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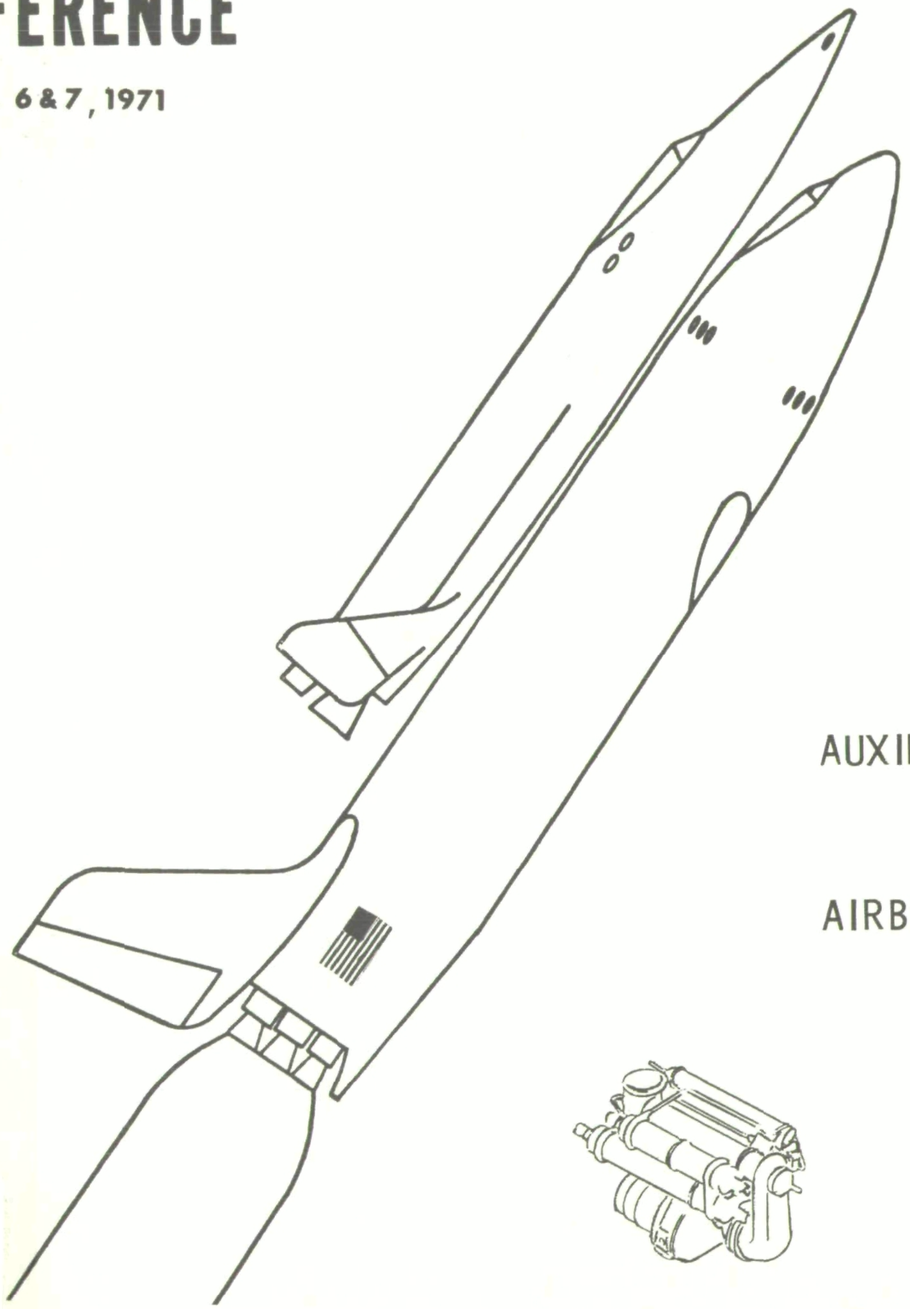
PROCEEDINGS

SPACE TRANSPORTATION SYSTEM PROPULSION TECHNOLOGY CONFERENCE

GEORGE C. MARSHALL SPACE FLIGHT CENTER

APRIL 6 & 7, 1971

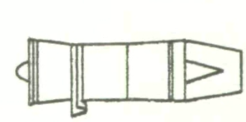


VOLUME III

AUXILIARY POWER UNIT

AND

AIRBREATHING PROPULSION



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PROCEEDINGS
SPACE TRANSPORTATION SYSTEM
PROPULSION TECHNOLOGY CONFERENCE

APRIL 6-7, 1971

MARSHALL SPACE FLIGHT CENTER

VOLUME III

SESSION III	AUXILIARY POWER UNIT	D. BEREMAND, CHAIRMAN
SESSION IV	AIRBREATHING PROPULSION	W. STEWART, CHAIRMAN

April 28, 1971

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AUXILIARY POWER UNIT

INTRODUCTION

by

Donald G. Beremand
Lewis Research Center

The Lewis Research Center currently is engaged in a program to develop the technology for a hydrogen-oxygen fueled Auxiliary Power Unit (APU) for the Space Shuttle vehicles. APUs are required to provide all hydraulic power on both the Booster and Orbiter vehicles. They will also provide all electrical power for the Booster and will supplement other electrical power sources during the launch, reentry, and landing phases for the Orbiter. Typical APU requirements are summarized briefly in figure 1. Peak power levels still vary substantially in the various vehicle studies and these values are representative of the low end of the range of peak powers of about 200 to 600 horsepower.

Critical requirements for the APU design imposed by the mission requirements are the combination of:

Large power ratios - $\frac{\text{maximum power}}{\text{minimum power}} \approx 10:1$

Rapid power transients - minimum power to maximum power in .075 seconds

Limited operation at peak power $< 10\%$ of mission

Peak power operation required from sea level to orbital altitudes

For reasons of cost, it is desired to provide a common APU for the Booster and Orbiter vehicles even though the specific requirements of the two vehicles may vary. In addition, a requirement has been established that the APU be self-cooling in order to avoid placing additional cooling loads on other vehicle systems.

In August 1970 preliminary APU design study contracts were placed with AiResearch Manufacturing Company, Los Angeles, and Rocketdyne, Canoga Park. Both contracts are two-phase efforts. The Phase I efforts, completed in December 1970, were aimed at studying and screening candidate systems to select a basic system concept. The objective of each of the Phase II efforts now underway is to generate a detailed APU design based on this concept. This design effort will be carried through engineering analysis and detailed layout drawings of the components and system.

The papers that follow present the major results and analyses generated in these contract efforts. The authors were requested to concentrate on somewhat different aspects of their studies to avoid undue duplication in the presentations.

H_2-O_2 APU REQUIREMENTS

POWER	
PEAK, HP	200
NORMAL, HP	45
HYDRAULIC	4000 PSI, 70 GPM
ELECTRIC	15 kW, 400 Hz, 120/208 V, 3 PHASE
LIFE, HR	1000
ACTIVE MISSION TIME, HR	UP TO 3
NO. OF MISSIONS	100

"AUXILIARY POWER UNIT DESIGN STUDIES"

W. L. BURRISS

M. L. HAMILTON

AIRESEARCH

TECHNICAL MANAGER

J. P. JOYCE

LEWIS RESEARCH CENTER

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SPACE SHUTTLE APU SYSTEM SELECTION AND PERFORMANCE

by: W.L. Burriss and M.L. Hamilton
AiResearch Manufacturing Company, Los Angeles

INTRODUCTION

This paper presents a portion of the studies performed by AiResearch for NASA Lewis Research Center under Contract NAS 3-14408. These studies have synthesized a number of possible system configurations for the Space Shuttle APU. All of these configurations meet the primary APU requirements, to provide output hydraulic and electric power to the vehicle in accordance with specified power and altitude profiles for the booster and orbiter missions. A principal study requirement is that common hardware be used on both stages of the Shuttle vehicle.

To select that system configuration best meeting the requirements, AiResearch has evaluated the performance of each of the candidate APU systems. Although primary emphasis has been placed on determining the amount of propellant consumed during a vehicle mission, other performance parameters, such as cost, reliability, ease of maintenance, etc., have also been considered in the NASA-supplied evaluation criteria. It should be noted that over 70 percent of the total APU system weight consists of the propellant and its storage tanks; therefore, optimization of propellant consumption is of primary concern.

The evaluation has indicated that recuperated hydrogen-oxygen APU using high-pressure gas supplied by the vehicle auxiliary propulsion system (APS) is the preferred concept. A self-contained APU system using high-pressure propellant storage tanks is a strong runnerup. Because of the strong similarity in components between these two concepts, and between a low pressure tank pump supplied APU, most of the information generated is applicable to all three systems.

CANDIDATE APU SYSTEM CONCEPTS

Preliminary analyses performed during the first two months of the study indicated that it was possible to reduce the number of candidate APU systems to the five shown on the accompanying chart. Three of these systems use hydrogen-oxygen propellants, one is a dual-mode system using hydrogen-oxygen during the space portion of the flight with a hydrogen-fueled air breathing gas turbine for the atmospheric portion, and one is a monopropellant system using 75-24-1 propellant (75 percent hydrazine, 24 percent hydrazine nitrate, 1 percent water). In the hydrogen-oxygen systems, the hydrogen- oxygen exhaust products are used to preheat the hydrogen entering the combustor, thus reducing the amount of oxygen required to bring the hydrogen up to the desired turbine inlet temperature.

Hydrogen has been used as a heat sink for component cooling because of its excellent heat transfer properties (heat capacity about 3.5 times that of water per deg F, thermal conductivity about 7 times that of steam at 600°F, 600 psia) and because the addition of the heat further reduces the cycle oxygen/fuel (O/F) ratio. When a hydrogen flow is not available, it is necessary to use either ram air (as done in the dual-mode cycle where an air duct is required for the air-breathing engine), or to boil water to obtain the necessary heat sink.

Since some of the candidate cycles store the propellants at a pressure less than that required by the combustor, it is necessary to provide some means of pressurizing the propellants. Examination of gearbox, pneumatic, hydraulic, and electric drives indicated that electric propellant pump drive is preferred because of the continuous availability of electric power and because of the self-start capability and installation flexibility with such a concept.

CANDIDATE APU SYSTEM CONCEPTS

FINAL CONCEPTS

CANDIDATE SYSTEM	PROPELLANTS	PROPELLANT TANKAGE	SYSTEM HEAT SINK	EXHAUST ENERGY UTILIZATION	COMBUSTOR PRESSURATION METHOD
LOW-PRESSURE CRYOGENIC LIQUID SUPPLIED	H_2-O_2	LOW-PRESSURE CRYOGENIC	H_2	RECUPERATION	ELECTRIC-DRIVE CRYOGENIC PUMPS
INTEGRAL HIGH-PRESSURE CRYOGENIC SUPPLIED	H_2-O_2	HIGH-PRESSURE CRYOGENIC	H_2	RECUPERATION	DIRECT FEED
HIGH-PRESSURE GASEOUS SUPPLIED	H_2-O_2	NONE	H_2	RECUPERATION	DIRECT FEED
DUAL-MODE	H_2-O_2 H_2-AIR	HIGH-PRESSURE CRYOGENIC	H_2 OR AIR	H_2 RECUPERATION, NONE WITH AIR	DIRECT FEED
MONOPROPELLANT	75-24-1	LOW-PRESSURE WITH BLADDERS	WATER	NONE	ELECTRIC-DRIVE MONOPROPELLANT PUMPS

PRELIMINARY, REJECTED CONCEPTS

- LOW-PRESSURE HYDROGEN-OXYGEN GAS FEED
- ALL NON-RECUPERATED HYDROGEN-OXYGEN CYCLES
- OPEN BRAYTON CYCLE USING HYDROGEN-OXYGEN

GARRETT

APU THERMAL MANAGEMENT

APU thermal management involves consideration of two functions:

(a) Propellant flow conditioning; and
(b) Dissipation of internally generated waste heat

At the combustor, the inlet propellant flow should be preheated sufficiently (to temperature on the order of 265°R for hydrogen and 315°R for oxygen) to avoid two-phase flow downstream of the propellant injectors. This thermal conditioning is performed most efficiently from a cycle thermodynamic standpoint by means of a recuperative heat exchanger using turbine exhaust gas to preheat the inlet cryogen flow. Two temperature limitations must be observed in the design and location of this recuperative heat exchanger as a result of the presence of steam in the turbine exhaust gas. First, to avoid freezing on the exhuast gas side, the wall temperature of the heat exchanger must be maintained above 460°R. Second, the bulk temperature of the exhuast gas from the recuperator must be above 700°R to avoid condensation of moisture and possible problems with water freezing in the discharge ducting. The first requirement dictates flow recirculation with propellants supplied at cryogenic temperatures. The second requirement is accomodated by proper sizing of the heat exchanger.

Dissipation of the internally generated heat is also facilitated by the recirculation loop to provide hydrogen at the proper temperature level to serve as the heat sinks for the APU loads (hydraulic fluid, gearbox lubricant, etc.). To avoid congealing of lubricant or hydraulic fluid in their heat exchangers, the hydrogen inlet temperatures to these components should be maintained above 460°R.

The importance of the freezing problem in the recuperator cannot be overestimated. If the recirculation loop is eliminated in the interests of system simplicity, it will seriously compromise reliability of that component. AiResearch experience with the glycol-to-cryogenic hydrogen heat exchangers in the Dyna-Soar (X-20) program indicates the desirability of this approach.

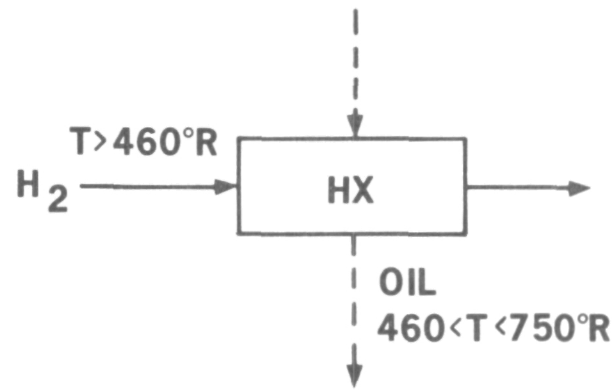
APU THERMAL MANAGEMENT

CONSIDERATIONS

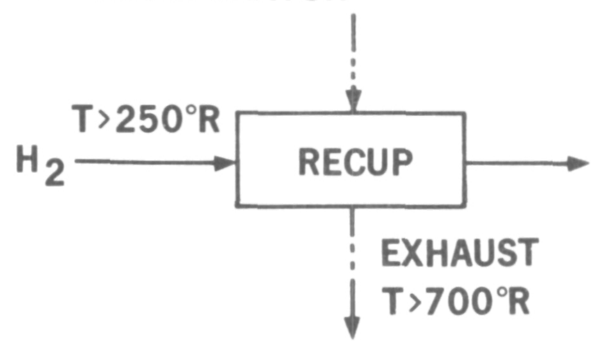
- APU MUST BE SELF-COOLED
- HEAT INPUT TO HYDROGEN WILL REDUCE PROPELLANT CONSUMPTION (LOWER O/F RATIO)
- HEAT SINK TEMPERATURES MUST BE CORRECT FOR EACH COMPONENT

RESTRAINTS

- LUBE/HYDRAULIC OIL HEAT EXCHANGERS



- RECUPERATOR



TYPICAL APU CANDIDATE SYSTEM SCHEMATIC

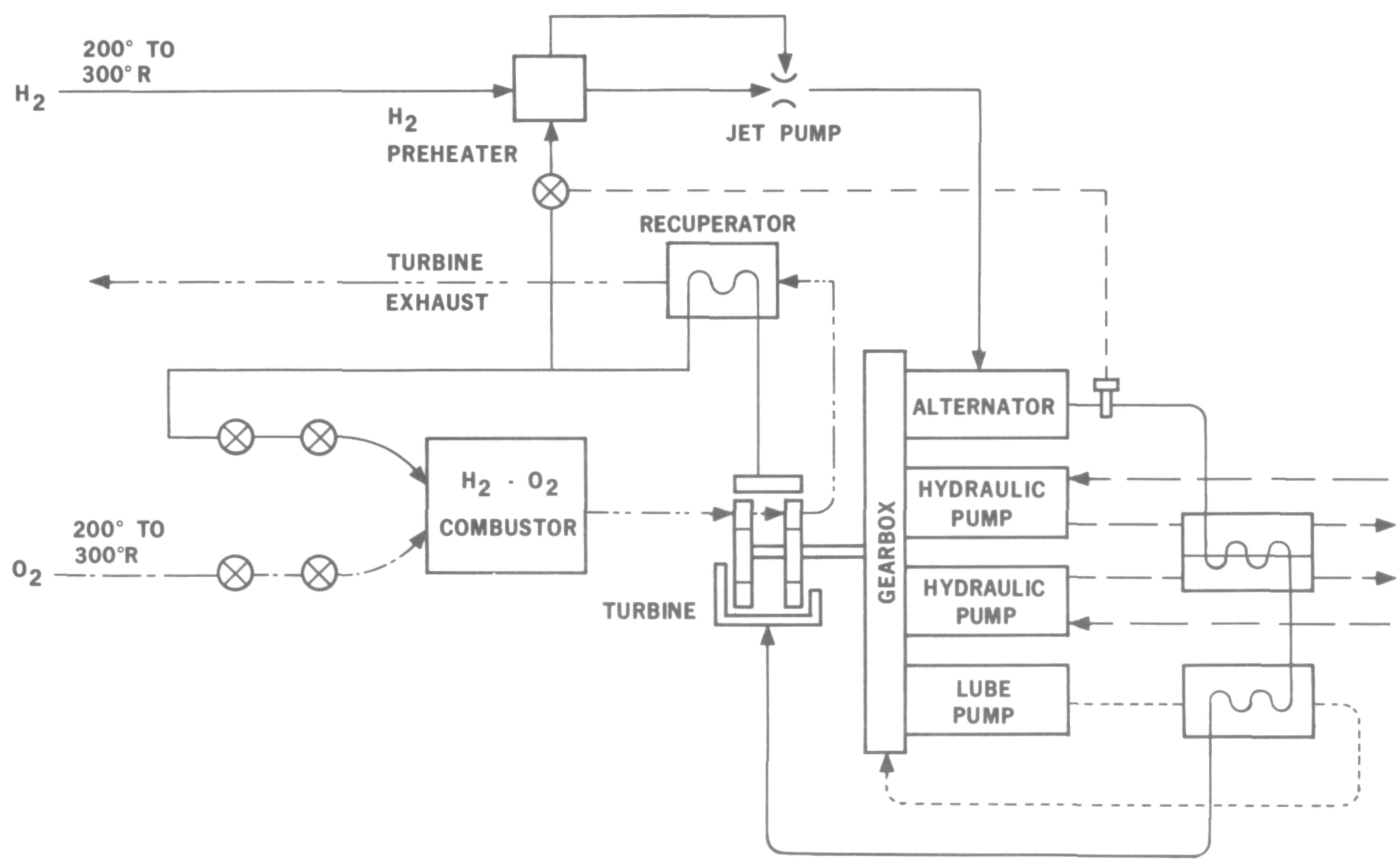
The accompanying chart shows a schematic of the selected APU system, a gas feed system in which pressurized gas from the APS is supplied to the APU. The system places the component heat loads (alternator, hydraulic pumps, gearbox, and turbine) upstream of the recuperator so that the hydrogen temperature inlet to the recuperator is well above 250^{0}R and will allow maximum cycle thermal efficiency (by maximizing the temperature at which the hydrogen enters the combustor). To provide a temperature of about 460^{0}R at the inlet to the hydraulic fluid heat exchanger, a portion of the hot hydrogen leaving the recuperator is recycled through the component cooling loop. Recycling is accomplished by equalizing the hot and cold hydrogen temperatures with a parallel flow hydrogen preheater, and then mixing the two flow streams in a jet pump. This method of obtaining a recycle flow is standardly used in numerous aircraft environmental control systems (ECS), such as the Huey Cobra ECS. In comparison to a fan or compressor, the jet pump method of providing recirculation requires only a single moving part, the valve controlling the amount of recycle flow.

The schematic indicates a hydrogen cooled alternator; however, an oil cooled unit could be used by using the gearbox oil flow to pass through the alternator. Also, turbine cooling is shown since the turbine inlet temperature is 1800^{0}F. NASA has lowered this to 1600^{0}F, thus eliminating the need for cooling.

System power/speed control is provided by modulating the combustor pressure using throttling valves on the combustor inlet. A low-pressure-drop diffusion-type combustor, similar to the one previously tested by AiResearch will be used to generate the hot gas for expansion across the two-stage pressure-compounded axial flow turbine. The turbine operates at 70,000 rpm and has a first-stage pitch-line velocity of 1800 fps with blades of about 0.265 in. height. Because of the lower temperature in the second stage, it is possible to operate it at a 2000 fps pitch-line velocity. Both stages are partial admission.

TYPICAL APU CANDIDATE SYSTEM SCHEMATIC

GAS FEED SYSTEM



METHOD OF DETERMINING APU SYSTEM PERFORMANCE

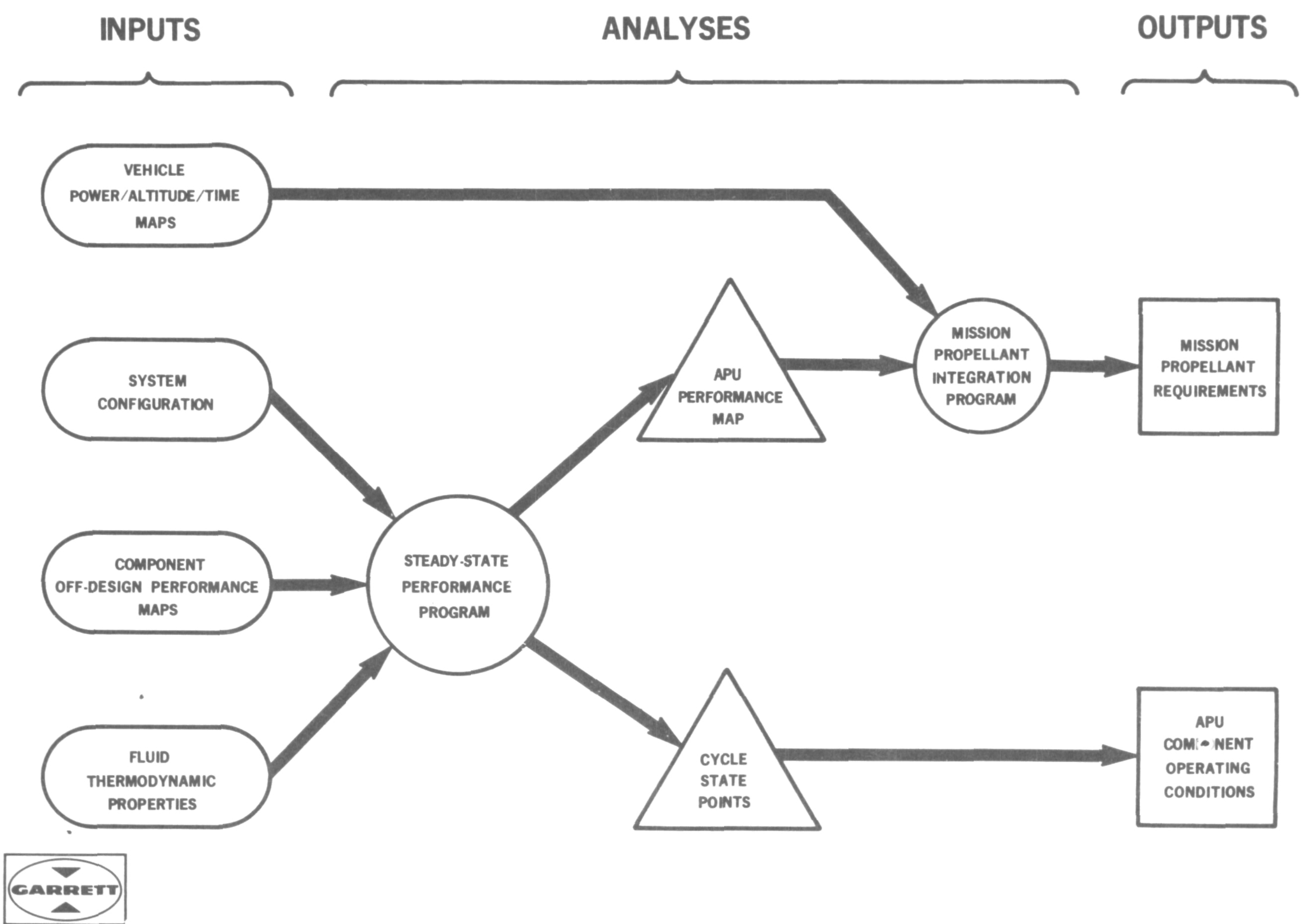
This chart illustrates the procedure used to establish the propellant consumption and operating conditions for the candidate APU cycles. Preliminary hand calculations were used to establish the approximate design points for each of the components and available off-design performance programs were used to predict the performance of the jet pump, the heat exchangers, and the turbine. This information, together with the logic establishing the system configuration, was used to generate a steady-state performance program modeling all of the components. The program includes the effects of pressure losses in the ducting and in the components. An additional input to the program was a subroutine generated under previous programs that obtained the complete thermodynamic properties of cryogenic fluids as a function of any two of four thermodynamic variables (temperature, pressure, enthalpy, and density).

The output of the steady-state program is a complete summary of the state points throughout the candidate system. By running a series of conditions (varying the load power and the ambient pressure), it is possible to generate an APU performance map that summarizes the propellant requirements as a function of the load power and the ambient pressure.

Finally, by combining the data from the performance map with the profiles for the required APU power output and ambient pressure (altitude) as a function of time, it is possible to establish APU propellant consumption for each segment of any vehicle mission. And, by varying some of the system primary design variables, such as turbine design point (power output, and discharge pressure) and maximum system pressure, it is possible to determine the sensitivity of the candidate systems to various design parameters, such as exhaust ducting size (hence pressure drop), combustor losses, heat exchanger effectiveness, etc.

This analytical process is similar to that developed at AiResearch for design of aircraft environmental control systems. Many of the APU components are also similar to those used in aircraft ECS.

METHOD OF DETERMINING APU SYSTEM PERFORMANCE



STEADY-STATE PERFORMANCE PROGRAM SUMMARY

The steady-state performance program is the key analytical tool used in evaluation of the candidate concepts. This program outputs the system cycle state points and component performance for any desired input load power and ambient pressure. It uses programming techniques developed over the past 10 years at AiResearch to support design of aircraft environmental control systems (which utilize components similar to those required for the APU; heat exchangers, jet pumps, control valves, pressure regulators, turbines, etc.). These techniques allow the program to iterate to convergence on the final answer, starting with "educated" guesses for the primary variables (in this case, turbine discharge pressure, O/F ratio, turbine nozzle inlet pressure, and alternator cooling flow discharge temperature). The convergence loops are nested so that each iteration on any given variable required iteration to convergence on all previously iterated variables.

The component performance data are input to the program as a series of performance maps which can be altered easily, either by being changed completely or by using scale factors to "rubberize" the component. About 50 component maps are input.

The hydrogen, oxygen, and water thermodynamic data are also stored as maps which must be read by the program. Each fluid map uses about 10,000 bytes of computer storage.

The principal advantages of this program are that it allows determination of exact system performance (within the accuracy of the performance maps) throughout the complete operating regime of the APU. Such variables as ducting losses, component pressure drops, etc. are all accounted for in the program. Additionally, by using actual fluid thermodynamic properties, the program accounts for the large changes in cryogenic fluid specific heats. It should be noted that such analytical sophistication would be virtually impossible on an analog computer, due to the large number of variables.

STEADY-STATE PERFORMANCE PROGRAM SUMMARY

PROGRAM

- DIGITAL PROGRAM WRITTEN IN FORTRAN V FOR UNIVAC 1108 COMPUTER, USES 13 SUBROUTINES

PRIMARY INPUTS

- COMPONENT OFF-DESIGN PERFORMANCE MAPS (ABOUT 50 MAPS TOTAL)
- FLUID THERODYNAMIC PROPERTIES (PRESSURE, TEMPERATURE, ENTHALPY, DENSITY) FOR HYDROGEN, OXYGEN, AND WATER

SOLUTION TECHNIQUE

- ITERATE TO CONVERGENCE USING 5 NESTED, ITERATIVE LOOPS
- APPROXIMATE EQUATIONS ARE USED TO ESTABLISH FIRST GUESSES FOR ITERATIONS

EVALUATIONS CONDUCTED

- ABOUT 400 PERFORMANCE POINTS EVALUATED
- 5 DIFFERENT SYSTEM CONFIGURATIONS CONSIDERED

ADVANTAGES OF THIS PROGRAMMING TECHNIQUE

- ALLOWS EXACT ANALYSIS INCLUDING COMPONENT OFF-DESIGN PERFORMANCE, DUCTING PRESSURE DROPS, EXPANSION LOSSES, AND REAL FLUID PROPERTIES

GARRETT

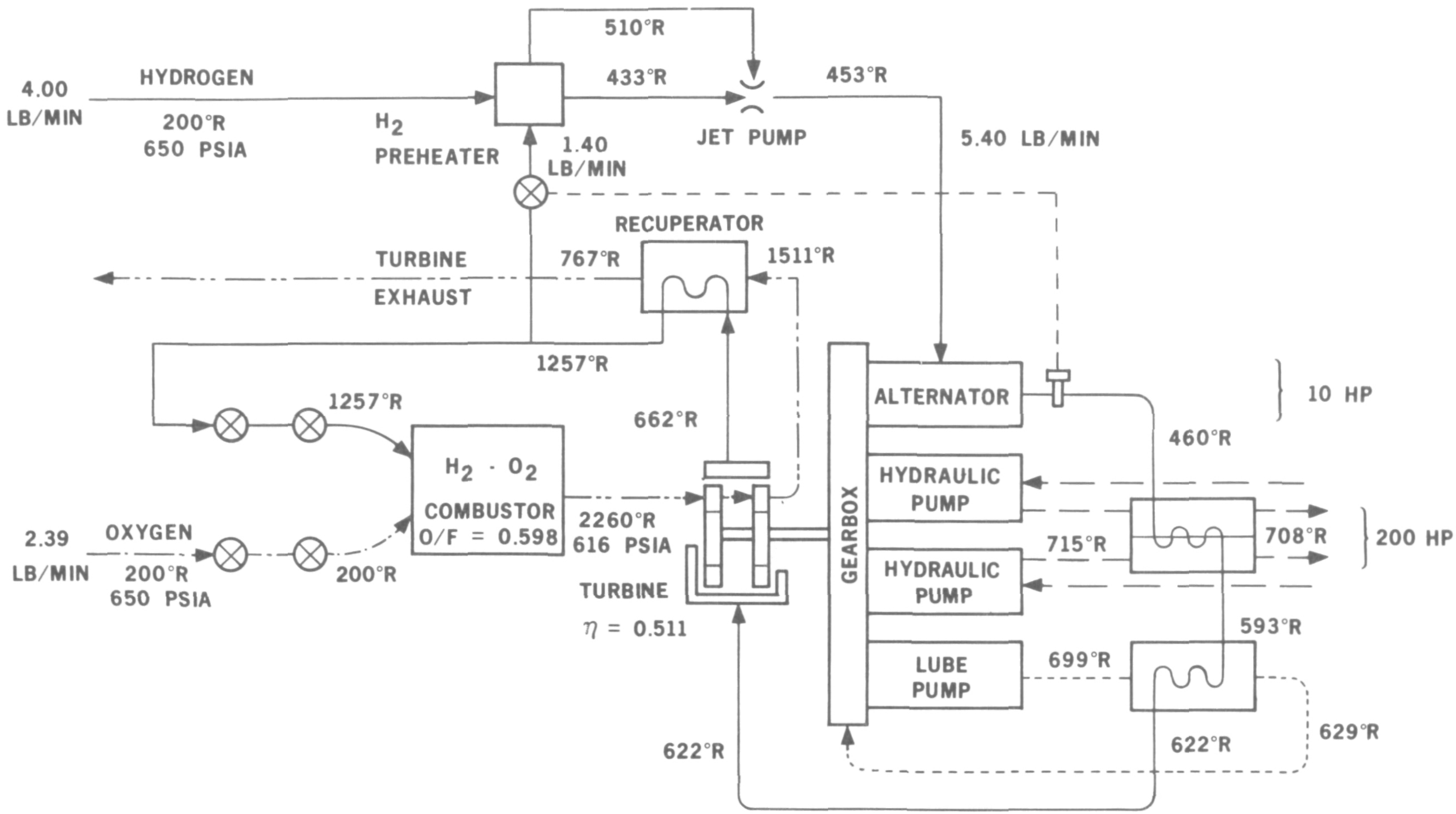
TYPICAL APU SYSTEM STATE POINTS

This chart shows the selected, gas feed APU system state points when the system is operating at full power at sea level. The state point data are used to generate the APU performance maps (next chart) and to insure that the individual components are operating at the desired temperature and performance levels. Typically, the initial state point data using the initial component off-design performance maps, indicate that changes in the component design points (such as increasing the lube oil heat exchanger effectiveness at its design point) will improve the system performance, either by providing more desiirable operating temperatures, or by reducing the fixed weight of the components.

For example, in the chart shown, the turbine exhaust gas temperature leaving the recuperator is 767°R. This value is 67°R in excess of the minimum discharge temperature desired to prevent freezing; thus, it would appear that the recuperator performance could be improved somewhat to add that additional 67°R of heat to the incoming hydrogen flow. However, at the low power, space ambient condition, the discharge temperature from the recuperator is only 700°R so that that point represents the design point for the recuperator.

TYPICAL APU SYSTEM STATE POINTS

SEA LEVEL FULL POWER



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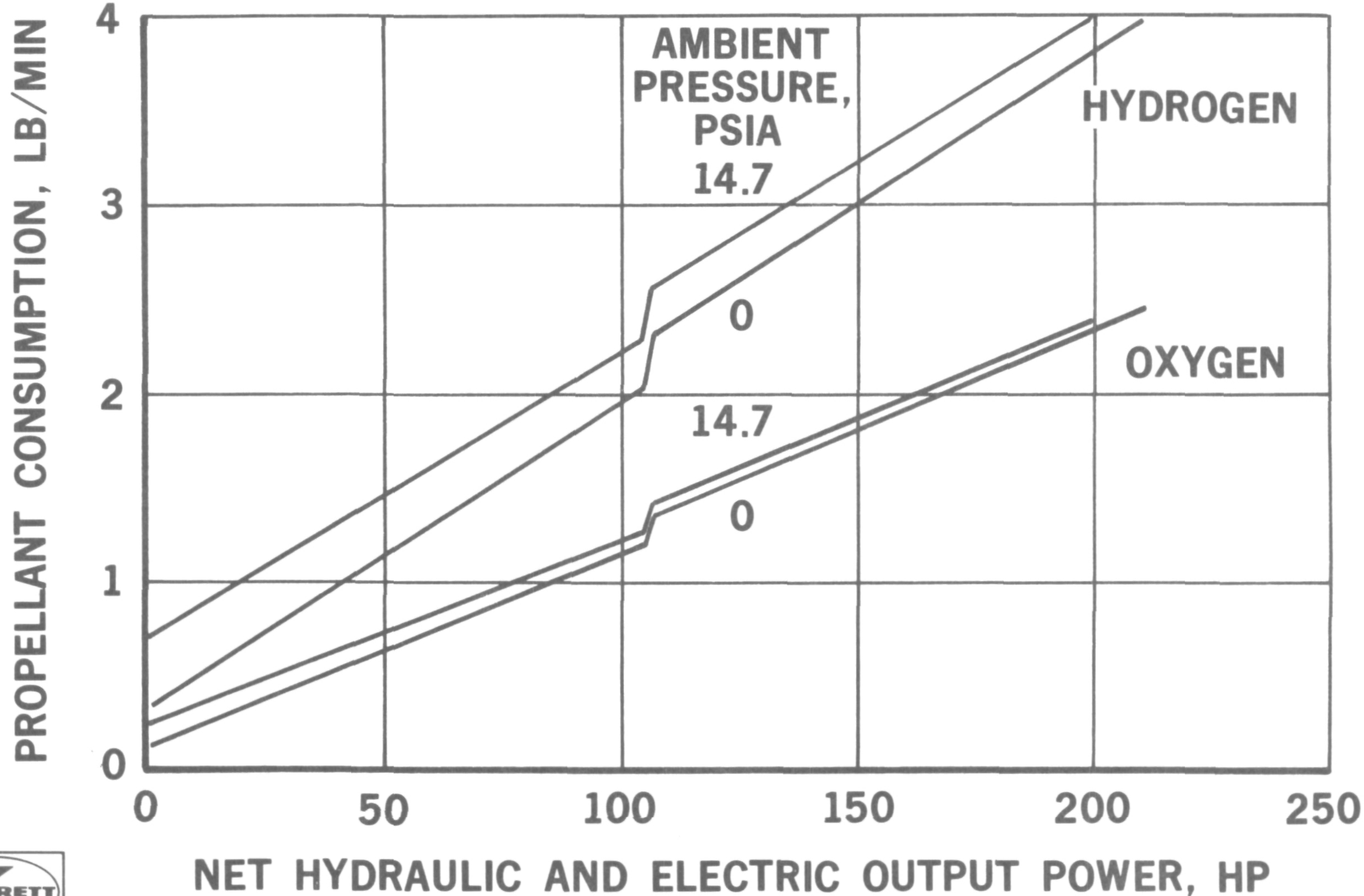
TYPICAL APU SYSTEM PERFORMANCE MAP

The cycle state point performance was analyzed as a function of turbine output power (and ambient pressure for a fixed system design) to obtain system performance maps of the type shown in the chart. This chart shows the hydrogen and oxygen flow required to provide a given APU net output power (0 to 210 hp) as a function of ambient pressure (0 to 14.7 psia). The discontinuities in the fuel consumption lines at 104-106 hp represent the power range in which one hydraulic pump is loaded or unloaded. At output powers below this range, system fixed losses and fuel consumption are reduced by approximately 10 percent by unloading one of the hydraulic pumps. Since the Space Shuttle APU will operate much of the time at low output power, performance in this range is important to minimizing propellant weight. It should be noted that the APU system performance map shown here is independent of shuttle vehicle mission profile and depends upon turbine design point parameters, propellant inlet state, and system component performance. During these studies, turbine design points were varied to include the effects of turbine inlet pressure, and sea-level full-power and altitude mode-power design points.

TYPICAL APU SYSTEM PERFORMANCE MAP

200°R, 650 PSIA DELIVERY GAS FEED SYSTEM WITH TURBINE DESIGNED AT ALTITUDE, MODE POWER

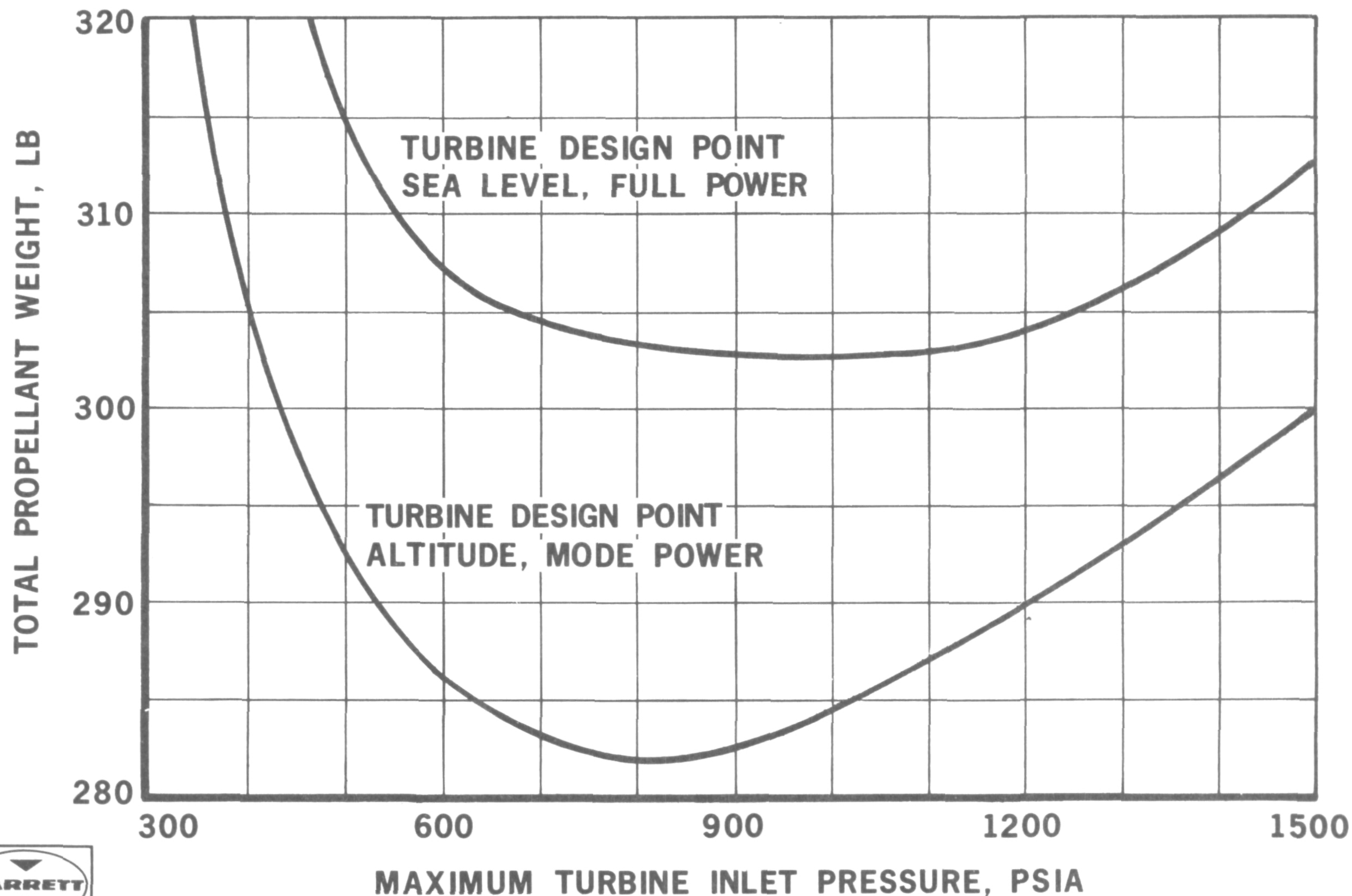


TYPICAL OPERATING PRESSURE/TURBINE DESIGN POINT OPTIMIZATION

The mission profile parameters (hydraulic and electric power output and vehicle altitude as functions of time) are used in the mission integration program with the APU performance maps to determine the total propellant requirements for a given cycle configuration and turbine design. By systematically varying the system design parameters to establish APU system performance maps which in turn are used in the mission integration program, the effect of turbine design parameters on total propellant requirement can be analyzed. The chart shows an example of this for one type of system (the low-pressure liquid supplied system). It gives the booster vehicle total propellant requirements as a function of turbine inlet design pressure for sea-level full-power and altitude mode-power design points. In this case, minimum propellant requirement is obtained for the altitude mode-power design point at a turbine design inlet pressure on the order of 850 psia. Use of the altitude design point leads to an approximate 8 percent reduction in fuel consumption over the sea level design point. Although the other systems will optimize at different pressure levels, all show an advantage for the altitude design point. The power level and the altitude design point for optimum performance depend upon the mission profile. In this case, the altitude design-point for the booster vehicle is 10,000 ft at average 80 shp output. More recent booster vehicle profiles indicate a higher altitude design point for an equivalent condition.

TYPICAL OPERATING PRESSURE/TURBINE DESIGN POINT OPTIMIZATION

LOW-PRESSURE LIQUID SUPPLIED SYSTEM FOR BOOSTER VEHICLE



TYPICAL APU SYSTEM EVALUATION MATRIX

Once the five candidate systems have been evaluated with respect to performance and analyzed in sufficient detail to determine near-optimum concepts for each type of system, it remains to compare these systems in accordance with the NASA-supplied evaluation criteria as follows:

Cost Factors

Factor	Weighting
Low weight	25
High flexibility	20
Ease of development	10
Ease of manufacturing	5
Ease of maintenance	5

Reliability Factors

Factor	Weighting
Simplicity	30
Experience	5

The chart shows a typical evaluation matrix, in this case for system weight. Total system weight is given for the booster and orbiter missions for each of the five systems. The five systems are then rated relative to the lowest weight system for the booster and orbiter missions. Then, a composite rating is obtained in which the orbiter system is weighted by a factor of six relative to the booster system (this reflects the 6 lb in booster weight required for an incremental one lb increase in orbiter weight). In this case, the liquid cryogen supplied system (supplied from low-pressure shared tankage) shows the highest rating for minimum weight.

TYPICAL APU SYSTEM EVALUATION MATRIX

SYSTEM	SYSTEM WEIGHT, LB		SYSTEM RATING		
	BOOSTER MISSION	ORBITER MISSION	BOOSTER MISSION	ORBITER MISSION	TOTAL MISSION
LOW-PRESSURE CRYOGENIC LIQUID SUPPLIED (SHARED TANKS)	537	310	1.000	0.990	1.000
INTEGRAL HIGH-PRESSURE CRYOGENIC SUPPLIED	648	425	0.786	0.723	0.741
HIGH-PRESSURE GASEOUS SUPPLIED (200°R, 650 PSIA)	573	307	0.938	1.000	0.993
DUAL MODE	629	473	0.854	0.650	0.691
MONOPROPELLANT	1242	557	0.433	0.551	0.523

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SUMMARY OF APU EVALUATION

This chart shows the ratings of the five candidate systems for the various evaluation factors given previously. The high-pressure gas feed system is found to have the highest overall total weighted rating and to be significantly better than the next three systems (low-pressure cryogenic liquid supplied system, integral high-pressure cryogenic supplied system, and monopropellant system) which are rated close together and appreciably higher than the fifth rated system (the dual-mode system). The low dual-mode system rating results to a large extent from its complexity and lack of weight advantage for the vehicle mission profiles used in the comparisons. The primary merit of the dual-mode system in facilitating maintenance and accomodating the ferry missions is not recognized in the evaluation matrix used here.

SUMMARY OF APU EVALUATION

	SYSTEM RATINGS				
	LOW-PRESSURE CRYO LIQUID	INTEGRAL HIGH-PRESSURE CRYO	HIGH-PRESSURE GAS FEED	DUAL MODE	MONOPROPELLANT
RELATIVE RANKING	3	2	1	5	4
TOTAL WEIGHTED RATING	73.0	73.1	93.4	52.7	69.9
WEIGHT	1.00	0.74	0.99	0.69	0.52
FLEXIBILITY	0.65	0.69	0.92	0.42	0.69
EASE OF DEVELOPMENT	0.53	0.73	0.89	0.53	1.00
EASE OF MANUFACTURING	0.63	0.76	0.90	0.49	0.76
EASE OF MAINTENANCE	1.00	0.56	1.00	0.42	0.71
SIMPLICITY	0.69	0.78	0.95	0.49	0.86
EXPERIENCE	0.17	0.50	0.50	0.50	1.00

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"AUXILIARY POWER UNIT DESIGN STUDIES"

R. S. SIEGLER

ROCKETDYNE

TECHNICAL MANAGER

H. M. CAMERON

LEWIS RESEARCH CENTER

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TITLE:	SPACE SHUTTLE AUXILIARY POWER UNIT
CONTRACT:	NAS3 - 14407
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COMPANY:	ROCKETDYNE, DIV. OF NORTH AMERICAN ROCKWELL
NASA PROJECT MANAGER	H. CAMERON

Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY POWER UNIT

FOR THE PAST YEAR, ROCKETDYNE DIVISION OF NORTH AMERICAN ROCKWELL HAS BEEN ENGAGED IN A DESIGN AND ANALYSIS STUDY OF THE AUXILIARY POWER UNITS REQUIRED FOR THE SPACE SHUTTLE VEHICLES (BOOSTER AND ORBITER). THE PROGRAM IS COMPROSED OF TWO PHASES AND PHASE I - SYSTEM SYNTHESIS AND SELECTION HAS BEEN COMPLETED. PHASE II - PRELIMINARY DESIGN OF SELECTED SYSTEMS IS CURRENTLY UNDER WAY AND IS IN THE PROCESS OF DEFINING THE OPTIMUM SYSTEM.

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PROGRAM OBJECTIVE

AUXILIARY POWER UNIT

- DESIGN/ANALYSIS STUDY
  - PHASE I - SYSTEM SYNTHESIS AND SELECTION
  - PHASE II - PRELIMINARY DESIGN OF SELECTED SYSTEM

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AUXILIARY POWER UNIT

THE APU PROVIDES THE PRIMARY HYDRAULIC POWER FOR THE SPACE SHUTTLE VEHICLES. THIS POWER IS USED TO ACTIVATE THE AERODYNAMIC SURFACES AND TO PROVIDE CONTROL FUNCTIONS FOR THE VEHICLES. IN THE DESIGN, RELIABILITY IS THE FOREMOST CONSIDERATION, AND REDUNDANCY IS UTILIZED TO ACHIEVE THIS. UP TO SIX APU'S ARE USED ON THE BOOSTER AND AT LEAST THREE ON THE ORBITER TO ACTUATE THE THREE INDEPENDENT HYDRAULIC SYSTEMS ON EACH VEHICLE.

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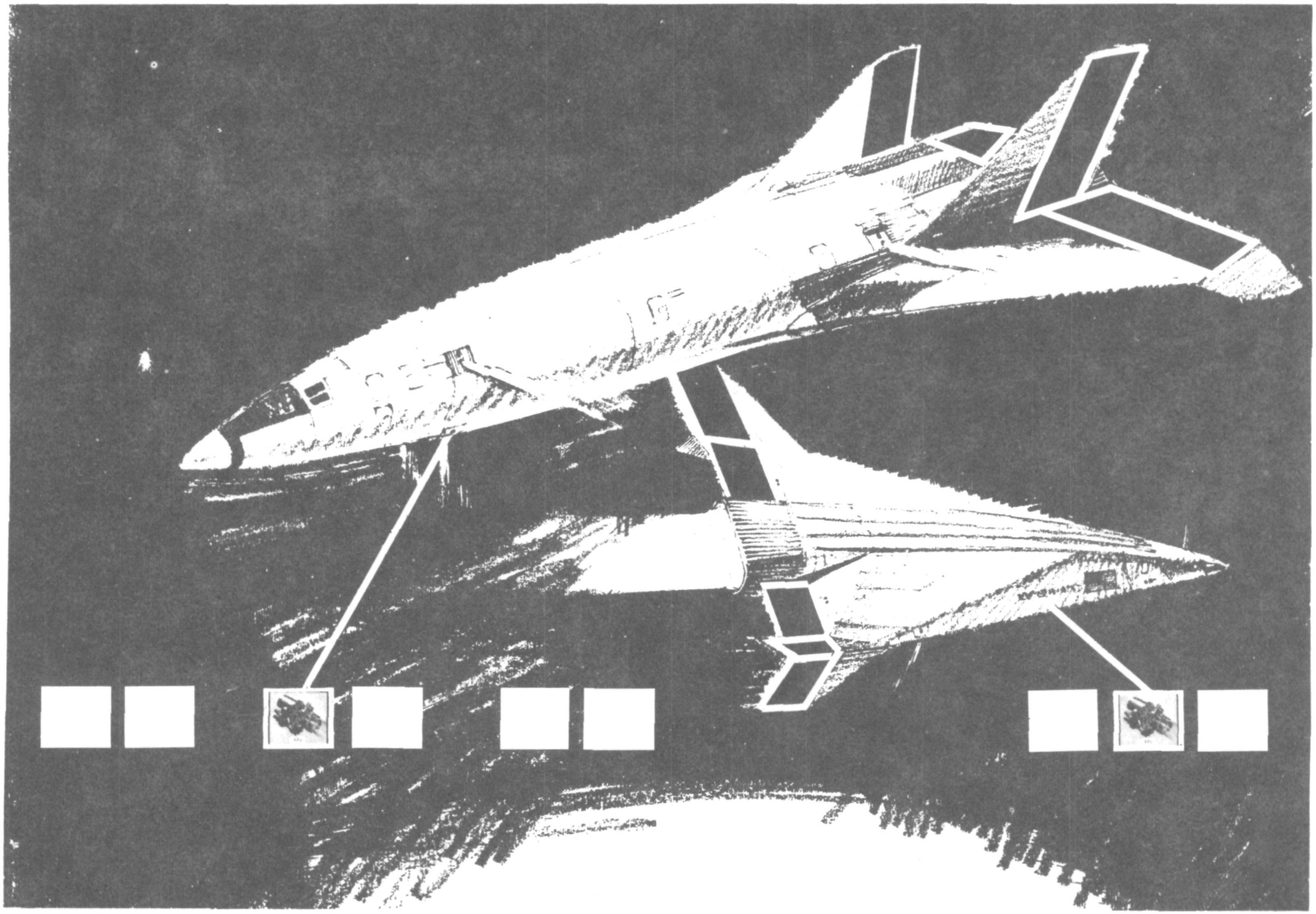
FLIGHT CONTROL POWER



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AUXILIARY POWER UNIT

IN SELECTING THE SYSTEM, EMPHASIS HAS BEEN PLACED ON MINIMUM WEIGHT. VEHICLE FLIGHT WEIGHT IS STRONGLY COST RELATED. WHEN COMPARING AN OPTIMIZED SYSTEM SUCH AS A HYDROGEN/OXYGEN FUELED APU WITH ONE IN WHICH STORABLE PROPELLANTS (E.G., HYDRAZINE) ARE USED, THE WEIGHT REQUIREMENT (ATTRIBUTABLE MOSTLY TO THE MUCH LARGER AMOUNT OF PROPELLANT REQUIRED) IS APPROXIMATELY 2:1.

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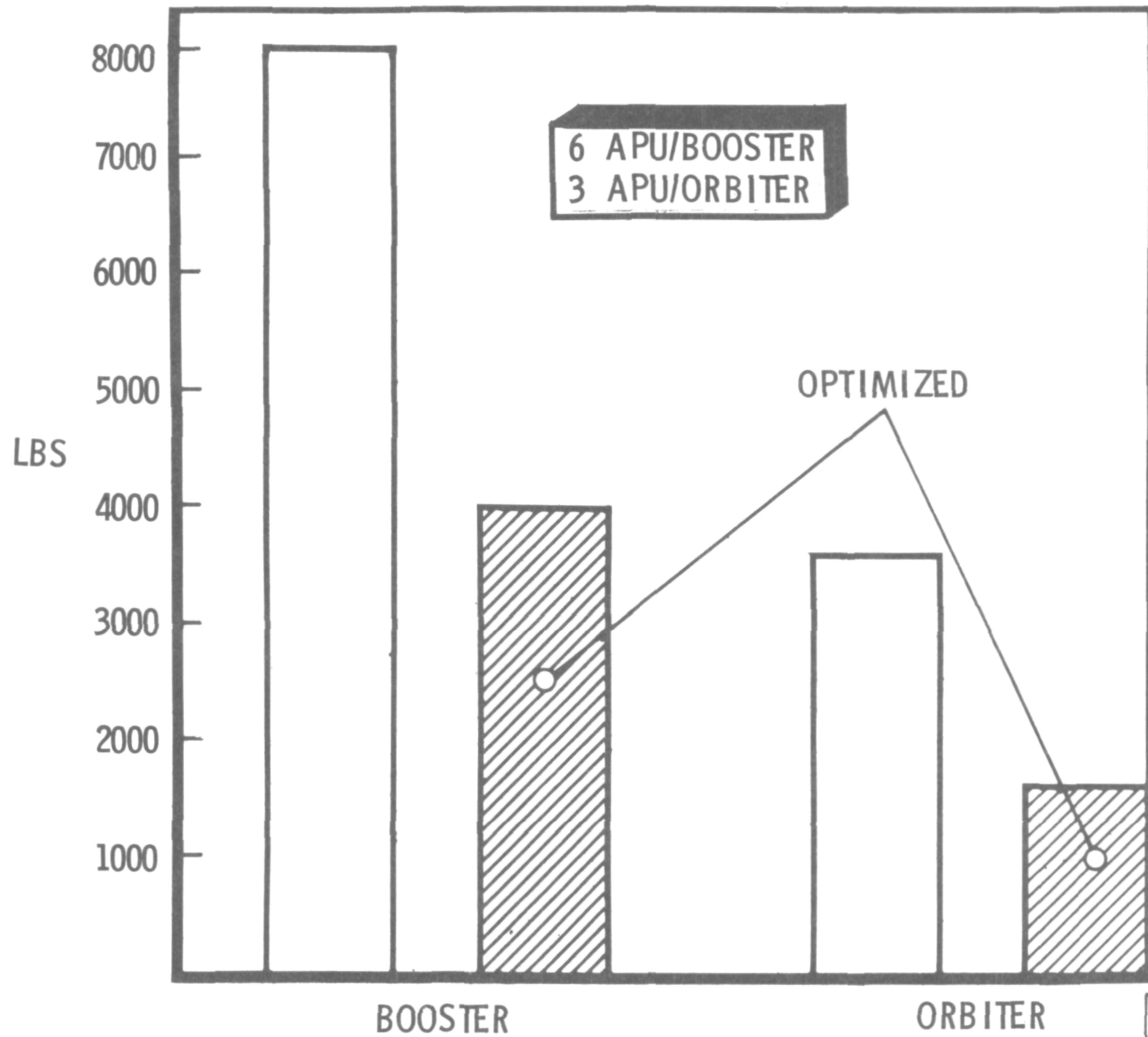
MINIMUM WEIGHT / COST



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AUXILIARY POWER UNIT

ONE OF THE KEY REQUIREMENTS FOR THE APU IS FLEXIBILITY. THIS RESULTS BECAUSE A WIDE RANGE OF MISSION CONDITIONS (E.G., SEA LEVEL TO SPACE, ENVIRONMENTAL TEMPERATURE) IS ENCOUNTERED IN THE LIFE OF THE SPACE SHUTTLE, AND BECAUSE OF LARGE VARIATIONS IN VEHICLE IMPOSED CONDITIONS (E.G., POWER OUTPUT, COOLING LOAD).

FLEXIBILITY

- MISSION ADAPTABILITY
- VEHICLE IMPOSED VARIATIONS

THE APU WILL BE REQUIRED TO RESPOND TO POWER DEMANDS VARYING FROM LESS THAN 25 TO 100 PERCENT WITHIN 75 MILLISECONDS. RAPID CHANGES IN PROPELLANT SUPPLY AND HYDRAULIC-OIL COOLING LOADS TOGETHER WITH CHANGING ENVIRONMENTAL CONDITIONS DURING FLIGHT (G LOADS, PRESSURE CHANGES) PRESENT A CHALLENGE TO THE DESIGNER, ESPECIALLY BECAUSE ALL OF THESE FLUCTUATIONS MAY OCCUR SIMULTANEOUSLY. HE MUST DESIGN FOR EFFICIENT OPERATION AND LOW PROPELLANT CONSUMPTION TO REDUCE WEIGHT, WHILE COPING WITH THE LARGE POWER TURNDOWN RATIO IMPOSED BY THE VEHICLE REQUIREMENTS.

VEHICLE/MISSION VARIATIONS

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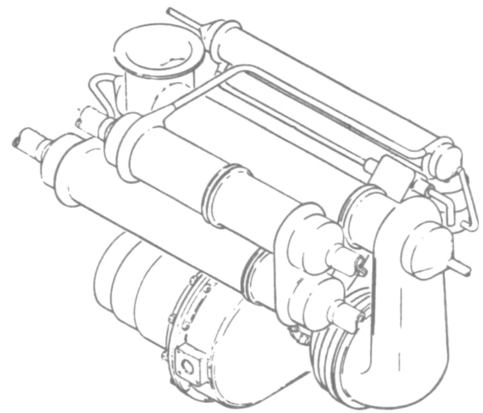
PROPELLANT SUPPLY
1000/500 PSIA
$300/75^{o}R$

HYDRAULIC
COOLING
40/15 HP



FLIGHT CONTROL
POWER
400/25 HP

EXHAUST PRESSURE
14.7/0 PSIA

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AUXILIARY POWER UNIT

ROCKETDYNE ENGINEERS HAVE, THEREFORE, DRAWN ON THE VAST RESERVOIR OF EXPERIENCE WITHIN THEIR OWN COMPANY AND THE EXPERIENCE AVAILABLE THROUGH QUALIFIED SUBCONTRACTORS. IN MANY INSTANCES DIRECTLY APPLICABLE EXPERIENCE IS AVAILABLE ON HARDWARE ALREADY DEVELOPED, OR CURRENTLY UNDER DEVELOPMENT, ON RELATED PROGRAMS. FOR EXAMPLE, DESIGN STUDIES AND TEST DATA AVAILABLE ON THE SUPERSONIC TURBINE DEVELOPMENT PROGRAM HAVE BEEN VALUABLE IN THE DESIGN ANALYSIS OF THE TURBINE REQUIRED.

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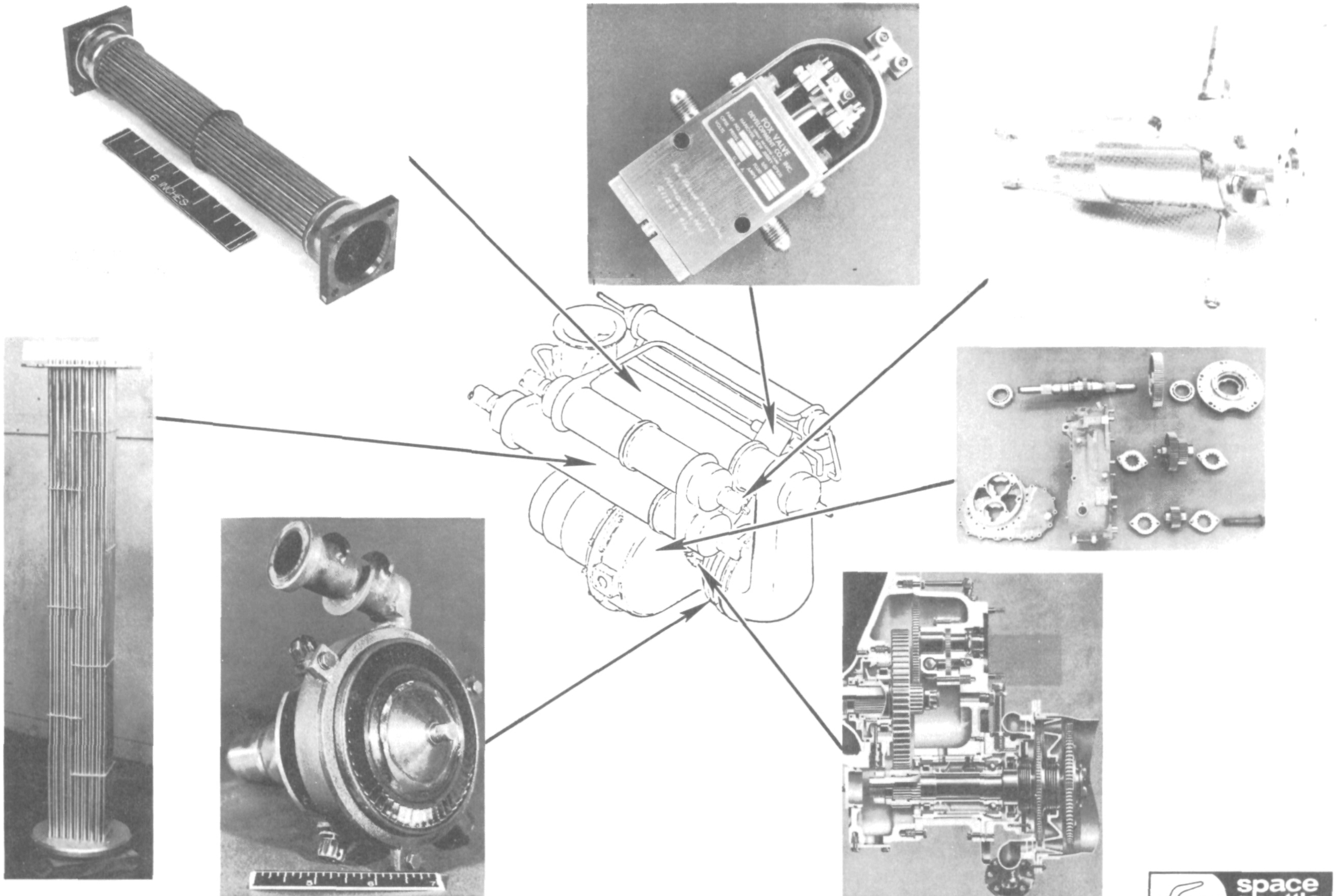
APU RELATED EXPERIENCE



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AUXILIARY POWER UNIT

A SUPERSONIC TURBINE DESIGN IS USED IN THE TURBO POWER UNIT. FOR THE BASELINE SYSTEM SELECTED IN THE STUDY, A TWO-DISK, PRESSURE STAGED TURBINE WAS SELECTED. THE HYDRAULIC PUMPS AND GENERATOR ARE PURCHASE ITEMS ATTACHED TO THE GEARBOX.

TURBO POWER UNIT

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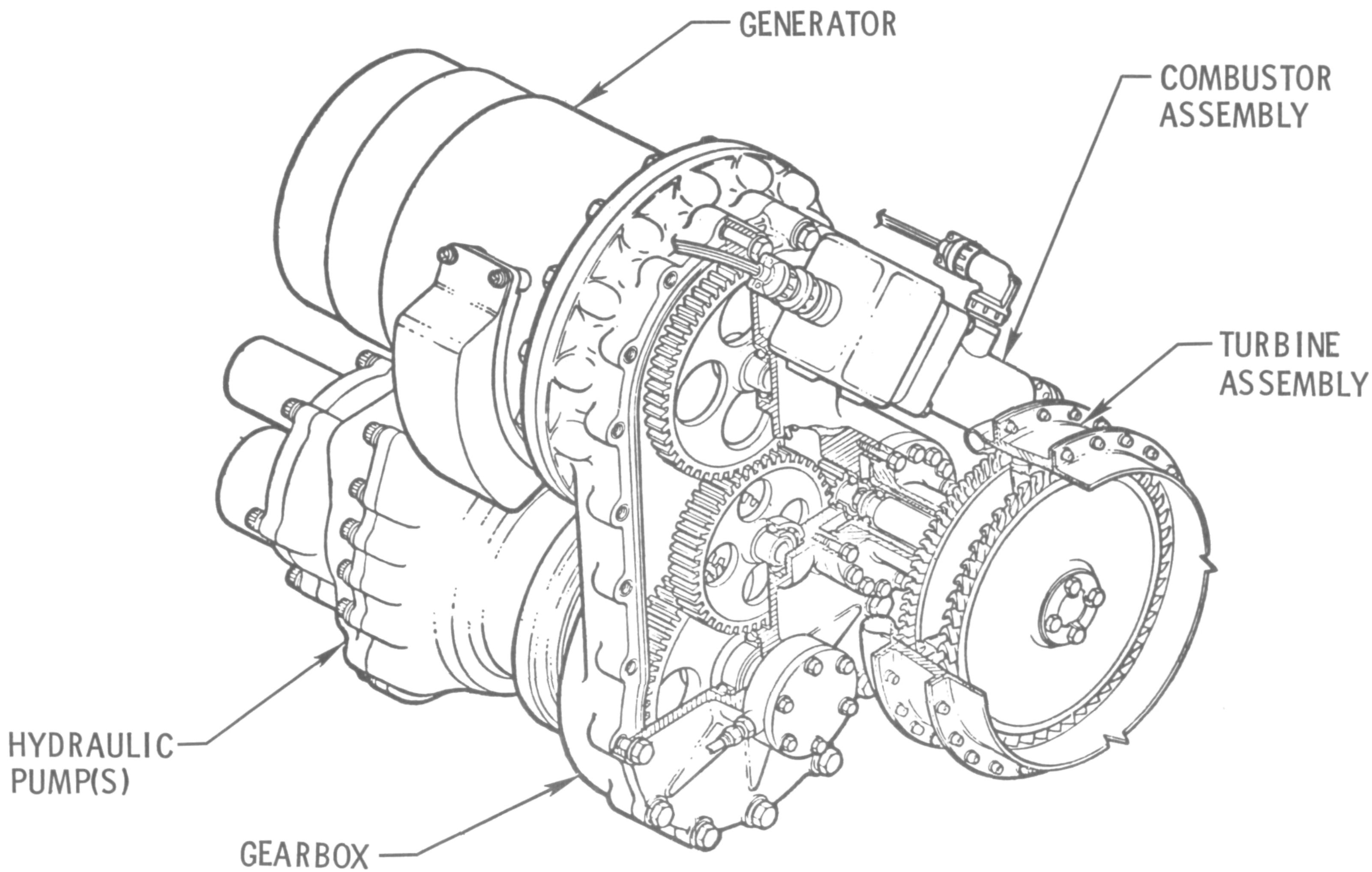

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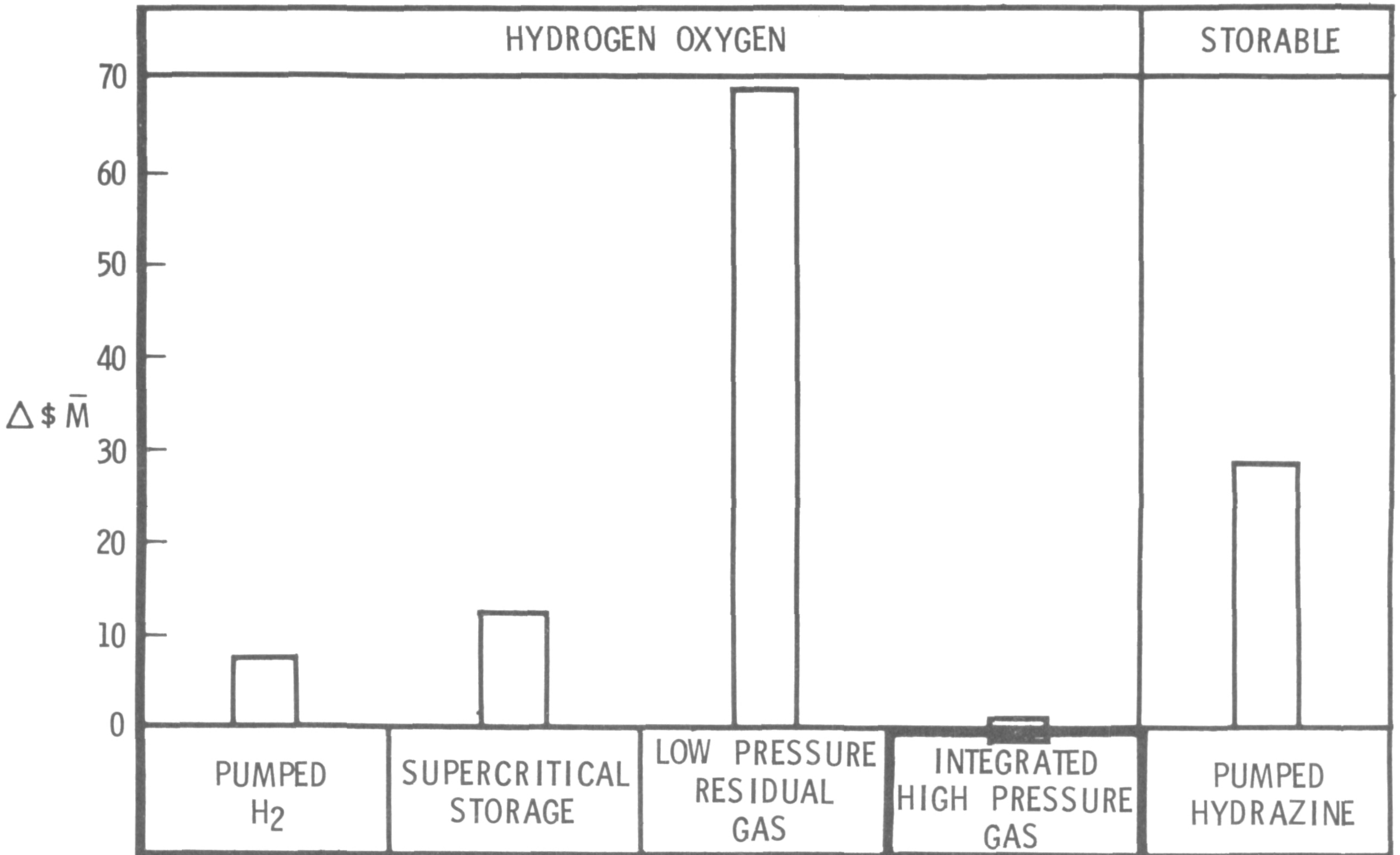
AUXILIARY POWER UNIT

SYSTEM OPTIMIZATION INCLUDED AN ASSESSMENT OF VEHICLE RELATED AND DEVELOPMENT COSTS. THESE COSTS INCLUDE ALL APU'S (6 ON BOOSTER AND 3 ON ORBITER) AND PROVIDE FOR A PRO RATA CHARGE FOR THE ORBITER RELATED WEIGHTS. THE INTEGRATED HIGH-PRESSURE GAS SYSTEM IS LOWEST IN COST. LOW PRESSURE RESIDUAL GAS SYSTEM IS EXTREMELY COSTLY BECAUSE APU WEIGHT IS HIGH, BUT PUMPED AND SUPERCRITICAL STORAGE SYSTEMS CARRY A MODEST COST PENALTY. STORABLE SYSTEMS SUCH AS HYDRAZINE WILL BE QUITE HEAVY SO THAT THE COST PENALTY, INCLUDING REDUCED DEVELOPMENT COSTS, WILL BE 30 MILLION DOLLARS.

TOTAL APU COST PENALTY

VEHICLE PAYLOAD + DEVELOPMENT COST



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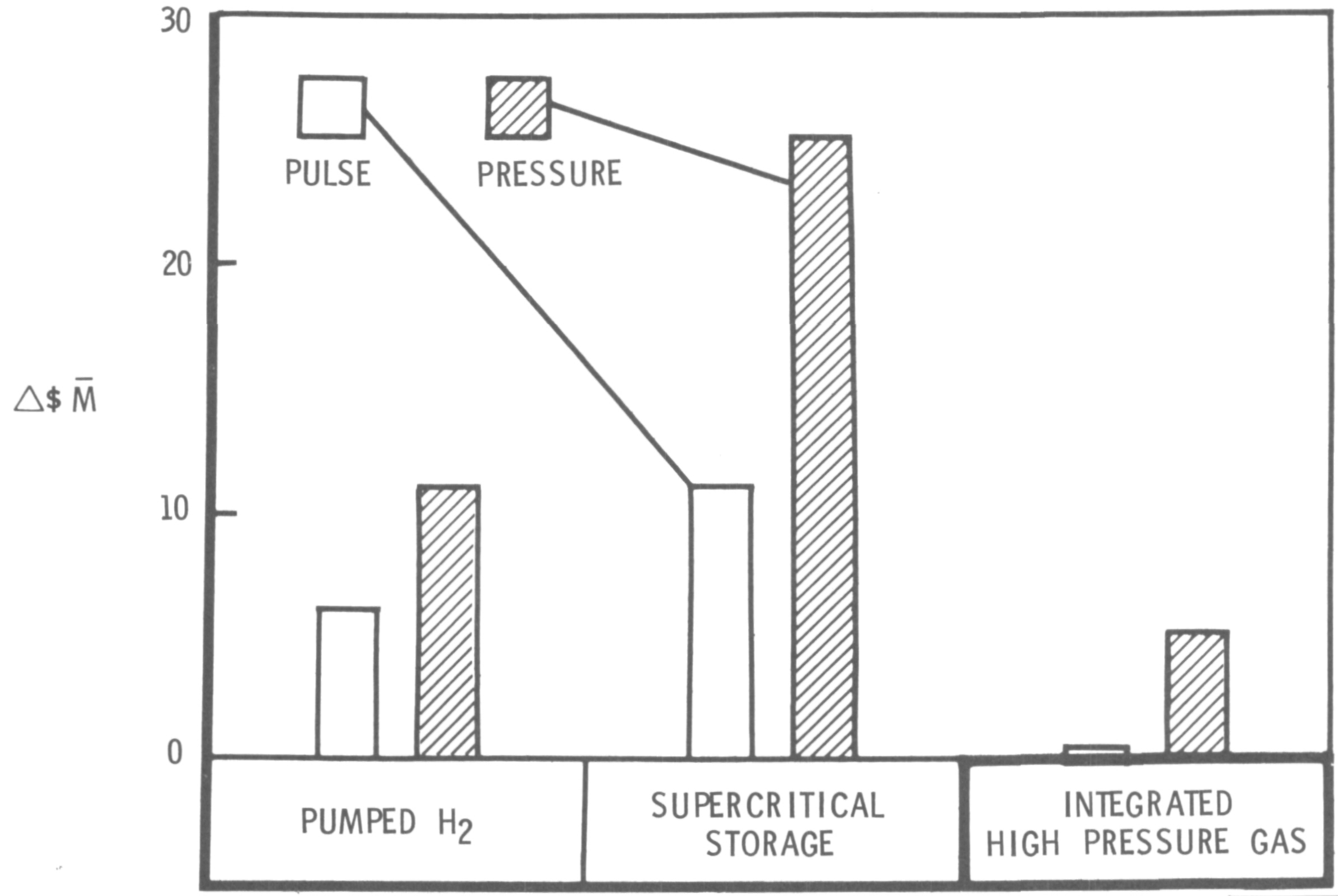
AUXILIARY POWER UNIT

THE COST PENALTIES ARE ALSO AFFECTED BY THE TYPE OF POWER CONTROL EMPLOYED. PULSE-WIDTH MODULATION RESULTS IN BETTER PERFORMANCE AND LOWER VEHICLE PAYLOAD RELATED PENALTY. THE FIGURES PRESENTED HERE ARE REPRESENTATIVE OF THE 250 MAXIMUM HORSEPOWER SYSTEM WHICH WAS SPECIFIED IN PHASE I OF THE STUDY, AND THE DIFFERENCES WILL BE INCREASED DUE TO LARGER ENERGY LEVELS OF THE HIGH CROSS RANGE ORBITER USED FOR THE PHASE II, 400 HORSEPOWER SYSTEM.

PRESSURE MODULATION COST PENALTY



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AUXILIARY POWER UNIT

THE APU CONSISTS OF THREE ELEMENTS

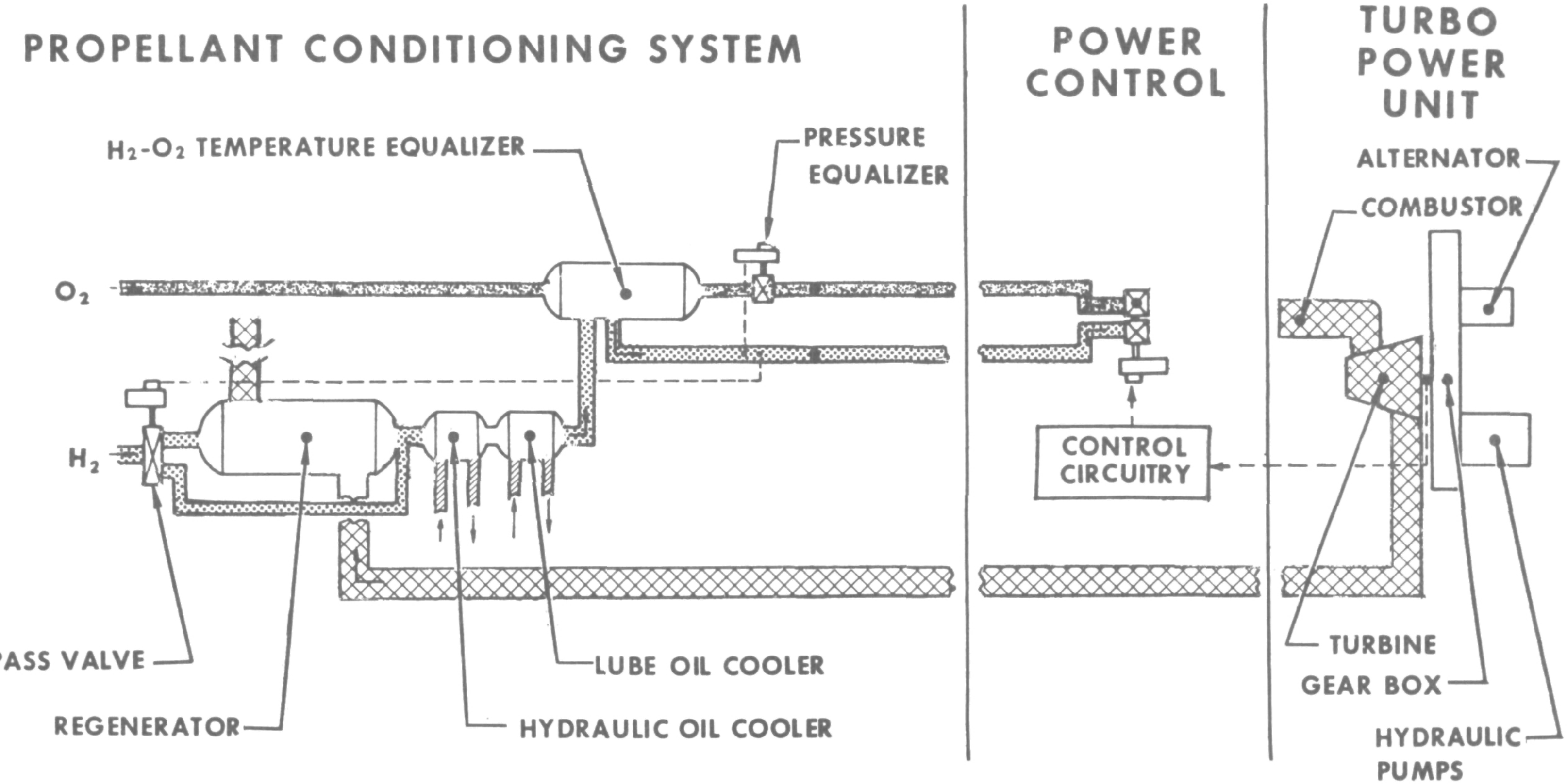
(1) A PROPELLANT CONDITIONING SYSTEM

(2) A POWER CONTROL SYSTEM

(3) A TURBOPOWER UNIT

HYDROGEN FLOWS FROM THE TANKAGE THROUGH A REGENERATOR WHERE IT IS CONDITIONED; THROUGH THE HYDRAULIC-OIL AND LUBE-OIL COOLERS; AND INTO AN EQUALIZER. HERE IT EXCHANGES HEAT WITH THE SUPPLIED OXYGEN, RESULTING IN THE TWO GASES REACHING AN EQUAL TEMPERATURE. THEIR PRESSURE IS EQUALIZED IN THE NEXT STEP PRIOR TO ENTERING THE POWER CONTROL SYSTEM, WHERE THE PROPELLANTS ARE METERED TO PROVIDE THE REQUIRED POWER. IN THE TURBOPOWER UNIT THE PROPELLANTS REACT AND EXPAND THROUGH A TURBINE WHOSE SPEED IS REDUCED IN A GEAR BOX TO DRIVE THE PUMPS AND THE ALTERNATOR. THE EXHAUST PRODUCTS ARE DUCTED THROUGH THE REGENERATOR TO CONDITION THE INCOMING HYDROGEN.

BASELINE APU



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AUXILIARY POWER UNIT

PROPELLANT CONDITIONING COMBINES THE FUNCTIONS OF PROVIDING TURBINE INLET TEMPERATURE CONTROL WITH SAFE AND CONTROLLED HYDRAULIC-AND LUBE-OIL COOLING. TURBINE INLET TEMPERATURE IS MAINTAINED BY PROVIDING CONTROLLED CONDITIONED PROPELLANTS OVER THE COMPLETE RANGE OF OPERATING CONDITIONS. THE POWER CONTROL ACCOMPLISHES POWER MODULATION OVER THE REQUIRED RANGE WITH A RESPONSE TIME OF 75 MILLISECONDS. IN THE TPU, THE ENERGY CONVERSION IS PERFORMED EFFICIENTLY BY A TWO-STAGE, PRESSURE-COMPOUND, SUPERSONIC TURBINE.

BASELINE SUBSYSTEM FUNCTIONS

- TURBINE INLET TEMPERATURE CONTROL
- HYDRAULIC/LUBE OIL COOLING
- POWER MODULATION
- EFFICIENT ENERGY CONVERSION

FLEXIBILITY IS THE FOREMOST FEATURE OF THE SYSTEM. THE PROPELLANT CONDITIONING SYSTEM IS DESIGNED TO ACCOMODATE RAPID CHANGES IN SUPPLY AND TO PROVIDE SMOOTH AND CONTROLLED FLOW. IT ALSO ASSURES SAFE HYDRAULIC OIL COOLING UNDER ALL OPERATING CONDITIONS ELIMINATING PROBLEMS OF HYDRAULIC OIL FREEZING AT ANY POINT IN THE DUTY CYCLE BY THE INHERENT NATURE OF THE DESIGN. TO PROVIDE FLEXIBILITY THE SYSTEM IS ADAPTABLE TO ALL PRIMARY METHODS OF POWER CONTROL, PULSE-WIDTH MODULATION, PRESSURE MODULATION, AND AREA MODULATION.

THE REFERENCE POWER CONTROL SYSTEM (PULSE-WIDTH MODULATION) IS DESIGNED TO OPERATE WITH EXTENSIVE POWER PROFILE VARIATION AT A MINIMUM SPC.

FLEXIBILITY

PROPELLANT SUPPLY

SAFE HYDRAULIC COOLING

POWER CONTROL ADAPTABILITY

POWER PROFILE VARIATION

ROCKETDYNE'S ANALYSIS OF THE APU SYSTEM INCLUDED A COMPREHENSIVE ANALOG SYSTEM SIMULATION.

THE INTERACTIONS BETWEEN

(1) PROPELLANT SUPPLY

(2) TURBINE INLET TEMPERATURE

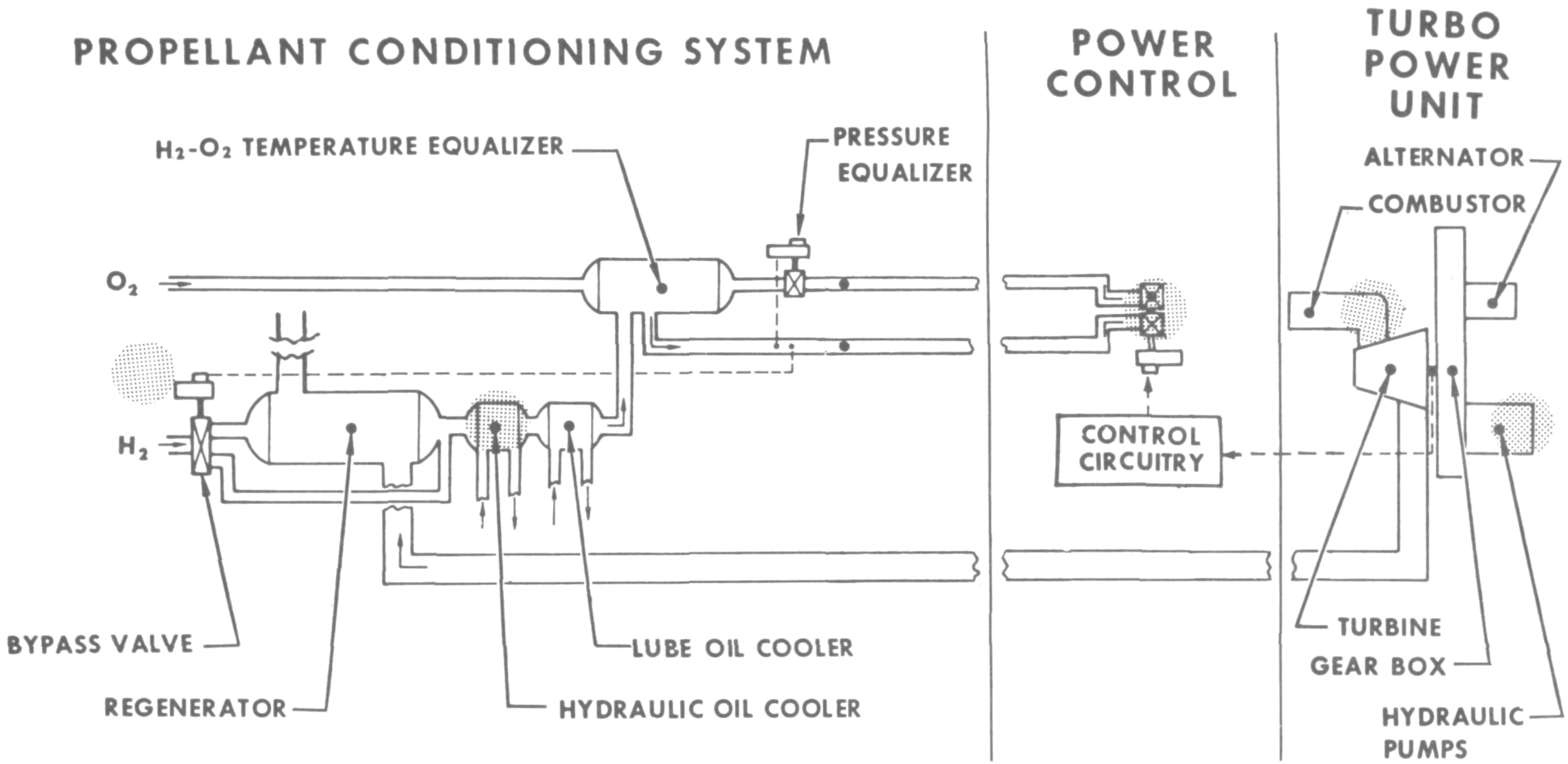
(3) VEHICLE HYDRAULIC COOLING

(4) PULSE AND PRESSURE MODULATION

(5) LOAD VARIATIONS

WERE INVESTIGATED.

SYSTEM OPERATION



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AUXILIARY PROPULSION SYSTEM

THE ANALOG SYSTEM SIMULATION INCORPORATES STEADY STATE AND DYNAMIC OPERATION OVER THE BROAD RANGE OF REQUIREMENTS;OVER THE FLIGHT PROFILE ON THE OUTPUT END; AND PROPELLANT SUPPLY VARIATION AT INPUT. THE SIMULATION PROVED THE SUCCESS OF THE VARIOUS CONCEPTS BECAUSE IT SHOWED THAT THE COMPLETE PROPELLANT SUPPLY RANGE COULD BE ACCOMODATED, TURBINE INLET TEMPERATURE COULD BE MAINTAINED AT VARYING LOADS AND HYDRAULIC COOLING REQUIREMENTS; THE TPU WAS CONTROLLED WITH BOTH PULSE AND PRESSURE MODULATION AND HAD ADEQUATE RESPONSE FOR THE 100 TO 25% LOAD VARIATION.

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SUCCESSFUL OPERATION SIMULATED

- ANALOG SYSTEM SIMULATION
- STEADY STATE AND DYNAMIC OPERATION

- PROPELLANT CONDITIONING SYSTEM
  - PROPELLANT SUPPLY RANGE ACCOMODATED
  - TURBINE INLET TEMPERATURE CONTROL
  - VEHICLE HYDRAULIC COOLING
- TURBOPOWER UNIT/POWER CONTROL
  - PULSE AND PRESSURE MODULATION
  - 100% TO 25% LOAD
  - ADEQUATE RESPONSE

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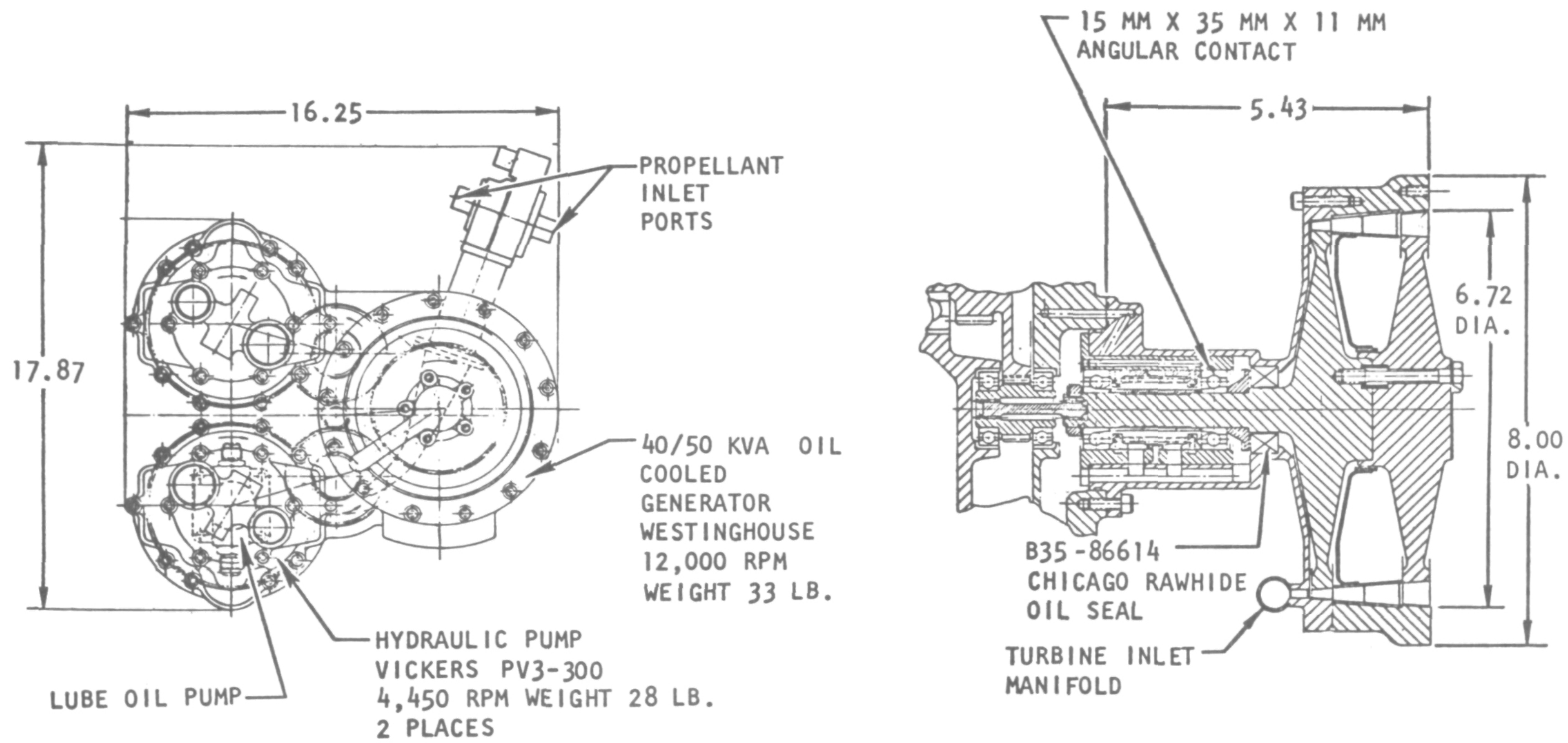
AUXILIARY POWER UNIT

THE ANALOG SYSTEM SIMULATION WAS PERFORMED USING REALISTIC MATHEMATICAL MODELS FOR ALL COMPONENTS. A PRELIMINARY TURBO POWER UNIT DESIGN WAS COMPLETED TO DETERMINE CHARACTERISTICS. PERFORMANCE CURVES BASED ON TEST DATA WERE USED FOR EXISTING COMPONENTS.

TURBO POWER UNIT



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AUXILIARY POWER UNIT

TEST DATA WERE OBTAINED ON A HYDROGEN/OXYGEN GAS GENERATOR DESIGNED FOR THE SPACE SHUTTLE APU AND OPERATED TO VERIFY PERFORMANCE. PULSING TESTS DEMONSTRATED REQUIRED RESPONSE TIME AND PERFORMANCE CONSISTENCY.

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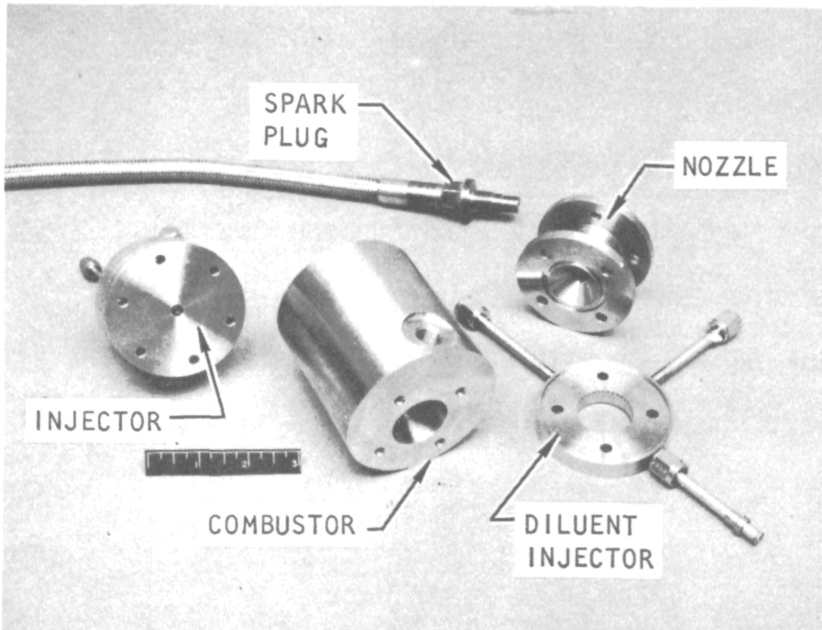
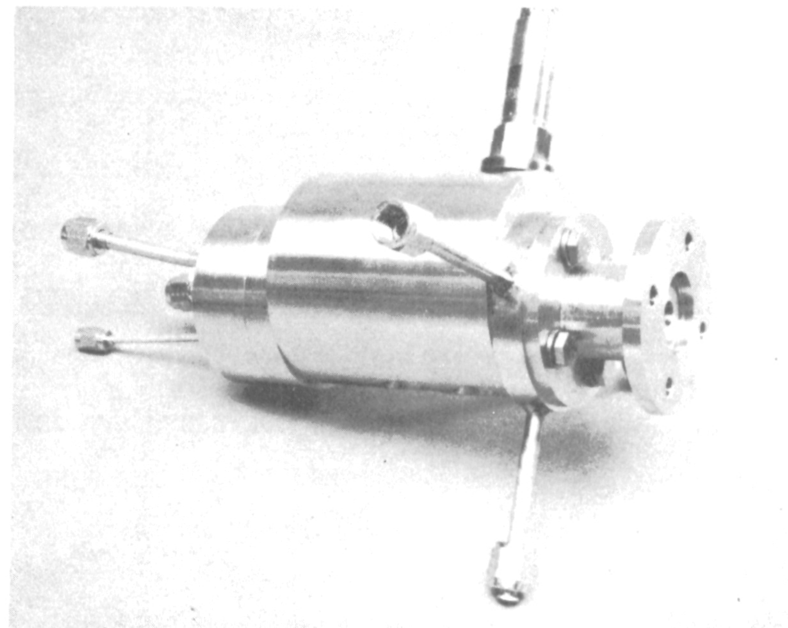
APU GAS GENERATOR

PULSING DEMONSTRATOR

COMPONENTS



ASSEMBLY



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AUXILIARY POWER UNIT

SUCCESSFUL PULSE MODULATED POWER CONTROL VALVE AND GAS GENERATOR ASSEMBLY OPERATION WAS DEMONSTRATED BY OPERATING WITHOUT FAILURE FOR 50,000 CYCLES, SIMULATING 5 TYPICAL SPACE SHUTTLE APU MISSIONS. THE TEST HARDWARE INCLUDED PROPELLANT VALVES, IGNITION SYSTEM, COMBUSTOR, AND A TURBINE BLADE. DYNAMIC RESPONSE CHARACTERISTICS WERE ALSO EVALUATED AND USED TO VERIFY THE DYNAMIC ANALOG RESULTS.

GAS GENERATOR TESTS

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AUXILIARY POWER UNIT

VALVE DATA WAS BASED ON EXISTING TECHNOLOGIES AND HARDWARE. LINKED BIPROPELLANT VALVES WITH RESPONSE CHARACTERISTICS COMPARABLE TO THOSE REQUIRED IN THE MODEL ARE AVAILABLE.

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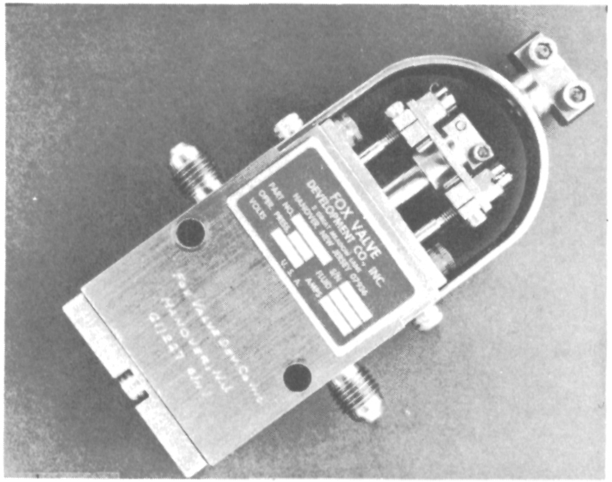
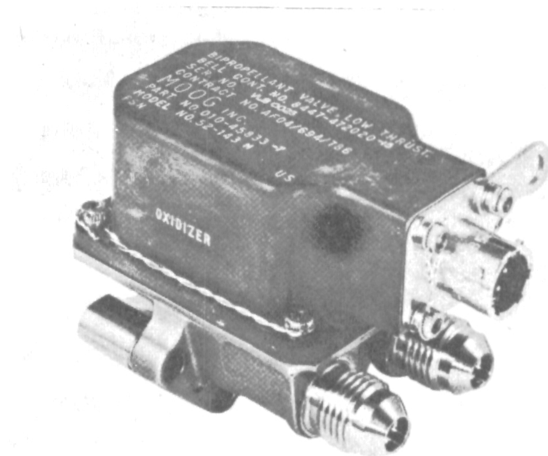
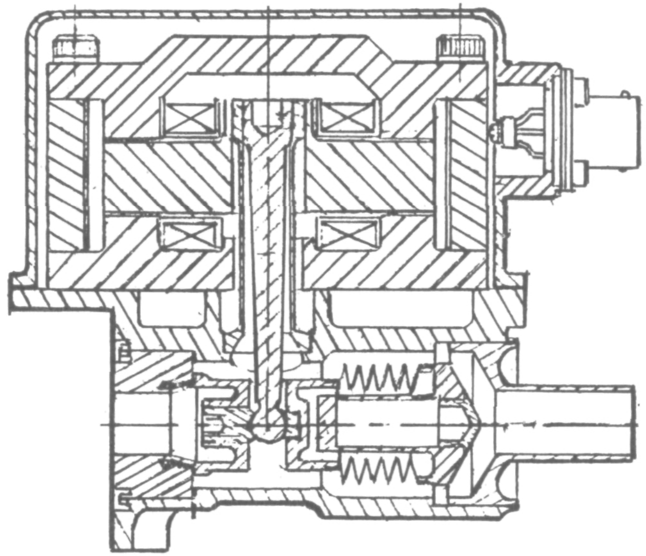
APU VALVE TECHNOLOGIES







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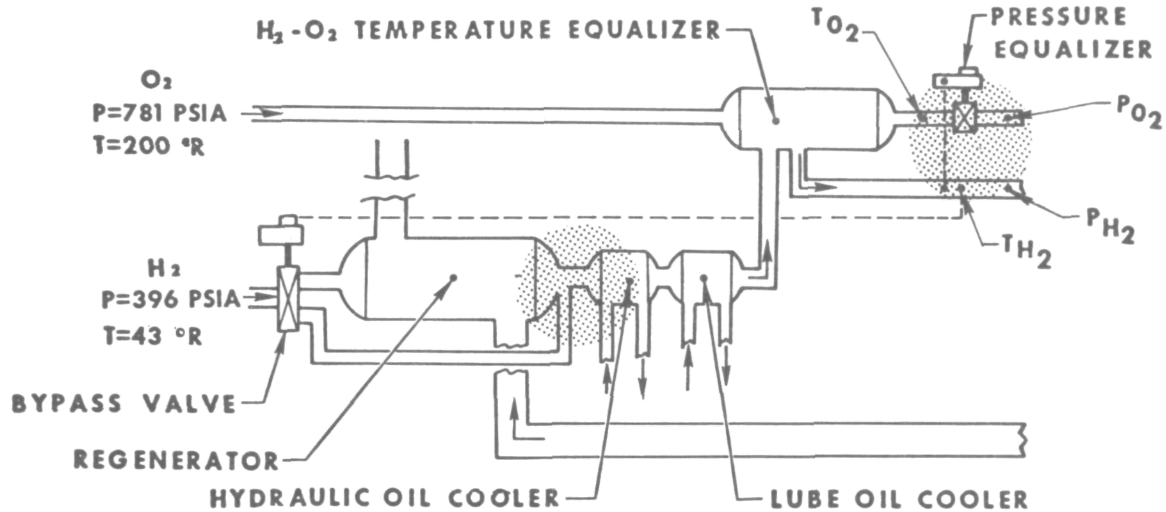
AUXILIARY POWER UNIT

SUPPLY OF CONTROLLED PROPELLANT FLOW TO THE CONTROL VALVES IS ACCOMPLISHED BY THE PROPELLANT CONDITIONING SYSTEM. PRIMARY CONTROL FUNCTION IS THE MAINTENANCE OF THE SUPPLIED HYDROGEN TEMPERATURE LEAVING THIS SYSTEM (T_{H2}) REGARDLESS OF FLOW AND HYDRAULIC AND LUBE-OIL COOLING RATES. THIS IS ACCOMPLISHED BY VARYING THE BYPASS FLOW AROUND THE REGENERATOR WHICH VARIES THE TEMPERATURE AT THE COOLER INLET. ALSO OF GREAT IMPORTANCE IS THE SEQUENCE OF REGENERATION AND COOLING. BY HEATING THE HYDROGEN IN THE REGENERATOR FIRST, THE WALL TEMPERATURE AT THE COOLER INLET WILL NEVER FALL LOW ENOUGH TO PERMIT FREEZING OF THE HYDRAULIC OIL. THE HYDROGEN LEAVING THE COOLERS IS USED TO PREHEAT THE OXYGEN IN THE TEMPERATURE/EQUALIZER AND THE OXYGEN PRESSURE IS EQUALIZED WITH THE OXYGEN PRESSURE USING A FOLLOWING PRESSURE REGULATOR.

PROPELLANT CONDITIONING SYSTEM



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AUXILIARY PROPULSION SYSTEM

TURBINE INLET TEMPERATURE CONTROL IS ACCOMPLISHED BY CONTROLLING BOTH PROPELLANT TEMPERATURE AND MIXTURE RATIO AND DELIVERING THE CONTROLLED PROPELLANTS TO THE POWER CONTROL VALVE AND TPU COMBUSTOR.

PROPELLANT TEMPERATURE IS ACTIVELY CONTROLLED BY USE OF THE REGENERATOR BYPASS VALVE. MIXTURE RATIO CONTROL IS ACCOMPLISHED WITH PASSIVE PROPELLANT TEMPERATURE EQUALIZATION AND ACTIVE PROPELLANT PRESSURE EQUALIZATION.

SAFE HYDRAULIC COOLING RESULTS FROM CONTROL OF THE PRE-HEATED HYDROGEN COOLANT WHICH AUTOMATICALLY (BUT PASSIVELY) ADJUSTS IT'S TEMPERATURE TO THE HYDRAULIC COOLING LOAD.

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TURBINE INLET TEMPERATURE CONTROL

PROPELLANT TEMPERATURE CONTROL

- REGENERATOR AND BYPASS VALVE — ACTIVE
  T_{H_2} = PRESET VALUE

MIXTURE RATIO CONTROL

- TEMPERATURE EQUALIZER — PASSIVE
- PRESSURE EQUALIZER — ACTIVE

HYDRAULIC/OIL COOLING

- CONTROLLED TEMPERATURE SINK — PASSIVE

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AUXILIARY POWER UNIT

THE POWER CONTROL SYSTEM PROVIDES FOR POWER MODULATION THROUGH CONTROL OF TURBOPOWER UNIT SPEED WITHIN $\pm$ 5%.

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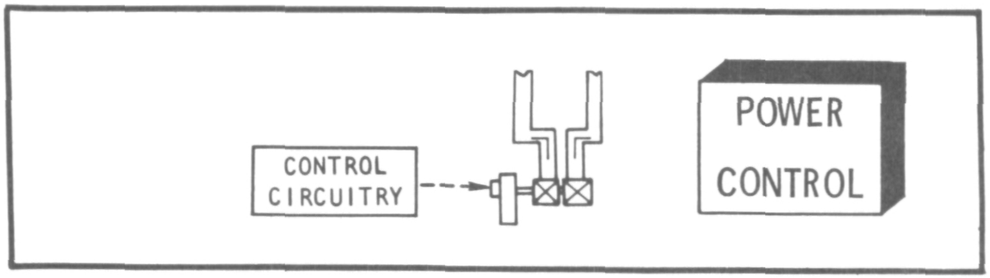
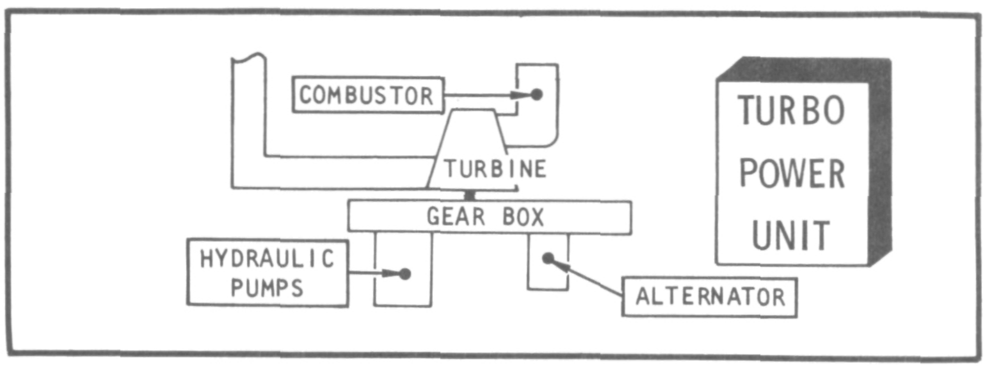
POWER CONTROL





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AUXILIARY POWER UNIT

PRIMARY ACTIVE CONTROL RESULTS FROM LINKED BI-PROPELLANT VALVES WHICH CAN BE USED FOR BOTH PULSE WIDTH AND PRESSURE MODULATION. FOR PULSE CONTROL, THE VALVE REMAINS ON LONG ENOUGH TO ACCELERATE THE TPU TO THE UPPER END OF THE SPEED BAND, THEN REMAINS OFF UNTIL THE MACHINE SLOWS TO THE LOWER LIMIT. FOR PRESSURE CONTROL, THE VALVE METERS THE FLOW AT A LEVEL PROPORTIONAL TO THE REQUIREMENTS TO MAINTAIN SPEED.

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POWER MODULATION

SPEED CONTROL

- LINKED BI-PROPELLANT VALVE — ACTIVE
  - PULSE CONTROL
  - PRESSURE MODULATED CONTROL
- SPEED SENSING AND CIRCUITRY — ACTIVE

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AUXILIARY POWER UNIT

THE RESULTS OF THE ANALOG SIMULATION STUDY INDICATED THAT PARALLEL FLOW SHOULD BE USED IN THE REGENERATOR TO ENSURE WALL TEMPERATURES HIGH ENOUGH TO PREVENT FREEZING ON THE EXHAUST GAS SIDE OF THE HEAT EXCHANGER. THE WALL TEMPERATURE SHOWN HERE IS THAT CORRESPONDING TO MAXIMUM HYDROGEN REGENERATOR FLOW (MINIMUM BYPASS AND MOST SEVERE FREEZING CONDITION) AND IT IS WELL ABOVE THE CONDENSATION LIMIT FOR THE HOT EXHAUST GAS. AS BYPASS FLOW IS INCREASED, THE WALL TEMPERATURE RISES BECAUSE OF LESSER REGENERATION, AND CONDENSATION WILL ALWAYS BE AVOIDED.

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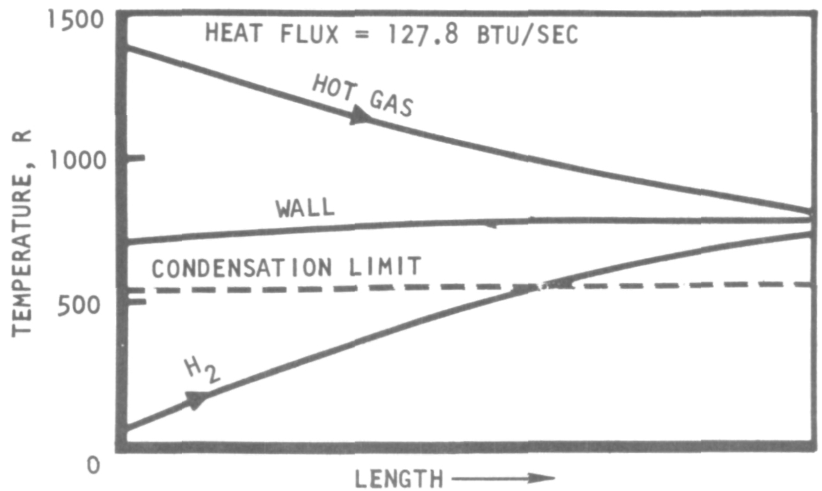
NON FREEZING REGENERATOR

PARALLEL FLOW



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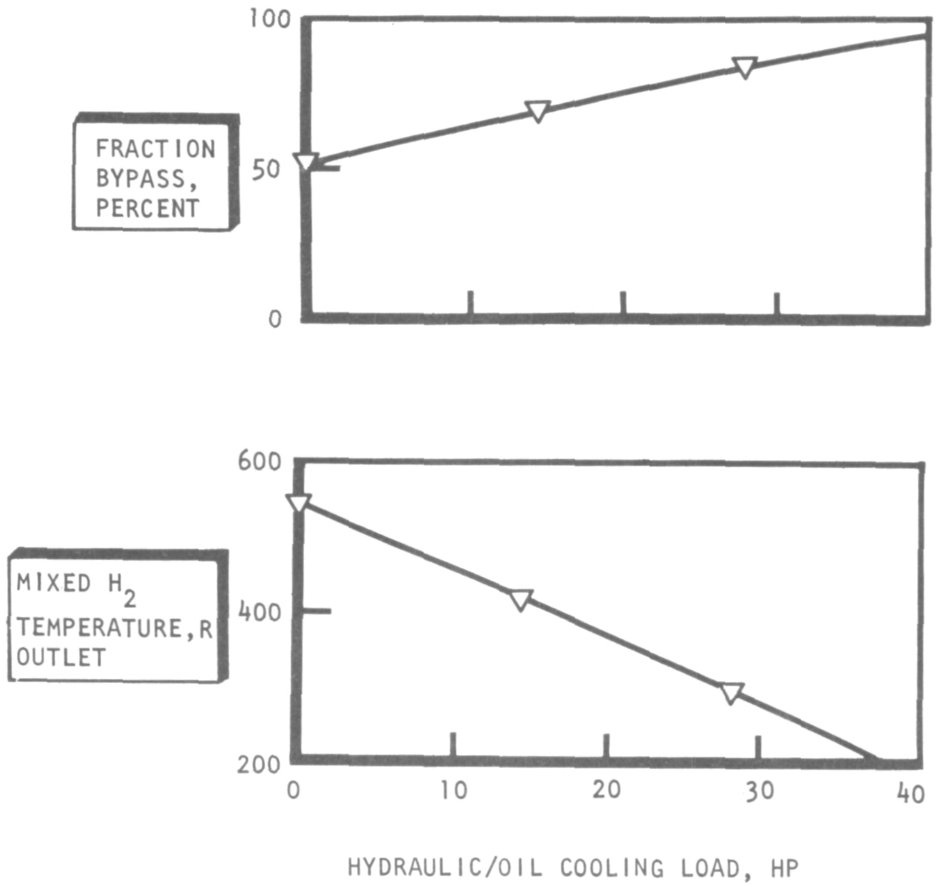
AUXILIARY POWER UNIT

FROM THE ANALOG RESULTS AN ASSESSMENT WAS MADE OF EFFECTS ON REGENERATOR PERFORMANCE AS HYDRAULIC LOAD VARIES. SINCE THE PROPELLANT TEMPERATURE IS CONTROLLED, THE REGENERATOR OUTLET TEMPERATURE MUST VARY TO ACCOMODATE REQUIRED COOLING LOAD. SOME BYPASS OCCURS EVEN WITHOUT HYDRAULIC COOLING SO THAT THE MIXED (BYPASS FLOW PLUS REGENERATOR FLOW) HYDROGEN TEMPERATURE IS 550 R. AS HYDRAULIC COOLING LOAD INCREASES, THE REGENERATOR MIXED OUTLET TEMPERATURE IS REDUCED BY BYPASS VARIATION BUT THE REGENERATOR OUTLET HYDROGEN TEMPERATURE PRIOR TO MIXING REMAINS HIGH SO NO FREEZ UP CAN OCCUR. IN THE HYDRAULIC COOLER THE LOW TEMPERATURE HYDROGEN IS USED FOR COOLING ONLY WHEN MAXIMUM COOLING IS REQUIRED. WHEN REDUCED COOLING IS NEEDED THE HYDROGEN COOLANT TEMPERATURE IS AUTOMATICALLY RAISED UP TO 550 R AT THE NO-COOLING-LOAD POINT.

SAFE HYDRAULIC COOLING



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space shuttle engine
AUXILIARY POWER UNIT

PRIOR TO ENTRY INTO THE POWER CONTROL SYSTEM, THE TWO PROPELLANTS ARE EQUALIZED IN TEMPERATURE. THIS ELIMINATES TEMPERATURE AS A POSSIBLE VARIABLE IN MIXTURE RATIO CONTROL, ENSURING MORE RELIABLE OPERATION WITH A PASSIVE CONTROL ELEMENT. A TYPICAL DESIGN, WEIGHING ONLY 2 POUNDS IS SHOWN HERE. THERE ARE NO JOINTS BETWEEN OXYGEN AND HYDROGEN PATHS SO THAT THE SAFETY CONSIDERATIONS RELATED TO ACCIDENTAL MIXING OF THE PROPELLANTS ARE ELIMINATED.

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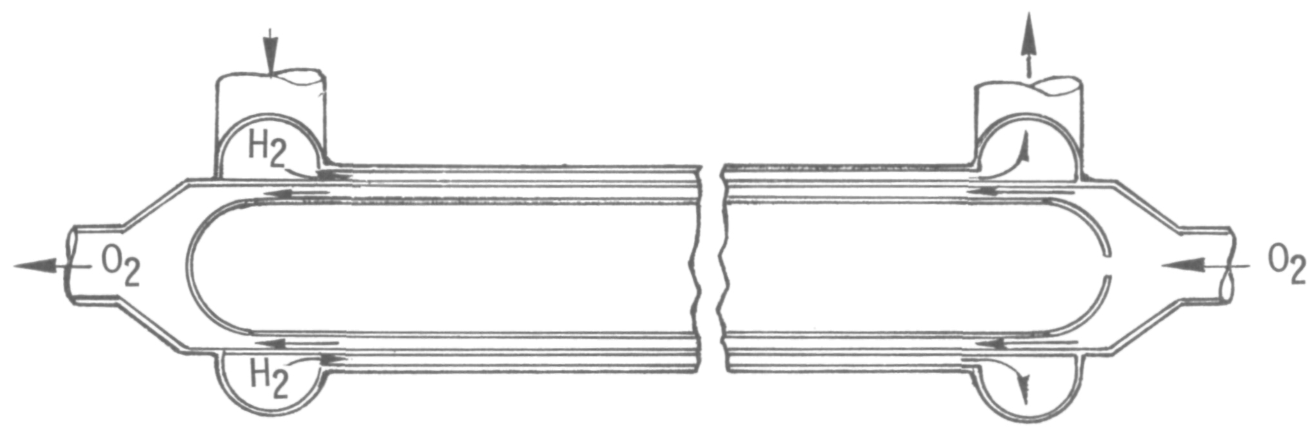
PROPELLANT TEMPERATURE EQUALIZER



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AUXILIARY POWER UNIT

PERFORMANCE RESULTS ON THE EQUALIZER ELEMENT SHOWS THAT THE DIFFERENTIAL BETWEEN HYDROGEN AND OXYGEN TEMPERATURE CAN BE HELD TO ± 20 F OVER THE COMPLETE POWER RANGE.

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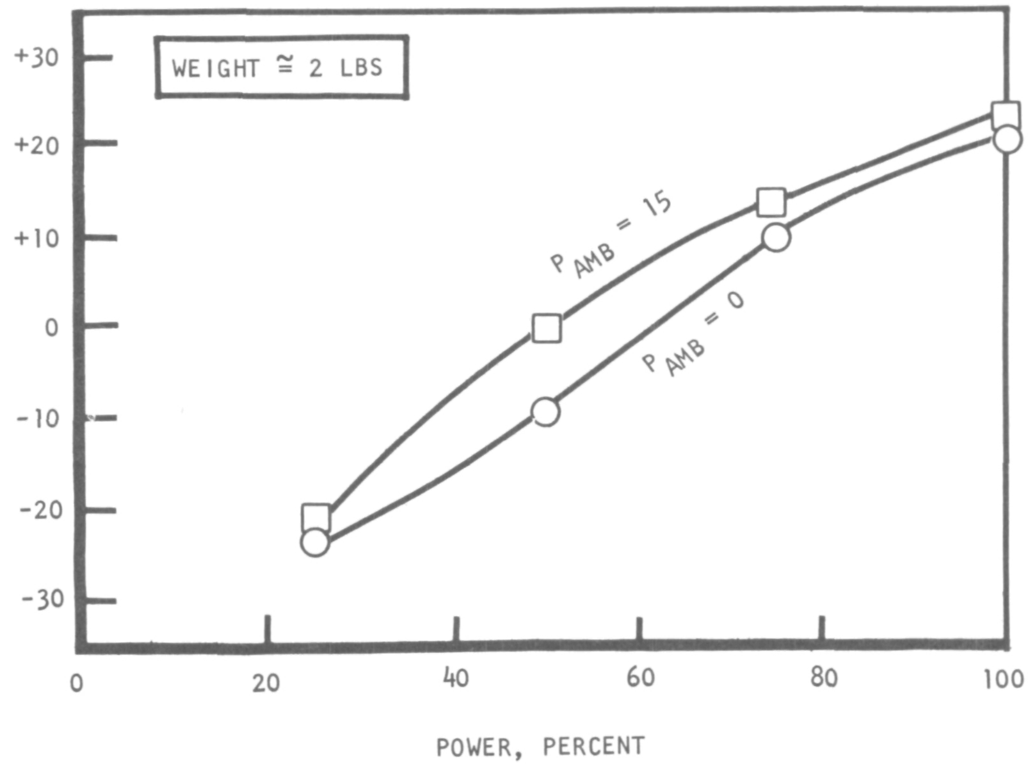
TEMPERATURES EQUALIZED

$\Delta T = \pm 20$

H_2-O_2
ΔT



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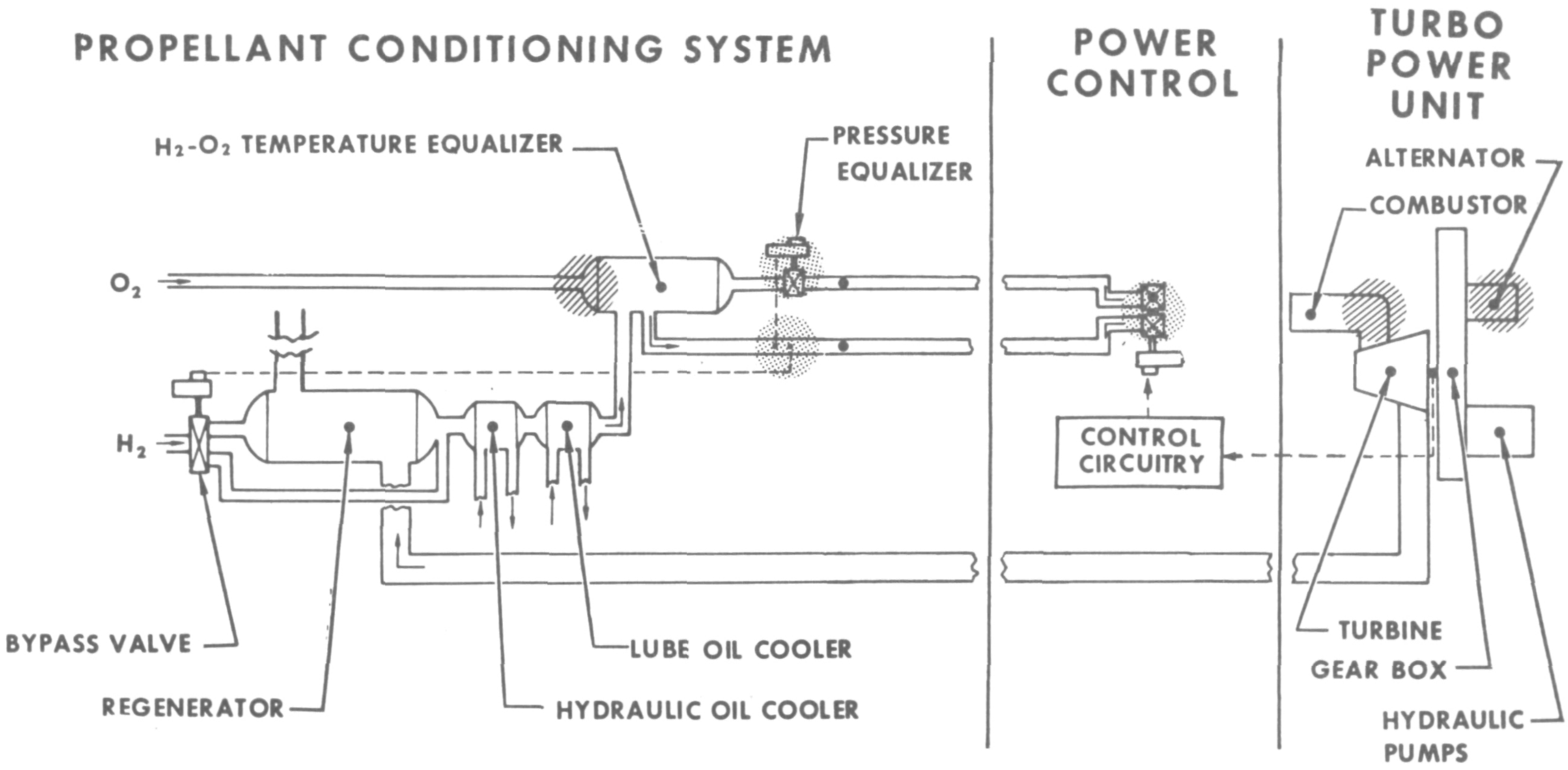
AUXILIARY POWER UNIT

THE ANALOG MODEL WAS USED FOR STEADY STATE AND TRANSIENT ANALYSES OF THE APU SYSTEM. PROPELLANT SUPPLY, POWER COOLING AND ALTITUDE VARIATIONS WERE ALL IMPOSED ON THE SYSTEM AND THE SYSTEM OPERATION MONITORED AT ALL KEY POINTS.

STEADY STATE AND TRANSIENT RESULTS



Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY PROPULSION SYSTEM

UNDER STEADY STATE CONDITIONS, THE CONTROL TOLERANCE BANDS ANTICIPATED WERE TRAVERSED TO DETERMINE MIXTURE RATIO AND TURBINE INLET TEMPERATURE VARIATION. THE RESULTING VALUES (3.1% ON MIXTURE RATIO AND 68 R ON TEMPERATURE) ARE WELL WITHIN THE ACCEPTABLE LIMITS OF OPERATION.

TURBINE INLET TEMPERATURE

MAXIMUM PROBABLE STEADY STATE VARIATIONS

TOLERANCE	MIXTURE RATIO VARIATION	TURBINE INLET TEMPERATURE VARIATION
$\Delta T_H = +5\%$ $\Delta P_O = +2\%$ $\Delta A_O = +1\%$	+3.1%	+68R

Rocketdyne
North American Rockwell

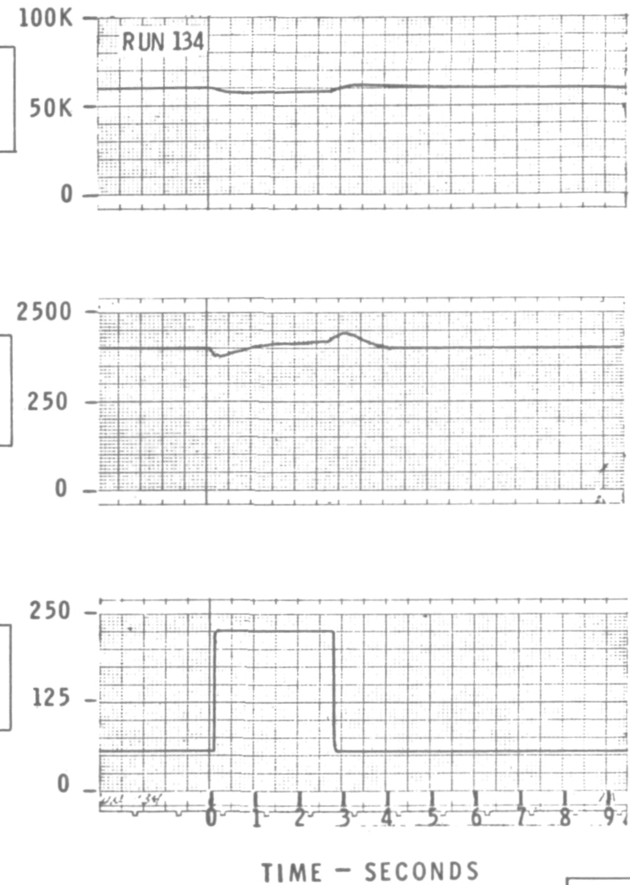
space shuttle engine
AUXILIARY POWER UNIT

UNDER TRANSIENT CONDITIONS SUCH AS A STEP POWER DEMAND (25 TO 100%), THE PRESSURE MODULATED SYSTEM ADJUSTS WITHIN APPROXIMATELY 1 SEC. AND THE VARIATION IN SYSTEM SPEED IS WELL WITHIN THE REQUIRED ±5%. THE TURBINE INLET TEMPERATURE VARIATION IS SHORT AND WELL WITHIN ACCEPTABLE LIMITS. SUBSEQUENT EVALUATIONS USING A HIGHER RESPONSE (BUT WITHIN THE STATE OF ART) PRESSURE EQUALIZER INDICATE THAT THE TRANSIENT TURBINE INLET TEMPERATURE VARIATION CAN BE REDUCED TO APPROXIMATELY 100 DEGREES.

PRESSURE MODULATED POWER CONTROL

TRANSIENT RESULTS

±4.2%	TPU SPEED (RPM)	
$+220^{0}R$ $-125^{0}R$	TURBINE INLET TEMP (R)	
25% AND 100%	TURBINE POWER (HP)	



THE PULSE MODULATED SYSTEM ACCOMODATES TO STEP POWER DEMAND WITHIN TOTALLY ACCEPTABLE LIMITS. IT SHOULD ALSO BE NOTED THAT WHILE COMBUSTOR FLOWRATE PULSES, THE PROPELLANT FLOWRATE (E.G., O_2) MODULATES WITH ONLY SMALL OSCILLATIONS. THIS RESULTS BECAUSE ACCUMULATORS ARE USED DOWNSTREAM OF THE PRESSURE EQUALIZER.

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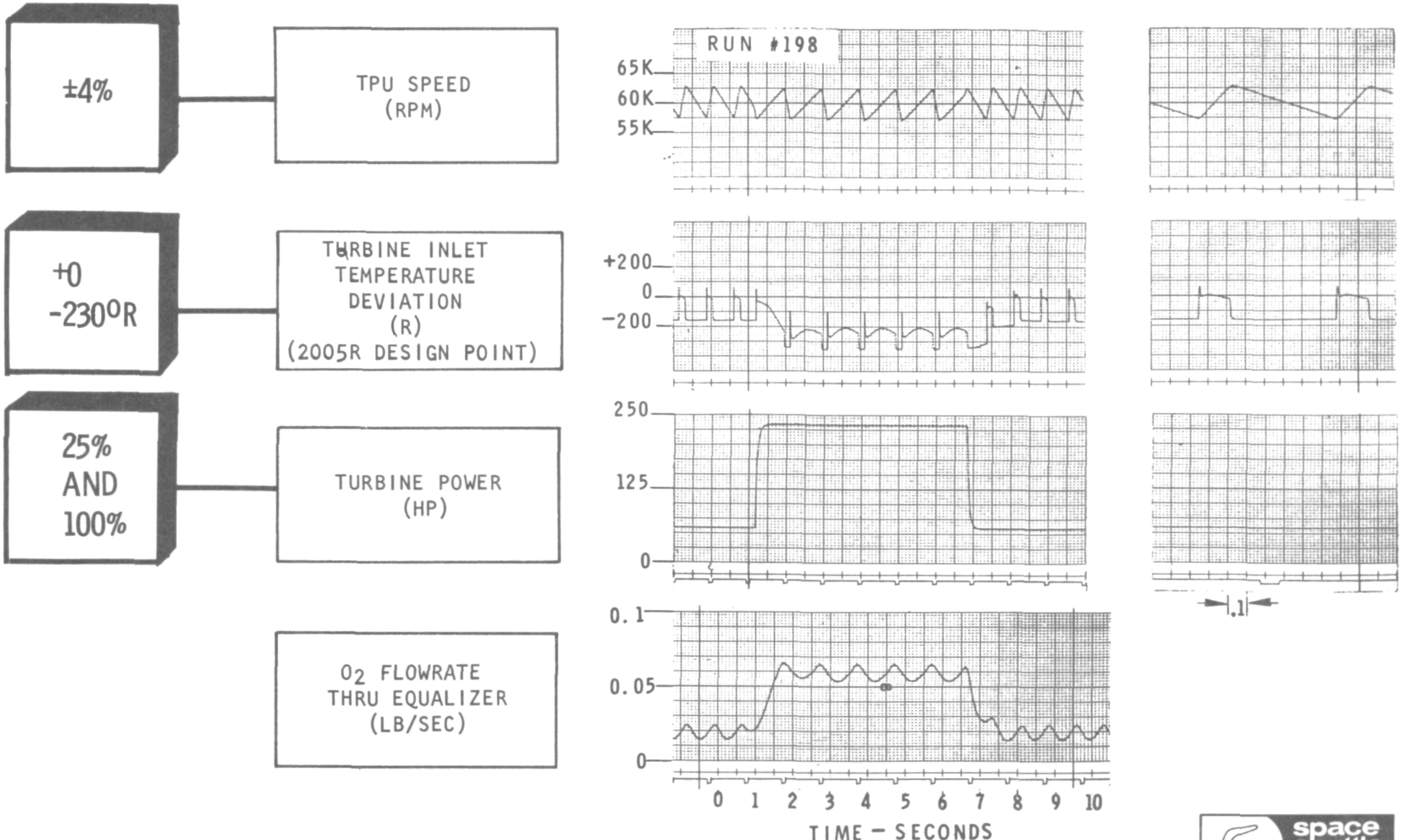
PULSE POWER CONTROL
TRANSIENT POWER



Rocketdyne
North American Rockwell

space shuttle engine
AUXILIARY POWER UNIT

THE RESULTS OF ROCKETDYNE'S COMPREHENSIVE STUDY HAVE SHOWN THAT THE APU SYSTEM EXHIBITS CHARACTERISTICS OF FLEXIBILITY AND FORGIVING DESIGN, AND THAT IT MAKES MAXIMUM USE OF EXISTING CONTROL ELEMENT TECHNOLOGY. IT IS ABLE TO MEET REQUIREMENTS FOR VARIATION IN PROPELLANT SUPPLY, COOLING LOAD, POWER LEVEL. MAXIMUM USE OF PASSIVE CONTROL IS MADE AND THE NUMBER AND COMPLEXITY OF ACTIVE CONTROLS IS REDUCED. THE CONTROL ELEMENTS ARE BASED ON THE USE OF EXISTING CRYOGENIC VALVES AND REGULATORS FROM ATTITUDE CONTROL SYSTEMS, GAS GENERATORS AND IGNITION SYSTEMS NOW IN USE AND UNDER DEVELOPMENT.

FEATURES

- **FLEXIBILITY**
  - PROPELLANT, COOLING LOAD, POWER LOAD VARIATIONS
- **FORGIVING DESIGN**
  - MAXIMIZE PASSIVE ELEMENTS
- **EXISTING COMPONENT TECHNOLOGY**
  - BY-PASS VALVE (J-2 PROPELLANT UTILIZATION)
  - PRESSURE REGULATOR (J-2 HELIUM REGULATOR)
  - ACS GAS GENERATOR AND IGNITER
  - BIPROPELLANT VALVE

FOLLOWING EXHAUSTIVE SYSTEM EVALUATION, AN OPTIMUM CONFIGURATION FOR THE APU HAS BEEN SELECTED. THE PRELIMINARY DESIGN HAS PROGRESSED TO WHERE THE SYSTEM HAS BEEN DEFINED, PERFORMANCE ESTABLISHED, AND THE TECHNOLOGY REQUIREMENTS IDENTIFIED. EXPERIMENTAL VERIFICATION OF THE BASELINE SYSTEM IS THE NEXT LOGICAL STEP OF THE PROGRAM.

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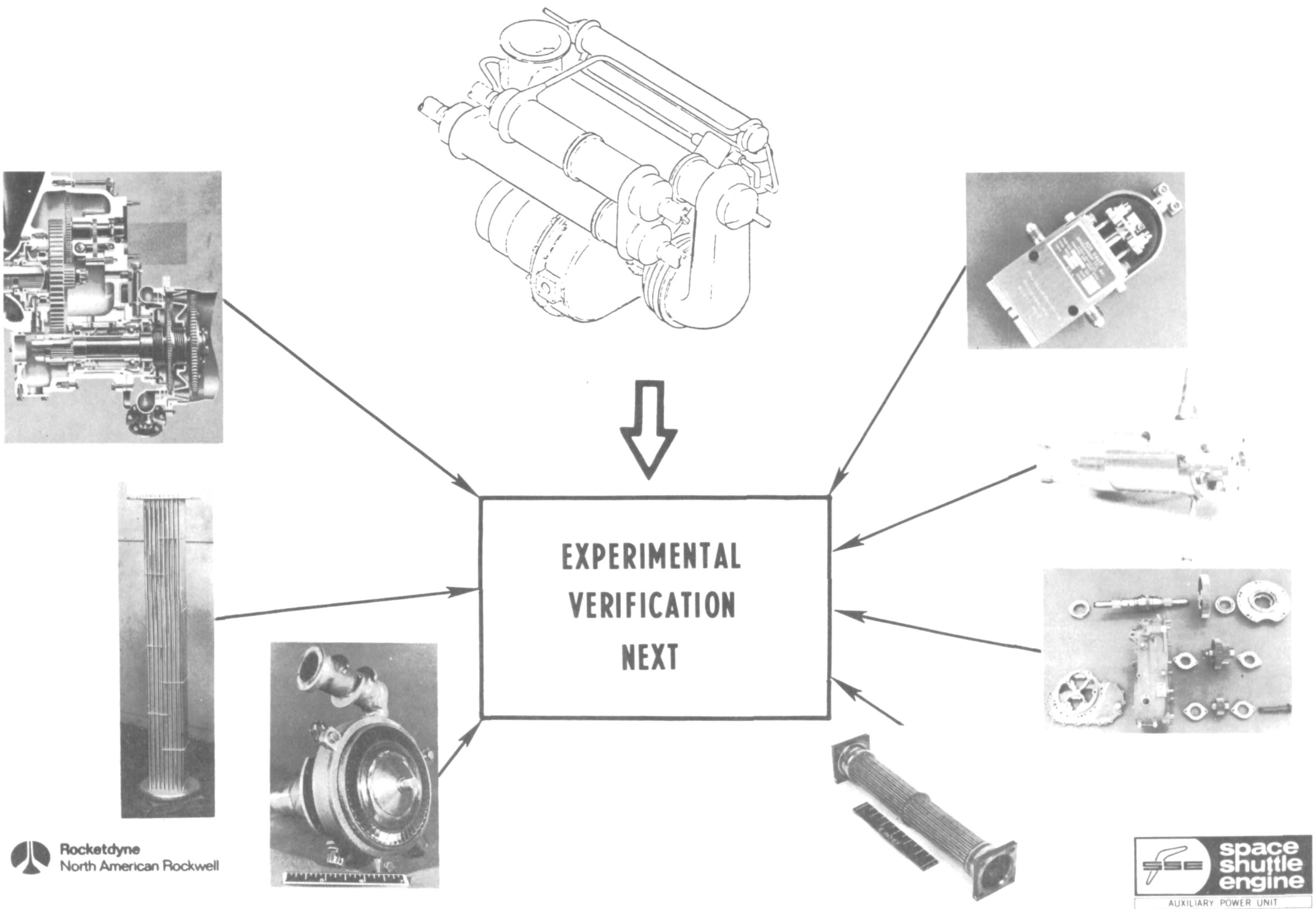
STATUS



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"H_2 FUEL SYSTEM INVESTIGATION"

W. COLLIER

GENERAL ELECTRIC

TECHNICAL MANAGER

S. M. NOSEK

LEWIS RESEARCH CENTER

CHART 1

This report covers the work done on the Space Shuttle Airbreathing Engine Systems Study for NASA Lewis. The NASA Project Manager was Mr. Stanley M. Nosek. The work was initiated in June 1970 and completed in March of this year.

SPACE SHUTTLE
AIRBREATHING ENGINE SYSTEM STUDY

GENERAL ELECTRIC COMPANY

W.R. COLLIER

CHART 2

The study program consisted of four tasks. Task I dealt with a parametric evaluation of various engine cycles employing hydrogen as the fuel. The primary variation in the cycles evaluated was the turbofan bypass ratio. The principle objective of this parametric study was to arrive at a recommended engine and cycle to be further evaluated during Task II. The principle areas of investigation of Task II covers problem identification, an investigation of modifications to employ the hydrogen fuel and an assessment of the technology level applicable to the timing of the Space Shuttle Program. Task III required the formulation of a recommended development plan for the selected engine and the preparation of a cost estimate to implement the recommended plan. Task IV dealt with the preparation of a preliminary performance specification for the selected engine.

The work on these tasks have been completed and the results presented to NASA Lewis.

CONTENTS OF STUDY

TASK I	Parametric Analysis
TASK II	Engine Design Study (F101/F12)
TASK III	Engine Development Plan and Cost Estimate
TASK IV	Performance Specification

CHART 3

The general approach and conduct of the Task I parametric engine study is outlined here. The first step was to identify a family of turbofan engines based on a technology level now in development. These engines covered a range of fan bypass ratios from 1 through 8. A composite mission and vehicle configuration was synthesized and the study engines were analyzed on a mission basis utilizing propulsion weight plus mission fuel as a merit factor. An important restraint was the commonality approach employed, in which identical engines were evaluated in both the booster and the orbiter missions and the engine merit factors were evaluated on a relative payload basis.

TASK I

PARAMETRIC ENGINE STUDY

APPROACH

- Define family of LH_2 fueled turbofan engines based on F101 technology level
  - Bypass 1 to 8
- Size engine to fit multi-engined boosters
- Assign installation factors
- Analyze booster reentry mission
- Use booster-sized engines in 4 engined orbiter
- Analyze orbiter reentry mission
- Evaluate engines on basis of equivalent payload to orbit

CHART 4

This chart shows some of the significant parameters of the parametric engine family. The engine designation numbers refer to the bypass ratios 1, 2, 4, 6, and 8 with a special case of a mixed flow engine referred to as SS-1M. Since this parametric family was treated at both the design and off-design operating conditions the engines were selected such that the low pressure components of the turbofan were varied about a common core or gas generator. A turbine inlet temperature of 2500°F was selected as this was believed consistent from a technology standpoint with the operational timing assumed. Other characteristics such as the thrust and air flow levels were the result of preliminary evaluation of the vehicle requirements. From a study of these engines, a fan pressure ratio was selected for each bypass ratio in an effort to maximize specific thrust and to minimize SFC.

PARAMETRIC ENGINE
SLS DESIGN COMPARISON

T_4—2500°F MAX

ENGINE	SS-1M	SS-1	SS-2	SS-4	SS-6	SS-8
W_2	302	302	390	525	705	885
β	.99	.99	1.97	3.92	5.86	7.70
F_n	18800	18400	19930	20450	23200	25620
SFC (H_2)	.229	.234	.200	.158	.136	.122
P_{23}/P_2	3.0	3.0	2.77	2.22	1.82	1.61
P_3/P_2	28.4	28.4	26.7	26.6	26.1	25.9

P_{23}/P_2 (Fan Pressure Ratio)
P_3/P_2 (Overall Pressure Ratio)

CHART 5

This chart illustrates as an example, one of the parametric engines employed. This particular design is a bypass 2 engine with separate exhaust for both the fan and the turbine exhaust flow. For each of these engines, installation configuration items such as the subsonic inlet and general nacelle approach were incorporated in the engine weight estimates.

TYPICAL PARAMETRIC ENGINE

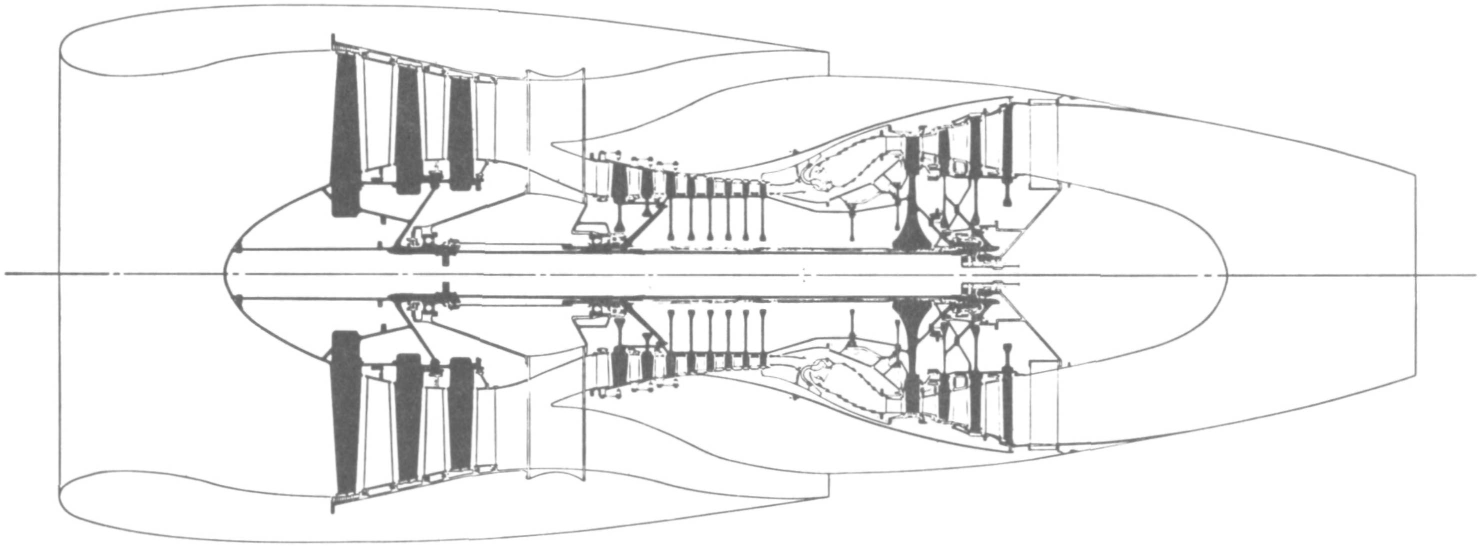

CHART 6

This chart shows the relative or equivalent payload decrement with varying turbofan bypass ratios. Concerning the booster alone, it is to be noted that the potential payload increases as bypass ratio is increased. Beyond approximately 4, however, the payload decrement becomes insignificant. The orbiter, on the other hand, shows a continuing decrease in payload capacity as turbofan bypass ratio is increased. Because of the relative weight sensitivity between the orbiter and the booster, the combined mission shows a minimum payload decrement in the vicinity of a bypass ratio of 2. This analysis was done on a hydrogen fuel basis. While the comparable JP fuel analysis has not been completed, the SFC effect and the attendant weight changes suggest that the best bypass ratio for JP fuel will shift to a higher value. The special case of a mixed flow engine did not show a significant impact in payload terms.

PAYLOAD EFFECTS

(COMMON ENGINES)

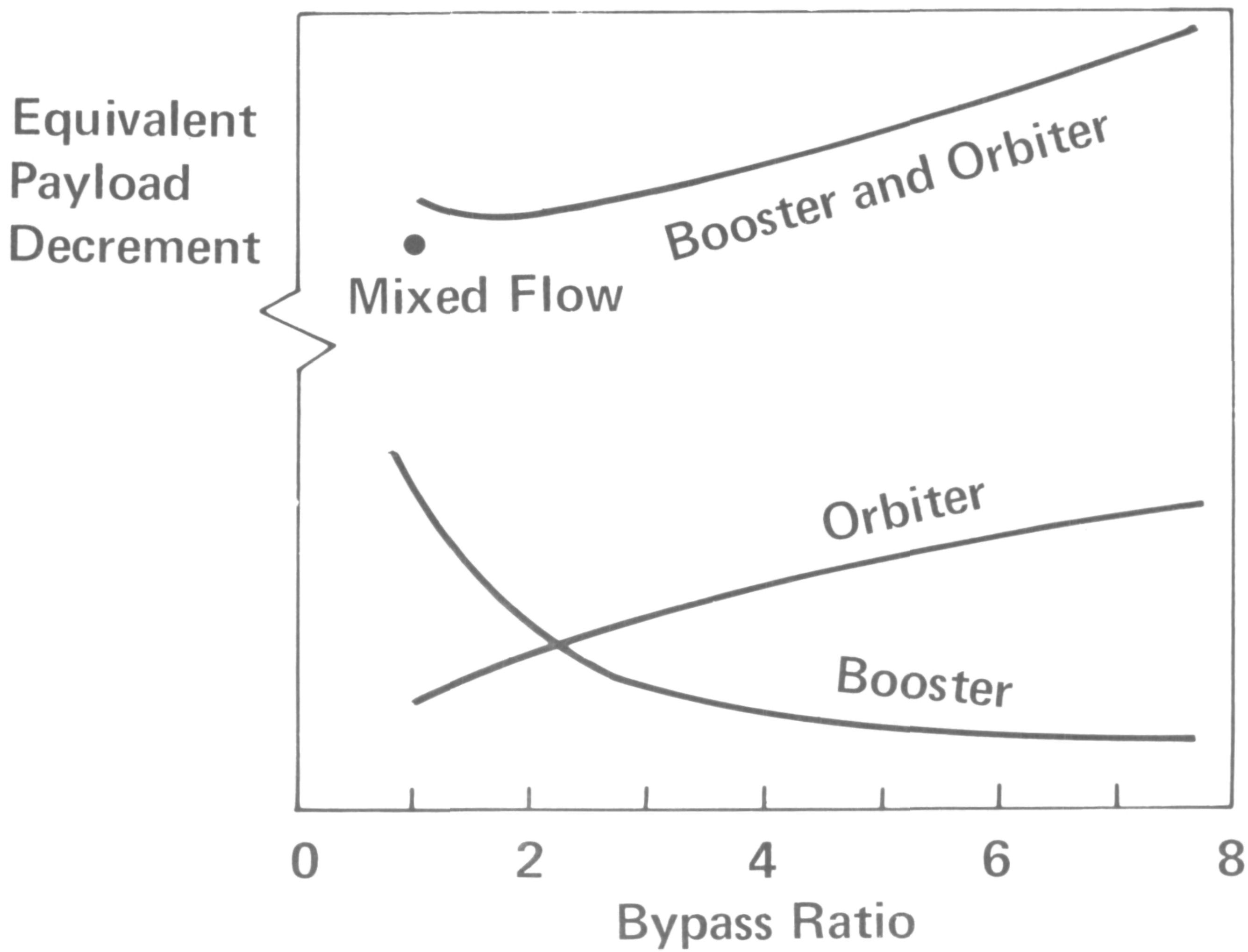

CHART 7

The conclusions reached as a result of the Task I effort suggests that for the case of a common engine, a turbofan having a bypass ratio near 2 would represent the best selection. Further, the consideration of the engine-out requirement indicates that in the case of the booster particularly, more than four engines should be selected in order to avoid a significant potential penalty due to the need to oversize the engine for the engine-out case. Following the evaluation of these parametric engines, specific engines were also exercised and a dry version of the GE-F101 engine was selected and recommended to the NASA Program Manager for continued study under Task II. This recommendation was accepted and the design study was initiated.

TASK I

CONCLUSIONS

- Optimum Booster Engine = Bypass Ratio above 4
- Optimum Orbiter Engine = Bypass Ratio Approximately 1.0
- Optimum Common Engine = Bypass Ratio in Range 1-2
- Engine-Out Requirement Suggests more than 4 Engines
- Dry F101 Engine Near Optimum
  - Selected for Subsequent Tasks

CHART 8

The major design considerations in Task II centered in four areas: The duty cycle and its effect on operating points and engine ratings, the mission environment with particular emphasis on lubrication and materials, the airstarting situation, and the reliability aspects of which airstarting appears to be of paramount importance.

MAJOR ENGINE DESIGN CONSIDERATIONS

- Duty Cycle
- Mission Environment
- Airstarting
- Reliability

CHART 9

The major design changes which evolved from the Task II study are listed here. It appears that an isolated oil tank, to minimize oil loss and contamination levels, could be accomplished without significant difficulty. Further, it appeared advantageous to incorporate a tank heater to maintain oil temperatures at approximately -25°F. Consideration has also been given to bearing and seal treatment with solid film lubricants as an insurance factor.

As part of the engine starting procedure, modifications of the fuel system were considered which would allow pre-fueling of the fuel system immediately before initiation of the start sequence. This would help in attaining a shorter time interval in attaining throttle-controlled operation. This item was primarily directed toward the use of JP fuel. With either fuel, hydrogen or JP, it was recommended that a fully redundant ignition system be adopted in that the number engine-supplied ignition circuits would be backed up by a separate ignition system utilizing spacecraft on-board power. In considering the use of liquid hydrogen the need to provide chill-down capability as well as purging, a new dual pump supply system and a new metering system would be required. With respect to engine materials, the current materials appear to be acceptable for the mission environment with the possible exception of some soft solder and some of the potting compounds which are normally employed in electrical control components. This latter point does not represent any particular barrier but does require further analysis.

DESIGN CHANGES

- Drain engine, seal oil in tank
- Heat oil to $-25^{o}F$ in orbit
- Possibly treat bearings and seals with solid film lubricants
- Prefill JP fuel system prior to start
- Replace soft solder and potting compounds in electric control components
- Fully redundant ignition
- LH_2 fuel system requires chilldown and purge cycles; new fuel supply and metering system

CHART 10

The most significant engine technical problem identified is that of airstarting. The problem is different than the usual air-start requirement in more conventional aircraft in that elapsed time and reliability aspects are particularly unique. As a result of analyzing this problem in further depth, stable combustion limits were assessed. As a result of the analysis and an examination of appropriate test data, it was concluded that for the flight trajectories currently defined, stable ignition and combustion could be obtained with either JP or hydrogen fuel below an altitude of 40,000 feet. Another aspect of the air starting problem, once ignition is obtained, is that of acceleration of the engines. Turbofans inherently require more unbalanced torque for acceleration and a higher core rpm for light-off than turbojet engines. Unbalanced torque, of course, has a significant effect on the starting time (the time increment from initiation of the start sequence to a throttle-controlled situation), hence an analysis was conducted for several trajectories and engine starter combinations. In this analysis the starting time and starter size were evaluated and it became apparent that a starter having a torque rating of approximately 400 to 500 ft/lbs would be required.

AIRSTARTING

- No combustion limits below 40,000 ft alt. (JP or LH_2)
- Turbofan windmilling characteristics poorer than turbojet at desired starting altitudes
  - Low core rpm for lightoff
  - Low unbalanced torque for acceleration
- Starting time analysis
  - Assumed family of starters
  - Assumed three reentry trajectories
  - Evaluated time to idle
  - Start time of 15-20 seconds requires oversized starter

CHART 11

To further illustrate the ignition and low speed combustion stability area, this chart shows both the estimated JP and hydrogen limits. Proper ignition and combustion is dependent upon the absolute level of pressure, temperature and velocity in the combustor dome. These conditions, represented by the severity parameter shown on the abscissa, are dependent upon the combustor equivalence ratio or the percentage of stoichiometric conditions that exist. The JP stability area is based on extensive testing on JP burners while the hydrogen limit region is less well defined although some test data do exist. Also shown is a representative ignition condition for the trajectories considered which suggest that stable ignition and combustion can be obtained with JP fuel.

COMBUSTION STABILITY LIMITS

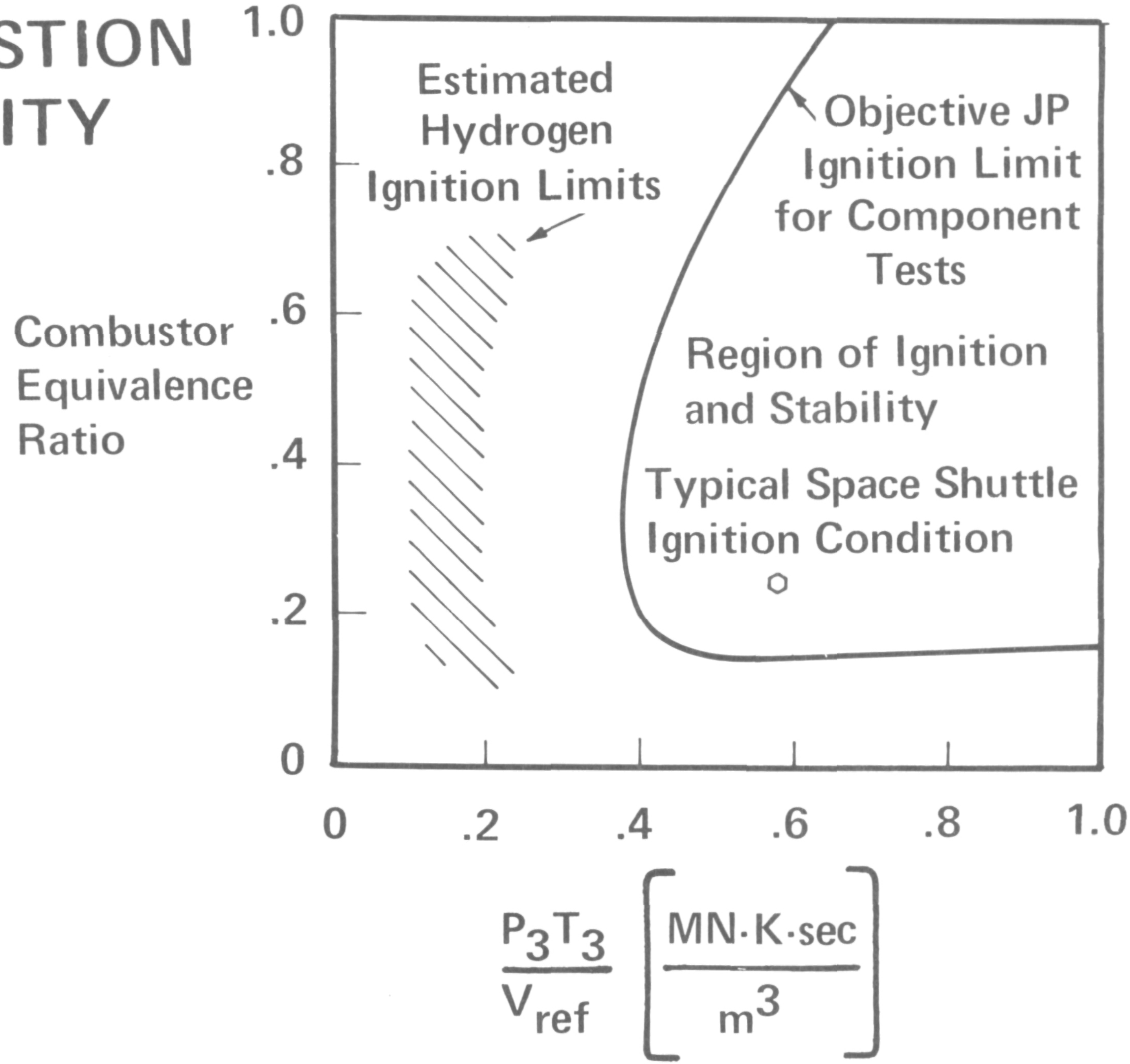

CHART 12

This chart amplifies earlier comments concerning starting time. The start time, in seconds, is the time from ignition sequence to a stable idle power setting. The starter size is represented by the torque levels shown on the ordinate. It appeared that the trajectories have little effect in attaining rapid start time, whereas starter size is the prime determinant in the analysis.

AIRSTARTING TIME TO IDLE

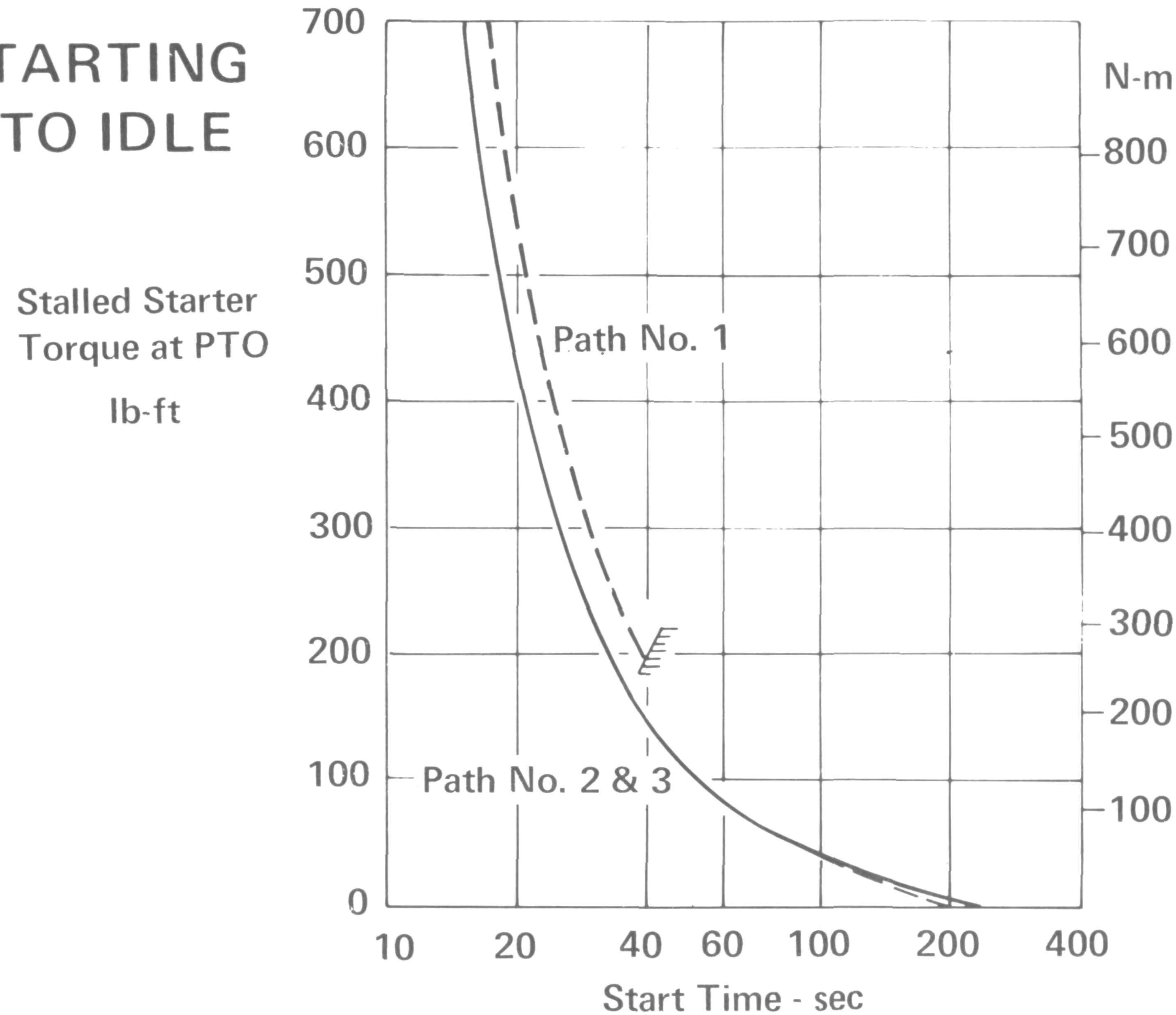

CHART 13

By far the most significant airstarting consideration is reliability. As mentioned earlier, conventional aircraft do not have the same reliability requirements as expected for the Shuttle Vehicles. The approach employed was that of a fault tree analysis in which all known factors were evaluated. Based on past experience each element or factor involved was assigned a reliability index and as a result, three items warranted specific attention. The need for a redundant ignition system and a suitably sized starter were indicated. In addition, because of the high ratio of full-power fuel flow to starting fuel flow, a more precise fuel flow control at low RPM would represent a significant improvement in the starting area. The current fuel control systems have not warranted this addition, however for JP system this additional feature can be accomplished. The fault tree analysis did indicate that the proof of starting reliability may be impractical by testing alone. Consequently, it was recommended that limited testing be done together with an expanded fault tree analysis.

RELIABILITY

FAULT TREE ANALYSIS

- Basis for quantitative analysis
- Identifies critical areas of engine design:
  - Fully redundant ignition
  - Ample starter size
  - Vernier fuel metering system

IMPRACTICAL TO PROVE STARTING RELIABILITY BY TEST
—RECOMMEND ANALYSIS SUPPORTED BY LIMITED TESTING

CHART 14

This chart amplifies the previous statement. Shown is the confidence level with which a number of successful starts could be accomplished for two different failure levels. As is apparent, the achievement of a 90% confidence level that the reliability index could be met with zero failures would require several thousand successful starts. Since these starts would have to be performed in a true altitude, speed environment, it would appear to be a major cost element. Therefore, it is believed that a judiciously defined test program with further fault tree type analysis will provide a satisfactory air start situation.

REQUIRED AIR STARTS TO ACHIEVE .999 STARTING RELIABILITY INDEX

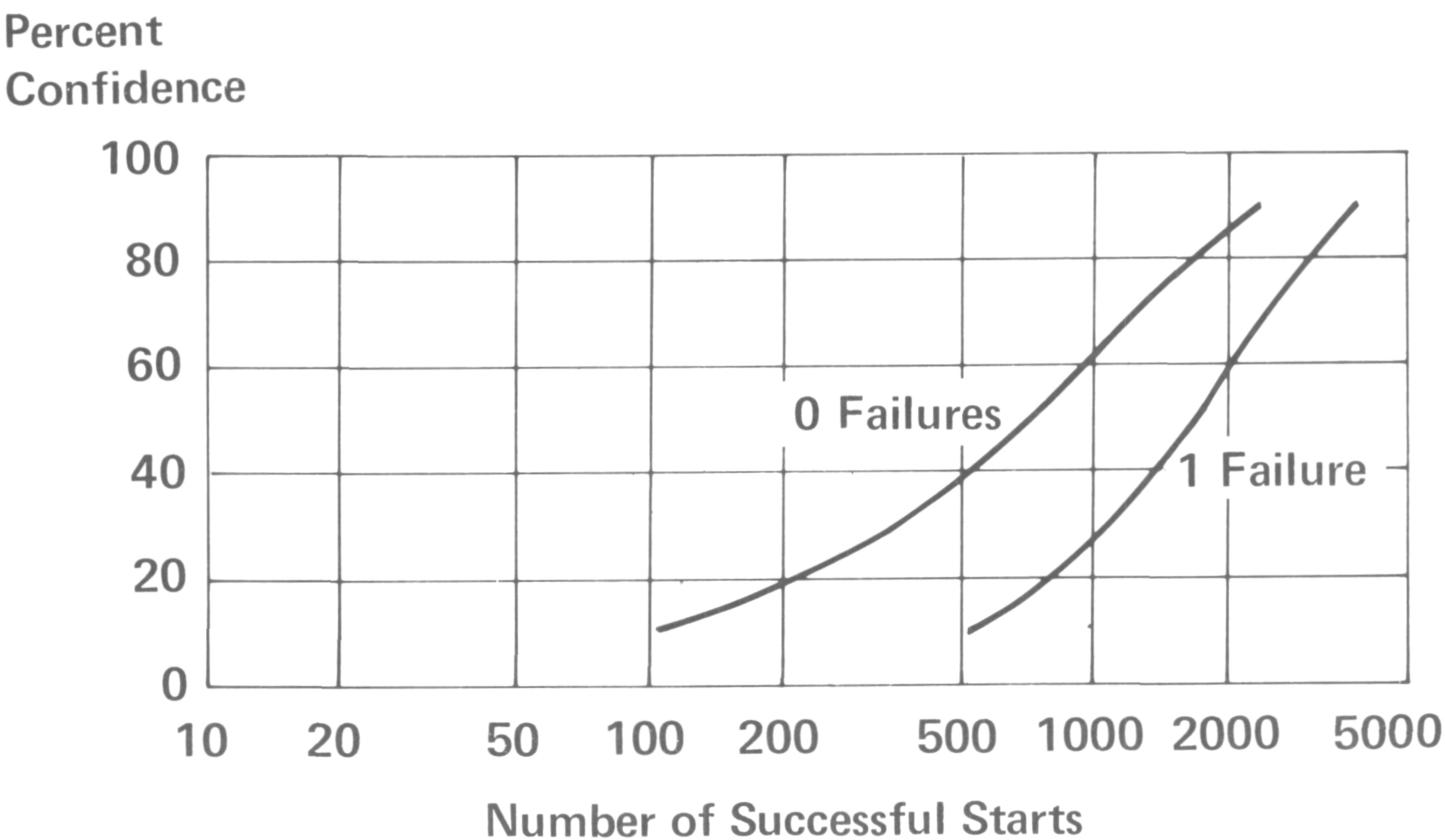

CHART 15

Task III of the program considered a recommended qualification procedure. Since the work was based on the derivative of an engine under development, the plan was based on those factors deemed unique to the application. A plan was evolved which concentrated on cyclic endurance representative of the mission duty cycle as well as altitude tank tests and flight test-bed engine operation. These tests were deemed essential, particularly for the airstarting area. Such flight testing was also considered essential in proof testing of the engine approximating closely the vehicle flight trajectories. In addition, items considered essential to space environment qualifications were evaluated. It was recommended that the engine be suggested to full mission range of pressures and temperatures particularly in the non-engine operating mode and currently facilities exist which would permit this testing. In addition, the launch vibration environment should be evaluated although the current estimated noise and vibration levels would not appear to represent a significant problem. As stated earlier, airstarting demonstrations and further reliability analyses are appropriate to the qualification plan. These recommendations have been provided to NASA Lewis.

ENGINE QUALIFICATION

(ASSUMES PRIOR QUALIFICATION OF BASE ENGINE)

PFRT (PRIOR TO HORIZONTAL FLIGHT TEST)

- Cyclic Endurance Test
- Altitude Performance Demonstration
- Flight Safety Proof Tests

SQT (PRIOR TO VERTICAL LAUNCH)

- Space Environmental Testing
- Static Vibration/Noise Testing
- Airstarting Demonstrations
- Reliability Analysis

CHART 16

To summarize the program effort to date the current study has been completed and a derivative of the F101 engine was recommended and used as the referenced engine. The modifications for the engine have been identified and a development plan, together with a preliminary performance specification, has been prepared and submitted to NASA Lewis. While the hydrogen versus JP fuel evaluation is not fully completed, it is clear that the use of hydrogen would require additional development, particularly in the fuel handling system and the related control and accessory area.

On the basis of the Space Shuttle mission as currently defined, it is believed that there are no technology barriers to fitting an engine of the type studied to this application. This is particularly true for a JP fuel system although some engineering development must be undertaken as part of the qualification plan.

SUMMARY

- Study effort completed and derivative F101 engine recommended
- Modifications to basic engine for JP use limited to:
  - Exhaust Nozzle and Mixer
  - Lube Tank
  - Control and Ignition Systems
- Development plan defined and recommended
- Preliminary specifications prepared
- LH_2 vs JP evaluation not fully completed, however LH_2 engine would require significant additional development
- No new technology for a JP system is required, some engineering development must be undertaken as part of the qualification plans

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"BOOSTER AND ORBITER ENGINE STUDIES"

H. G. MOORE

PRATT & WHITNEY AIRCRAFT

TECHNICAL MANAGER

A. J. GLASSMAN

LEWIS RESEARCH CENTER

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AIRBREATHING ENGINE STUDIES FOR SPACE SHUTTLE APPLICATION

Prepared By: H. G. Moore
Pratt & Whitney Aircraft
West Palm Beach, Florida

Contract: NAS3-14403
Arthur J. Glassman, Project Manager

Baseline concepts of both the booster and orbiter stages of the Space Shuttle presently require recovery and landing to be accomplished in a manner similar to conventional aircraft. To complete this portion of the mission, and provide booster flyback, airbreathing engines must be provided.

Since June 30, 1970, Pratt & Whitney Aircraft (FRDC) has been under contract with the NASA Lewis Research Center to conduct studies of hydrogen fueled airbreathing engine for the Space Shuttle. As of March 11, 1971, major tasks of this study have been completed. Hydrogen fuel was initially selected as the airbreathing engine fuel because of payload sensitivity to booster cruise fuel weight and vehicle inert weight. Since initiation of this study contract, the airbreathing engine fuel has been changed to JP-4. The magnitude of this change on total vehicle Gross Liftoff Weight (GLOW) is still being studied, however, preliminary results have indicated that GLOW may have increased as much as 1,000,000 pounds due to the change to JP-4. The costs associated with the increased weights may prove to be greater than costs required for a hydrogen fueled airbreathing engine system. Potentially large GLOW differences have prompted completion of the hydrogen engine studies to identify new technology areas, required engine modifications, and estimated costs. It is noted that much of the design study work in areas such as space proofing the lubrication system, in-flight starting, hot section trades, etc., are applicable to either JP or H_2 fueled Space Shuttle airbreathing engines.

This presentation reviews the study results and conclusions obtained from the hydrogen fueled airbreathing engine studies.

SPACE SHUTTLE BOOSTER/ORBITER AIRBREATHING ENGINE STUDY

The Space Shuttle airbreathing engine study was divided into four major tasks.

Task I was a comparative evaluation of several Pratt & Whitney Aircraft candidate engines for both booster and orbiter vehicles including a new engine optimized for the Space Shuttle. High and low cross-range booster and orbiter missions were considered.

Task II consisted of design studies for the JTF22A-4(H) engine. This is a nonaugmented F401-PW-400 engine derivative that was selected for study by the NASA Lewis Research Center. Design studies include potential weight reduction and performance improvement studies, definition of a hydrogen fuel and control system, and definition of critical research areas for hydrogen fueled airbreathing engines.

Task III required definition of a development program including the estimated cost and schedule to qualify the JTF22A-4(H) hydrogen fueled engine for the Space Shuttle.

Task IV consisted of the generation of a performance specification for the JTF22A-4(H) engine.

SPACE SHUTTLE BOOSTER/ORBITER AIRBREATHING ENGINE STUDY

Program Objectives:

Task I Comparative Evaluation of Pratt & Whitney Aircraft Candidate Space Shuttle Booster and Orbiter Engines

Task II Design Study of Engine Selected by NASA (Nonaugmented Derivative of F401-PW-400 Engine, P&WA Designation: JTF22A-4(H))

Task III Development and Cost Schedule

Task IV Specification for Selected Engine

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Pratt & Whitney Aircraft DIVISION OF UNITED AIRCRAFT CORPORATION
FLORIDA RESEARCH AND DEVELOPMENT CENTER

SCOPE OF TASK I

To assess the relative importance of engine parameters, technology, and cycle optimization on overall mission performance, ten separate engine configurations were studied. These engines included current production, current development, current development derivative, and new engines. Both augmented and nonaugmented engines were studied. New engine optimization studies included both turbojet and turbofan engines with a broad range of bypass ratio, fan pressure ratio, and high compressor pressure ratio.

Vehicles considered for study were both high and low cross-range booster and orbiter configurations. Missions selected were representative of concurrent Phase B study mission profiles.

SCOPE OF TASK I

Study Included:

- Ten (10) Candidate Engines
  - Current Production
  - Current Development
  - Current Development Derivatives
  - New Engines Optimized for Space Shuttle Mission
- High and Low Cross-Range Booster and Orbiter Missions

CANDIDATE ENGINE SCREENING PROCEDURES

Shuttle payload sensitivity to inert weight requires that candidate engine comparison be made with particular attention to total system effects. Total installed engine plus fuel weights must be considered for accurate comparative results. To assess the applicability of each engine for the booster, orbiter, and overall Space Shuttle, an engine screening procedure was used which compared the ten candidate engines on an installed weight basis for each vehicle/mission combination.

Relative engine development costs were estimated for each engine to provide the NASA material for cost vs payload trades.

CANDIDATE ENGINE SCREENING PROCEDURE

- Generated Performance for Ten Candidate Engines
- Defined High and Low Cross-Range Booster and Orbiter Mission Profiles
- Defined Engine Weights
- Completed Parametric Study for New Engine
- Established Number of Engines Required for Each Vehicle
- Determined Installed Engine Plus Fuel Plus Fuel Tank Weight
- Estimated Relative Engine Costs
- Compared Candidate Engines

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712403

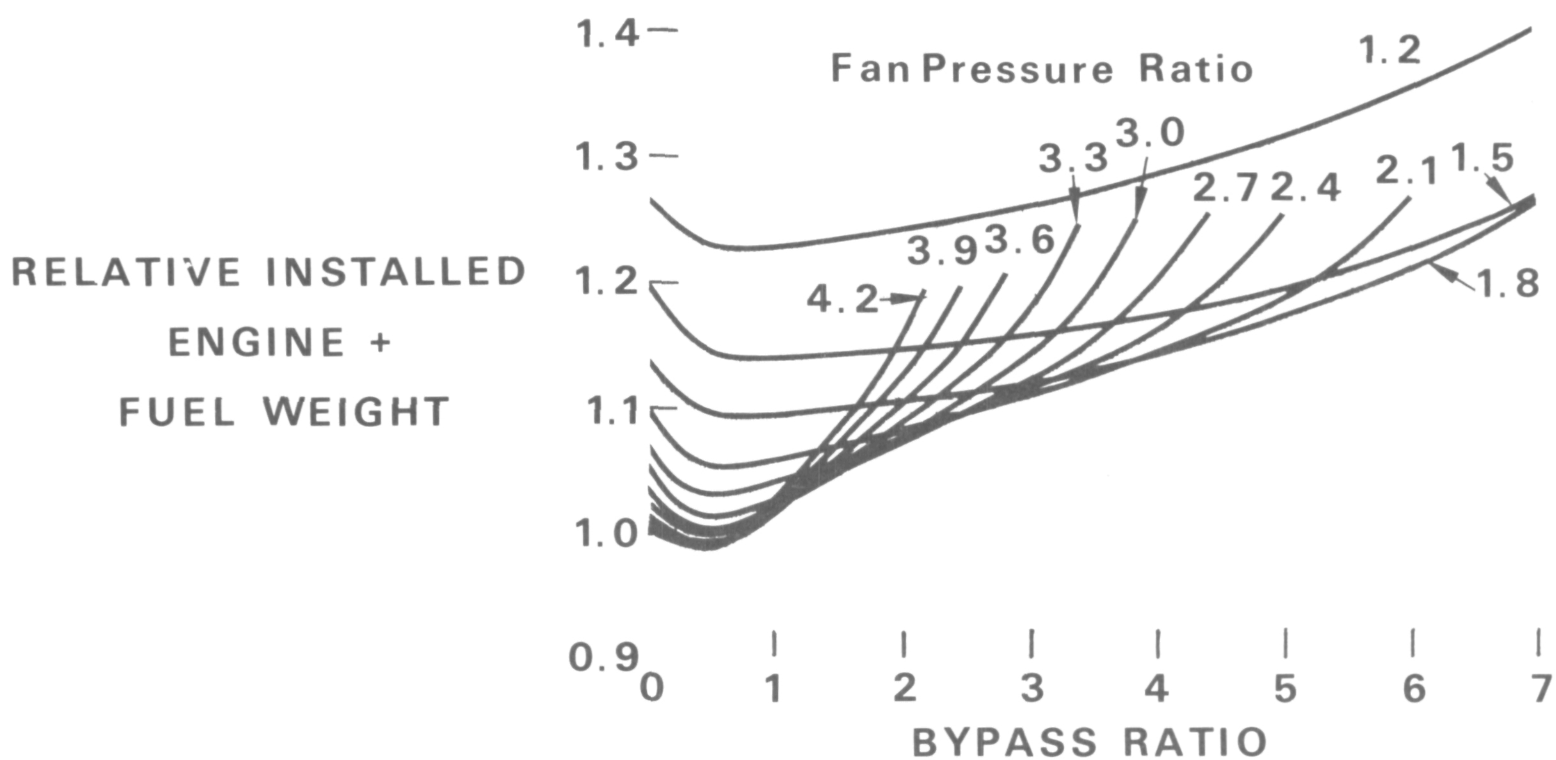
NEW ENGINE STUDY
(ORBITER)

The selection of a new engine cycle best suited to meet a specific set of vehicle requirements requires parametric studies of allowable cycle variables. New engine studies for the orbiter included evaluation of both turbojet and turbofan engine configurations. Cycle variables were bypass ratio, fan pressure ratio, high compressor pressure ratio, and augmentation. Relative engine weights were defined parametrically as a function of thrust size, specific thrust, and cycle variables. Critical engine sizing thrust requirement for the high cross-range orbiter was that required for one-engine-out approach abort. Total system weight comparisons were made of engine weight, installation weight including tankage, and fuel weight.

The parametric studies showed thrust/weight to be the predominate parameter which influences orbiter total airbreathing engine system weight. The parametric plot of installed engine plus fuel weight on the facing page is representative of the results obtained for both the high and low cross-range orbiter, and showed that a low bypass ratio turbofan engine is optimum.

NEW ENGINE
Orbiter

● **Conclusion -Low Bypass Ratio Optimum**



Pratt & Whitney Aircraft
FLORIDA RESEARCH AND DEVELOPMENT CENTER

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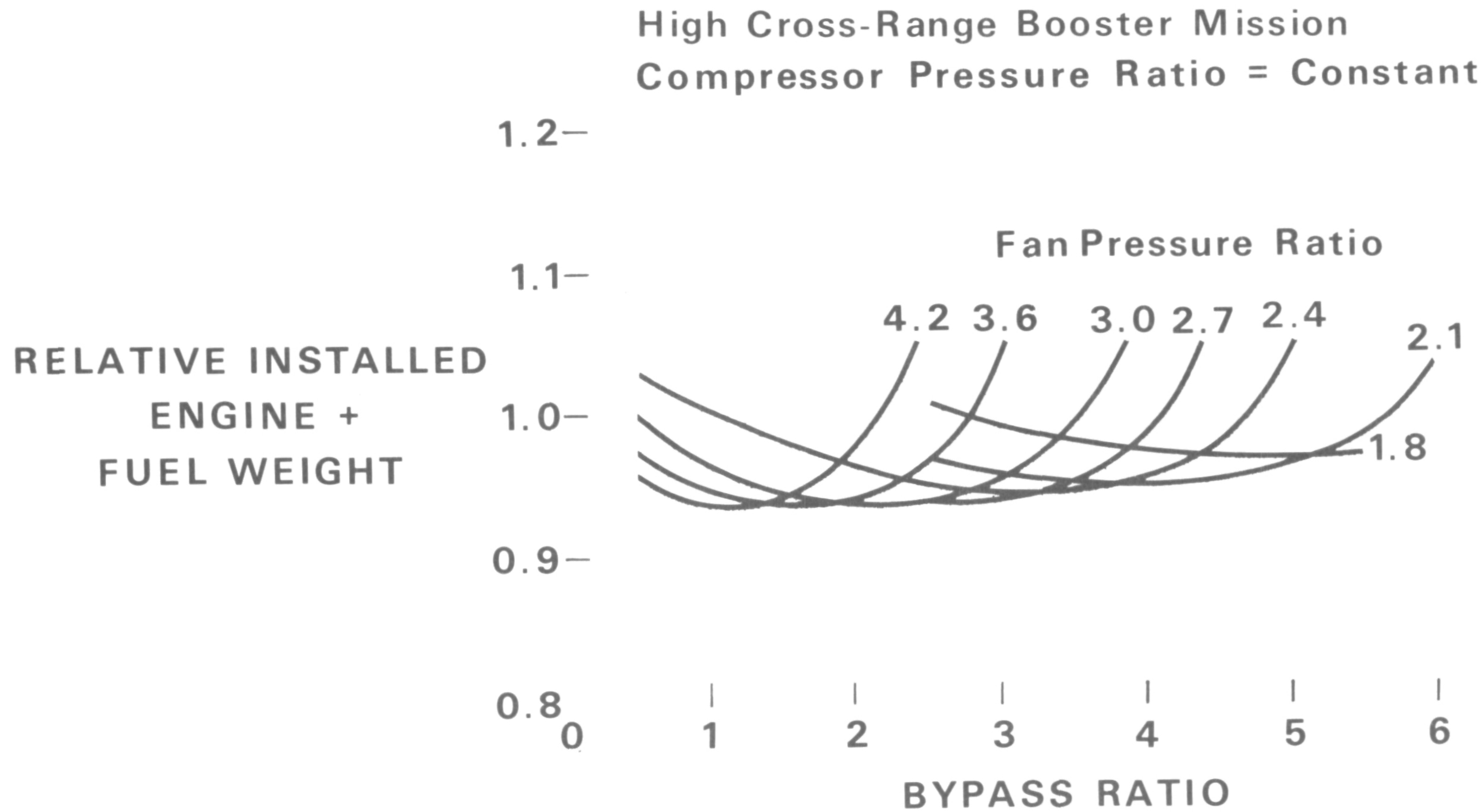
NEW ENGINE STUDY (BOOSTER)

Similar parametric studies were completed to optimize a new engine cycle for the booster vehicle. Although specific fuel consumption (TSFC) is a much stronger influencing parameter for the booster cruise mission, fuel load differences with hydrogen do not off-set the higher installed engine weights of the high bypass ratio engines. This is due to the short booster cruise range (350-450 NM.) and the low basic TSFC with hydrogen fuel. The booster study showed that system weight differences are small over a broad range of bypass ratio, with the optimum again occuring at low BPR. A factor which must also be considered by the vehicle manufacturer is engine diameter. Since the airbreathing engines must be thermally protected during reentry, small diameter engine configurations can significantly simplify engine installation. Accordingly, the low bypass ratio, small diameter, turbofan engine is best suited to the hydrogen fueled booster vehicle.

NEW ENGINE
Booster

● Conclusion - Low Bypass Ratio Optimum



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NEW ENGINE STUDY
(COMMON ENGINE)

Development of two separate new engines, one for the booster and one for the orbiter, would double development costs. The more reasonable comparison of a "new engine" optimum cycle is, therefore, on a common engine basis for the booster and orbiter vehicles. To define the optimum common engine cycle, payload sensitivities of each vehicle derived from Phase B vehicle data were defined. These factors are:

$$\text{Orbiter:} \quad \frac{\text{Payload}}{\text{lb inert weight}} = 1.0$$

$$\text{Booster:} \quad \frac{\text{Payload}}{\text{lb inert weight}} = 0.18$$

The chart shown on the facing page illustrates overall vehicle payload sensitivity to engine cycle selection. This chart shows the optimum new common engine cycle to be a low bypass ratio, moderately high fan pressure ratio configuration.

NEW ENGINE
(Common Vehicle)

Conclusion:
Low Bypass Ratio
Optimum for Common
Booster/Orbiter Engine

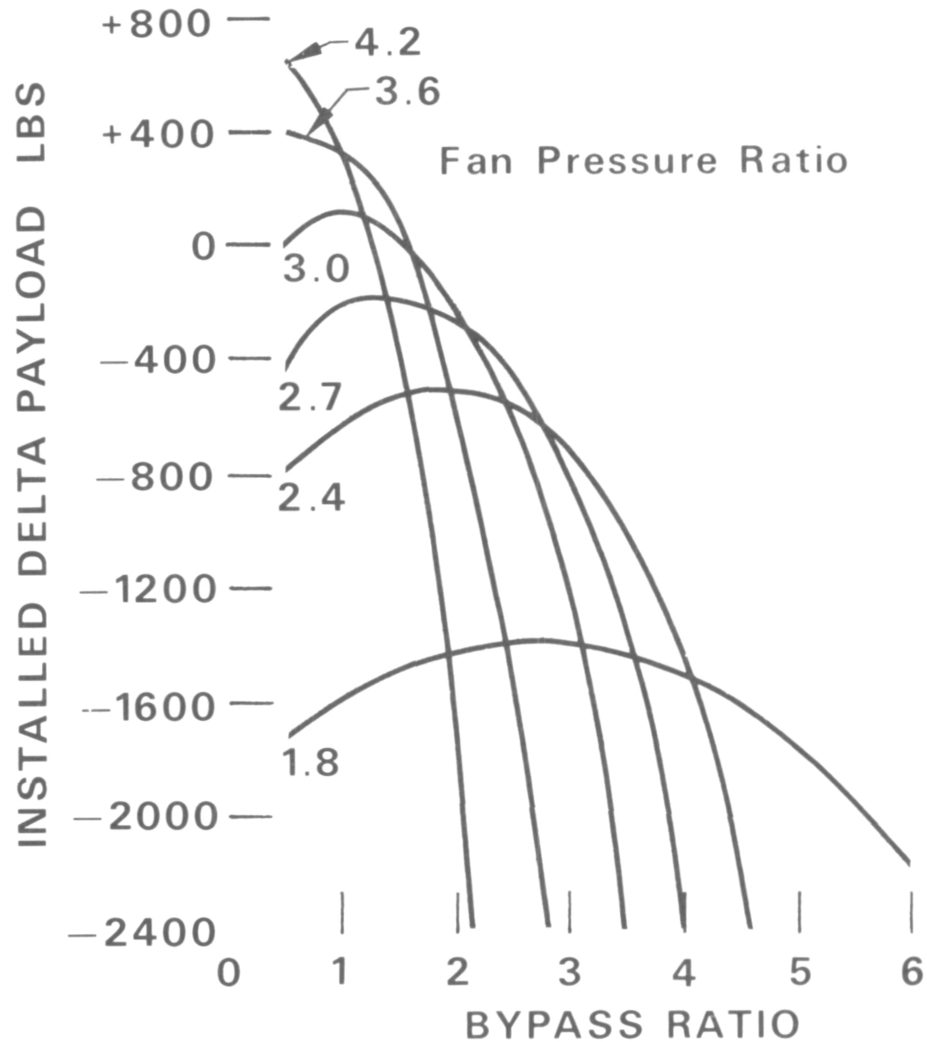
High Cross-Range Mission

Compressor Pressure
Ratio = Constant

Payload Sensitivity:

Orbiter: $\frac{\Delta \text{ Payload}}{\text{lb Inert wt}} = 1.0$

Booster: $\frac{\Delta \text{ Payload}}{\text{lb Inert wt}} = 0.18$



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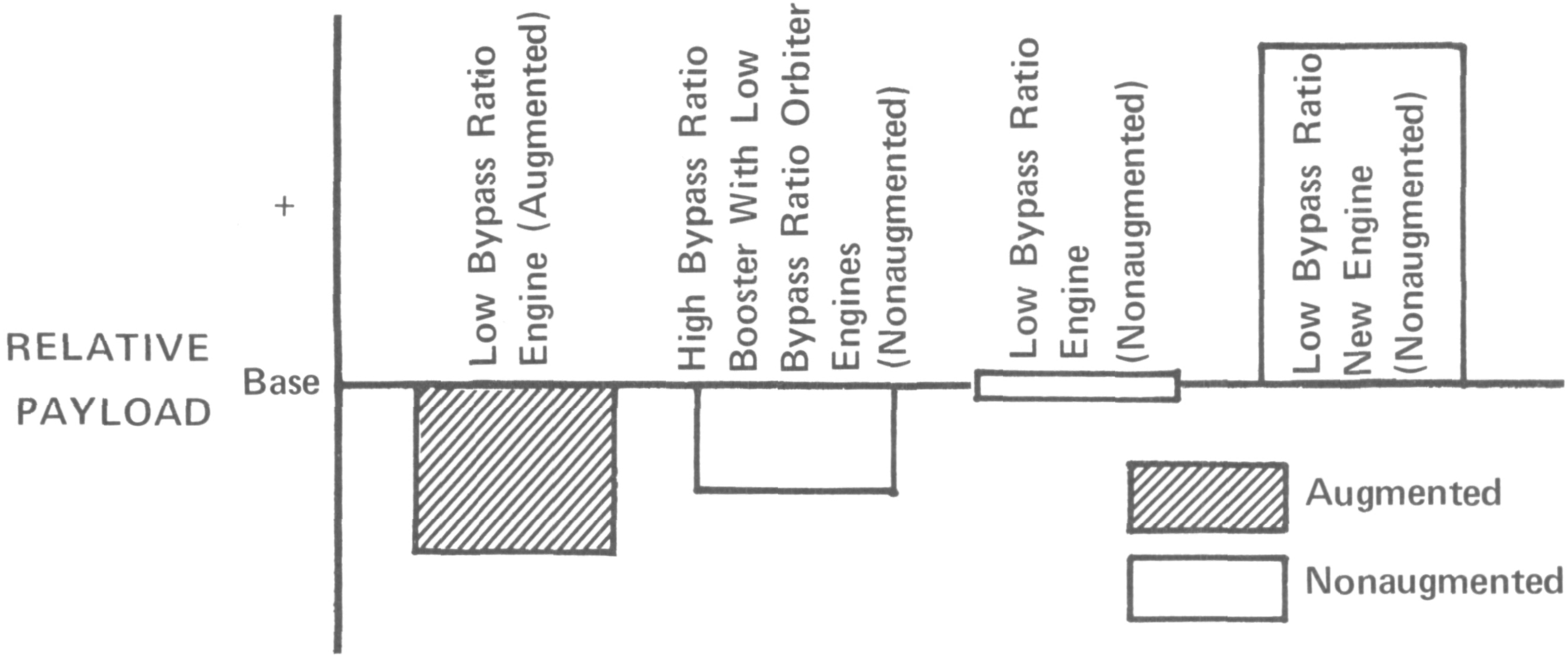
SUMMARY OF CANDIDATE ENGINE COMPARISON

P&WA candidate engines were evaluated for booster only, orbiter only, and common engine application based on total installed engine plus fuel weight and comparative payload gain.

Results of the Task I engine comparison were:

1. Low bypass ratio turbofan engine yields minimum system weight,
2. Augmentation not attractive.
3. High thrust/weight important due to payload sensitivity to inert vehicle weight.
4. New engine optimized for the Space Shuttle could improve payload approximately 4000 to 5000 pounds.

SUMMARY OF CANDIDATE ENGINE COMPARISON



- Low Bypass Ratio Turbofan Yields Minimum System Weight
- Augmentation Not Attractive
- High Thrust/Weight Important
- New Engine Developed for Space Shuttle Would Improve Payload 4000 to 5000 lb
- Low Bypass Ratio, Nonaugmented, Turbofan Selected by NASA for Continued Design Studies

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TASK II STUDIES

Task II consisted of design studies for the JTF22A-4(H), nonaugmented F401-PW-400 engine derivative. Design studies included potential weight reduction and performance improvement studies, definition of a hydrogen fuel and control system, and definition of critical research areas for hydrogen fueled airbreathing engines. In addition, studies were completed to define engine hot section trades, required lubrication system modifications, inflight starting methods, and engine system weights.

TASK II STUDIES

- Define Vehicle Requirements and Engine Operating Environment
- Complete Engine Hot-Section Trade Studies
- Conduct Stress and Vibration Analysis
- Define Required Lubrication System Modifications
- Define Hydrogen Fuel System Including Control Requirements
- Evaluate In-Flight Starting
- Define Hydrogen Engine Configuration and Weight
- Identify Critical Development and Research Areas

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LUBRICATION SYSTEM

Design studies of the JTF22A-4(H) engine lubrication system have shown that modifications required for space application are minor. These modifications include the addition of a separate oil storage tank, and oil shutoff valve, and three more air/oil coolers. The JP fuel/oil cooler would be eliminated. To simplify draining prior to prelaunch vertical positioning, a common engine drain would be added.

Storage of lubrication oil in a separate tank during prelaunch, launch, orbit, and initial reentry solves potential problems associated with the space environment and seal leakage. Use of a separate storage tank also permits thermal conditioning of lubrication oil during orbit if compartment temperatures are below oil freezing levels.

LUBRICATION SYSTEM

- JTF22 Lubrication System Modifications Required for Space Application Are Minimal

- Modifications Required Include:

  1. Addition of Oil Storage Tank and Tank Shutoff Valve
  2. Addition of Three Air/Oil Coolers. (Three Air/Oil Coolers on Present F401-PW-400 Engine)
  3. Addition of Common Drain Fitting
  4. Removal of JP Fuel/Oil Cooler

- Lubricating Oil Stored in Seperate Tank From Prelaunch to Reentry

FUEL SYSTEM SELECTION

Development of a hydrogen fuel and control system will be the only significant effort required to adapt the present F401-PW-400 engine for hydrogen fuel. A primary task of the P&WA air-breathing engine study was to screen possible candidate pump and drive systems, select the system which best meets shuttle requirements, and complete preliminary design studies of the selected system. Stringent limits on NPSP (2.0 psia minimum) and high flow turndown ratios (50:1 minimum) were applied to each candidate system. A required attribute of each design was that it provide proper flow and pressure at each flight condition within the Space Shuttle operating envelope for all power settings from idle to maximum power.

Candidate fuel pump and drive systems studied are shown on the facing chart.

Alternative schematics of a vane pump and a centrifugal pump fuel system were provided.

FUEL SYSTEM SELECTION

- Candidate Fuel Pumps
  - Staged Centrifugal
  - Fixed Displacement Vane
  - Variable Displacement Vane
  - Gear
  - Hybrid

- Candidate Drive Systems
  - Direct Engine Gearbox
  - Engine Gearbox With Variable Speed Transmission
  - Direct, Variable Speed, Air Turbine
  - Geared, Variable Speed, Air Turbine
  - Regenerative, Hydrogen Turbine

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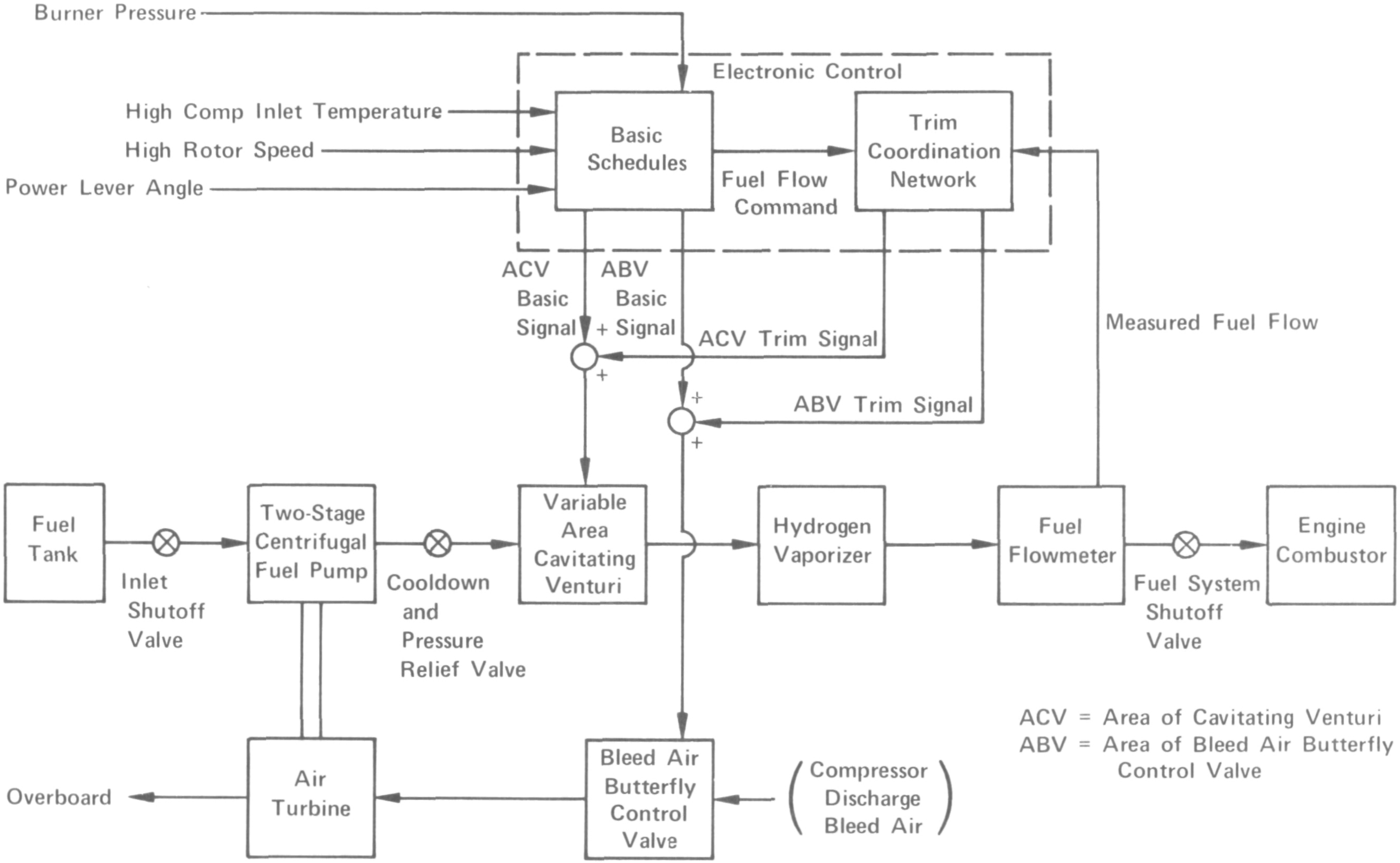
HYDROGEN FUEL SYSTEM SCHEMATIC

The lowest risk fuel and control system for the hydrogen engines would use a centrifugal pump as shown on the facing chart. To minimize development risk and cost, components (where possible) are scaled from proven rocket or gas turbine engine designs. Examples are:

Inlet Shutoff Valve	- Scaled RL10 fuel pump inlet valve.
Two-Stage Centrifugal Fuel Pump	- Scaled 5K Flox-Methane fuel pump. Blade angles and inducer design same as RL10A-3-7 fuel pump which demonstrated 0 NPSP pump operation (liquid at inlet)
Cavitating Venturi	- Scaled RL10A-3-7 design.
Cooldown and Pressure Relief	- Scaled RL10 C/D valve.
Fuel System Shutoff Valve	- Scaled RL10 fuel pump inlet valve.
Prestart, Start, Starter Bleed, and Purge Valves	- RL10 configuration.
Bleed Air Pump Speed Control Valve	- 1. Buterfly valve used for P&WA 550K main oxidizer valve controller. 2. Scaled from F401-PW-400 augmentor pump controller valve. 3. Similar to J58 augmentor pump controller valve.
Air Turbine	- Similar to augmentor pump drive turbine for J58 and F401-PW-400 engines.

By utilizing proven design concepts and existing cryogenic test facilities, development cost of the hydrogen fuel system can be reduced.

HYDROGEN FUEL SYSTEM SCHEMATIC WITH CENTRIFUGAL PUMP



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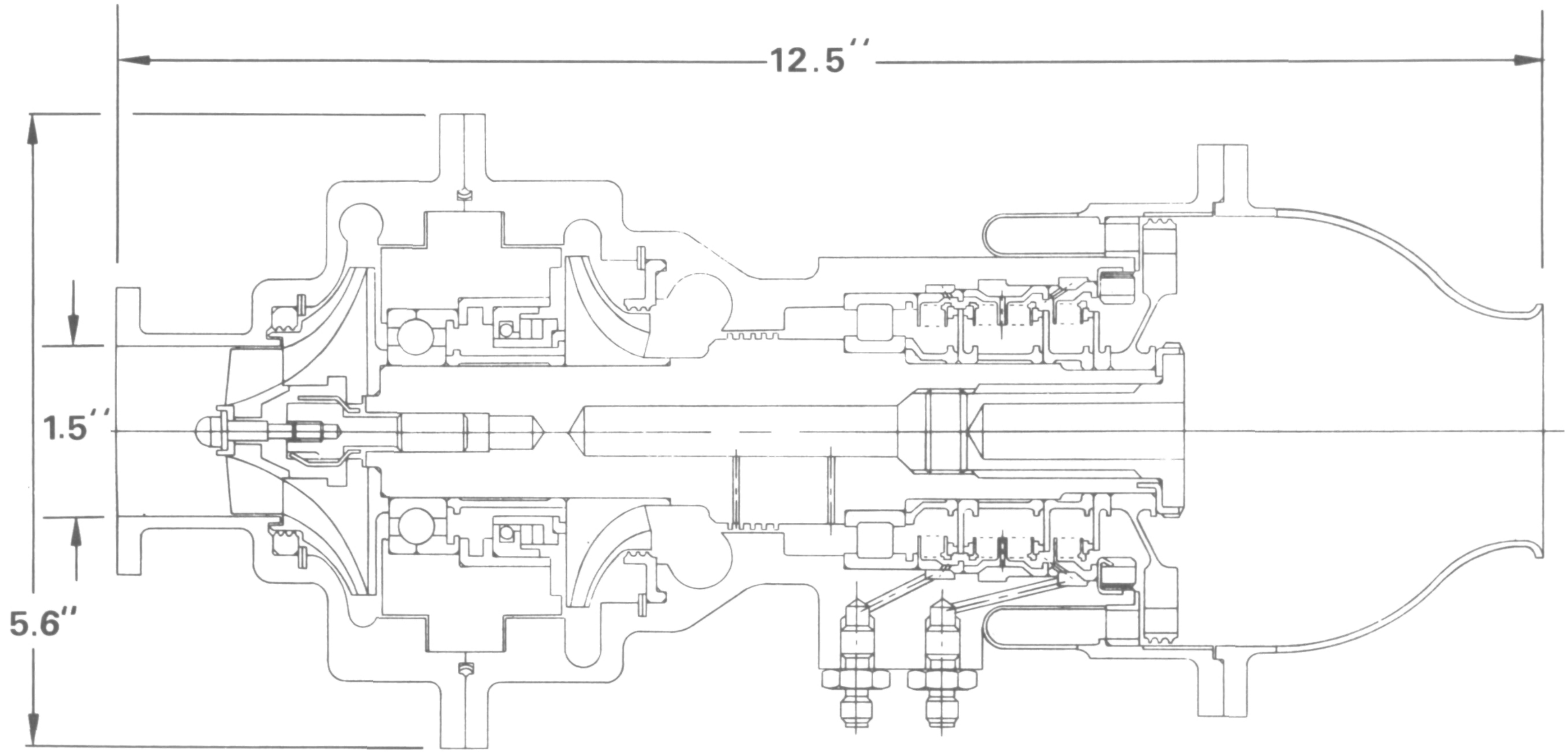
JTF22A-4(H) HYDROGEN TURBOPUMP

A preliminary layout of a proposed JTF22A-4(H) turbopump assembly is shown on the facing page. This design is essentially a scaled version of the 5K Flox-Methane pump. The turbopump assembly consists of a two-stage centrifugal pump driven by a single-stage, full admission impulse turbine. Turbine drive air is supplied from compressor discharge. P&WA currently utilizes air turbines driven by compressor discharge bleed to drive augmentor JP fuel pumps on both the J58 and F100/F401 engines. This turbopump drive concept has proved to be highly reliable.

TURBOPUMP



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JTF22A-4(H) HYDROGEN FUELED ENGINE DEVELOPMENT SCHEDULE

A primary objective of the Space Shuttle airbreathing engine study has been to minimize overall development costs while insuring delivery of qualified engines consistent with shuttle milestones. As part of this study, P&WA has outlined a development program for qualification of the hydrogen fueled JTF22A-4(H) engine based on shuttle schedules which include first horizontal flight in June 1976, first manned orbital flight in April 1978, and orbital capability in July 1979.

Scope of the development program includes development of required engine modifications, delivery of mockup and ground test engines for vehicle installation compatibility tests, and completion of Preliminary Flight Rating Test (PFRT) and Qualification Test (QT). Development milestones and component test schedules are shown on the qualification program schedule in the facing chart.

JTF22A-4(H) QUALIFICATION SCHEDULE MILESTONES

- F401-PW-400 Qualification Test - 1st Quarter 1973
- JTF22A-4(H) Go-Ahead - September 1972
- Component Testing - March 1973 to September 1974
- Engine Testing - September 1974 to September 1976
- Preliminary Flight Rating Test - November 1975
- Delivery of Prototype Engines for Horizontal Flight Test - December 1975
- Qualification Test - September 1976
- Production Engine Delivery - October 1976

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CRITICAL TECHNOLOGY AREAS

Currently developed hydrogen turbopumps with fuel cooled bearings are designed for very low life compared to the life requirements of an airbreathing engine. The development of long life, hydrogen cooled bearings and seals represents a significant new area of technology for a hydrogen fueled airbreathing engine. Development of an electronic control system becomes critical with respect to design and procurement times to meet overall engine development schedules.

Other critical design areas which should be explored are possible brinelling of engine bearings due to launch vibration, and the effects of space environment on engine materials. Since engine bearings are designed for much longer life than required for Space Shuttle application, brinelling is not expected to be a significant problem. Vacuum testing of a complete engine system would identify potential vacuum effects on engine seals and materials.

Early investigation of the fuel system and in-flight starting techniques could be most beneficial to the overall development plan.

CRITICAL TECHNOLOGY AREAS

- Hydrogen Fuel System Concept Definition and In-Flight Starting Procedure
- Turbopump Life:
  - Fuel Cooled Bearings
  - Seals
- Electronic Control System
- Engine Bearings:
  - Possible Brinelling Due to Launch Vibration
- Vacuum Effect on Engine Seals and Materials

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