TECHNOLOGY CONSIDERATIONS FOR MAIN ENGINE SPECIFIC IMPULSE

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TOPICS TO BE DISCUSSED

- IMPORTANCE OF SPECIFIC IMPULSE
- SPECIFIC IMPULSE CONSIDERATIONS IN CYCLE SELECTION
- SOURCES OF SPECIFIC IMPULSE INEFFICIENCY
- COMBUSTION PROCESS
- THRUST CHAMBER COOLING PROCESS
- NOZZLE EXPANSION PROCESS
- SPECIFIC IMPULSE PREDICTION TECHNOLOGY STATUS

TECHNOLOGY CONSIDERATIONS FOR MAIN ENGINE SPECIFIC IMPULSE

INTRODUCTION

The space shuttle vehicle is being designed for requirements not considered on previous man-rated rocket launch systems. Criteria such as airline-type checkout, maintenance, and turn around operations for a fully recoverable vehicle impose stringent performance requirements on the propulsion system. The specific impulse value required for mission success has resulted in selection of a high chamber pressure, staged combustion cycle and a high energy cryogenic propellant combination. Even with these favorable decisions from a performance standpoint, the NASA Phase B Work Statement [1] specifies a minimum delivered specific impulse which is in excess of 96% of theoretical at the 3000 psia chamber pressure design point. Consequently, the highest possible specific impulse efficiency must be realized from each engine component.

Another factor which makes specific impulse an extremely important design parameter is the sensitivity of payload to a reduction in delivered specific impulse. This effect is illustrated quantitatively by the data presented in Figure 3, which is based on mission analysis data for a typical space shuttle vehicle. It shows almost a 3% reduction in allowable payload for each second loss in specific impulse. Some vehicle studies show an even larger effect of specific impulse on mission payload. Although these data are preliminary at this state of development, the trend suffices to point out the extreme importance of achieving the specific impulse requirement.

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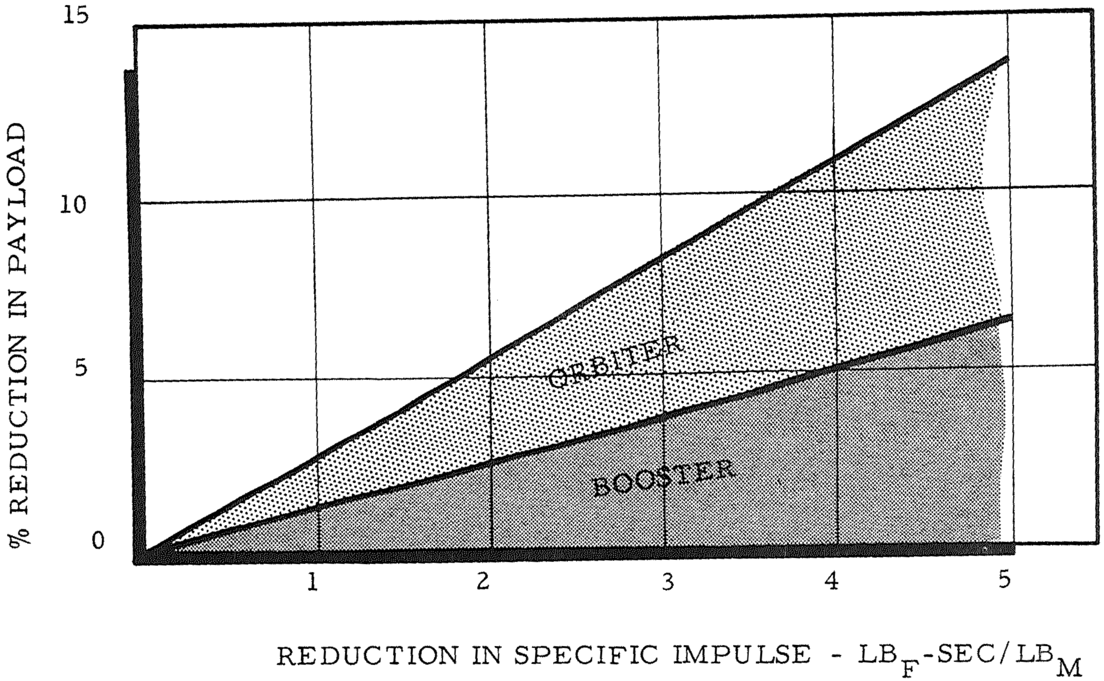
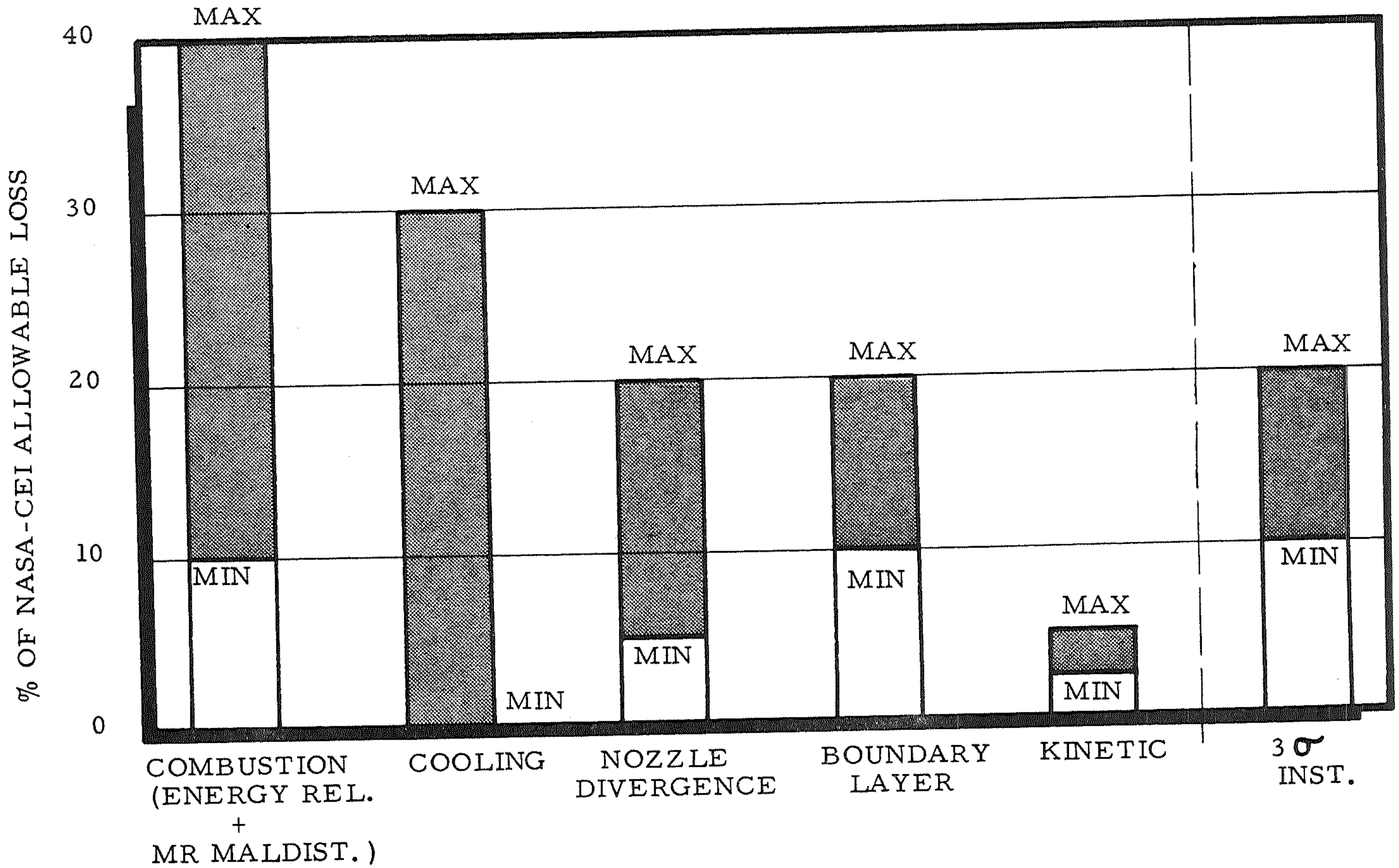
[1] Space Shuttle Main Engine, Statement of Work - Phase B, George C. Marshall Space Flight Center, NASA, Feb. 16, 1970

EFFECT OF SPECIFIC IMPULSE ON MISSION PAYLOAD



THRUST CHAMBER SPECIFIC IMPULSE LOSSES



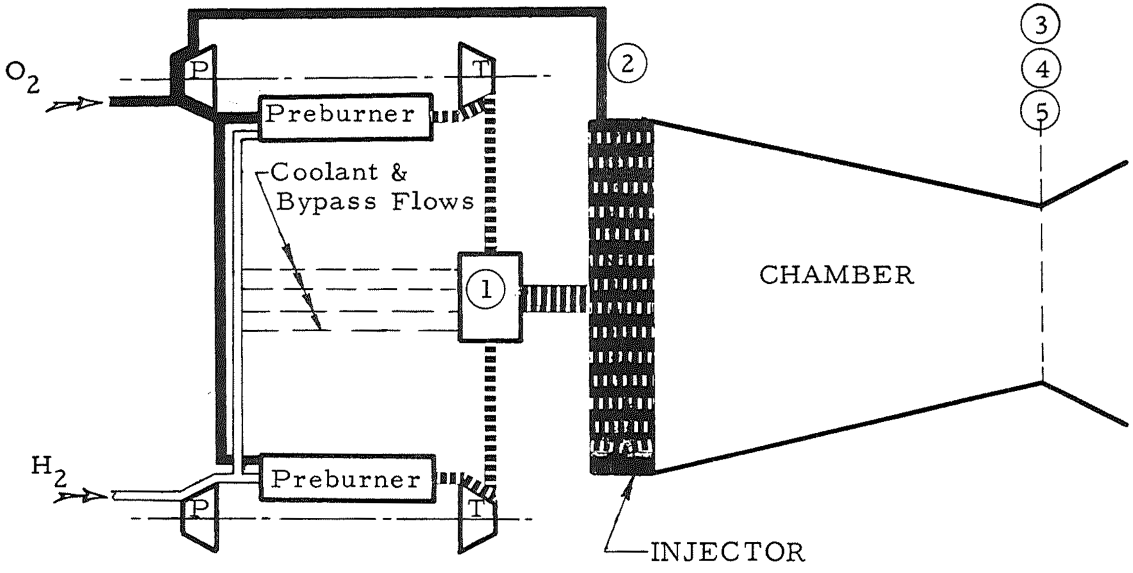
CRITERIA FOR HIGH COMBUSTION EFFICIENCY

Five design criteria must be satisfied to assure attainment of high combustion efficiency on the space shuttle main engine. These are listed on Figure 5 along with a schematic drawing indicating the location where each criterion must be satisfied.

Preburner gas homogeneity must be satisfied prior to injection into the thrust chamber. In addition to possibly different mixture ratio preburner gases, significant quantities of coolant and/or bypass flows of hydrogen at various energy levels are introduced into the fuel-rich gaseous propellant stream. This flow must be mixed prior to injection into the thrust chamber in order to prevent mixture ratio maldistribution performance effects. Next, both propellants must be uniformly distributed across the injector face so that mass and mixture ratio gradients are not significant. Then the liquid oxygen must complete the supercritical vaporization process within the chamber. Also propellant mixing must be completed on a fine basis so that the chemical reaction process can go to completion on a molecular scale within the chamber.

CRITERIA FOR HIGH COMBUSTION EFFICIENCY

1. PREBURNER GAS HOMOGENEITY
2. BOTH PROPELLANTS UNIFORMLY DISTRIBUTED
3. COMPLETE LIQUID OXIDIZER VAPORIZATION
4. COMPLETE PROPELLANT MIXING
5. COMPLETE CHEMICAL REACTION



TECHNIQUES FOR ACHIEVING HIGH COMBUSTION EFFICIENCY

Although the combustion criteria represent difficult design requirements for the engine designer, there are both analytical and experimental techniques available which will permit him to accomplish this task. Some of the primary techniques are tabulated on Figure 6 for each combustion criterion.

Analysis of the combustion process must still depend to some extent on empirical techniques. As noted on Figure 6, those criteria which are rate limiting can be characterized by cold flow experimental techniques. These techniques use simulated propellants which are non-reactive and permit mixing and flow distribution processes to be directly measured. Cold flow testing has been demonstrated on numerous engine development programs and provides an economical method for verifying that important design criteria have been met early in the development program. Proper utilization of this experimental technique will greatly assist in the achievement of high combustion efficiency on the space shuttle main engines.

TECHNIQUES FOR ACHIEVING HIGH COMBUSTION EFFICIENCY

CRITERIA	ANALYTIC TECHNIQUE	EXPERIMENTAL TECHNIQUE
PREBURNER GAS HOMOGENEITY	GEOMETRIC & FLOW SIMILITUDE	GAS-GAS COLD FLOW TESTING
UNIFORM PROPELLANT DISTRIBUTION	MANIFOLD FLOW DISTRIBUTION ANALYSIS	GAS/LIQUID COLD FLOW TESTING
PROPELLANT MIXING	1. TURBULENT MIXING MODEL 2. SPRAY CORRELATIONS	GAS/LIQUID COLD FLOW TESTING
*OXIDIZER VAPORIZATION	SUPERCRITICAL DROPLET VAPORIZATION MODEL	HOT FIRE TESTING
*CHEMICAL REACTION	TURB. MIXING AND REACTION MODEL	HOT FIRE TESTING

* PROBABLY NOT RATE LIMITING ON THE MAIN ENGINE

GAS COLD FLOW TEST INSTALLATION

An example of the type of cold flow installation for testing the preburner gas flow distribution is shown in Figure 7. This particular installation was used on the ARES Program [4] to assure that the preburner gases were uniformly distributed immediately downstream of the vane type thrust chamber injector. The ARES was a high pressure, staged combustion cycle, transpiration-cooled engine developed by Aerojet for the Air Force. Since only one fluid was involved in the cold flow experiment, pressure profile instrumentation was sufficient to determine mass distribution.

A test to determine preburner gas homogeneity would of course, require measurements which can be related to mixture ratio distribution. To accomplish this an additional parameter such as temperature distribution can be used to infer the uniformity of the mixture; or more directly, the mixture ratio distribution can be measured with a mass spectrometer using two cold gas simulants of varying composition. In setting up this type of experiment, care must be exercised to maintain simulitude with the actual engine geometry, flow rate ratio and static-to-dynamic pressure relationships.

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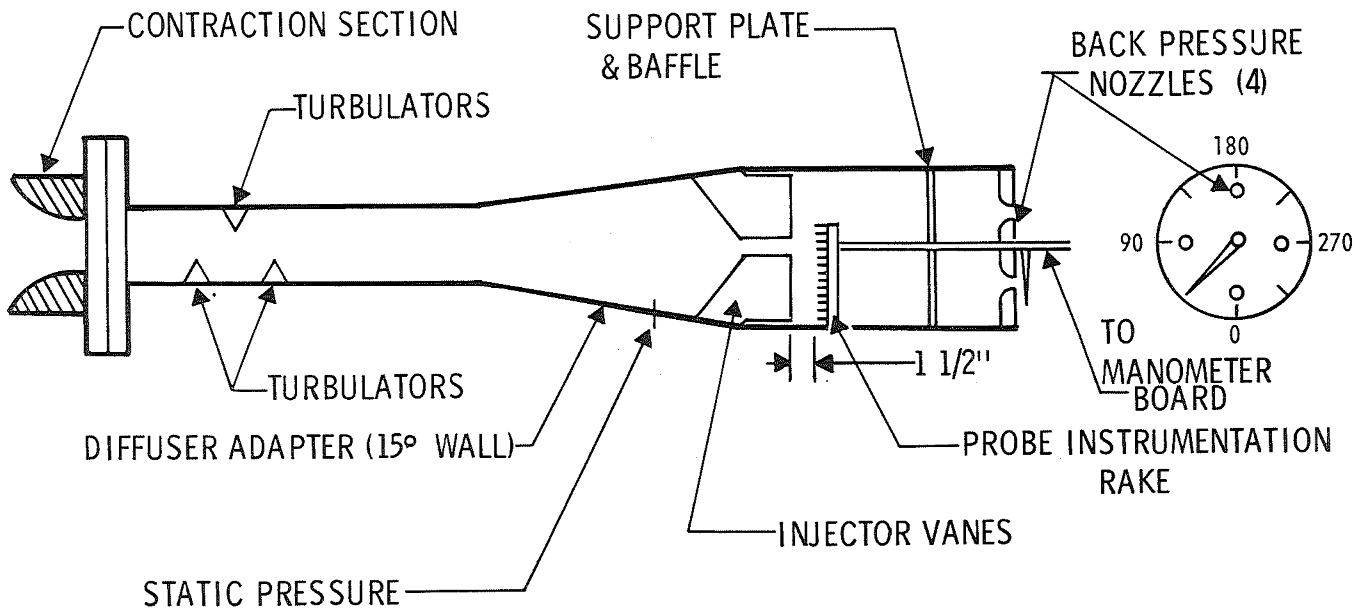
[4] Beichel, R., Gibb, J. A., Hankins, R. A., Advanced Rocket Engine-Storable, Phase 1 Final Report, Supplement 1, Report 10830-F-1, Phase 1, Supplement 1, Aerojet General Corporation, AFRPL-TR-68-70, May 1968 (Confidential)

GAS COLD FLOW TEST INSTALLATION

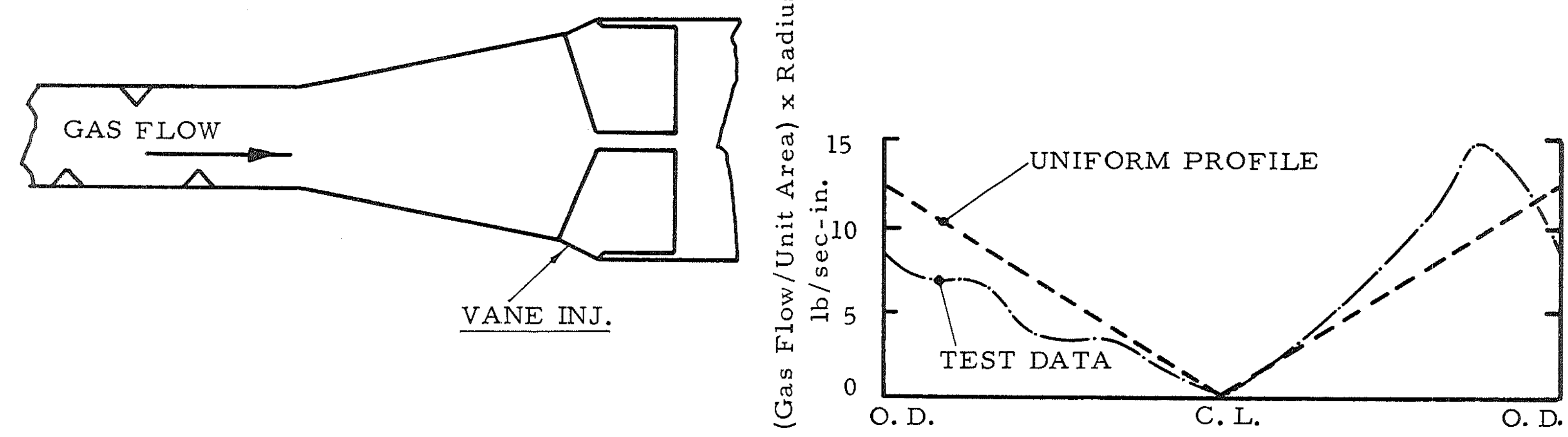


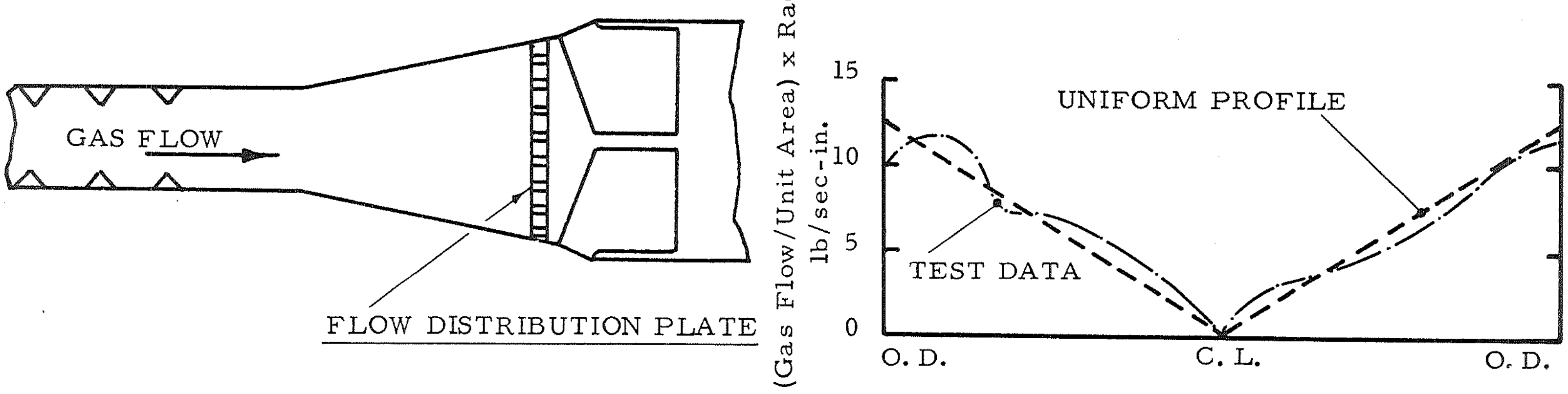
GAS COLD FLOW EXPERIMENTAL RESULTS

The test installation shown in Figure 7 was used to measure the preburner gas distribution downstream of the ARES vane injector. Figure 8, at the top, shows a typical test flow distribution which resulted from the original configuration. The flow parameter, (Gas Flow/Unit Area) x Radius, permits the variation of flow area with radial distance to be included in the data. The flow distribution deviated from the uniform profile to the extent that design modifications were required to prevent a significant mixture ratio distribution specific impulse loss. The bottom portion of Figure 8 shows the flow distribution plate and additional turbulators which were incorporated into the revised design. An example of the flow distribution determined from subsequent cold flow testing is also shown on Figure 8, which indicates that the design modification was successful in achieving uniform preburner gas flow distribution.

The other propellant, which was liquid and injected through the vaned injector, was also cold flow tested to verify that its mass distribution was uniform immediately downstream of the injection plane. For this ARES configuration both propellants were uniformly distributed and subsequent hot fire testing demonstrated high combustion performance which resulted in a specific impulse value that exceeded the contract requirement.

GAS COLD FLOW EXPERIMENTAL RESULTS

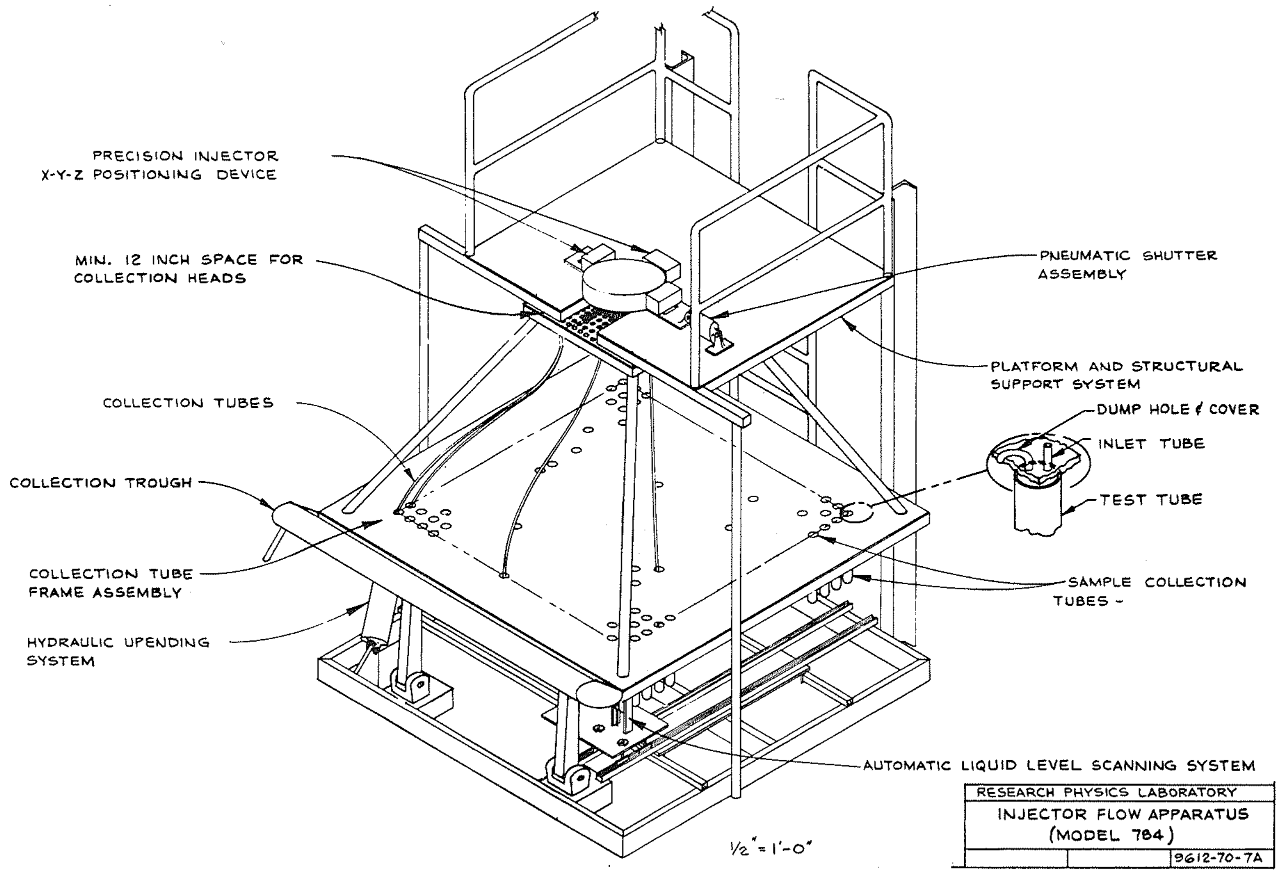




LIQUID COLD FLOW APPARATUS

Figure 9 is a drawing of a cold flow apparatus which can be employed to determine liquid mass distribution from a gas/liquid injector such as proposed for the space shuttle main engines. The injector is positioned face down on the top platform and the simulated cold flow propellant is collected a few inches downstream with a matrix-type collection head which has several hundred individual probes. Each probe is plumbed to a separate tube at the bottom of the fixture with flexible tubing. After each test an automatic liquid-level scanning device records the quantity of simulated propellant in each tube and feeds it into a computer program which prints out the mass distribution across the injector face.

LIQUID COLD FLOW APPARATUS

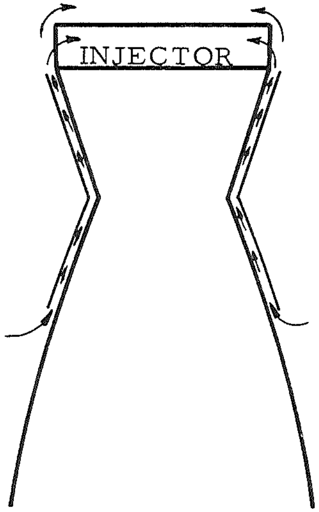
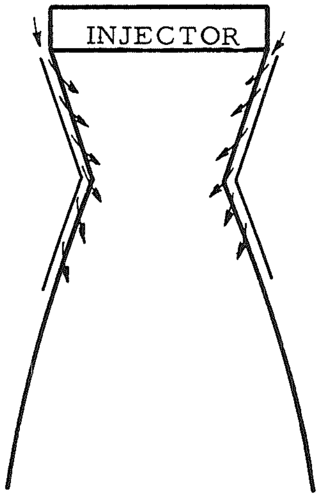


THRUST CHAMBER COOLING TECHNOLOGY

The second area of technology relating to specific impulse is the thrust chamber cooling concept. Two basic types are proposed for the main engine; both concepts use hydrogen as the coolant. One is the regenerative design which introduces the coolant into the preburner gases upstream of injection into the thrust chamber, thus eliminating a propellant mixing process within the thrust chamber.

The other concept is a transpiration coolant design which introduces the coolant flow through discrete apertures into the combustion chamber walls. The coolant flow carries away heat transferred to the wall from the combustion gases and subsequently mixes with the boundary flow to produce a lower energy wall barrier. This cooling concept causes a specific impulse loss due to (1) mixture ratio maldistribution and (2) reduced expansion of that portion of the coolant flow injected downstream of the throat. The loss due to reduced expansion appears to be quite minor based on calculations using the main engine design and operating parameters. Consequently, transpiration coolant specific impulse loss primarily results from mixture ratio maldistribution caused by incomplete mixing of the hydrogen coolant with the primary combustion gases.

THRUST CHAMBER COOLING TECHNOLOGY

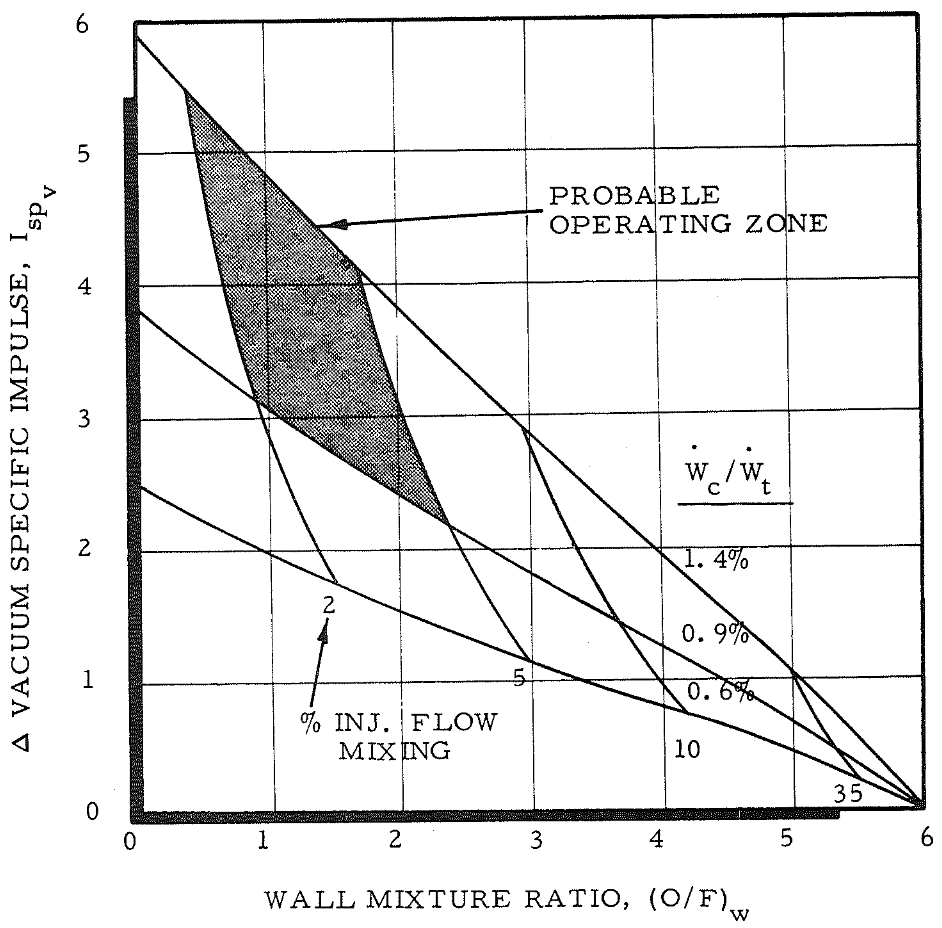
DESIGN CONCEPT:	REGENERATIVE	TRANSPIRATION
		
TYPE OF MIXING PROCESS:	COOLING FLOW MIXED WITH FUEL RICH GASES PRIOR TO INJECTION	COOLING FLOW MIXED WITH COMB. GASES IN CHAMBER
PERFORMANCE EFFECT:	NO DIRECT PERFORMANCE LOSS	PERFORMANCE LOSS DUE TO MIXTURE RATIO MALDISTRIBUTION AND REDUCED EXPANSION

TRANSPIRATION COOLANT PERFORMANCE LOSS

Since the specific impulse loss due to transpiration coolant is primarily due to mixture ratio maldistribution, it is a function of two parameters. One is the quantity of coolant injected and the other is the effectiveness of the subsequent mixing process with the injector flow. The specific impulse loss for the main engine as a function of these two parameters is parametrically plotted on Figure 11. The "% Inj. Flow Mixing" parameter is the fraction of the total injector flow that is assumed to have mixed with the coolant. The specific impulse effect was then calculated for the two stream tubes - (1) mixed coolant and fraction of injector flow and (2) remaining injector flow - and compared to the specific impulse for a completely mixed gas composition.

A "probable operating zone" is also included in Figure 11. The upper and lower boundaries are based on a range of estimated coolant quantity, $\dot{W}_c/\dot{W}_t$, which is expected to be required to maintain the chamber wall temperature at a low enough temperature to permit the long-life design for the space shuttle. The left boundary (2% injector Flow Mixing) represents an estimated value of mixing for the space shuttle engine design based on analysis of test data conducted by Aerojet on various transpiration coolant designs. The right boundary (5% Inj. Flow Mixing) represents a value which makes allowance for possible measurement tolerances in the test data and variations due to injector flow composition at the chamber wall.

TRANSPIRATION COOLANT PERFORMANCE LOSS



TECHNIQUES FOR MINIMIZATION OF COOLANT PERFORMANCE LOSS

No performance loss occurs for the regenerative coolant flow if it is mixed with the preburner gases prior to injection into the thrust chamber (Combustion Criterion No. 1 on Figure 5). Preliminary analysis indicates that this can be accomplished without a significant penalty for mixer weight and/or pressure drop.

Various modeling techniques are available for the transpiration coolant process.[5] Due to the extremely complex flow mechanisms involved, these models are largely supported by empirical correlations which are somewhat configuration sensitive. They have been successfully applied to provide initial design values including the coolant flow rate required. However, specific impulse loss for the transpiration design can be reduced primarily by experimental techniques which establish a gas side wall temperature vs. coolant flow rate relationship. Flow rate can thereby be lowered to a value that will result in a gas side wall temperature which represents the maximum value consistent with the long life requirements of the space shuttle main engines.

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[5] ibid.

TECHNIQUES FOR MINIMIZATION OF COOLANT PERFORMANCE LOSS

REGENERATIVE DESIGN

NO PERFORMANCE LOSS IF COOLANT FLOW IS MIXED WITH PREBURNER GASES (COMBUSTION EFFICIENCY CRITERION)

TRANSPIRATION DESIGN

TEMPERATURE - FLOW RATE TEST DATA CORRELATIONS EMPLOYED TO MINIMIZE COOLANT FLOW

TECHNIQUES FOR NOZZLE PERFORMANCE OPTIMIZATIONS

The third area of technology which affects the main engine specific impulse is the nozzle flow expansion process. The nozzle design parameters which influence specific impulse are listed at the center-top of Figure 13. The component performance effects are listed in the left column, and applicable JANNAF computer program (s) available for evaluating each performance effect are listed in the corresponding row of the right hand column. A large dot has been placed under each nozzle design parameter if it influences the corresponding performance effect.

Except for limitations which will be discussed later, the nozzle design parameters can be adequately optimized using the applicable JANNAF* Computer Programs listed on Figure 13, except for the vehicle specific impulse trade-off effects for weight and envelope. Since 3 out of the 4 nozzle design parameters identified on Figure 13 are influenced by vehicle trade-off factors, it is apparent that accurate values must be available to the engine designer if the highest effective specific impulse is to result.

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* Rocket engine contractors have other state-of-the-art nozzle analysis programs which may be equally applicable to this type of analysis. However, use of other programs sacrifices commonality and requires government agencies to assess the relative merit of the analysis as well as the design when comparing 2 or more engine concepts.

TECHNIQUES FOR NOZZLE PERFORMANCE OPTIMIZATIONS

PERFORMANCE EFFECT	NOZZLE DESIGN PARAMETER: AREA RATIO	LENGTH	CONTOUR (INITIAL DIV.)	WEIGHT PER IN^2	JANNAF COMPUTER PROGRAMS
POTENTIAL THERMO-CHEMICAL PERFORMANCE	●				ONE DIMENSIONAL EQUILIBRIUM
DIVERGENCE LOSS	●	●	●		TWO DIMENSIONAL KINETIC TWO DIMENSIONAL EQUILIBRIUM
BOUNDARY LAYER LOSS	●	●			TURBULENT BOUNDARY LAYER, TBL
KINETIC LOSS	●		●		ONE DIMENSIONAL KINETIC TWO DIMENSIONAL KINETIC
VEHICLE TRADE-OFFS	●	●		●	(VEHICLE OPTIMIZATION ANALYSIS)

JANNAF PERFORMANCE PREDICTION TECHNIQUES

Applicability of the JANNAF Performance Methodology for analyzing the space shuttle main engine was the subject of a joint NASA-JANNAF Liquid Rocket Performance Committee meeting earlier this year . All three main engine contractors presented their evaluation of the JANNAF methodology when applied to space shuttle main engine analysis. The information tabulated on Figure 14 was primarily synthesized from information presented at that meeting and documented in the minutes.[6] It should be emphasized that Figure 14 is applicable to JANNAF methodology only and the "ADDITIONAL DEVELOPMENT" column does not necessarily account for more highly developed performance prediction techniques which may be in the possession of one or more of the rocket engine contractors.

The performance effects which are not adequately modeled by the JANNAF methodology are noted in the right hand column of Figure 14. Some work is presently being conducted to provide initial modeling of the energy release loss and the transpiration/film cooling loss. However, major efforts could be profitably utilized to improve these techniques even though a completely generalized performance model based exclusively on engine design and operating parameters could not be synthesized within the time-table presently envisioned for the space shuttle.

The mixture ratio maldistribution model listed on Figure 14 is a stream tube model without any provision for mass, momentum or energy transport between stream tubes. This assumption is probably not valid under conditions where significant velocity gradients exist between stream tubes. The other major development area listed in Figure 14 is nozzle flow separation, which may become a performance influence for the main engines when they are operated at a back pressure exceeding the nozzle exit pressure. Nozzle flow separation mechanisms are not well understood and represent a technology area which needs additional effort before the engine designer can quantitatively evaluate this effort on his design.

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[6] Liquid Rocket Performance Committee, 2nd Meeting Minutes, JANNAF Performance Standardization Working Group, NASA Lewis Research Center, April 30, May 1, 1970

JANNAF PERFORMANCE PREDICTION TECHNIQUES

SSE PERFORMANCE EFFECT	APPLICABLE JANNAF PROGRAM	ADDITIONAL DEV. REC. FOR SSE
THEORETICAL PERFORMANCE	ONE DIMENSIONAL EQUILIBRIUM	NO
ENERGY RELEASE LOSS	(NOT AVAILABLE)	MAJOR DEV.
MIXTURE RATIO MALDISTRIBUTION	STREAM TUBE MODEL	MAJOR DEV.
COOLING LOSS (TRANSPIRATION/ FILM)	(NOT AVAILABLE)	MAJOR DEV.
DIVERGENCE LOSS	TWO DIMENSIONAL KINETIC	NO
	TWO DIMENSIONAL EQUILIBRIUM	
BOUNDARY LAYER LOSS	TURBULENT BOUNDARY LAYER	MINOR DEV.
KINETIC LOSS	ONE DIMENSIONAL KINETIC	NO
	TWO DIMENSIONAL KINETIC	MINOR DEV.
SEPARATION CRITERIA	(NOT AVAILABLE)	MAJOR DEV.

SUMMARY AND CONCLUSIONS

Four major technology areas which influence main engine specific impulse were discussed. This included a description of the techniques which are available for achieving success in each of these areas and some of the limitations of the techniques. A brief summary for each of these areas is given below.

Combustion Process

High combustion performance can be achieved by using interrelating analytical and experimental techniques to satisfy five combustion criteria. The experimental data can be, to a large extent, obtained from cold flow testing using models or subscale hardware. This will permit the initial designs to be evaluated early in the program and revisions incorporated without the time loss associated with full scale hot firing testing.

Cooling Process

The regenerative design will not result in a specific impulse penalty if the coolant flow is mixed with the preburner gases prior to injection into the thrust chamber. The specific impulse loss for a transpiration coolant design is dependent on the coolant flow rate and the effectiveness of the downstream mixing process. The primary technique for reducing the loss due to transpiration cooling is to conduct tests in which thermal data and local coolant flow rate data can be obtained. This will permit correlations to be developed which will assure minimum coolant quantity (and specific impulse loss) consistent with wall temperatures required for long chamber life.

Nozzle Process

Optimization analysis can be conducted using JANNAF Computer Programs to determine the effect of nozzle design parameters on engine delivered specific impulse. In order to assure that a nozzle is designed for maximum effective specific impulse, it is imperative that accurate vehicle trade-off data be available to the nozzle designer.

SUMMARY

HIGH COMBUSTION PERFORMANCE

- INTERRELATED ANALYTICAL
- AND EXPERIMENTAL TECHNIQUES

MIN. COOLING PERFORMANCE LOSS

- REGENERATIVE DESIGN
  (NO LOSS)
- TRANSPIRATION DESIGN
  DESIGN/TEST ITERATION

HIGH NOZZLE PERFORMANCE

- JANNAF COMPUTER PROGRAMS
- ACCURATE TRADE-OFF FACTORS

JANNAF PREDICTION TECHNIQUES

- ACCURATE ± 1% (REGEN) WITH COLD OR HOT FLOW COMBUSTION DATA
- COMBUSTION AND TRANSPIRATION COOLING MODELS INADEQUATE

Summary and Conclusions (cont.)

JANNAF Prediction Techniques

Although a rigorous error analysis has not been conducted, it is estimated that specific impulse predictions using the JANNAF methodology with cold or hot flow combustion data should be within $\pm$ 1% of subsequent test data values. The performance losses which at present cannot be analyzed with adequate JANNAF models are the combustion and transpiration cooling losses. Some limited development efforts are progressing in these areas but additional effort will be required to develop models which will be adequate to evaluate main engine specific impulse potential without extensive experimental data input.

RECOMMENDATION FOR ADDITIONAL PERFORMANCE TECHNOLOGY EFFORTS

Most of the technology effort required to assure the space shuttle main engines will achieve their specific impulse potential has been identified and appropriate development effort is being conducted. However, it is recommended that JANNAF prediction techniques be further developed particularly in the areas of the combustion process, transpiration cooling process and the nozzle separation process. In addition, it is recommended that the techniques used for cold flow testing be standardized, so that the government agencies can be assured of consistent data from each rocket contractor. This is a task for which the JANNAF Liquid Rocket Performance Committee could also assume cognizance.

Each of these items have an impact on delivered specific impulse, and models should be available to the engine designer and government agency monitor at the earliest possible data. Although the schedule for completing these models may not be compatible with Phase B, they would be valuable in the subsequent development phase to help interpret test data and assure that specific impulse will not limit mission success for the space shuttle.

INVESTIGATION OF NOZZLE FLOW SEPARATION PHENOMENON

JANNAF COMBUSTION PERFORMANCE MODEL

JANNAF TRANSPIRATION/FILM COOLING PERFORMANCE MODEL

STANDARDIZATION OF COLD FLOW PERFORMANCE EXTRAPOLATIONS