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Selecting Cooling Techniques for Liquid Rockets for Spacecraft

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Nomenclature

A_* = nozzle throat area, in.^{2}
A_c = combustion chamber cross-sectional area, in.^{2}
A_e = nozzle exit plane area, in.^{2}
F = thrust, lb
h = heat-transfer coefficient, Btu/hr-ft^2-°R
I_{sp} = specific impulse, lbf/(lbm/sec)
I_t = total impulse, lb-sec
k = thermal conductivity, Btu/hr-ft-°F
L^* = characteristic length, in.
P_c = chamber pressure, psia
P_{inj} = propellant injection pressure, psia
q/A = heat flux, Btu/hr-ft^2
r_i = inside radius, in., ft
T_g = gas recovery temperature, °R
T_w = wall temperature, °R
T_∞ = effective heat sink temperature, °R
t = wall thickness, in.
ΔT = temperature difference between gas and wall
V = velocity, fps
W_{fc} = coolant flow
W_T = propellant plus coolant flow, lb/sec
θ = time, sec
ϵ = exterior wall emissivity
α = effective thermal diffusivity, in.^{2}/sec
δ = char depth, in.
σ = Stefan-Boltzmann constant, Btu/ft^2-°R^4-hr

Subscripts

0 = initial condition
f_c = film coolant

Introduction

LIQUID-PROPELLANT rocket engines for spacecraft propulsion may range in thrust from a few pounds to many thousands of pounds, with burning times from fractions of a second to many minutes. Some of these engines may be externally mounted and free to radiate heat to space, and others may be buried within the spacecraft and completely insulated.[1, 2] The more promising propellants for space propulsion include both storable hypergolics and high energy cyrogenics.[3] This broad spectrum of engine characteristics and requirements results from the great variety of space maneuvers and vehicle sizes envisioned. Typical space maneuvers will include trajectory correction, orbital rendezvous, station keeping, lunar and planetary landing and takeoff.[4–9]

All rocket engines have one problem in common: The energy released by the propellants must be contained, and the thrust chamber and any surrounding structure must be protected and/or cooled. Concepts developed to cope with this problem, either singly or in combination, include regenerative cooling, radiation cooling, film or transpiration cooling, ablation, and inert or endothermic heat sinks. In addition to this primary requirement for cooling during firing, the design of thrust chambers for intermittent operation in space poses problems related to 1) storage in the space environment,[10–12] 2) starting and intermittent operation in a space vacuum,[12] 3) shutdown and postrun soaking in the

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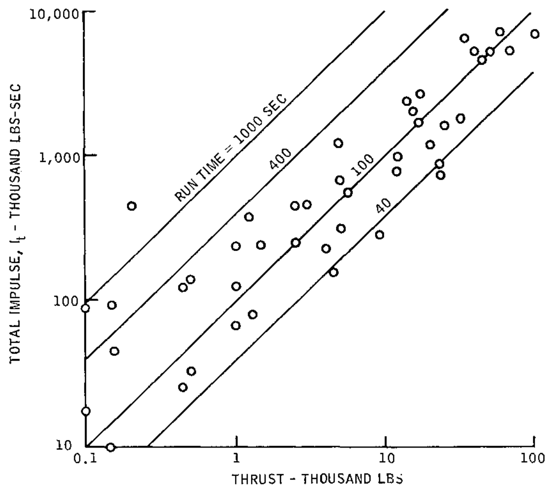

Fig. 1 Typical liquid-rocket engine thrust, time, and impulse values for space missions.

space environment,[14] and 4) entry at high velocity into planetary atmospheres.[7]

The point being made here is that many factors affect thrust chamber design, and there are a large number of cooling concepts from which to choose. The various cooling concepts may be used separately or combined in different ways for each of the thrust chamber components, such as the combustion chamber, the nozzle throat, and the expansion nozzle. The objectives of this paper are 1) to present the ranges of applicability and the limitations imposed on these cooling concepts, and 2) to propose a basis for selecting the most suitable methods for specific space applications. The results of a number of thrust chamber cooling design studies are presented graphically to fulfill partially the first objective and to facilitate the second.

A five-step approach to the selection of a cooling method is proposed:

1) Specification of propulsion requirements
2) Screening and review of various cooling techniques for applicability
3) Completion of a preliminary weight analysis for the applicable methods
4) Evaluation of propulsion performance penalties
5) Selection of one or more promising cooling techniques for a more complete design study

The selection procedure outlined can be carried out with a minimum of analysis and calculation. Optimization and final choice between two or more applicable thrust chamber designs may be based finally on factors beyond the scope of this paper. Many of the more detailed design considerations and cooling limitations are covered in a recent Marquardt report[15] and in separate reports on individual cooling techniques.[16–21]

Continued research and development in the areas of thrust chamber materials and rocket engine design will modify any specific conclusions drawn from these studies, but the approach to evaluating the different cooling concepts may continue to be useful.

Specification of Propulsion Requirements

The initial specification of the propulsion requirement may be quite general, e.g., an initial spacecraft mass and velocity change, or a thrust and burning time.[4–8] If these are the only restrictions given, several designs will have to be carried far enough to establish the advantage of one propulsion system over another. To get a preliminary design under way, as many as possible of the propulsion system requirements outlined below should be specified or bounded.

Mission

The function of the engine establishes several important cooling parameters, such as the engine location,[1] thrust level, burn time, and duty cycle. Of particular interest are the burn times, which for many missions range from 40–400 sec, with the majority of the maneuvers requiring burn times of less than 100 sec, as shown in Fig. 1.[15] It may also be desirable for the same engine to fulfill more than one mission or be reused on subsequent missions. Typical thrust-time histories for various maneuvers are shown in Fig. 2. The propulsion requirements, which evolve from such mission requirements, should be detailed as completely as possible in terms of the following:

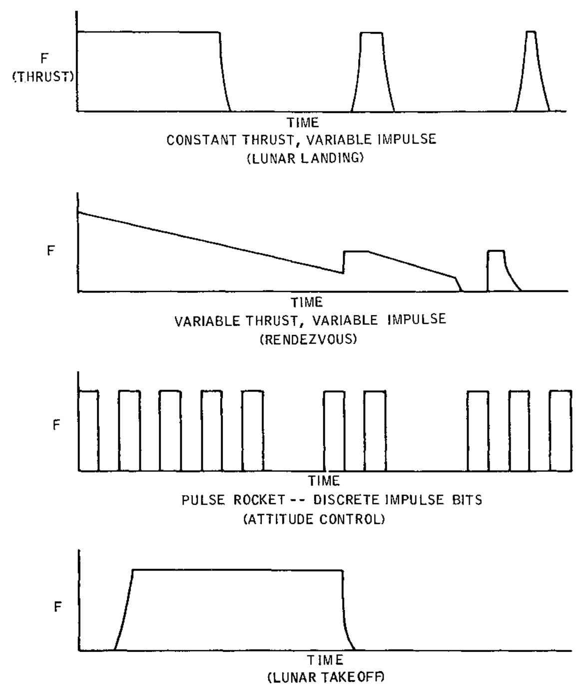

Fig. 2 Typical thrust-time plots for space engine missions.

Total impulse and/or velocity changes
Throttling range
Thrust level (as a function of time, if possible)
Maximum single run time
Number of starts and repetition rate
Cutoff accuracy and minimum impulse bit
Number of engines and vectoring requirement

Propellants

The choice of propellants may be based on a specific impulse requirement to accomplish a given mission.[22] The propellant choice (including O/F ratio or range), in turn, determines the combustion gas temperature and composition. Several propellants are excellent coolants, while others have little cooling capability,[23–29] and the combustion gas constituents vary widely in their compatibility with candidate thrust chamber materials.

Environmental and Operational Requirements

As many as possible of the following characteristics also should be specified with respect to their effect on the thrust chamber design. Many of these are also derived from the mission, of course; the groupings are arbitrary.

- Engine location in or on spacecraft
- Engine envelope limitations (affects choice of nozzle type, allowable expansion ratio, etc.)
- Exterior temperature limits or heat loss limits
- Pressure limits (or propellant supply pressure)
- Storage time in space
- Distance and attitude with respect to the sun during mission life
- Maximum acceleration and vibration loads
- Onboard nuclear emission
- Re-entry environment
- Reliability requirements
- Ground and inflight checkout requirements

Screening and Review of Various Cooling Techniques

Several of the requirements just specified directly affect cooling and may strongly favor one or more cooling methods while wholly eliminating others. Furthermore, the severity of heat fluxes will vary over a wide range in the combustion chamber, the nozzle throat, and the nozzle exit cone or skirt. Hence, the thrust chamber design may incorporate two or more basic cooling methods, either combined or applied separately to the different chamber components. Preliminary screening charts have been prepared in Tables 1 and 2. These charts indicate whether or not the propellant choice, or an engine operating requirement or range of application, may be a limiting factor in the choice of that cooling method. Since it is obviously impractical to cover all possibilities in such charts, some *arbitrary ground rules* have been assumed:

Table 1 Effect of propellant choice on cooling method selection

	Earth storable, hypergolic		Cryogenic			Space storable	
	N_2O_4/N_2H_4	N_2O_4/Aerozine 50	O_2/H_2	F_2/H_2	OF_2/H_2	OF_2/B_2H_6	OF_2/CH_4
$I_{sp(max)}$ at 300 psia	342	338	456	478	477	437	430
O/F	1.4	2.0	4.5	10.0	7.0	4.0	5.3
Flame temp., °R	5700	5700	5600	7300	6600	8000	7600
Principal products	N_2, H_2, H_2O	N_2, H_2, H_2O, CO	H_2, H_2O	HF, H_2	HF, H_2, H, H_2O	HF, H, H_2, BOF	HF, CO, H
Regenerative cooling cap'y of fuel (typ.), Btu/in.^{2}-sec. at 600 psia	12 (both for nucleate boiling, V = 30 fps)	6	15 (for convective cooling with H_2, V = 1580 fps, ΔT = 3000°F)	15	15	Unsuitable	
Open tube cooling	Unsuitable		With H_2, above 10,000 lb-thrust			Unsuitable	
Radiation cooling	Coated refractories at 3000°F		≤3300°F coated refrac.	>4000°F uncoated refrac.	P_c < 50 psia	For P_c <20 psia, or for nozzle skirts	
Heat sink cooling	For times ≤ 2 min, coated graphite, pyrographite, or silicon carbide		For times ≤ 2 min at low P_c; graphites or carbides			Limited run time; best materials unknown, probably pyrographite	
Ablative cooling	Silica-phenolics		Silica pheno.	Carbon pheno.	Carbon pheno.	Limited run times, throat insert required	
Film or transpiration cooling	May be required at high P_c		H_2 is a good coolant, but it reacts with graphite at high temperature.			Coolant capabilities of propellants unknown	

Table 2 Cooling method screening chart

Cooling method	Regenerative cooling	Radiation cooling	Ablation cooling	Film cooling (by fuel)	Transpiration (stored)	Open Tube (H_2)	Inert heat sink	Endothermic heat sink
For restarts	OK	OK	OK	Ltd.	Ltd.	OK	OK	Ltd.
For pulsing	No	OK	OK	No	No	Ltd.	OK	Ltd.
For throttling	Ltd.	OK	OK	Ltd.	Ltd.	OK	OK	Ltd.
Run time θ limit	None	hr	5–20 min	None	Ltd. by coolant	None	min	Ltd. by coolant
Pressure P_c limits	Ltd. by $P_{propell.}$	50 psia; or 90 psia, low thrust	Ltd. θ for throat	None	None	Possible tube-diam. limit	Ltd. θ	Ltd. θ
Vac. operation	Ignition problem	Possible coating life ltd.	Some residual thrust	Residual thrust	Residual thrust	OK	OK	Residual thrust
Att. re: sun	Trapped coolant heating	OK	Soak T ltd. to 500°F	OK	OK	OK	OK	Soak T ltd.
Meteoroids	May puncture	May erode coating	OK	OK	OK	May puncture	OK	OK
Exterior temp.	≤1000°F	≤3300°F	≤800°F	Can be ltd.	Can be ltd.	May → 1500°F	Can be ltd., may be >4000°F	Can be ltd.
External heat flux	Min.	Max.	Soak transient	Min.	Min.	Ltd. by coolant temp.	$f(\theta)$ & soaking	Can be ltd.
Adv. nozzle geometry	Ltd. by passage sizes	Ltd. by P_c and conf.	Throat erosion critical	OK	OK	OK, but ltd. by passages	Ltd. θ	Ltd. θ
Penalties	P losses	Big size, low P_c	Heavy, ltd. θ	Possible I_{sp} loss	Possible I_{sp} loss	Possible I_{sp} loss	Ltd. θ vs w.	Ltd. θ

1) Pressurized propellant feed.
2) For film cooling, the fuel is used as the coolant, whereas for transpiration cooling, a separate coolant is used, and its flow is controlled by heat flux in the chamber.
3) Hydrogen is the only coolant considered in the open tube concept.

From these charts, one or several thrust chamber design approaches may appear promising for a particular application. A more detailed review of the advantages and limitations of each cooling concept is presented below to facilitate preliminary evaluations and the preparation of a thrust chamber weight comparison. Possible penalties due to heat losses, mass addition, or nozzle erosion are also evaluated, because any requirement for extra propellant or coolant must be considered as part of the engine weight.

Regenerative Cooling

Regenerative cooling has long been the conventional method for cooling liquid rockets.[16] The coolant(s) (one or both of the propellants) is (are) circulated through cooling jackets or through contoured thin-walled tubes that form the chamber walls.[30] The metal walls operate at relatively low temperatures (<2000°F) compared to the gas temperatures (4500°–6000°F); heat fluxes range from 1–10 Btu/in.^{2}-sec.[31, 32] Basic advantages are: 1) continuous long run times are possible, 2) there is negligible heat loss from the propulsion cycle due to cooling, and 3) with modern fabrication techniques, the chamber structure is relatively light compared to typical uncooled structure.

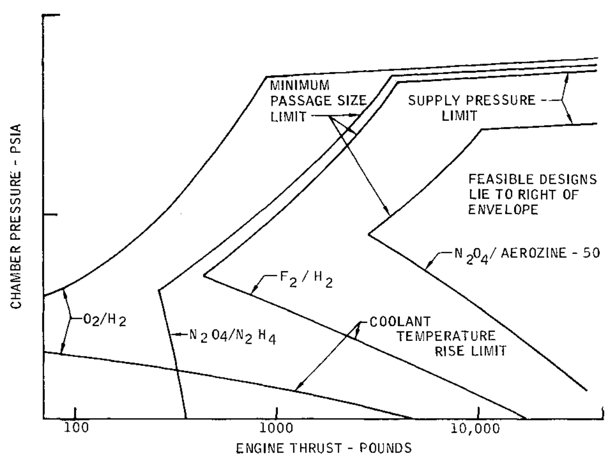

Fig. 3 Feasibility map for regenerative cooling with propellants O_2/H_2, F_2/H_2, N_2O_4/N_2H_4, and N_2O_4/Aerozine 50.

Three factors describe the nature of the operating limits for regenerative cooling: pressure, fabrication limits, and coolant heat capacity. The results of specific studies of chamber pressure limits vs engine thrust capabilities for various propellant combinations are shown in Fig. 3.

Three lines define the boundaries of feasibility in Fig. 3. The maximum chamber pressure line is fixed by the available propellant supply pressure, which must be greater than the sum of the chamber pressure, the propellant injector pressure, and the coolant circuit pressure loss. Propellant supply pressure, in turn, depends on the result of a system weight optimization, and whether the propellants are pump-fed or pressure-fed. The upper-left line which limits minimum thrust depends on the minimum practical coolant passage height at the nozzle throat, assumed to be 0.060 in. for this plot. (The heat flux capability of the liquid propellants is directly related to coolant pressure and velocity[23]; velocity may be increased by decreasing the flow area, subject to pressure loss and fabrication limitations.) The lower line which limits minimum thrust is determined by the allowable bulk temperature rise for the coolant. For propellants such as N_2O_4, the bulk temperature limit is the saturation temperature, whereas for fuels such as pentaborane, hydrazine or RP-1, the limit is the thermal decomposition temperature which would cause solids formation or an explosion;[23, 33, 34] e.g., the coolant-side wall temperature limitation for Aerozine 50 (0.5 N_2H_4–0.5 UDMH) is 500°–600°F.[16] For hydrogen, the limit is determined only by the required temperature difference (to achieve heat transfer) between the wall and the hydrogen; the lower curve in Fig. 3 is drawn for a hydrogen temperature limit of 1000°F. These coolant temperature limitations determine how much of the thrust chamber area can be cooled. In Fig. 3 the thrust limit curves are drawn for the case of cooling the chamber, the nozzle throat, and the expansion section to an area ratio of 5; greater relative areas can be cooled for engines falling above the line, and vice versa.[15]

Specific design conditions could alter the locations of these regenerative cooling envelopes, but the trends would remain the same. It is seen that regenerative cooling is best suited to high-thrust, pump-fed, main-propulsion engines. Current design studies[16] indicate that chamber pressures from 2000–5000 psia are feasible and that cooling limits may be extended further through the use of auxiliary film cooling and/or refractory metals.

Operational limitations are related to possible requirements for multiple starts, throttling, and space storage. Starting and stopping is feasible (e.g., Bell Agena engine), but response time and impulse accuracy are poor due to the time associated with filling and draining coolant passages; if the coolant isn't drained, then freezing in the space environment becomes a problem. In general, practical throttling ratios are $\leq$10:1 for hypergolic storable fuels[15] but may be as high as 30:1 for hydrogen. Referring again to Fig. 3, one may plot a line of engine thrust vs chamber pressure for constant engine size to determine (within the feasibility boundaries) what the maximum throttling range might be. One space storage aspect is the possibility of meteoroid penetration of the cooling jacket, but this aspect is so speculative that it can only be noted as a potential problem.

A preliminary thrust chamber weight estimate as a function of thrust, chamber pressure, and nozzle contraction and expansion ratio is shown in Fig. 4 for a thrust chamber composed of 0.010-in.-wall stainless steel tubes with conventional propellant manifolds and pressure banding, exclusive of the propellant injector and accessories.[30]

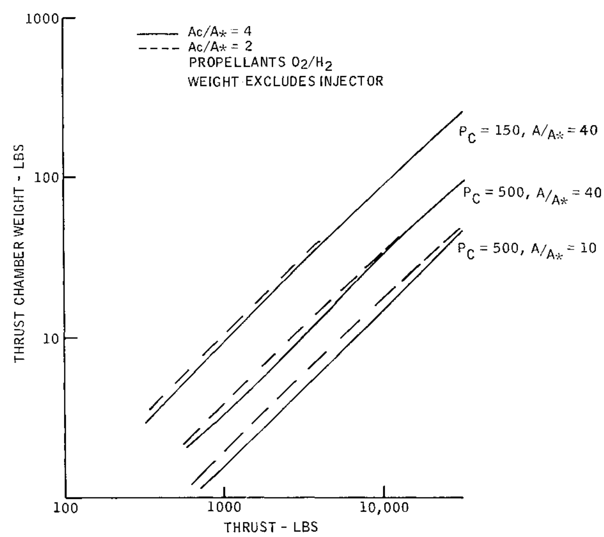

Fig. 4 Weight of regeneratively cooled thrust chamber of thin-walled stainless steel tubes.

Open Tube Cooling

In open tube or "dump" cooling, a small percentage of one of the propellants is passed through coolant passages and dumped overboard in an axial direction to produce thrust. This technique reduces the complexity of the engine design, but, generally speaking, it is attractive only for large engines (above 10,000-lb thrust) cooled by hydrogen. Attempts to cool only a critical part of a small engine with a liquid propellant encounter limitations due to the very small passage dimensions required. Operational limitations in space may be less severe than those for regenerative cooling, due to the independent characteristic of the cooling circuit, which is always vented to space. For large hydrogen-cooled engines, thrust chamber weights would be comparable to those for a regeneratively cooled chamber.

Radiation Cooling

The equilibrium temperatures reached by the walls of a refractory metal thrust chamber cooled only by radiation to the surroundings may be expressed in simplified form by the familiar steady-state equation

$$q/A = h(T_g - T_w) = \epsilon\sigma(T_w^4 - T_\infty^4)$$

A graphical presentation of this relationship for a N_2O_4/N_2H_4-UDMH system is shown in Fig. 5, where heat-transfer coefficients were determined from equations by Bartz.[35] From curves such as these, it is obvious that radiation cooling for an entire engine is feasible only when a low chamber pressure (<100 psia) can be tolerated (i.e., a low-thrust requirement). For such applications, radiation-cooled engines have the advantages of lightweight, simple structure, and long operating life. They are capable of high-frequency pulsing or variable-thrust operation and are not subject to dimensional change during operation.

Radiation-cooled engines of thrust levels less than 100 lb have been developed to run steady-state for over an hour at chamber pressures of 90 psia.[17] Experimental heat-transfer rates in small thrust chambers can be controlled by injector design[14, 23] (leading to combustion stratification) to permit chamber pressures well above theoretical limits. Approximate weights as functions of chamber pressure and thrust are shown in Fig. 6 for radiation-cooled thrust chambers of

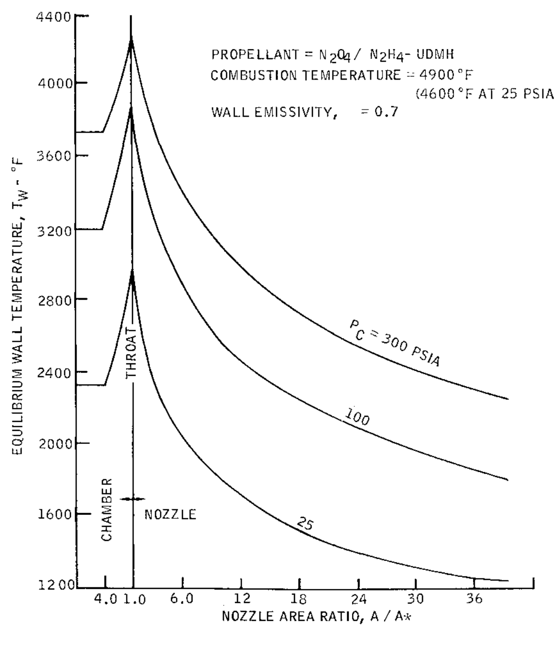

Fig. 5 Equilibrium wall temperatures for thin-walled radiation-cooled chamber and exit nozzle vs nozzle area ratio.

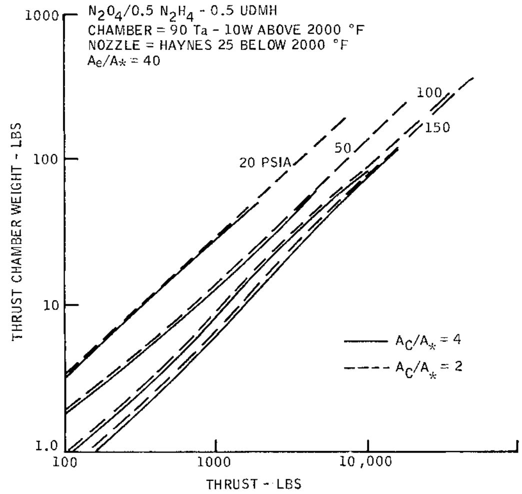

Fig. 6 Weight of radiation-cooled thrust chamber using 90Ta-10W and Haynes 25 alloys.

minimum gage tantalum alloy with an expansion skirt of Haynes 25 superalloy.

The characteristic limitation on radiation cooling is the availability of materials that can operate in the combustion environment at the equilibrium wall temperatures. The most common area of application in large engines is the expansion nozzle skirt; here the heat fluxes and static pressures are low, and surface areas are large, so that radiation-cooled skirt extensions can be employed to gain increased engine thrust at small increases in structural weight.

The firing duration limit for a radiation-cooled refractory metal is usually determined by the life of its protective coating. Engine runs of several hours have been demonstrated with molybdenum-disilicide-coated molybdenum chambers at metal temperatures above 3000°F.[17] The life of silicide coatings on other refractory metals is comparable, based on test samples in oxy-acetylene and plasma flames.[36–39] In space operation, the outside surface of the engine, which may be above 2500°F, is exposed to a hard vacuum. Over a prolonged period, the thin oxidation-resistant coating may evaporate.[38] For pulsed engines, the inside surfaces may be repeatedly subjected to vacuum while at high temperature. With current designs, however, burning times of less than 1000 sec have not posed an evaporation problem.

The propellants also establish limits of applicability, which depend on compatibility of the combustion gas with the motor walls or coatings, and on the combustion gas temperature. Most of the propellant combinations considered contain water vapor as the most reactive gas, but F_2/H_2 and OF_2/B_2H_6 products are primarily HF, H_2,[12] or other unusual species, many of which have not been completely evaluated as to their reactions with bare refractory metals and graphite. Silicide coatings are not compatible with oxidizers containing chlorine or fluorine. However, a radiation-cooled chamber of bare tungsten or pyrolytic graphite may be feasible for F_2/H_2 at some chamber pressures and mixture ratios.

The problem of meteorite penetration of the chamber or coating is still the subject of research and speculation. A current evaluation of the problem is that it is not critical, as attested to by the selection of radiation-cooled reaction-control engines for the Apollo and lunar landing manned spacecraft.[40]

A major consideration in the application of radiation-cooled engines is the protection of adjacent vehicle structure. In installations where the engine is essentially free to radiate to space but is near the vehicle structure, some radiation shielding and insulation on the structure may suffice. In other installations, the engine may be buried within the vehicle structure[1] and an exterior temperature limit specified. (For short run times this is an obvious application for an ablative or heat sink engine, but for runs ≥ 400 sec, engine weights may become excessive.) To permit use of a radiation-cooled engine in such cases, the concept of a fuel-cooled radiation shield has been developed[17] and successfully tested with both 25- and 100-lb thrust engines. This concept differs from a regeneratively cooled engine in that the cooling coils operate at normal fuel inlet temperatures, while the engine wall operates at temperatures up to 2800°F.

An alternate approach (at less severe combustion conditions) is to use a radiation shield, insulated on the back side, to successively reflect and hence emit the heat from the vehicle through the annular space between the engine and the shield.[41] For a shield temperature ≤ 1500°F, the increase in temperature of the thrust chamber wall will be less than 100°F.

Still another approach is to use a solid material, such as Teflon or lithium hydride, as an evaporating heat sink surrounding the thrust chamber. This technique minimizes the problem of postrun soaking temperatures, and radiation to the vehicle structure could be limited to any degree desired; the weight and volume of shield material would be proportional to run time and the protection desired. The feasibility of the concept has been demonstrated with a polyethylene shield around a radiation-cooled molybdenum chamber.[17]

Ablative Cooling (Char-Forming Plastics)

The thrust chamber can be lined with a solid matrix containing a substance that pyrolyzes to form gases which in turn act as a transpiration coolant. (This technique has been extensively used to protect re-entry vehicles.)[42–48] For engines using N_2O_4/hydrazine blends and O_2/H_2 propellants, the oriented-silica-fiber reinforced phenolics have consistently shown superior performances over many other ablative materials.[3, 15, 18, 49, 50] This has been attributed to the formation of a very viscous protective film of molten silica and to the favorable structural and insulating qualities of the char. Numerous thermodynamic processes are involved: heat absorption, heat conduction, resin depolymerization, silica-carbon reaction, transpiration cooling, evaporation, radiation, etc. A variety of analyses have been developed to account for these phenomena and facilitate calculation of charring and surface recession rates.[15, 18, 46, 51, 52]

An analysis of transient temperature response data on several ablative nozzle firings at the NASA Lewis Research Laboratory[49] and at Marquardt,[15] and a comparison with a slab heat conduction analysis[53] carried out with the IBM 704 digital computer, assuming only conduction heat transfer, revealed that the experimental char depth and transient temperature response could be predicted approximately, neglecting the other phenomena. The correlation of a large amount of data has produced the following relationship between char depth δ, an effective thermal diffusivity α, and run time θ:

$$\delta = 2.0(\alpha\theta)^{1/2}$$

The successful prediction of surface recession rates on so simple a basis has not been accomplished for any general case, but useful empirical correlations have been achieved for limited ranges of applications.[42]

Successful runs of ablative throats for 10 min or more have been reported,[50] but the throat erosion rates appear to be very sensitive to chamber pressure, engine size, and injector design. At the present time there is no general agreement on the best resin system or on silica reinforcement orientation. For small ablative thrust chambers, the use of a hard, nonablating, heat-sink type of throat insert has proved feasible for chamber pressures to 150 psia and run times to 10 min with the earth-storable hypergolics.[55]

It is recognized that the use of silica-phenolics with high energy propellants or at higher chamber pressures will impose drastic reductions in performance.[51] For ablative chamber applications involving fluorine containing oxidizers, the carbon reinforced phenolics have proved superior to the silica-phenolics.[54]

Operational considerations reveal certain advantages for ablative techniques: 1) there does not appear to be any inherent limitation on restart or multiple firing capability (but an additional char thickness allowance, up to twice, must be provided for repetitive pulse operation[55]); 2) they can be used at very high chamber pressures, except in the nozzle throat where erosion is critical (on the other hand, there is no significant reduction in charring during throttled operation, except at very low pressures); and 3) they are very stable and shock resistant, permitting handling and storage over a wide range of temperatures. Silica-phenolics have been extensively evaluated under heat, vacuum, radio activity, and ultraviolet radiation.[10]

A preliminary thrust chamber structural weight estimate may be made based on the forementioned char depth data correlation. The thrust chamber weights used in the example at the end of this paper were based on wall thickness in the chamber equal to total char depth plus a 0.35-in. margin for insulation and structural strength. Actual measured char depths in the expansion section of the nozzle are considerably reduced. For preliminary design purposes, the required wall thickness at the end of a 30:1 expansion section is about one-half that in the combustion chamber and at the downstream edge of the throat insert, with a linear thickness variation from the latter point. Char thickness behind a nozzle throat insert is about the same as along the combustion chamber ahead of the insert.[55]

Film and Transpiration Cooling

In film cooling, the fluid is introduced directly into the thrust chamber through slots or holes and directed along the walls. This layer of liquid or gas absorbs heat and thickens the effective boundary layer, thus reducing the heat flux to the walls.[56–74] Cooling films may be supplied from several sources. One source is a propellant injector designed such that a separate supply of fuel is injected along the chamber walls during firing, thus producing a fuel-rich boundary zone. Alternatively, a fraction of the fuel or oxidizer can be routed directly to slots or holes in the chamber wall just ahead of the critical nozzle cooling area. Or, a separate supply of coolant (liquid, gas, or solid) can be metered separately or controlled by the heat flux (causing melting, expansion, or evaporation) to provide flow through slots or holes in the nozzle surface. Cooling of this latter type has been studied for solid-propellant motor application.[75–77]

Transpiration cooling may be thought of as a special case of film cooling in which the slot or hole spacing becomes very small (porous surface). Many of the same design considerations apply, but the available analytical design equations differ.[57, 59, 78–80] Coolant sources are usually 1) pressure-fed fuel, water, hydrogen, or other coolant or 2) a material such as copper, lithium, or a subliming salt with which the porous refractory wall has been impregnated. Transpiration cooling is most applicable to one-shot, constant-thrust engines due to the problems of flow control and shutdown effects.

There are no inherent limitations on the coolant capability run time, or chamber pressure with either film or transpiration cooling. If one of the propellants (usually the fuel) or an inert fluid is used as a coolant at the nozzle throat, there may be a performance penalty (I_{sp} loss, where coolant flow is counted as added propellant flow in the I_{sp} calculation). However, coolant introduced well ahead of the nozzle throat may completely mix and react, thus causing no loss.

Application to pulsing engines is limited by possible coolant waste in starting and residual flow from coolant passages after shutdown. Plugging of film cooling passages or transpiration media may be caused by either thermal decomposition during operation or by freezing between firings.

Heat Sink Cooling

The primary limitation on this transient heating approach is the run time available before one of two limiting surface temperatures is reached: 1) the melting, subliming, or softening temperature at which the material would flow or erode rapidly, or 2) the temperature at which the oxidation or reaction rate with the combustion gases would be excessive. Heat sink materials should have high heat capacity, high thermal conductivity, high structural temperature limits, and compatibility with combustion gases. Pyrolytic graphite, isotropic graphite, and tungsten top the list for use with high-temperature propellants. Oxidation of these materials is the critical problem with all combustion gases containing CO_2 and H_2O. Surface coatings for graphite and tungsten offer only a partial solution to this problem, since available coatings are limited to temperatures of less than 4000°F.[36–38]

Effects of various parameters on heat-sink run time capability are shown by Fig. 7. A dimensionless heat-transfer parameter, hr_i/k, is plotted against a dimensionless run-time parameter, $\alpha\theta/r_i^2$, for various values of the wall-thickness to inside-radius ratio, t/r_i. Here θ is the time required to raise the wall temperature by an amount equal to half the initial temperature difference between the gas and the wall. The limiting or "envelope" curve represents the locus of points for which increased wall thickness could not increase the run time parameter; in other words, for a given value of hr_i/k, this curve gives the maximum value of $\alpha\theta/r_i^2$ achievable regardless of wall thickness. For example, if $\alpha\theta/r_i^2 = 0.1$ is desired, nothing would be gained by making t/r_i greater than 0.67, and the maximum hr_i/k that could be handled for that run time would be 3. From the individual curves, it can be seen that for the relatively thin-walled cylinders that one would consider acceptable for the combustion chamber ($t/r_i \rightarrow 0.1$), the run time parameter is inversely proportional to the heating parameter. For very thick walls, which might be acceptable only in the nozzle throat, i.e., working along the envelope curve, it is seen that the run time can be increased by a factor near 5 if the convective heat-transfer coefficient is halved.

The over-all conclusion from Fig. 7 is that the inert heat sink approach is applicable only for small engines with low to moderate chamber pressures (low h). The use of limited film cooling to extend run time may be attractive for special applications. Theoretically, run times for heat sink nozzles can also be extended through the use of endothermic materials, such as subliming salts, lithium compounds, and low-melting-point metals, which can absorb large amounts of heat through a phase change. The endothermic materials may be impregnated into porous refractory wall materials or used to back up the walls as an insulator as well as a heat sink.[51, 77]

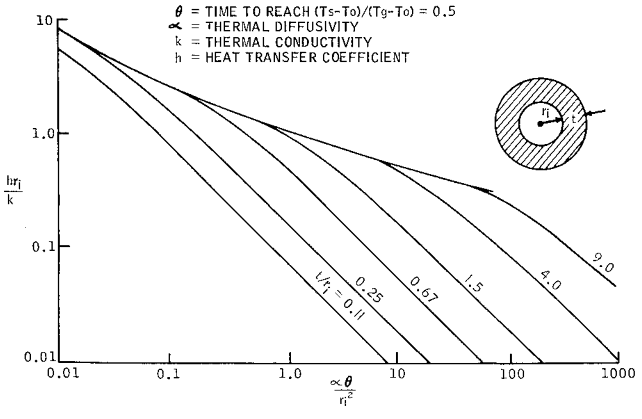

Fig. 7 Time required for heat sink throat insert to reach a surface temperature of $(Ts\text{-}T_0)/(Tg\text{-}T_0) = 0.5$.

Inert heat sinks would be suitable for pulsing and throttling operation within their practical total run times. Thermal stress and oxidation would be the important design considerations. Endothermic heat sinks would be most attractive for short runs at high chamber pressure and gas temperature.

Propulsion Performance Penalties

Since thrust chamber weights may be as little as 1% of the total propellant weight, it can be readily appreciated that a 1% loss in effective I_{sp} due to the cooling technique employed could override any chamber weight comparisons made for different cooling techniques. For this reason, a number of potential performance loss factors are examined below.

Losses due to Film and Transpiration Cooling

If a boundary-layer film of liquid or gas flows through a rocket nozzle throat at a temperature different than the main bulk of exhaust gas, the net thrust of the engine will be less than that which would result if these gases had been thoroughly mixed while retaining the same over-all total enthalpy.[15] This calculated gas stratification effect is independent of the effective chemical combustion efficiency. The magnitude of this effect on I_{sp} is presented for various boundary-layer film temperatures and film thicknesses in Fig. 8. Experimental data in Fig. 9, reported by Welsh,[83] confirm a performance loss approximately equal to the percentage of coolant flow. For preliminary design, this is a recommended value to use.

Losses due to Throat Erosion

Nozzle throat erosion, if controlled and predictable, could be acceptable in some engine applications. The effect on thrust, propellant flow rate, and I_{sp} have been calculated for throat enlargements up to 25%. For fixed-area injectors and fixed propellant supply pressure, propellant flow and engine thrust would increase while I_{sp} performance would decrease. For a 40:1 expansion nozzle, an I_{sp} loss of only 0.5% would be incurred for as much as 10% increase in throat area, if the aerodynamic performance of the nozzle contour did not deteriorate (Fig. 10).

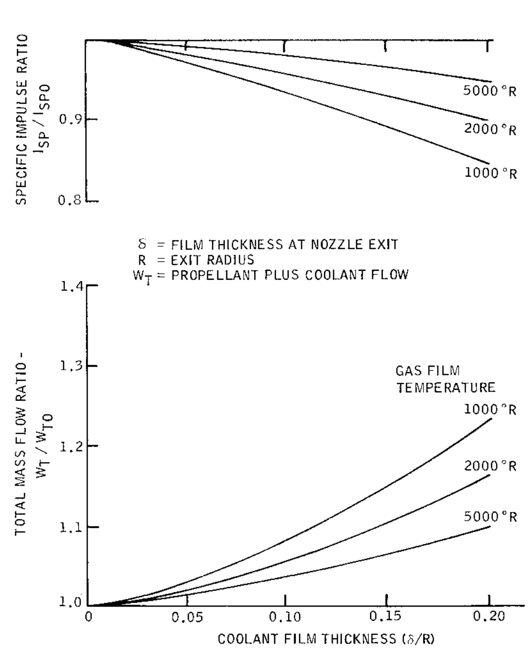

Fig. 8 Effect of gaseous cooling film on rocket engine specific impulse.

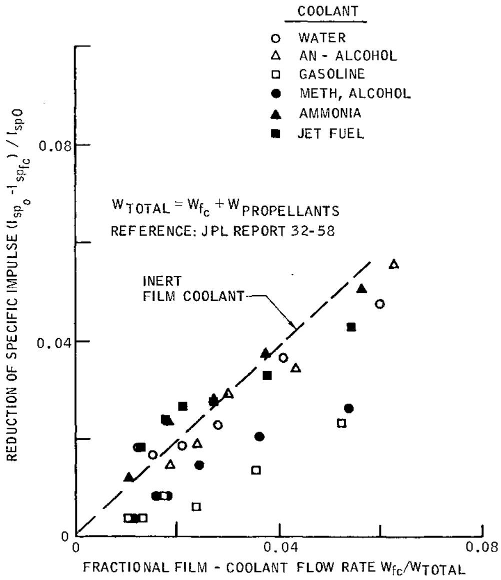

Fig. 9 Experimental data on reduction in specific impulse with liquid film cooling.

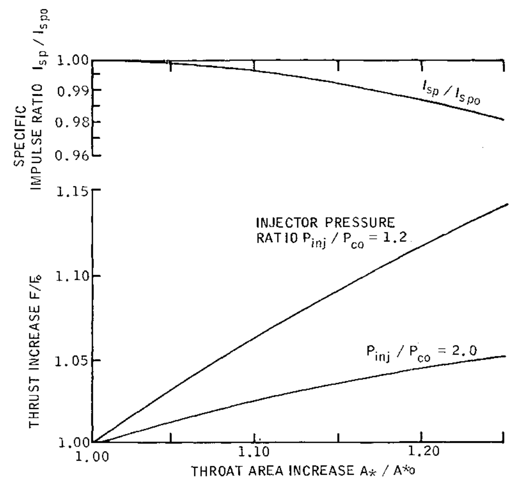

Fig. 10 Thrust and specific impulse variation due to rocket nozzle throat enlargement.

Heat Losses and Pressure Losses

Heat losses from combustion gases to chamber walls and the pumping energy required to overcome pressure losses in propellant and coolant lines result in an I_{sp} loss equal to one-half of the ratio of the energy loss to the total gas enthalpy. For example, in a typical 2000-lb thrust radiation-cooled engine, the total heat flux lost through the combustion chamber walls would be 72 Btu/sec. This is approximately 0.6% of the total gas flow enthalpy. Hence, the I_{sp} loss due to radiant heat transfer from the thrust chamber would be 0.3%.

Residual Thrust in Ablative Engines

After an ablative thrust chamber has been stopped, the heat stored in the charred phenolic and silica reinforcement soaks into the virgin material. Thermal degradation continues (for perhaps 100 sec) until the mean temperature of the char is reduced to $\sim$500°F. The weight of gas generated due to charring is $\sim$15% of the weight of ablative material charred. For an assumed average postrun gas temperature (e.g., 1100°F) and pressure tail-off, a postrun impulse can be calculated; e.g., 0.062 in. of charring in a 100-lb thrust engine gives 3 lbf-sec, equivalent to a 30-msec pulse, which may or may not be greater than the desired minimum typical impulse bit. The magnitude of this effect decreases for larger engines.

Design Study Examples

Variable-Thrust, Deep-Space Engine

Specification of propulsion system

Mission requirements: Midcourse propulsion in deep-space environment only; thrust range, 500–2000 lbf; run time, 725 sec; space starts, 8; and propellant mass, 3480 lb. Propellants: N_2O_4/(N_2H_4 + 10% EDA). Engine configuration: A single engine with convergent-divergent nozzle and no inherent envelope limitations (hence a 40:1 expansion ratio) was postulated.

General applicability screening

Scanning Tables 1 and 2 and reviewing factors relative to run time, restarts, engine envelope, and the chosen propellants, the following cooling concepts appear to be applicable: regenerative, radiative, ablative, and combinations of these. Film cooling was ruled out due to its inherent complexity and performance penalty that put it at a disadvantage when radiation and regenerative cooling are both possible.

Weight analysis

Preliminary thrust chamber weights were calculated for a range of chamber pressures as shown in Fig. 11.

Factors affecting final choice

The long run time of 725 sec and the 4:1 throttling range are the most demanding requirements. The lightest chamber design shown by Fig. 11 is the regeneratively cooled chamber operating at 250 psia at the 2000-lb thrust level, but two factors that require further study are the requirements for higher propellant supply pressure and purging of the cooling passages after shutdown. The second choice could be either the radiation-cooled engine or the ablative engine with a radiation-cooled skirt. The choice may be based on a sys-

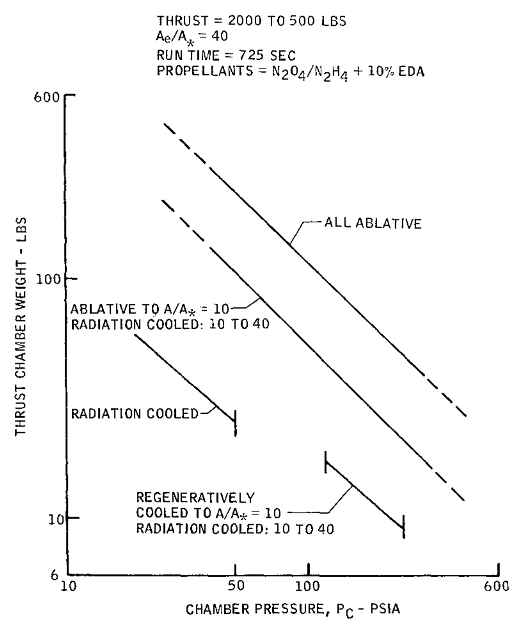

Fig. 11 Thrust chamber weights for long run, throttling engine.

tem study that would include possible engine envelope restrictions and propellant supply system weights. Meteoroid effects could have an effect on chamber design choice if more data were available.

Constant-Thrust, Oxygen-Hydrogen Space Engines

Specification of propulsion system

This study was made primarily to demonstrate the variation of cooling method applicability with thrust level (constant for given engine) and run time for a minimum weight thrust chamber. The ranges of parameters studied were: thrust range, 20–19,000 lb; burning times, 3–1000 sec; and number of starts, 1. Maximum thrust chamber pressures: radiation cooled, 50 psi; ablative cooled, 150 psia; heat sink, 150 psia; and regenerative, see Fig. 3.

Results

Figure 12 shows that on a weight basis, the best applications for the cooling methods shown are: radiation—low thrust, long run times; ablative—low thrust, run times from 10–300 sec; regenerative—high thrust, medium to long run times; and heat sink—short run times.

Combined Cooling Techniques and Advanced Concepts

There are several propellant systems for which no completely satisfactory cooling technique for reasonable run times has yet been developed. Examples are fluorine-based oxidizers, such as OF_2 and ClF_3, used with fuels containing such metals as boron, aluminum, beryllium, and lithium. These propellants give combustion gas temperatures in the 6000°–8000°F range,[15] and the combustion products are usually highly erosive and corrosive on the available refractory metals and carbides. Throat heat fluxes may fall in the 15–25 Btu/in.^{2} second range for a chamber pressure of 600 psia (and increase with pressure, of course). At these conditions, the very best inert heat sinks would reach temperatures of 5000°F in less than 20 sec. The other cooling techniques, which do not involve a performance loss such as regenerative, ablative, and radiation, will not do the job alone; some form of film or transpiration cooling is required.

When film or transpiration cooling is required, the objective is to minimize the required coolant flow, which depends on the surface area to be cooled and also the wall temperature. Therefore, the engine surfaces should be held at the highest temperatures consistent with structural integrity. Materials with the highest temperature capabilities are the graphites, tungsten, and the carbides of hafnium and tantalum. Structurally, graphite and tungsten are capable of operation above 5000°F[17] (the structural capability of the carbides has not been demonstrated), but all of these materials are subject to oxidation and erosion by the combustion gases even at 5000°F, so that they must be cooled to somewhat lower temperatures.

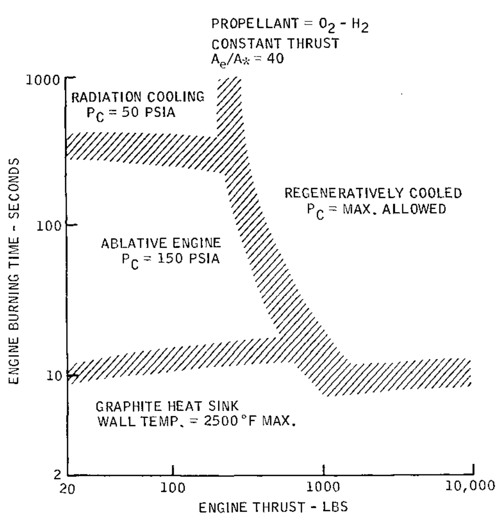

Fig. 12 Thrust vs burning time envelopes for minimum-weight space engines.

Most advanced cooling studies now in progress[51, 54, 77, 84–86] are related to ways of generating coolant films either on a transient or steady-state basis. Development problems lie in the areas of refractory material formulation, nozzle design and fabrication, coolant selection and supply techniques. Particular problems include passage plugging by coolant or combustion products, coolant distribution, starting and shutdown phenomena, limits on run time and thrust variability, and thermal expansion and sealing provisions.

Combined cooling concepts which may be promising, but which either have not yet been demonstrated or have been demonstrated only for short run times, include:

1) Porous refractories impregnated with lower melting metals or endothermic solids such as a subliming salt[77]
2) Porous throat inserts backed by a reservoir of endothermic heat sink material which absorbs heat in gasification[75, 76] and provides a transpiration cooling effect
3) Sacrificial inserts ahead of the nozzle throat
4) Coolant in a separate liquid or gas reservoir which is pumped directly to the nozzle
5) Liquid-metal reservoir to supply convective coolant to the back side of a thin-walled refractory metal nozzle[86]
6) Radiation-cooled heat sink of pyrolytic graphite[51]
7) Film-cooled heat sink to extend inert heat sink run times with a minimum performance penalty
8) Film-cooled convective nozzle with coolant injected upstream of throat after being used to cool the throat convectively

The limitations of these cooling concepts have not been established. Continued research is required in the development of refractory materials, in the development of optimum film and transpiration coolant supply systems, and in experimentally defining the actual combustion environments.

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Prediction of Design Reliability of Very Large Solid-Rocket Motors

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A technique is presented for predicting the a priori design reliability of a large solid-rocket motor. It is postulated that the motor reliability R_m is a function of its structural reliability R_s and its performance reliability R_p. The structural reliability, which also can be termed the probability that the structure will successfully retain the motive elements for the intended duration within the environmental envelope, is computed using the distributions of environmental stresses and material strengths. Performance reliability, or the probability that the motor will perform within established performance limits, is estimated using computer-simulated firings of the motor. The independent computer input variables are expressed stochastically and used in conjunction with a Monte Carlo type of random sampling. Any number of such "computer firings" are run, and the performance reliability calculated is based on the number of times specified performance parameters have been satisfied. An outline is also given on the use of the prediction technique for purposes of design tradeoff and system optimization studies.

Introduction

DURING the history of solid-rocket motor development, numerous attempts were made to establish a measure for the rate of program progress toward a target fixed by time. These attempts were fostered by customer requirements to establish success probabilities of the related weapon system. Two principal methods evolved, namely, the statistically demonstrated reliability with an associated confidence level and a nonstatistical measure of reliability based on learning curve concepts.

The statistical demonstration of reliability with the customary confidence limits (90–95%) requires large sample sizes even for moderate values of reliability target values. For man-rated NOVA-type vehicles, where single motor reliability requirement in a cluster may be as high as 99%, it would be necessary to test fire without failure 230 motors (90% confidence) or 295 motors (95% confidence). The limitations of time and cost obviously prohibit such test programs.

The second method, based on learning curve concepts, requires the definition of four critical measures: 1) unit of measurement, 2) value of the measure at onset of development, 3) predicted value for the rate of growth, and 4) technique for estimating program status at a given time. Many

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