LOW-THRUST CHEMICAL PROPULSION

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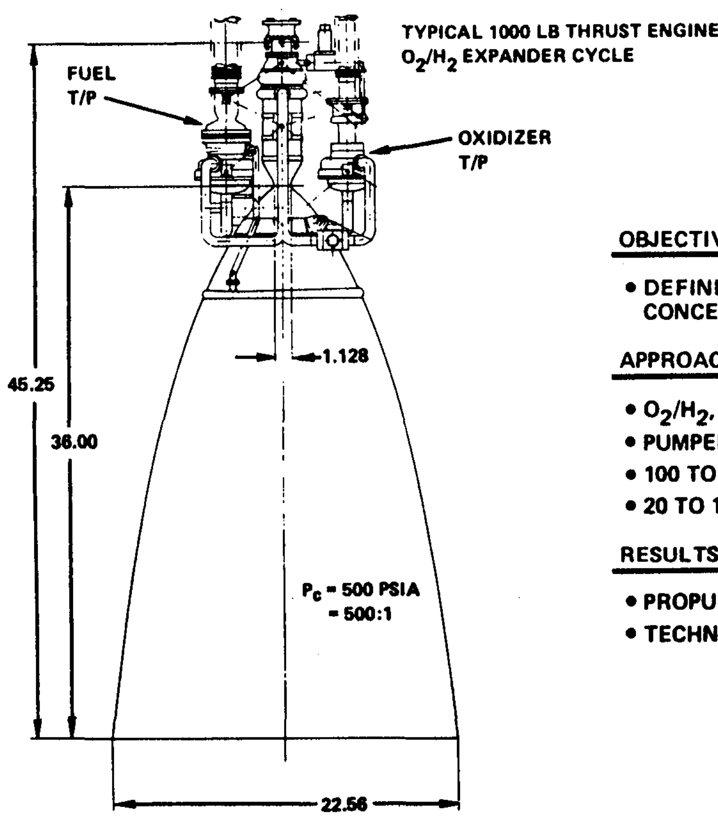
This presentation will summarize the results of an on-going contract with NASA-LeRC. The NASA-LeRC Project Manager is Dean Scheer and the Rocketdyne Program Manager is Hal Diem. The results will include: (1) Thrust chamber cooling analysis and results; and (2) Engine cycle/configuration limits; and (3) Engine performance data.

This chart presents the basic objective, approach, and the desired results of the program. The primary program objective is to define low-thrust chemical engine concepts. The approach is to consider three candidate propellant combinations (O_2/H_2, O_2/CH_4, and O_2/RP-1) for both pump and pressure-fed engines with a thrust range of 100 lb. to 3000 lb. and a chamber pressure range of 20 to 1000 psia. The program results are to include a formulation of the propulsion system concept and a definition of required technology.

LOW THRUST CHEMICAL ROCKET ENGINE STUDY



OBJECTIVE

- DEFINE LOW-THRUST CHEMICAL ENGINE CONCEPTS

APPROACH

- O_2/H_2, O_2/CH_4, O_2/RP-1 PROPELLANTS
- PUMPED AND PRESSURE FED
- 100 TO 3000 LB THRUST RANGE
- 20 TO 1000 PSIA CHAMBER PRESSURE RANGE

RESULTS

- PROPULSION SYSTEM CONCEPT FORMULATION
- TECHNOLOGY PROGRAM DEFINITION

For the low thrust engine two conventional thrust chamber cooling techniques were to be evaluated. These were regenerative/radiation and film/radiation cooling which utilized the fuel as the coolant. With the three propellant combinations and the two cooling techniques, a total of six cases can be configured.

LOW THRUST RANGE OF INTEREST*

CASE NO.	PROPELLANTS	MIXTURE RATIO	COOLING METHOD	COOLANT	THRUST STUDY RANGE, POUNDS	CHAMBER PRESSURE STUDY RANGE, PSIA
1	O_2/H_2	6.0	REGEN	H_2	100 TO 3000	20 TO 1000
2	O_2/H_2	6.0	FILM	H_2	100 TO 3000	20 TO 1000
3	O_2/RP-1	3.0	REGEN	RP-1	100 TO 3000	20 TO 1000
4	O_2/RP-1	3.0	FILM	RP-1	100 TO 3000	20 TO 1000
5	O_2/CH_4	3.7	REGEN	CH_4	100 TO 3000	20 TO 1000
6	O_2/CH_4	4.7	FILM	CH_4	100 TO 3000	20 TO 1000

*FROM TABLE I OF THE RFP

This chart presents the analysis guidelines primarily associated with the thrust chamber cooling evaluation. A nozzle with a 400-to-1 area ratio and 90-percent length was specified for this portion of the study. Combustion chamber lengths and contraction ratios were sized to achieve a minimum combustion efficiency of 98-percent. The film/radiation-cooled thrust chambers were permitted a maximum of 10-percent cooling loss. For hydrocarbon fueled propellants, the benefit of the gas-side carbon layer was to be neglected although current add-on studies will evaluate its influence. For the regenerative/radiation-cooled thrust chambers, a milled-channel wall combustor using NARloy-Z (Twg_{max} = 1000°F) or nickel (Twg_{max} = 1300°F) was used. These temperature limits were set based on a hardware durability standpoint. The nozzle was to be a stainless steel tubular construction. For regenerative-cooling, the maximum coolant velocity and the coking temperature limits for the hydrocarbon fuels were specified as shown. Also the coolant flow within the thrust chamber must be stable. For film/radiation-cooling, conventional wall materials and their respective maximum temperature limits were used. The thrust chamber cycle life required was five thermal cycles times a safety factor of four.

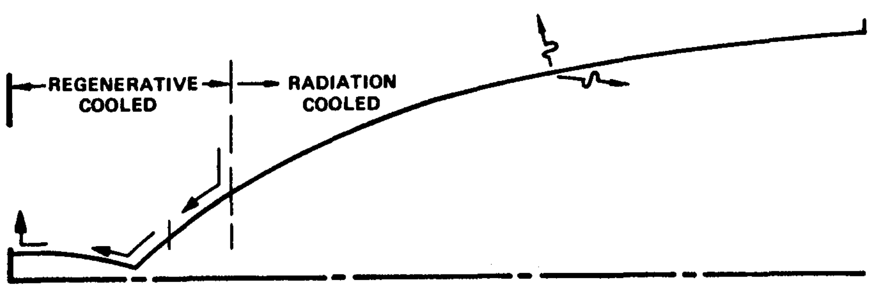
ANALYSIS GUIDELINES

- THRUST: 100 TO 3000 LB
- PROPELLANTS:
  - O_2/H_2 AT MR = 6
  - O_2/CH_4 AT MR = 3.7
  - O_2/RP-1 TO MR = 3.0
- CHAMBER PRESSURE: 20 TO 1000 PSIA
- PERFORMANCE:
  - 98% COMBUSTION EFFICIENCY
  - FILM/RADIATION COOLED
    - $(P_C)_{MAX}$: $(\eta_{IS})_{FILM}$ = 0.90
- THRUST CHAMBER COOLING:
  - HOT-GAS HEAT TRANSFER
    - NEGLECT CARBON LAYER BENEFIT
  - REGENERATIVE/RADIATION
    - MATERIAL
      - COMBUSTOR: CHANNEL WALL [COPPER ALLOY: $(T_{WG})_{MAX}$ = 1000 F; NICKEL: $(T_{WG})_{MAX}$ = 1300 F]
      - NOZZLE: TUBULAR
    - COOLANT (FUEL)
      - MAXIMUM COOLANT VELOCITY (REGENERATIVE-COOLED)
        GAS: MACH NO. = 0.3
        LIQUID: 200 FT/SEC
      - COKING LIMIT (REGENERATIVE-COOLED)
        RP-1: $(T_{WC})_{MAX}$ = 550 F
        CH_4: $(T_{WC})_{MAX}$ = 1300 F
      - COOLANT FLOW MUST BE STABLE
  - FILM/RADIATION
    - MATERIAL
      - L605: $(T_{WG})_{MAX}$ = 2000 F
      - MOLY: $(T_{WG})_{MAX}$ = 2500 F
- CYCLE LIFE:
  - FIVE THERMAL CYCLES TIMES A SAFETY FACTOR OF FOUR
  - ACCUMULATIVE RUN TIME (FUNCTION OF THRUST)
- GENERAL
  - STRUCTURAL
    - YIELD SAFETY FACTOR = 1.1
    - ULTIMATE SAFETY FACTOR = 1.4

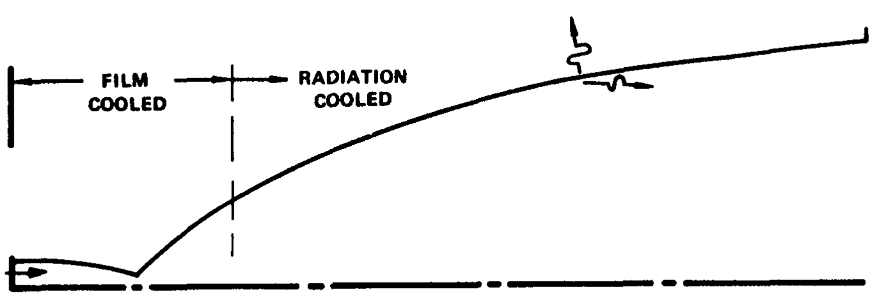
- THRUST CHAMBER GEOMETRY: 400-TO-1 AREA RATIO (90% LENGTH)

This chart presents the two candidate thrust chamber cooling methods evaluated. The regenerative/radiation-cooled thrust chamber had a portion of nozzle and the combustion chamber regeneratively-cooled and the remainder of the nozzle was radiation cooled. The film/radiation-cooled thrust chamber had the film coolant injected at the injector face.

CANDIDATE THRUST CHAMBER COOLING METHODS



(A) REGENERATIVE/RADIATION COOLED CONFIGURATION



(B) FILM/RADIATION COOLED CONFIGURATION

The method of analysis for the radiation-cooled portion of the nozzle utilized an integral boundary layer computer program with conventional wall materials to determine the nozzle wall temperature profile and define parametric nozzle attach area ratio data. For regenerative-cooling the gas-side heat transfer coefficient distribution was determined utilizing a combination of the integral boundary layer computer program results and extrapolated test data. The test data are used to provide a more realistic distribution near the injector. The coolant-side heat transfer coefficient was determined using existing coolant correlations. For example, for hydrogen the modified Dipprey -Sabersky coolant correlation was used. For methane a generalized coolant correlation was assumed; and for RP-1, the coolant correlation developed from the F-1 and Atlas Program was used. The thrust chamber coolant passage design utilized the regenerative-cooling design/analysis computer program. This computer program is capable of both design and analysis of channel wall or tubular coolant passages and is capable of performing two-dimensional wall temperature calculations as well as structural analysis of the coolant passage and predicts thrust chamber cycle life.

THRUST CHAMBER COOLING: ANALYSIS APPROACH

- RADIATION COOLING
  - METHOD OF ANALYSIS
    - ROCKETDYNE INTEGRAL BOUNDARY LAYER COMPUTER PROGRAM
  - CONVENTIONAL WALL MATERIALS
    - L605
    - MOLYBDENUM WITH OXIDATION PROTECTION COATING
  - DETERMINE WALL TEMPERATURE PROFILE
    - DEFINE NOZZLE ATTACH AREA RATIO
- REGENERATIVE-COOLING
  - METHOD OF ANALYSIS
    - GAS-SIDE HEAT TRANSFER COEFFICIENT
      - ROCKETDYNE INTEGRAL BOUNDARY LAYER COMPUTER PROGRAM
      - EXTRAPOLATED TEST DATA
    - COOLANT-SIDE HEAT TRANSFER COEFFICIENT
      - EXISTING COOLANT CORRELATIONS
    - COOLANT PASSAGE DESIGN
      - ROCKETDYNE REGENERATIVE-COOLING DESIGN/ANALYSIS COMPUTER PROGRAM

The wall materials considered for regenerative-cooling included NARloy-Z, cres, and nickel. The regenerative-cooling analysis defined the cooling limits based on the analysis guidelines, determined coolant passage design, and provided parametric data on thrust chamber coolant heat input and coolant pressure drop.

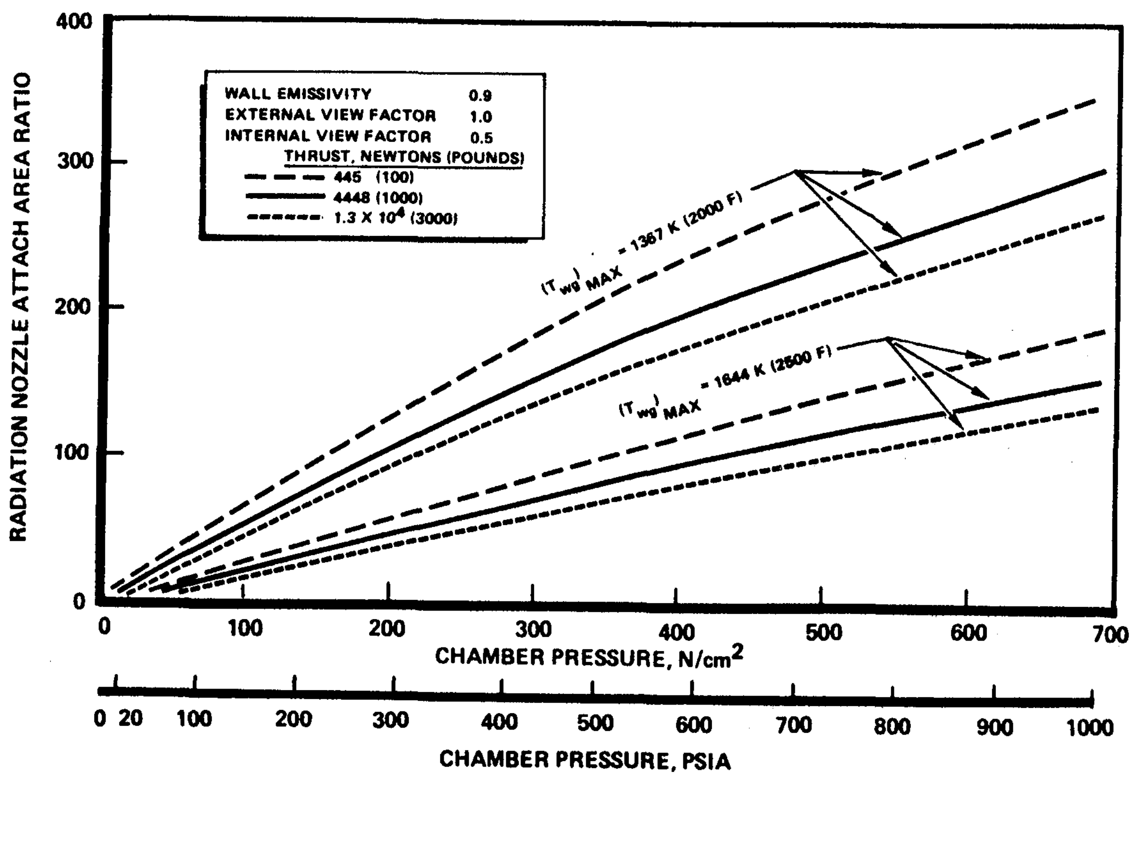
For film cooling, the linear mixture ratio profile model (simplified JANNAF analysis approach) was utilized to determine the maximum allowable film-coolant flow (10-percent cooling loss). The thrust chamber film-cooling heat transfer analysis to obtain wall temperatures and cooling limits utilized a gaseous film-cooling model for supercritical pressures and a liquid film-cooling model for subcritical pressures.

THRUST CHAMBER COOLING: ANALYSIS APPROACH

- REGENERATIVE-COOLING
  - NARLOY-Z, CRES AND/OR NICKEL
  - HEAT TRANSFER DATA
    - DEFINE COOLING LIMITS
    - DETERMINE COOLANT PASSAGE DESIGN
    - DETERMINE COOLANT HEAT INPUT AND COOLANT PRESSURE DROP
- FILM-COOLING
  - METHOD OF ANALYSIS
    - LINEAR MR PROFILE FILM COOLING MODEL
    - ROCKETDYNE GASEOUS AND LIQUID FILM-COOLING COMPUTER PROGRAMS
  - WALL MATERIALS
    - L605 OR MOLYBDENUM WITH OXIDATION PROTECTION COATING
  - HEAT TRANSFER DATA
    - DETERMINE REQUIRED COOLANT FLOW
    - DEFINE COOLING LIMITS

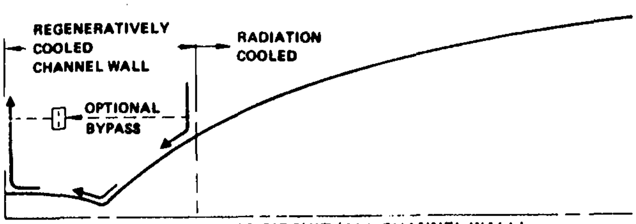
This chart presents the results of the radiation-cooled nozzle analysis for O_2/H_2. Radiation nozzle attach area ratios for two maximum wall temperatures (2000°F and 2500°F) are presented for thrust levels of 100, 1000, and 3000 lbs. Results of a preliminary in-house design effort indicated that for a retractable nozzle (to achieve a reduce engine length), a convenient cutoff area ratio was approximately 200-to-1 area ratio. If this value is selected, all O_2/H_2 thrust chambers in the thrust and chamber pressure range of interest will have a maximum wall temperature less than 2500°F for the radiation-cooled portion of the nozzle. Also since O_2/H_2 is the most energetic of the three propellant combinations, the radiation-cooled nozzle wall temperatures would even be lower for O_2/CH_4 and O_2/RP-1.

RADIATION NOZZLE ATTACH AREA RATIO VARIATION WITH CHAMBER PRESSURE AND THRUST FOR LO_2/H_2

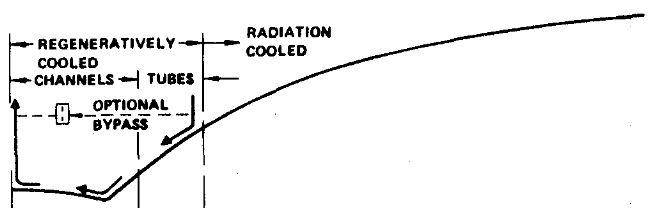


For the regenerative/radiation-cooled thrust chamber, four regenerative cooling circuits were initially evaluated. Cooling circuits A and B are single uppass circuits. Circuit C is a split-flow cooling circuit in which the coolant flows through the combustor and nozzle in parallel. The series cooling circuit (Circuit D) was selected as the baseline due to its lower coolant pressure drop for the low thrust conditions of interest.

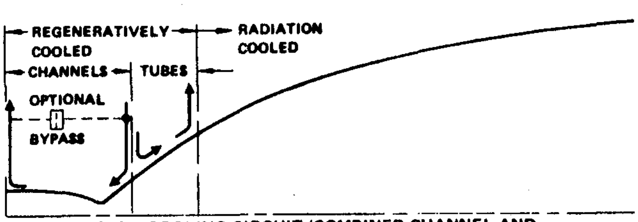
TYPICAL REGENERATIVE COOLING CIRCUITS



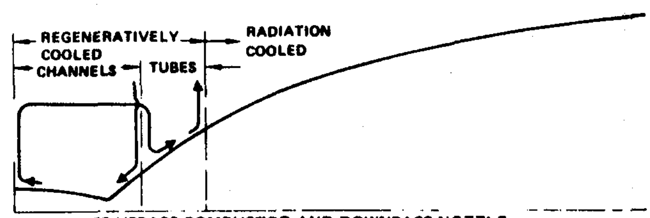
(A) SINGLE UPPASS COOLING CIRCUIT (ALL CHANNEL WALL)



(B) SINGLE UPPASS COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONFIGURATION)



(C) SPLIT-FLOW COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONFIGURATION)

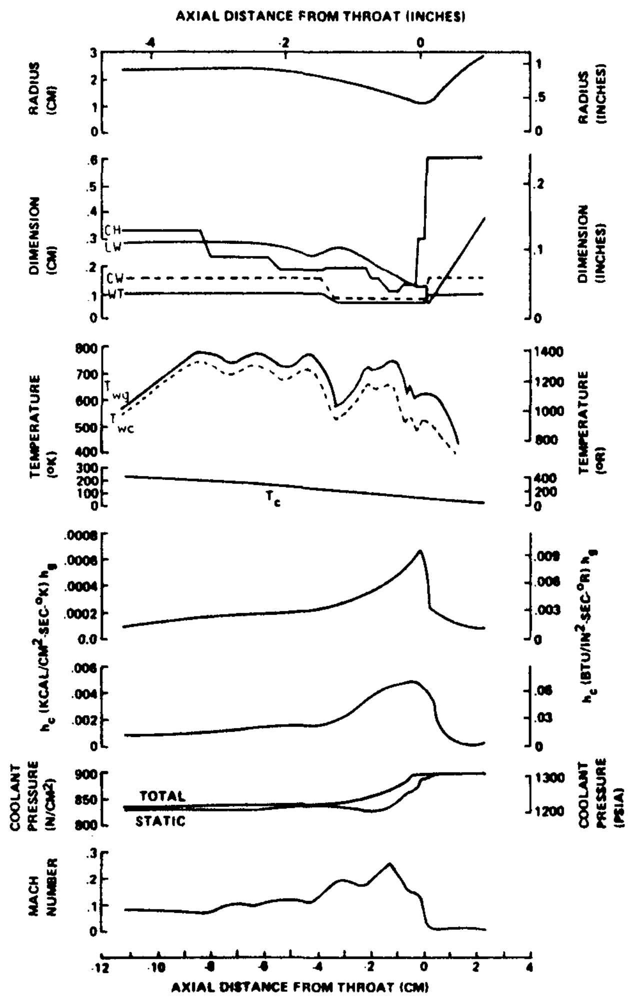


(D) SERIES UPPASS COMBUSTOR AND DOWNPASS NOZZLE COOLING CIRCUIT (COMBINED CHANNEL AND TUBULAR WALL CONSTRUCTION)

Detailed regenerative-cooled thrust chamber analyses were performed for a discrete number of cases to define the cooling limits and obtain heat transfer data for input into the engine cycle analysis. This chart presents the detail analysis results for a typical LO_2/H_2 combustor (injector to a low supersonic area ratio). The design condition was 1000 LB_f thrust and a chamber pressure of 1000 psia at a mixture ratio of 6.0. The combustor contour along with coolant channel dimensions, wall temperatures (two-dimensional), gas-side and coolant-side film coefficients, coolant pressures and coolant Mach number distributions are presented. As noted in this chart, the maximum wall temperature is below the $1460^{\circ}R$ maximum allowable for NARloy-Z and the coolant Mach number is slightly below the maximum allowable of 0.3. Therefore this condition represents a thrust chamber on the regenerative-cooling limit.

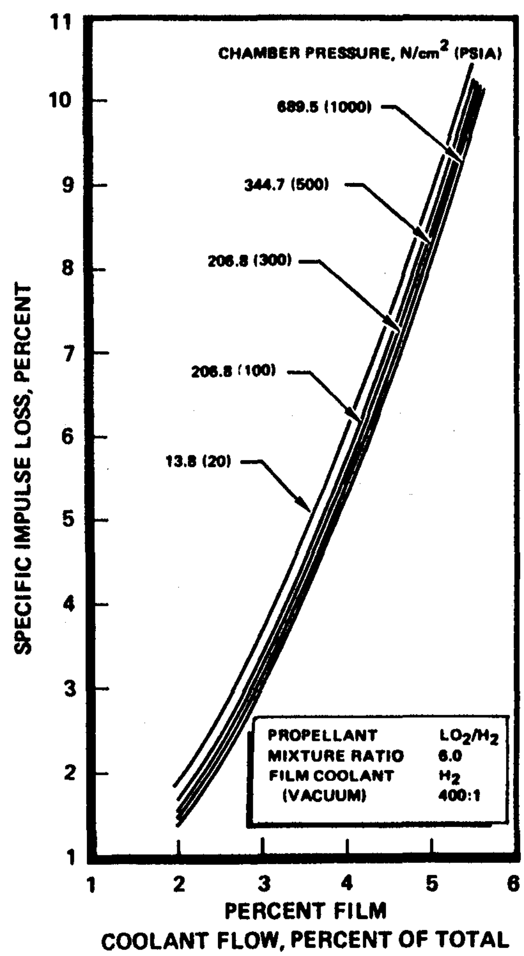
PARAMETERS FOR THE O_2/H_2 LOW THRUST COMBUSTOR

MIXTURE RATIO	6.0
THRUST, NEWTONS (LBF)	4448 (1000)
CHAMBER PRESSURE, N/CM^2 (PSIA)	689.5 (1000)



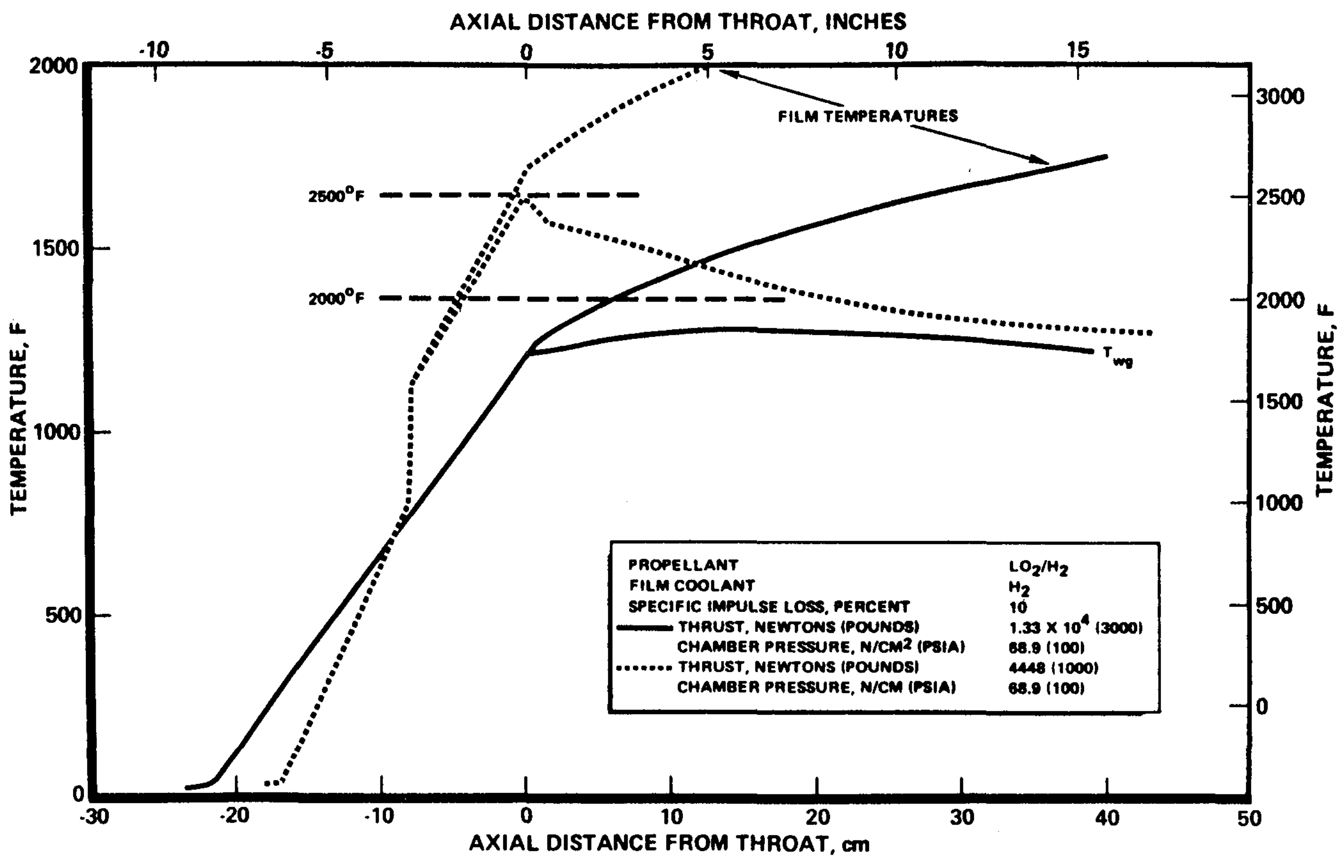
For the film/radiation-cooled thrust chamber, the maximum allowable film coolant flowrate was determined by using the linear mixture ratio profile film cooling performance loss model. For the maximum 10-percent performance loss (see Study Guidelines), a film coolant flow of approximately 5.5-percent resulted for LO_2/H_2 with a nozzle area ratio of 400-to-1. Also note that the resulting film coolant flow was rather insensitive to chamber pressure.

LO_2/H_2 FILM COOLING PERFORMANCE LOSS



Using these allowable film coolant flowrates, detailed heat transfer analyses were performed for a number of design conditions to define the film/radiation-cooled thrust chamber cooling limits. Two typical analysis results are presented in this chart for LO_2/H_2 at a chamber pressure of 100 psia. Axial film and wall temperature distributions are shown. The lower thrust (1000 LB_f) resulted in a higher wall temperature (approximately 2500°F) due to the lower hydraulic diameter causing higher heat fluxes. The deviation of the film and the wall temperature downstream of the throat is due to radiation-cooling. For a maximum allowable temperature of 2500°F, the 1000 LB_f thrust design condition is on the cooling limit for the film/radiation-cooled thrust chamber.

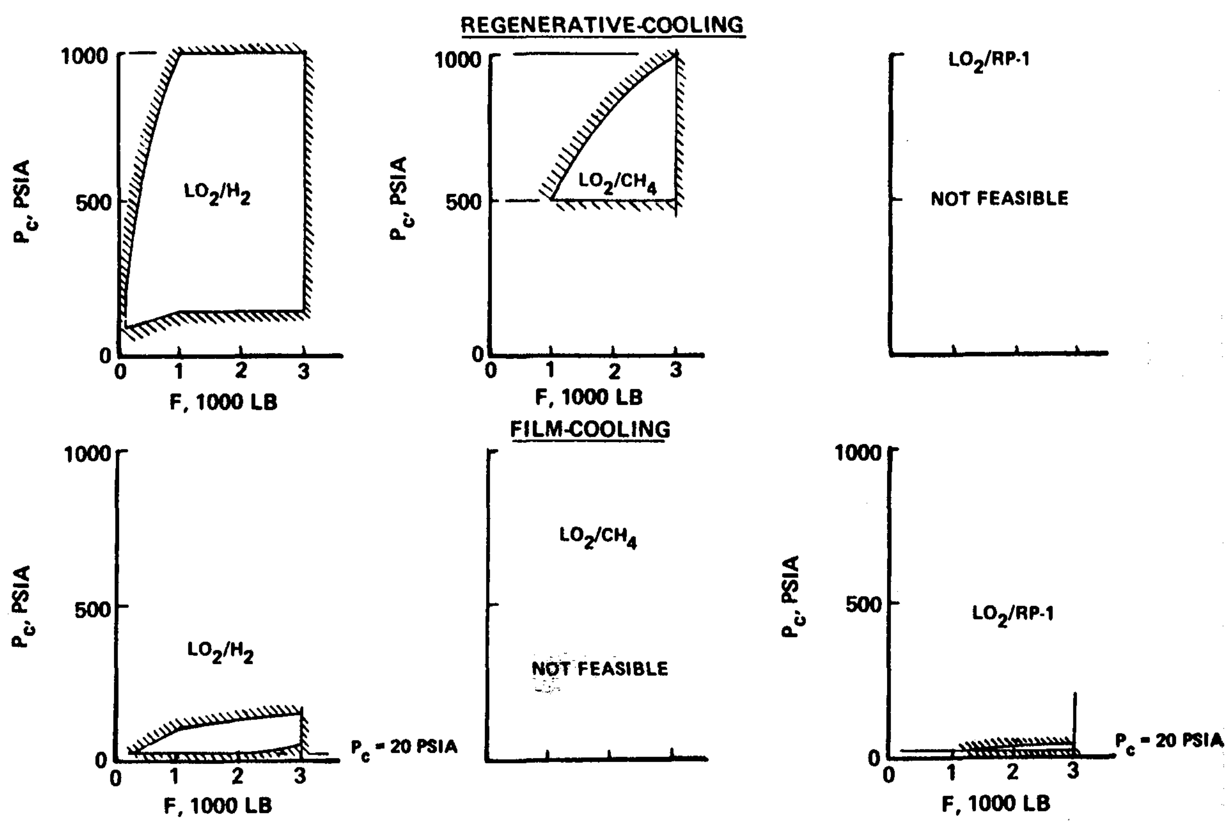
LO_2/H_2 FILM-COOLED THRUST CHAMBER RESULTS



This chart presents a summary of the thrust chamber cooling limits for both regenerative and film cooling. Above 1000 LB thrust, the LO_2/H_2 regenerative-cooled thrust chamber maximum chamber pressures exceeded the maximum study chamber pressure of 1000 psia; however, below 1000 LB thrust the maximum chamber pressure decreased to 200 psia at 100 LB thrust. The minimum chamber pressure was set to maintain a coolant pressure above the critical pressure due to coolant flow instability resulting from two-phase flow. For LO_2/CH_4 the operational envelope was considerably less for LO_2/H_2 due to the poorer cooling capability of Methane and higher critical pressure. Regenerative-cooling for LO_2/RP-1 was found to be not feasible, primarily the result of neglecting the gas-side carbon layer. This influence will be evaluated as part of the program add-on effort.

The operational envelopes for film cooling were limited to a maximum chamber pressure of approximately 150 psia which was for LO_2/H_2. The LO_2/CH_4 film-cooled thrust chambers were found to be not feasible and the operational envelope for LO_2/RP-1 thrust chambers was extremely limited.

THRUST CHAMBER COOLING LIMIT SUMMARY



The engine cycle/configuration analyses approach consisted of first a definition of candidate cycles including the work statement specified configurations and the incorporation of the heat transfer analysis results. The analyses of the resultant engine cycle/configurations was performed using the Rocketdyne Low Thrust Engine Cycle Balance Computer Program which is capable of simultaneously optimizing up to eight parameters. The alternator, electric motor, and fuel cell data and design relationships were incorporated in the computer program. These analyses defined the engine cycle limits (maximum design chamber pressure) and provided the engine balance data. Parametric thrust chamber performance data were also generated.

Currently the screening and evaluation of the engine cycle/configurations are being performed by determining the cycle operational capability, performance, envelope, weight, complexity, and technology advancement required.

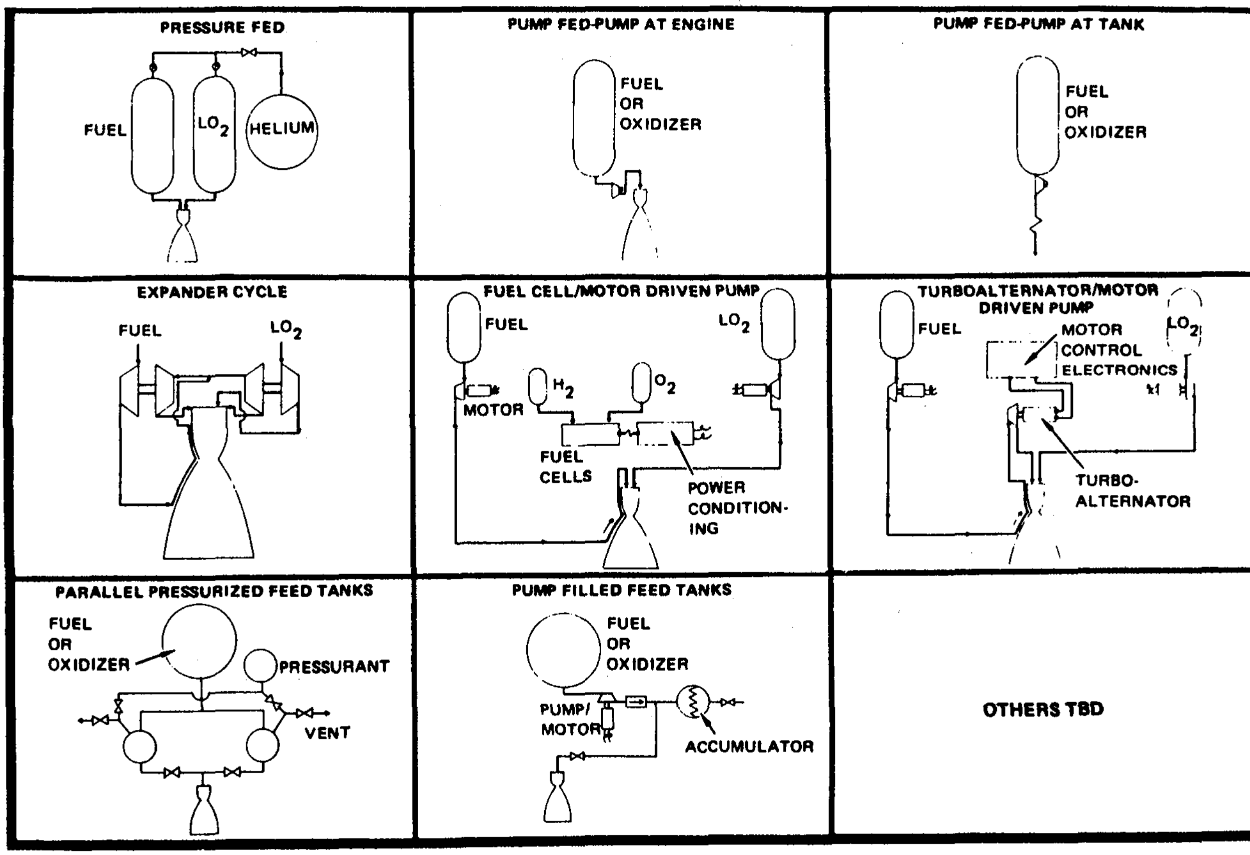
ENGINE CYCLE/CONFIGURATION EVALUATION:ANALYSIS APPROACH

- ENGINE CYCLE/CONFIGURATION DEFINITION AND MATRIX REFINEMENT
  - WORK STATEMENT SPECIFIED CONFIGURATIONS
  - INCORPORATION OF HEAT TRANSFER ANALYSIS RESULTS
- ENGINE CYCLE/CONFIGURATION ANALYSIS
  - METHOD OF ANALYSIS
    - ROCKETDYNE LOW THRUST ENGINE CYCLE BALANCE COMPUTER PROGRAM
      - INCORPORATION OF ALTERNATOR, ELECTRIC MOTOR, AND FUEL CELL DATA AND DESIGN RELATIONSHIPS
  - DETERMINE PARAMETRIC THRUST CHAMBER PERFORMANCE DATA
  - DEFINE ENGINE CYCLE LIMITS
    - ENGINE BALANCE DATA
- ENGINE CYCLE/CONFIGURATION SCREENING EVALUATION AND SELECTION
  - CYCLE OPERATIONAL CAPABILITIES
  - PERFORMANCE
  - ENVELOPE
  - WEIGHT
  - COMPLEXITY
  - TECHNOLOGY ADVANCES REQUIRED

This chart schematically illustrates the candidate engine cycle/configurations. The engines include both pressure-fed and pump-fed engines. The pump-fed engines have the pumps located on the engine or at the tank. Conventional gas driven turbine cycles such as the direct expander cycle are candidates as well as unconventional cycles such as the fuel cell/motor driven pump cycle, turboalternator cycles, parallel pressurized feed tank, and pump-filled tank cycle.

ENGINE SYSTEM CONCEPTS TO BE STUDIED

(O_2/H_2, O_2/RP-1, O_2/CH_4 PROPELLANTS; REGEN. AND FILM COOLING)



The resulting engine cycle/configuration matrix for the three propellant combinations and two cooling approaches is presented in this chart. The open boxes indicate the candidate engine cycles and the shaded boxes depict cycles which have been eliminated due to technical unfeasibility noted in the chart. Majority of eliminations occurred as a result of the incorporation of heat transfer results.

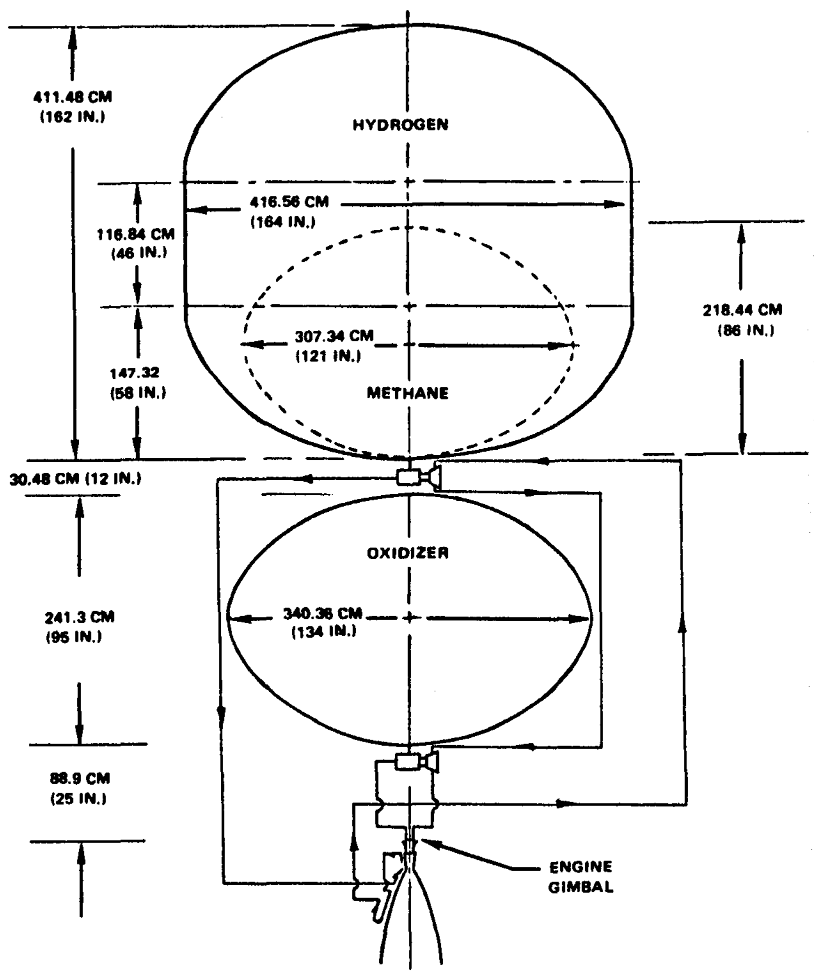
ENGINE CONFIGURATION MATRIX

PROPELLANT	O_2/H_2 (MR = 6.0)		O_2/CH_4 (MR = 3.7)		O_2/RP-1 (MR = 3.0)	
COOLING	REGEN. COOLED	FILM COOLED	REGEN. COOLED	FILM COOLED	REGEN. COOLED	FILM COOLED
ENGINE MOUNTED PUMP-FED						
EXPANDER CYCLE		(1)		(1)	(2)	(1)
GAS GENERATOR CYCLE				(4)	(3)	(5)
STAGED COMBUSTION CYCLE				(4)	(3)	(5)
TANK-MOUNTED PUMP-FED DIRECTLY POWERED PUMPS						
EXPANDER CYCLE		(1)		(1)	(2)	(1)
GAS GENERATOR CYCLE				(4)	(3)	(5)
STAGED COMBUSTION CYCLE				(4)	(3)	(5)
TANK-MOUNTED PUMP-FED INDIRECTLY POWERED PUMPS TURBO ALTERNATOR (WITH OR WITHOUT PUMP-FILLED FEED TANK)						
EXPANDER CYCLE		(1)		(1)	(2)	(1)
GAS GENERATOR CYCLE				(4)	(3)	(5)
STAGED COMBUSTION CYCLE				(4)	(3)	(5)
O_2/H_2 FUEL CELL POWERED SYSTEM				(4)	(3)	(5)
PRESSURE-FED						
CONVENTIONAL			(6)	(4)	(3)	
PARALLEL PRESSURIZED TANK			(6)	(4)	(3)	

NOTES
(1) EXPANDER CYCLE REQUIRES HEATED PROPELLANT TO DRIVE TURBINES
(2) RP-1 EXPANDER CYCLE NOT FEASIBLE DUE TO COKING
(3) RP-1 REGENERATIVE-COOLING NOT FEASIBLE } FROM HEAT TRANSFER RESULTS
(4) CH_4 FILM COOLING NOT FEASIBLE } FROM HEAT TRANSFER RESULTS
(5) MAXIMUM P_c (~25 PSIA) TOO LOW FOR PUMP FED LO_2/RP-1 ENGINES
(6) 500 PSIA CHAMBER PRESSURE TOO HIGH FOR PRESSURE-FED LO_2/CH_4 ENGINE

For the tank-mounted pump/turbine engine cycles, the NASA-LeRC specified propellant tank configurations are illustrated. Both LO_2/H_2 and LO_2/CH_4 tank configurations are presented. An expander cycle with tank-mounted pumps and turbine is shown. These tank configurations enable the calculation of line lengths.

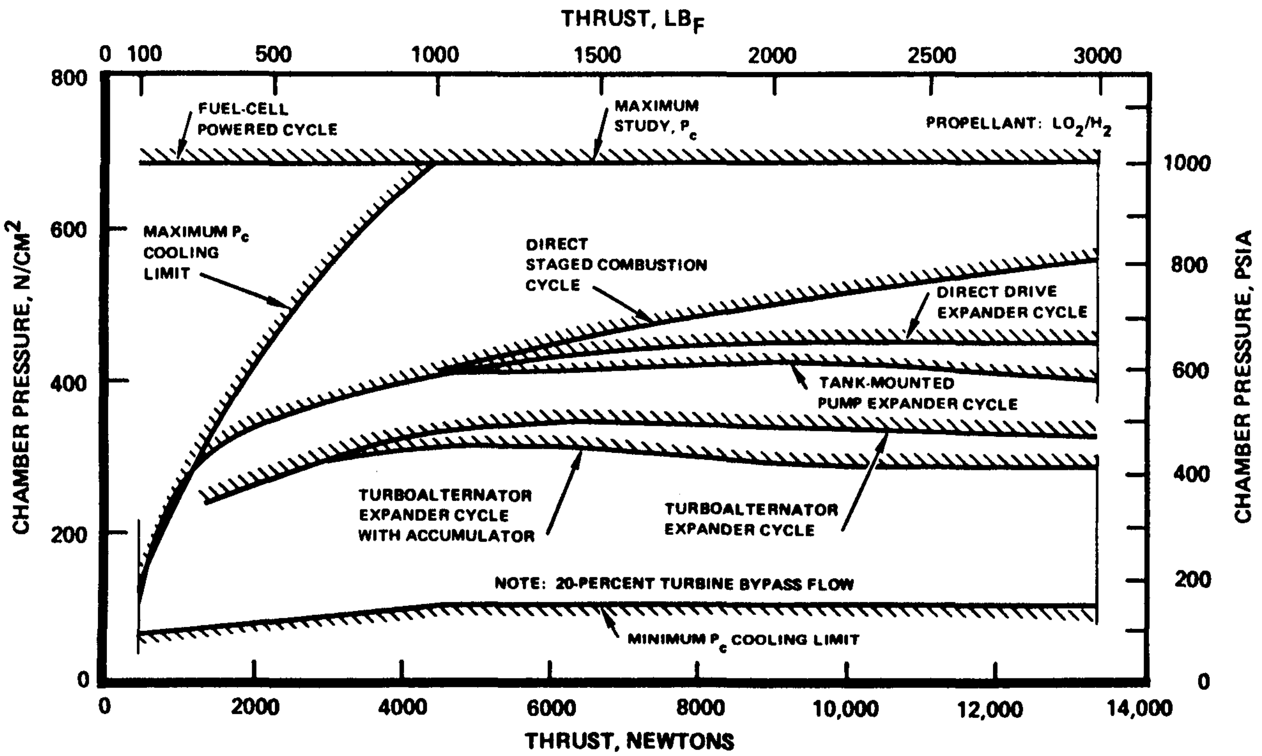
PROPELLANT TANK CONFIGURATIONS



This chart presents the regenerative-cooling and cycle limits for LO_2/H_2 engines. The fuel-cell powered cycle was capable of achieving the maximum study chamber pressure of 1000 psia for any thrust due to an almost unlimited available power. Whatever power was required to drive the pumps, a bigger fuel cell was incorporated. As a result the fuel cell system weight was, in general, an order of magnitude higher than the other engine concepts. The direct staged combustion cycle achieved the next highest chamber pressure; however, this cycle resulted in a marginal combustion stability for the preburners which could be detrimental.

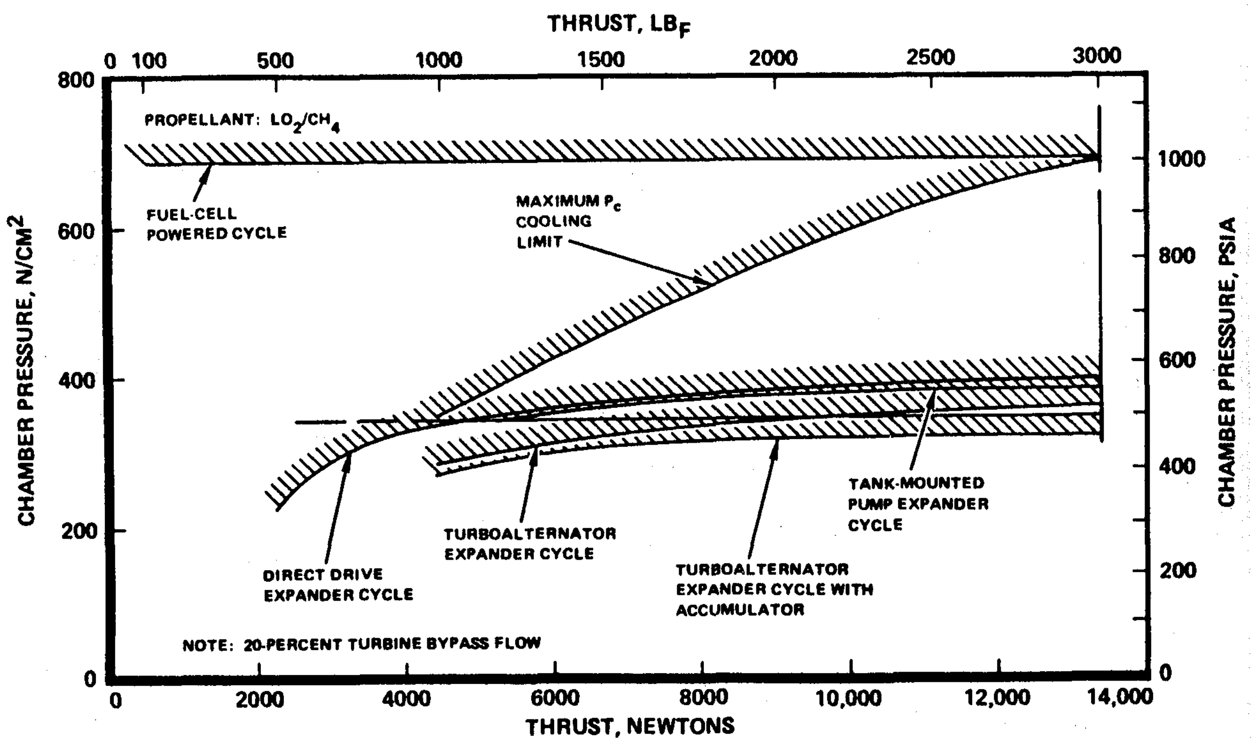
The next highest chamber pressure was achieved by the direct drive expander cycle. This cycle achieved a maximum chamber pressure of approximately 650 psia which remains essentially constant with decrease in thrust until 1000 LB. Modifications to the expander cycle all lead to a decrease in maximum chamber pressure at a given thrust. The tank-mounted pump expander cycle resulted in a lower maximum chamber pressure due to the additional pressure drop of the long hot-gas ducts. The inefficiencies of the added components (alternator and electric motors) decreased the maximum chamber pressure of the turboalternator expander cycle. The addition of the accumulator (pump-filled feed tank) improved the pump efficiencies but due to the increased propellant flow, required an increase in horsepower and therefore a decrease in chamber pressure resulted.

REGENERATIVE-COOLING AND CYCLE LIMITS FOR LO_2/H_2 ENGINES



Similar results occurred for the regenerative-cooled LO_2/CH_4 engines although the cycle limits were not as sensitive as for the LO_2/H_2 engines. Current analyses efforts indicate that the minimum chamber pressure limit for LO_2/CH_4 regenerative-cooling may be lower due to the increase in the actual coolant discharge pressure as a result of the turbine pressure ratios.

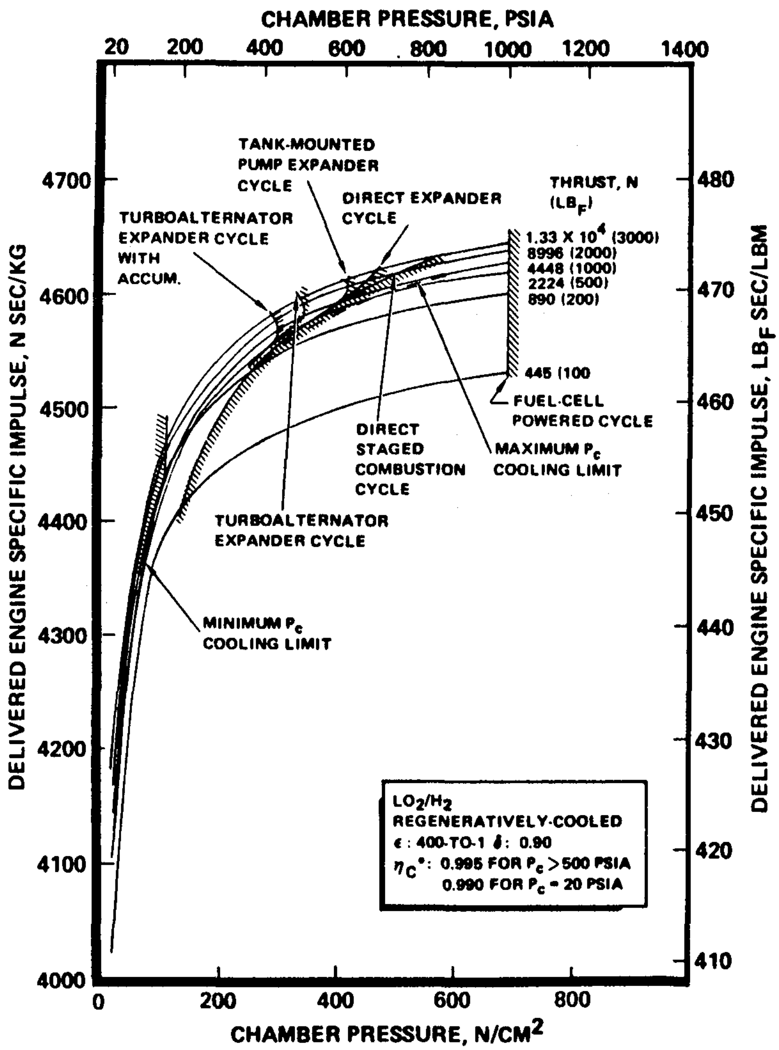
REGENERATIVE-COOLING AND CYCLE LIMITS FOR LO_2/CH_4 ENGINES



Parametric delivered engine specific impulse data are shown in this chart for regenerative-cooled LO_2/H_2 engines with both the cooling and cycle limits superimposed; and therefore clearly shows the maximum attainable engine specific impulse. These curves also show the rapid decrease in specific impulse below approximately 400 psia chamber pressure. Delivered specific impulses for the direct expander cycle engine can exceed 470 LB_f sec/LB_m.

LO_2/H_2 REGENERATIVELY-COOLED ENGINE DELIVERED SPECIFIC IMPULSE

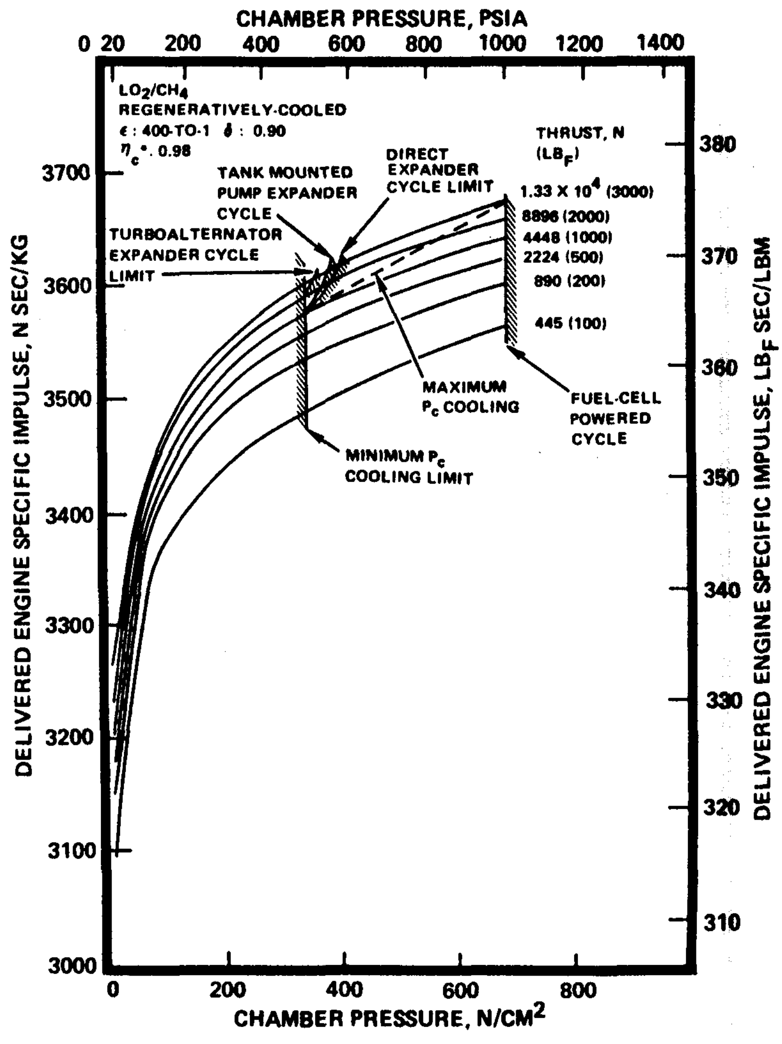
COOLING AND CYCLE LIMITS SUPERIMPOSED



Similar results for regenerative-cooled LO_2/CH_4 engines are presented in this chart. Delivered engine specific impulses are approximately 100 -LB_f sec/LB lower than these for the LO_2/H_2 engines.

LO_2/CH_4 REGENERATIVELY-COOLED ENGINE DELIVERED SPECIFIC IMPULSE

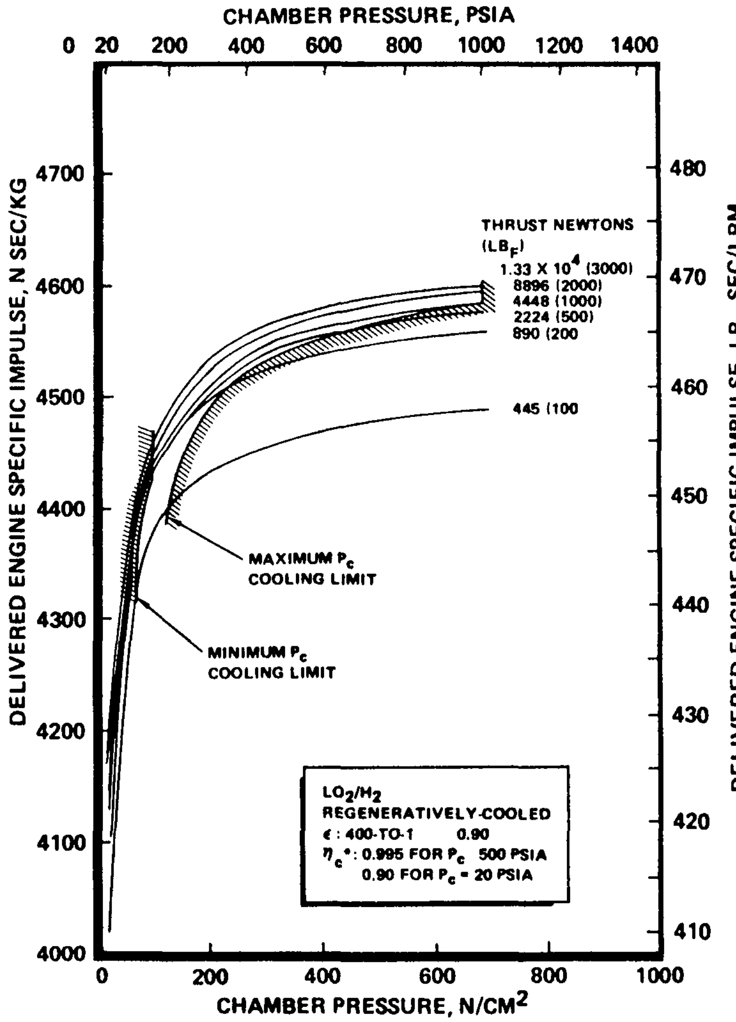
COOLING AND CYCLE LIMITS SUPERIMPOSED



Delivered engine specific impulse curves for the regenerative-cooled LO_2/H_2 gas generator cycle engines are presented in this chart. The specific impulse values were approximately 1-percent lower than for the expander cycle engines.

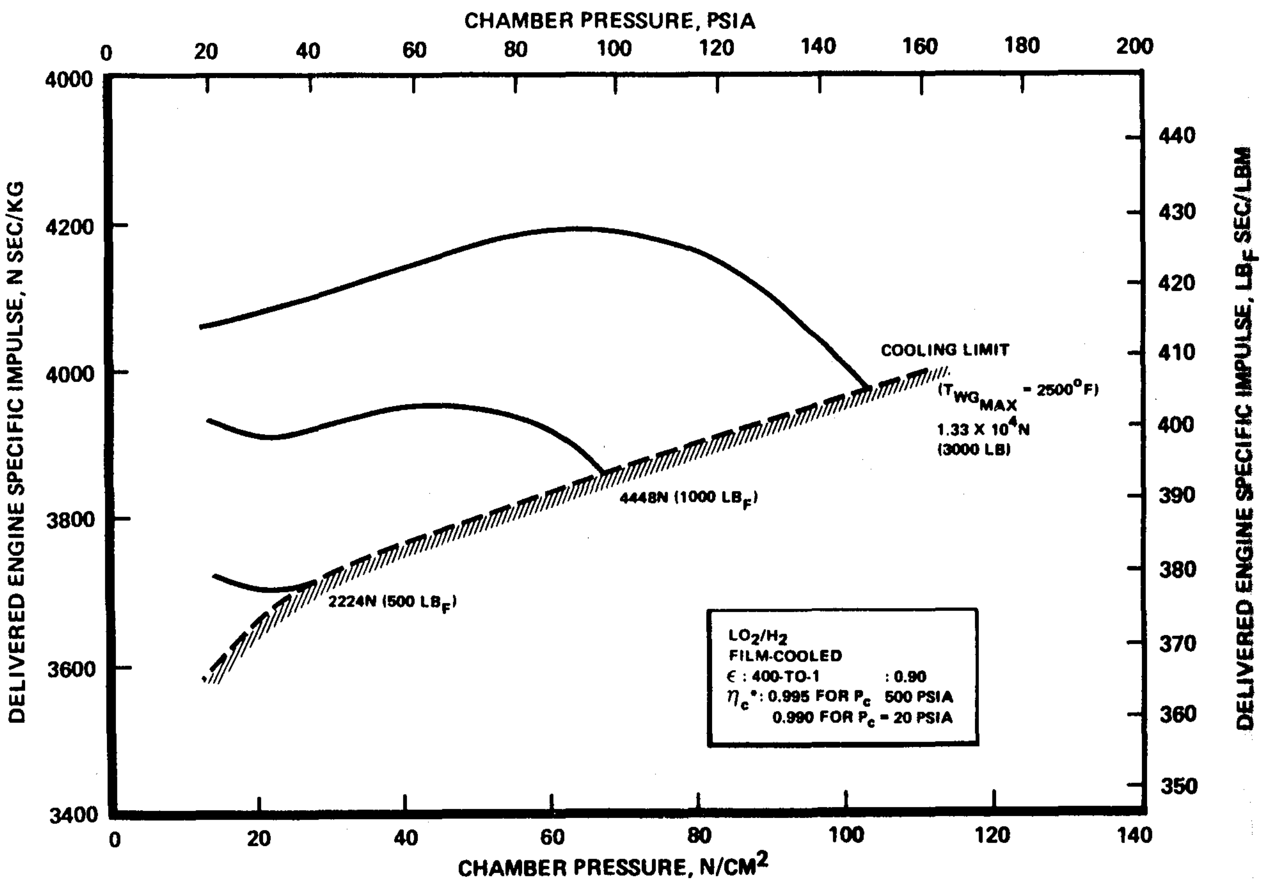
LO_2/H_2 REGENERATIVELY-COOLED GAS GENERATOR ENGINE DELIVERED SPECIFIC IMPULSE

COOLING AND CYCLE LIMITS SUPERIMPOSED



The delivered engine specific impulse for film/radiation-cooled LO_2/H_2 engines is shown in this chart. The specific impulse initially increased with chamber pressure but as the wall temperatures increased, additional film coolant was required which decreased the specific impulse with increase in chamber pressure until the maximum allowable film-cooling performance loss of 10-percent is reached (cooling limit). The maximum delivered specific impulse is approximately 428 LB_f sec/LB_m which is significantly lower than that for the regenerative-cooled engines.

LO_2/H_2 FILM-COOLED DELIVERED ENGINE SPECIFIC IMPULSE

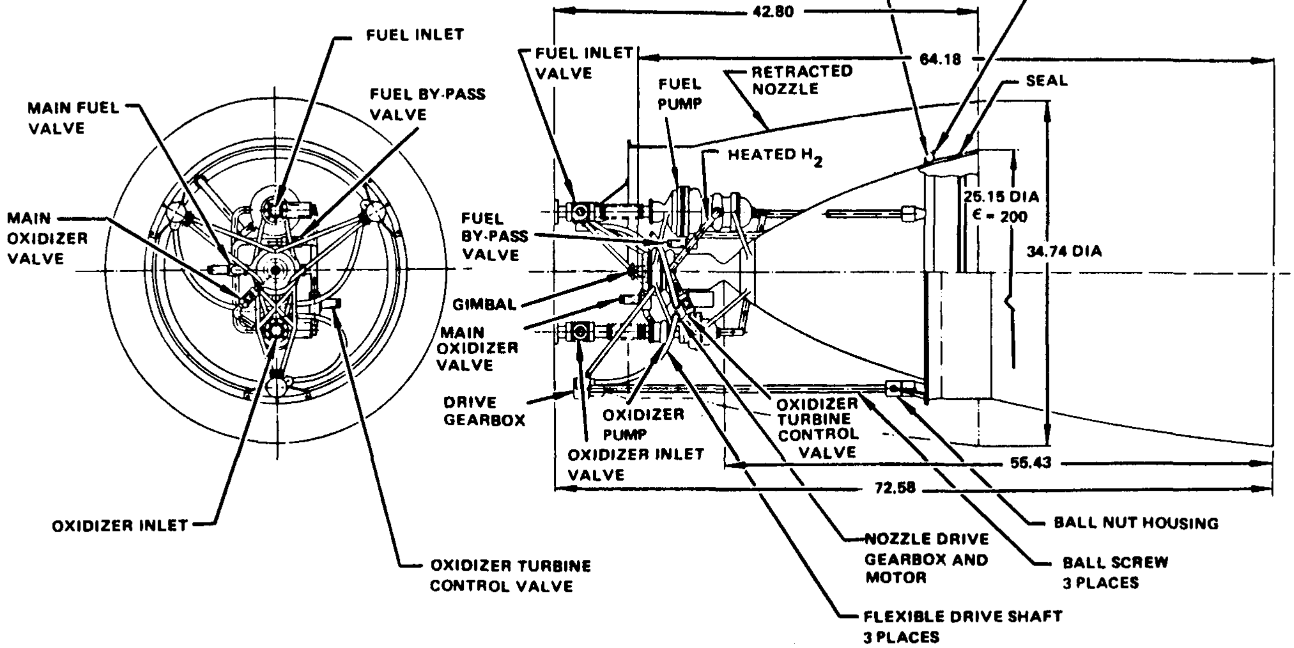


Typically one might expect that low thrust engines are all small in size. As shown in this chart, the engine length can vary from 16 inches to 340 inches. A typical LO_2/H_2 expander cycle engine at 3000-LB thrust and 660 psia chamber pressure is illustrated. The engine length is 72.6 inches and the utilization of a retractable nozzle resulted in a 42.8 inch length (a 41-percent length reduction). Since the launch vehicle is most likely the Space Shuttle, engine length can be extremely important.

EXPANDER CYCLE LOW THRUST ENGINE

801.2

PROPELLANT	LO_2/H_2
THRUST, NEWTONS (POUNDS)	1.33×10^4 (3000)
CHAMBER PRESSURE, N/CM^2 (PSIA)	455 (660)
	400



RANGE OF ENGINE DIMENSIONS FOR F & P_c RANGE:
DIAMETER, CM (IN.): 12.98 (5.11) TO 516.4 (201.34)
LENGTH, CM (IN.): 41.35 (16.28) TO 863.65 (340.02)

The summary of results to date are presented in this chart. From the thrust chamber cooling analyses, regenerative/radiation-cooled LO_2/H_2 thrust chambers offerred the largest thrust and chamber pressure operational envelope primarily due to the superior cooling capability of hydrogen and its low critical pressure. Regenerative/radiation-cooled LO_2/CH_4 offerred the next largest operational envelope. LO_2/RP-1 regenerative-cooling was found not to be feasible over the study range due to RP-1 coking. The inclusion of the carbon layer benefit would make LO_2/RP-1 cooling feasible; this is currently being evaluated. The maximum chamber pressure for film/radiation-cooling was significantly lower than for regenerative/radiation-cooling. As in regenerative/radiation-cooling, LO_2/H_2 thrust chambers achieved the highest maximum chamber pressure. LO_2/CH_4 film/radiation-cooling was found not feasible and LO_2/RP-1 film/radiation-cooling was extremely limited.

In the engine cycle/configuration evaluation, the engine cycle matrix was defined through the incorporation of the heat transfer results. Engine cycle limits were established with the fuel-cell power cycle achieving the highest chamber pressure; however, the fuel cell system weights were excessive. The staged combustion cycle achieved the next highest chamber pressure but the preburner operational feasibility was in question. The next highest chamber pressure was achieved by the direct drive expander cycle.

Currently in addition to finalizing the cycle limits, the complexity and weight of the engine cycles are currently being determined. This engine cycle/configuration evaluation is to lead to the selection of one LO_2/Hz and one LO_2/hydrocarbon fuel engine for preliminary design and analysis.

SUMMARY OF RESULTS TO DATE

- HEAT TRANSFER
  - REGENERATIVE/RADIATION COOLING
    - LO_2/H_2 OFFERED LARGEST F AND P_c OPERATIONAL ENVELOPE
      - H_2 COOLING CAPABILITY
      - LOW H_2 CRITICAL PRESSURE
    - LO_2/RP-1
      - NOT FEASIBLE OVER STUDY F AND P_c RANGE DUE TO RP-1 COKING LIMIT
  - FILM/RADIATION COOLING
    - MAXIMUM P_c LOWER THAN REGENERATIVE/RADIATION COOLING
    - LO_2/H_2: ACHIEVED HIGHEST MAXIMUM P_c
    - LO_2/CH_4: NOT FEASIBLE OVER STUDY RANGE
    - LO_2/RP-1: LOW P_c
- ENGINE CONFIGURATION EVALUATION
  - DEFINED ENGINE CYCLE/CONFIGURATION MATRIX
    - INCORPORATED HEAT TRANSFER RESULTS
  - ENGINE CYCLE/CONFIGURATION LIMIT (ORDER OF HIGHEST P_c TO LOWEST AT A GIVEN THRUST)
    - FUEL-CELL POWERED CYCLE
    - STAGED COMBUSTION CYCLE (FOR LO_2/H_2)
    - DIRECT DRIVE EXPANDER CYCLE
  - FUEL-CELL RESULTED IN EXCESSIVE WEIGHT
  - STAGED COMBUSTION PREBURNER DESIGN FEASIBILITY BEING EVALUATED
  - ENGINE CYCLE/CONFIGURATION COMPLEXITY ANALYSIS IN PROGRESS