid flow termination • Required: thrust < 0.7 burnout weight
Safety and reliability	Identical for manned and unmanned
Life cycle costs	• 14-year operational phase • 4-year linear growth • 10-year constant rate: - 1 flight per month - 1 flight per week
Utilization	Expendable-reusable

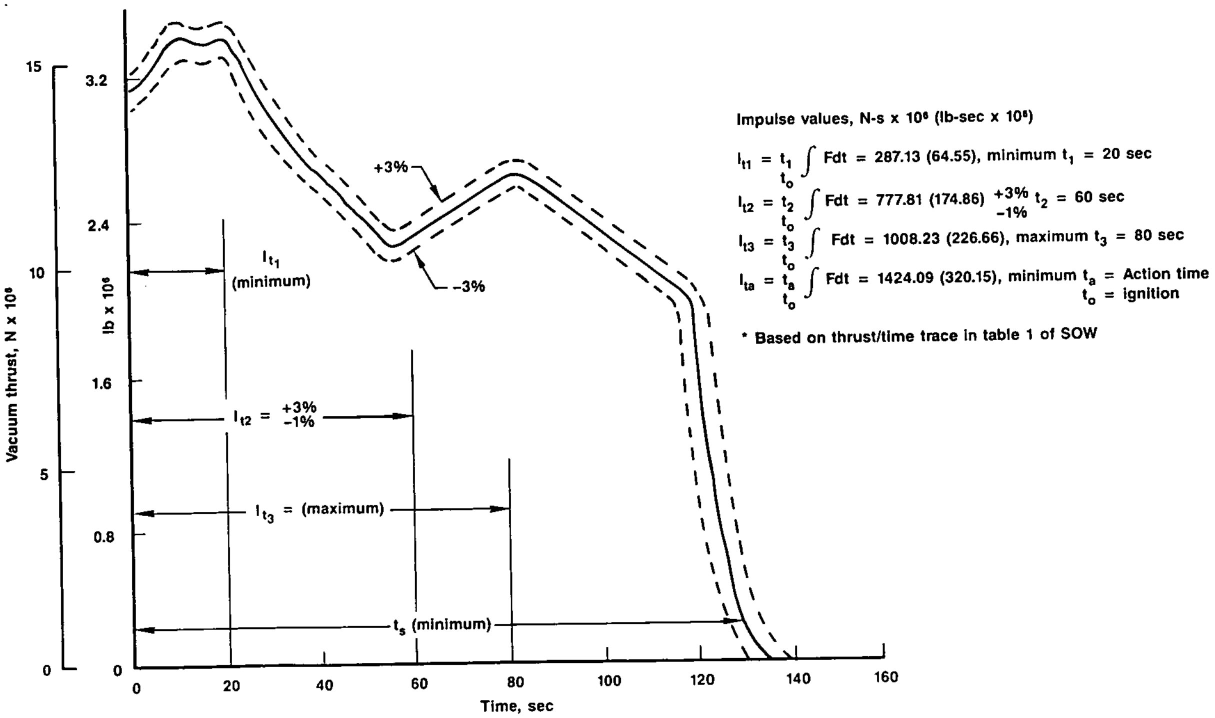


Fig. 1. ASRM Reference Vacuum Thrust -Time Trace and Impulse Values at 60°F

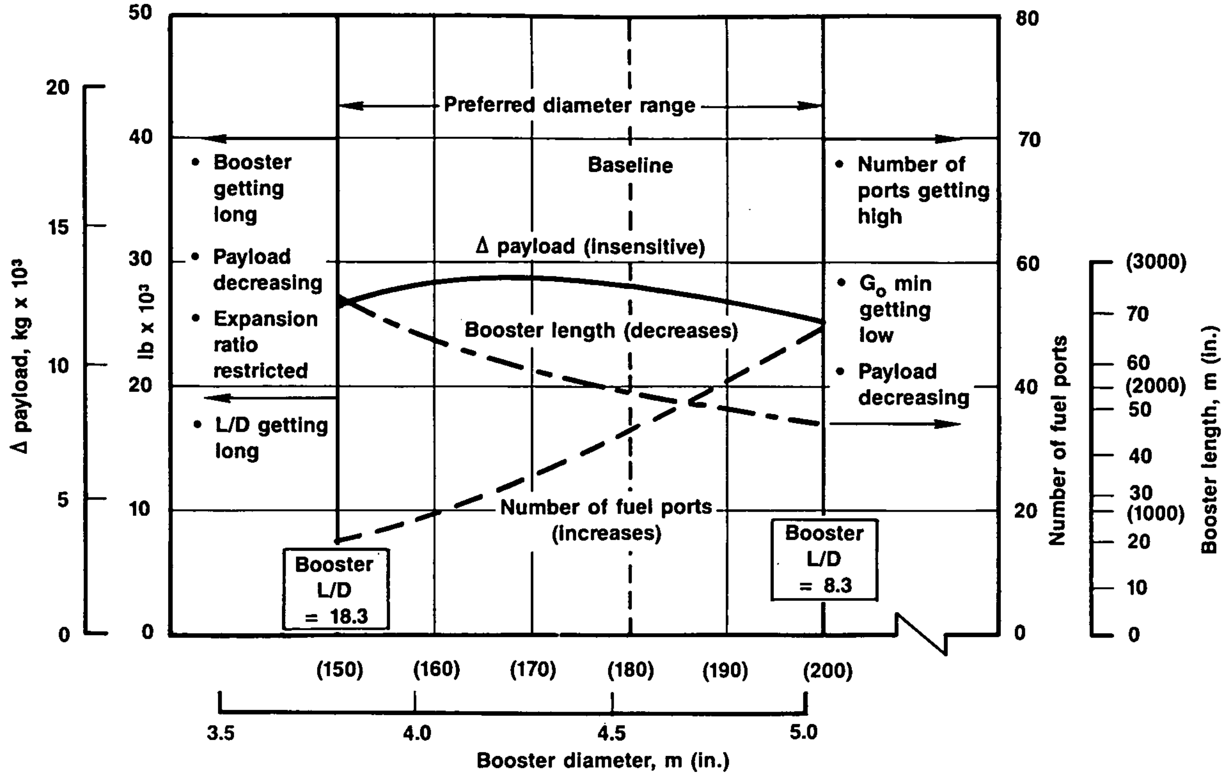
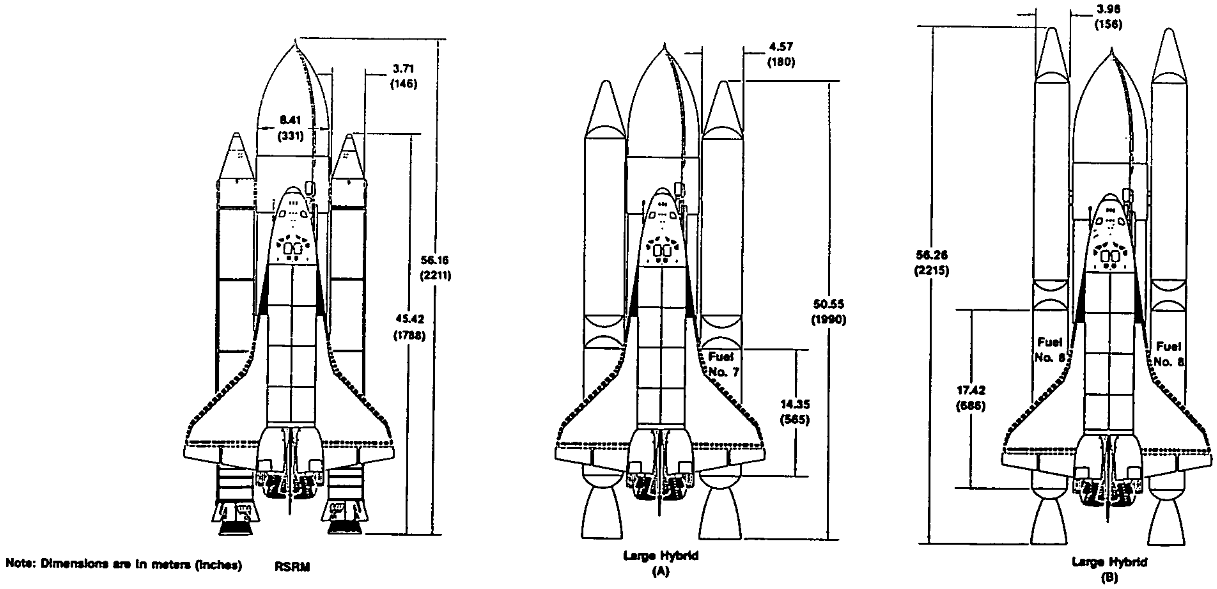


Fig. 2. Hybrid Booster Sizing Trends

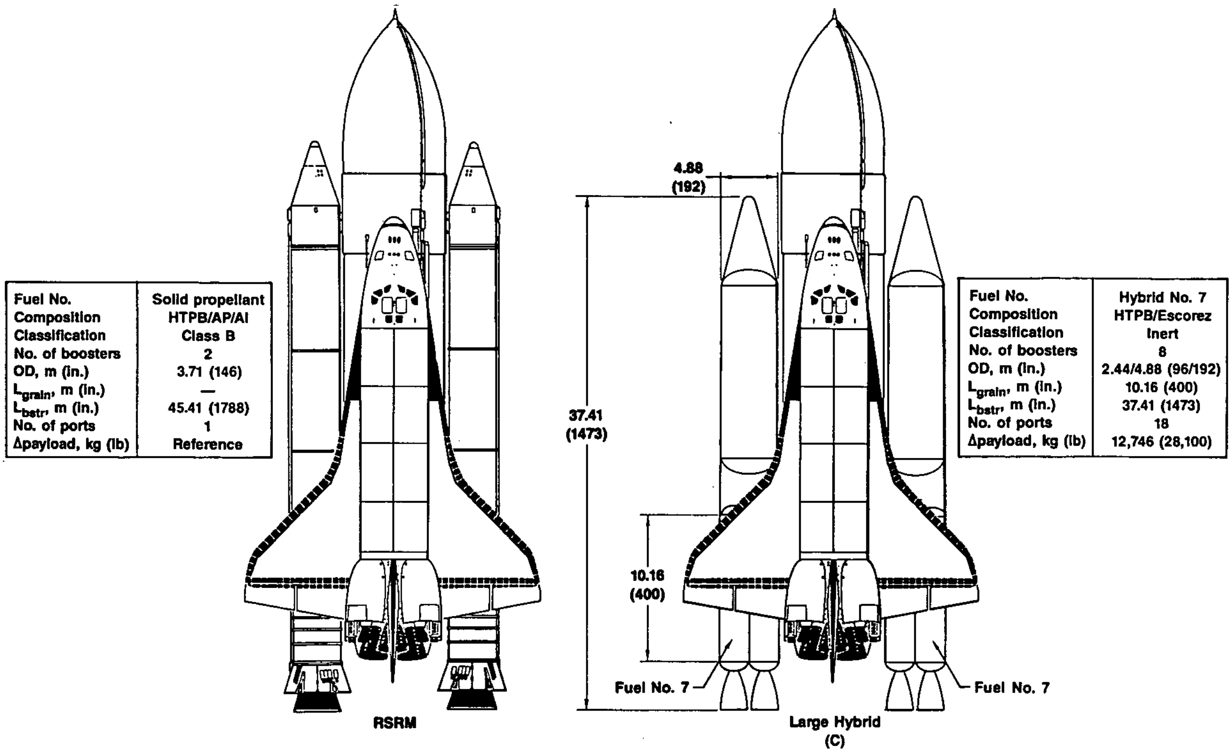


Fuel No.	Solid propellant
Composition	HTPB/AP/Al
Classification	Class B
No. of boosters	2
OD, m (in.)	3.71 (146)
L_{grain}, m (in.)	—
L_{bstr}, m (in.)	45.42 (1788)
No. of ports	1
Δpayload, kg (lb)	Reference

Fuel No.	Hybrid No. 7
Composition	HTPB/Escorez
Classification	Inert
No. of boosters	2
OD, m (in.)	4.57 (180)
L_{grain}, m (in.)	14.35 (565)
L_{bstr}, m (in.)	50.55 (1990)
No. of ports	34
Δpayload, kg (lb)	11,533 (25,425)

Fuel No.	Hybrid No. 8
Composition	HTPB/Escorez/AP
Classification	Class B
No. of boosters	2
OD, m (in.)	3.96 (156)
L_{grain}, m (in.)	17.42 (686)
L_{bstr}, m (in.)	56.26 (2215)
No. of ports	9
Δpayload, kg (lb)	9959 (21,955)

Fig. 3. Full-Scale Boosters

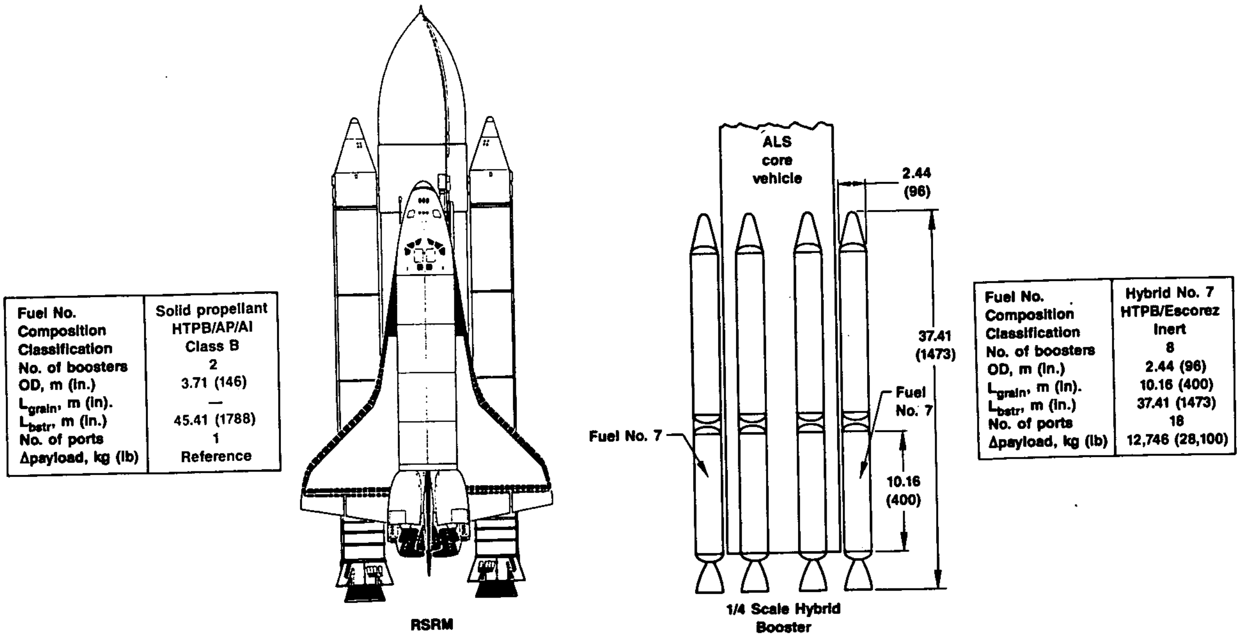


Fuel No.	Solid propellant
Composition	HTPB/AP/Al
Classification	Class B
No. of boosters	2
OD, m (in.)	3.71 (146)
L_{grain}, m (in.)	—
L_{bstr}, m (in.)	45.41 (1788)
No. of ports	1
Δpayload, kg (lb)	Reference

Fuel No.	Hybrid No. 7
Composition	HTPB/Escorez
Classification	Inert
No. of boosters	8
OD, m (in.)	2.44/4.88 (96/192)
L_{grain}, m (in.)	10.16 (400)
L_{bstr}, m (in.)	37.41 (1473)
No. of ports	18
Δpayload, kg (lb)	12,746 (28,100)

Note: Dimensions are in meters (inches)

Fig. 4. Full-Scale Booster/Quad Combustor



Fuel No.	Solid propellant
Composition	HTPB/AP/Al
Classification	Class B
No. of boosters	2
OD, m (in.)	3.71 (146)
L_{grain}, m (in).	—
L_{bstr}, m (in.)	45.41 (1788)
No. of ports	1
Δpayload, kg (lb)	Reference

Fuel No.	Hybrid No. 7
Composition	HTPB/Escorez
Classification	Inert
No. of boosters	8
OD, m (in.)	2.44 (96)
L_{grain}, m (in).	10.16 (400)
L_{bstr}, m (in.)	37.41 (1473)
No. of ports	18
Δpayload, kg (lb)	12,746 (28,100)

Note: Dimensions are in meters (inches)

Fig. 5. Quarter-Scale Hybrid Booster

TABLE II. HYBRID BOOSTER PERFORMANCE SUMMARY

Parameter	Booster Size		
	4.57-m (180-in.) Diameter*	3.96-m (156-in.) Diameter†	2.44-m (96-in.) Diameter*
Booster weight, kg (lb)	569,503 (1,255,546)	592,016 (1,305,179)	141,229 (311,358)
Length, m (in.)	50.5 (1990)	56.3 (2215)	37.4 (1473)
Δ payload, kg (lb)	11,532 (25,425)	9977 (21,995)	12,746 (28,100)
$\overline{O/F}$	2.64	1.84	2.76
$\overline{I_{sp}}$ (vacuum), N-s/kg (sec)	2965 (302.3)	2825 (288.1)	2947 (300.5)
$\bar{P}$, MPa (psi)	5.18 (750)	5.18 (750)	5.18 (750)
MEOP, MPa (psi)	7.01 (1027)	7.14 (1035)	7.27 (1053)
Mass fraction, %	85.3	86.1	86.6
Life cycle cost, $ x 10^6			
One launch per month	6008	Not determined	8757
One launch per week	18,468	Not determined	27,178
$ per pound payload			
One launch per month	2032	Not determined	2252
One launch per week	1943	Not determined	2082

* HTPB/Escorez fuel
† HTPB/Escorez AP/Al fuel

ORIGINAL PAGE IS
OF POOR QUALITY

TABLE III. FINAL DESIGN WEIGHT SCHEDULE - FUEL NO. 7 PUMP-FED LOX FLEXSEAL TVC

Item	Weight, kg (lb)			
	4.57-M (180-in.) Diameter		2.44-M (96-in.) Diameter One Quarter Scale	
Hybrid rocket motor	569,506	(1,255,546)	141,230	(311,358)
Fuel (including 6% residual)	138,069	(304,389)	33,645	(74,175)
Oxidizer - hybrid (including 0.5% residual LOX and 0.53% residual GOX)	355,791	(784,384)	90,690	(199,937)
Subsystems, recovery separation motors, standard structures	16,500	(36,377)	5151	(11,355)
LOX tank (composite with metal liner)	2703	(5958)	985	(2171)
Interstage structure	1425	(3142)	423	(932)
Ignition system	227	(500)	68	(150)
Motor case	26,912	(59,331) (steel)	2766	(6098) (composite)
Case insulation	5594	(12,335)	1794	(3955)
Flexseal nozzle	9867	(21,753)	2503	(5518)
Mass fraction	0.853		0.866	
Total propellant (fuel, oxidizer	497,621	(1,097,066)	125,242	(276,112)
Total inerts (does not include residual propellants)	72,838	(160,580)	18,210	(40,145)

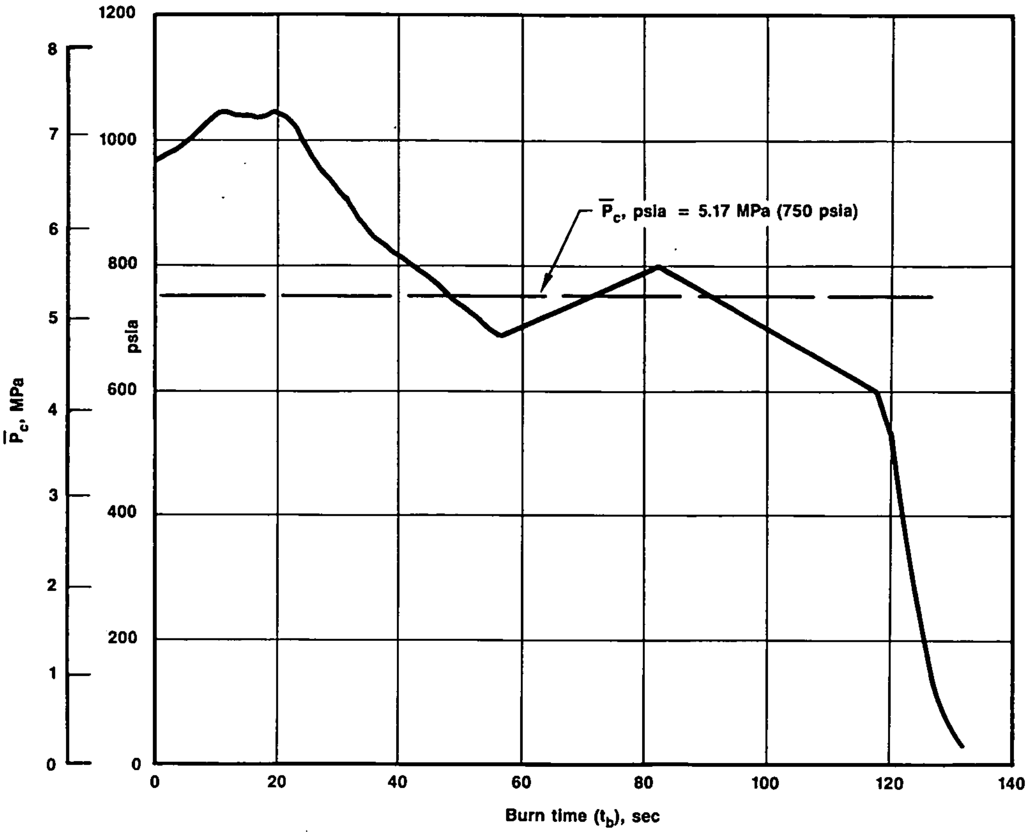


Fig. 6. Large Hybrid Chamber Pressure as a Function of Time

Fig. 7. Large Hybrid Booster Thrust-Time Schedule

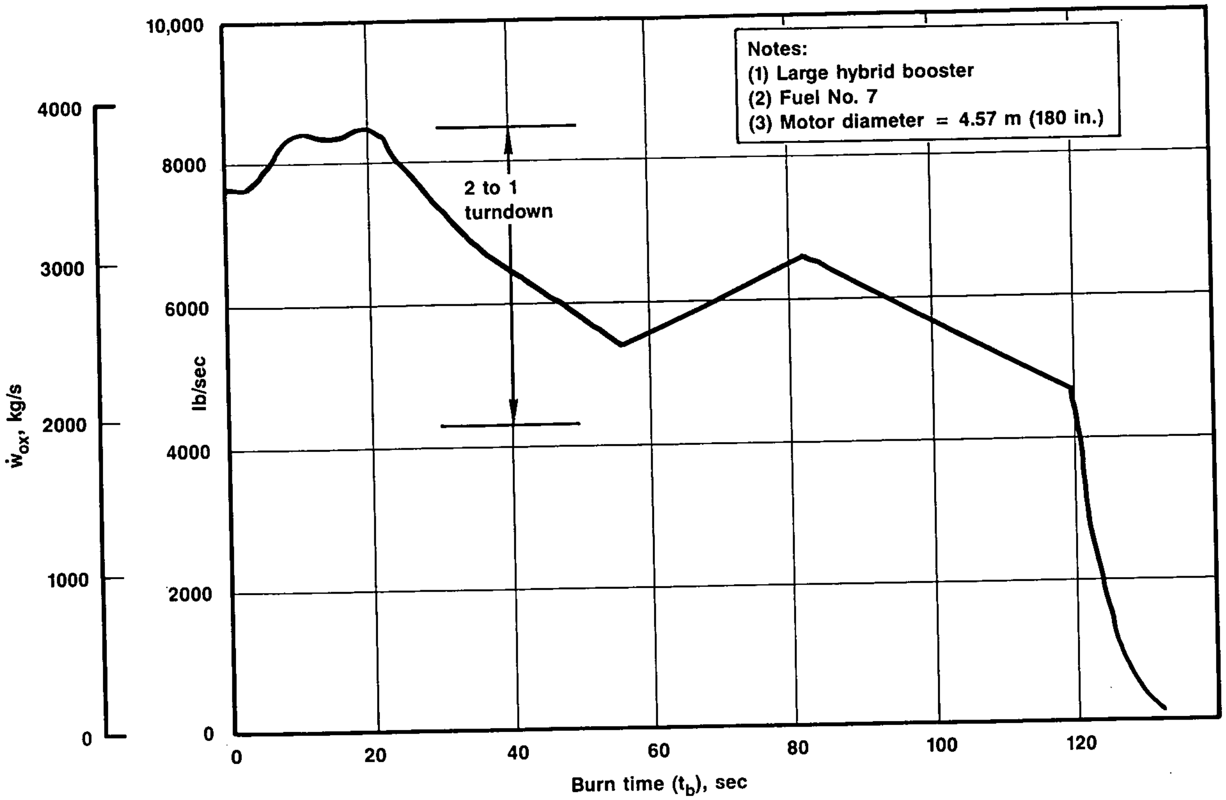


Fig. 8. Oxygen Flow Rate as a Function of Time

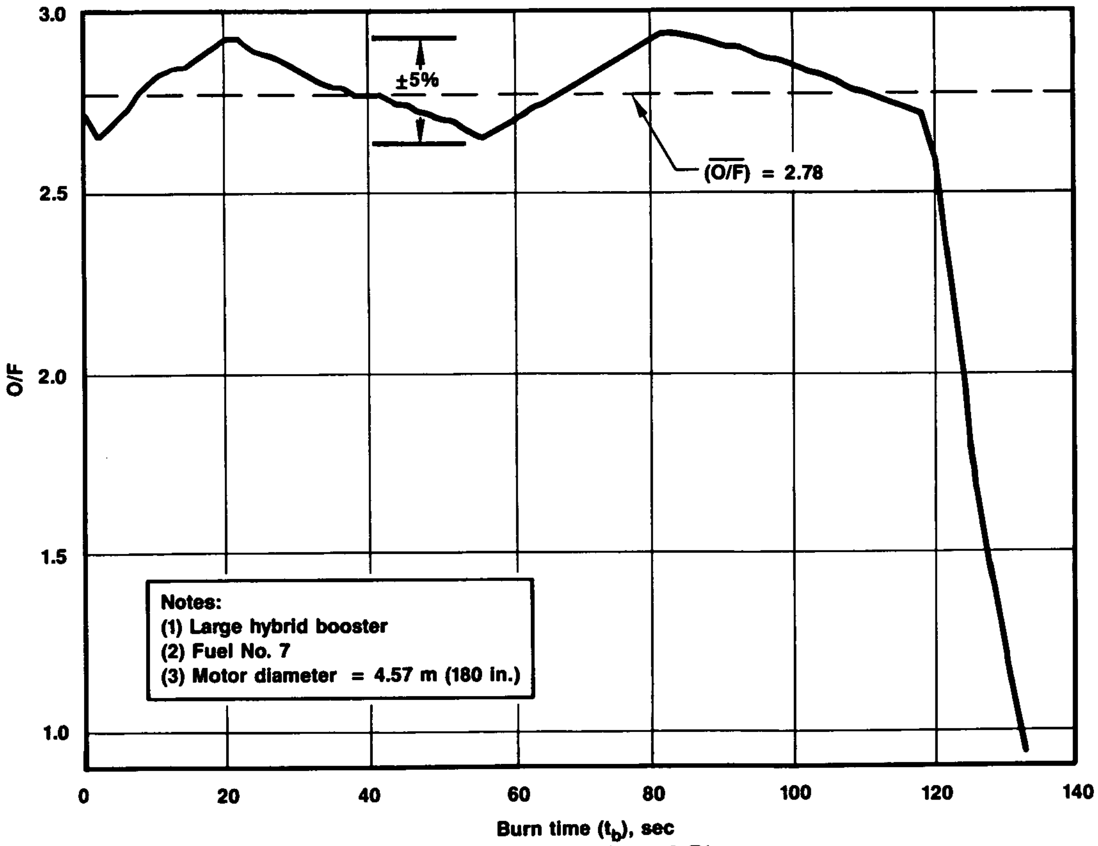


Fig. 9. Mixture Ratio as a Function of Time

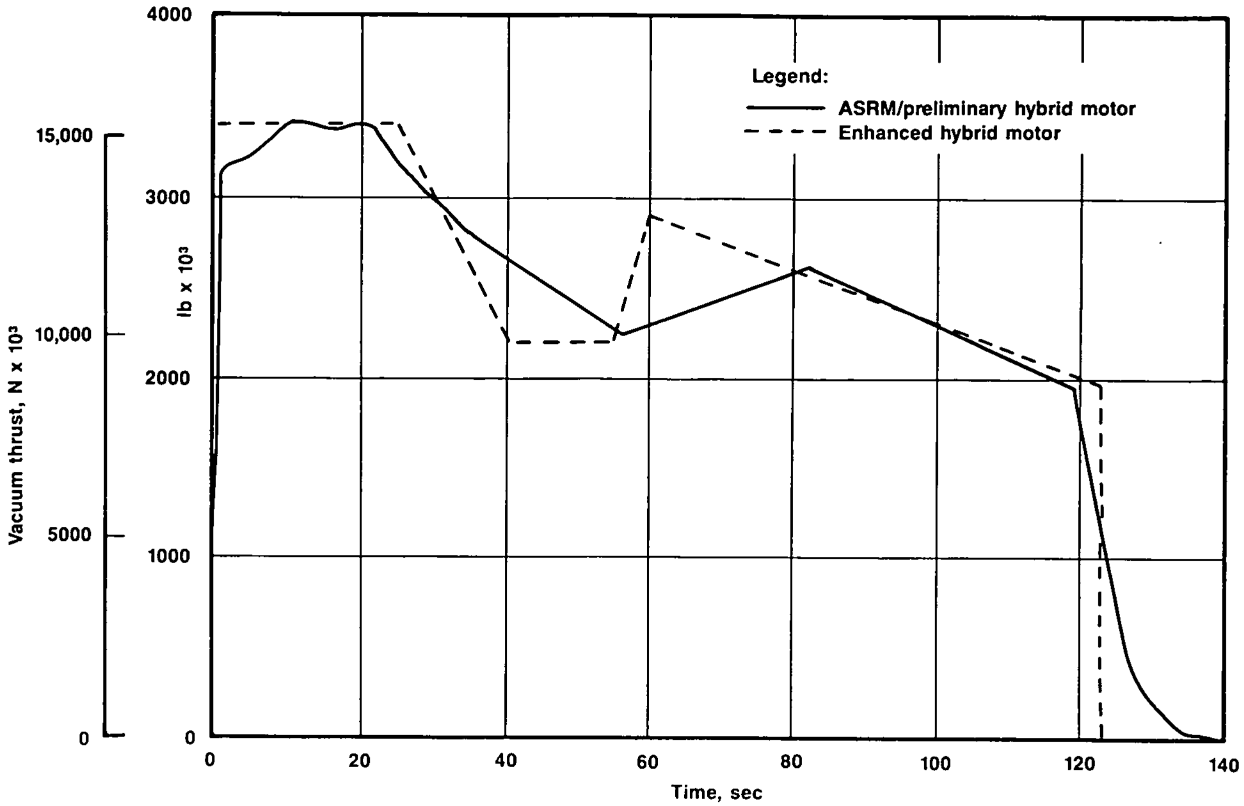


Fig. 10. Thrust Profile

TABLE IV. PERFORMANCE COMPARISON OF SOLID AND HYBRID BOOSTERS

Parameter	NASA's ASRM		Hybrid 1*		Hybrid 2†	
Burnout weight, kg (lb)	75,323	(166,059)	85,075	(187,558)	85,075	(187,558)
Expanded weight, kg (lb)	548,286	(1,208,763)	490,970	(1,082,404)	490,970	(1,082,404)
Total weight, kg (lb)	623,609	(1,374,822)	576,045	(1,269,962)	576,045	(1,269,962)
Effective I_{sp}, N-s/kg (sec)	2023	(267.45)	2929	(298.67)	2929	(298.67)
Total impulse, N-sec (lb-sec)	1.43803×10^9	(323,283,664)	1.43802×10^9	(323,281,603)	1.43802×10^9	(323,281,603)
MECO weight, kg (lb)	166,624	(367,344)	168,388	(371,231)	171,568	(378,243)
Payload improvement, kg (lb)			1763	(3887)	4939	(10,889)

* Duplicates ASRM's thrust profile
† Enhanced thrust profile that uses the HRB's throttleability