ROCKET ENGINE FUNDAMENTALS

SESSION 1

April 1, 1991

JAY R. COBIA/
VINCE GARCIA

COURSE SCHEDULE

Eleven Sessions for Rocket Fundamentals

Date	Session	Topic	Instructors
4/1	1	Intro to Rockets, Rocket Programs, History	Cobia/Garcia
4/8	2	Vehicles, Application & UTC Involvement	Cobia/Garcia
4/15	3	Rocket Engine Cycles & System Controls	Spinn/Petrino
4/22	4	Rocket Engine Performance I - Pumps	Lawing
4/29	5	Rocket Engine Performance II - Turbines	Spryer
5/6	6	Trajectory & Vehicle Analysis	Joyner
5/13	7	Materials & Fabrication Technology	Bales
5/20	8	Component Design & Structural Analysis I	Hudson
6/3	9	Component Design & Structural Analysis II	Haluck/Bonnell
6/10	10	Component Design & Structural Analysis III	Palgon
6/17	11	Test Area "E" Facilities, Programs & Tour	Gillis

GENERAL

- ELEVEN WEEK COURSE
- CLASSES IN ROOM C IN MOD 4
- LAST CLASS WILL MEET IN TEST AREA E
- CLASSES 4:30 TO 6:30 PM ON MONDAYS
- NO CREDIT FOR CLASSES AFTER 5:00
- TWO MISSES MAXIMUM FOR COURSE CREDIT

ROCKET PROGRAMS – INTRODUCTION

OBJECTIVES

1. Students will learn the basic difference between rocket engine and gas turbines to be able to relate this course to their experience.
2. Students will learn a brief history of rocketry with emphasis on P&W History to be able to know what experience exists at P&W and how it forms the basis for ongoing programs.
3. Students will learn about the different ongoing rocket program at P&W and be able to understand the goal of each.

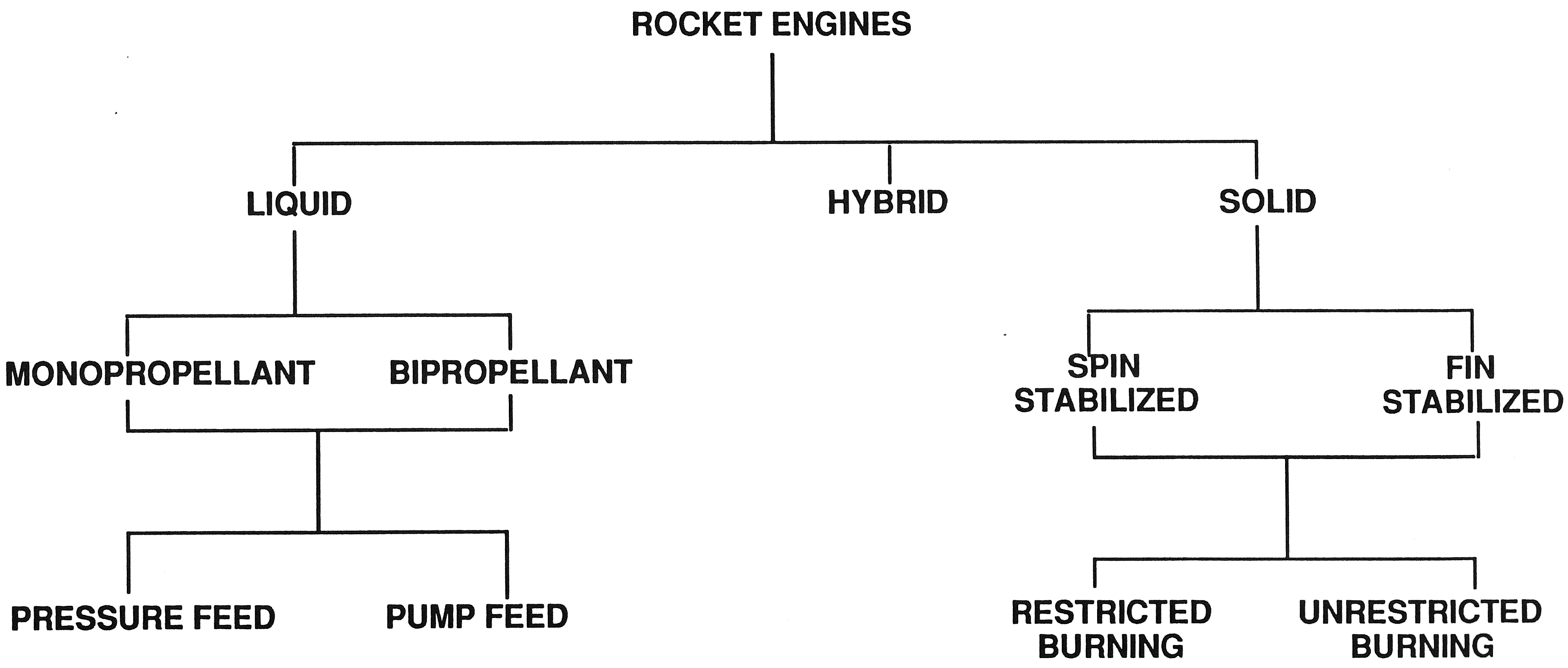
LIQUID ROCKET ENGINE

A device for converting the thermochemical potential energy of one or more propellants into exhaust jet kinetic engergy.

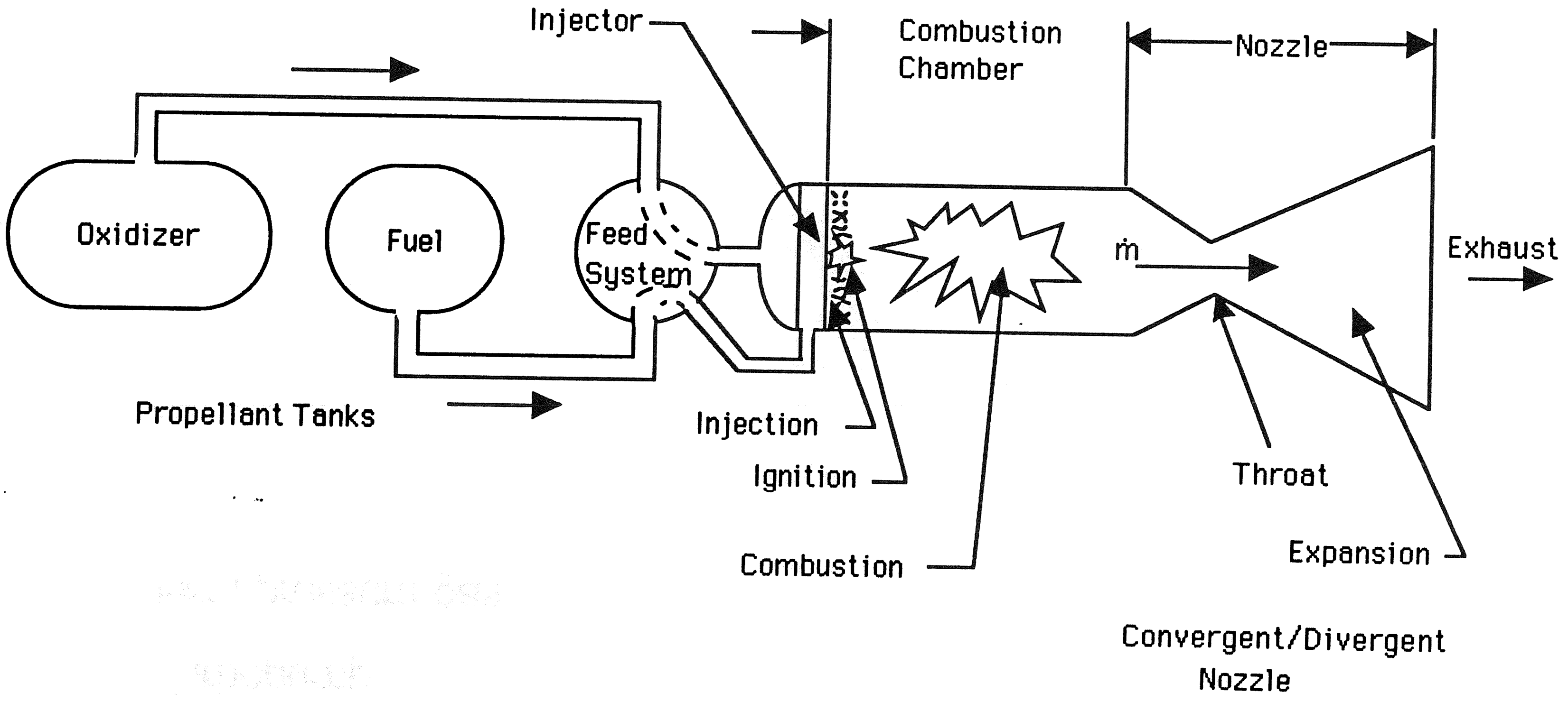
Conversion is accomplished through several distinct steps:

- propellant feed
- injection
- ignition
- combustion
- expansion

ROCKET ENGINE CLASSIFICATION



SIMPLIFIED LIQUID ROCKET ENGINE



PROPELLANT FEED

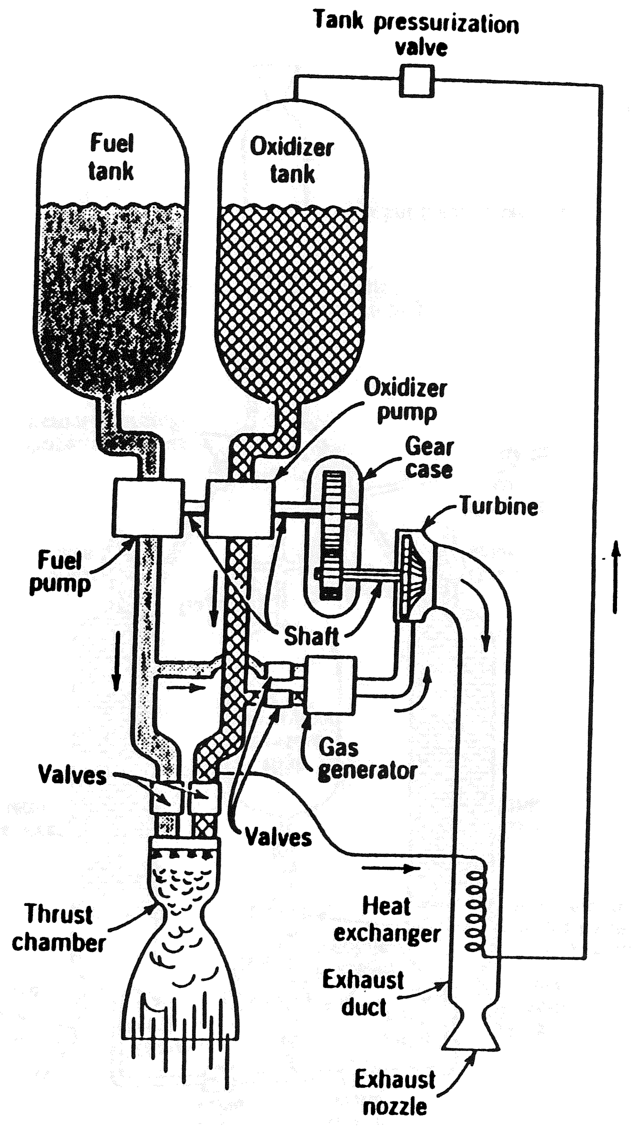
A propellant feed system forces the liquid propellant from the tanks and into the injector.

METHOD

Turbopump

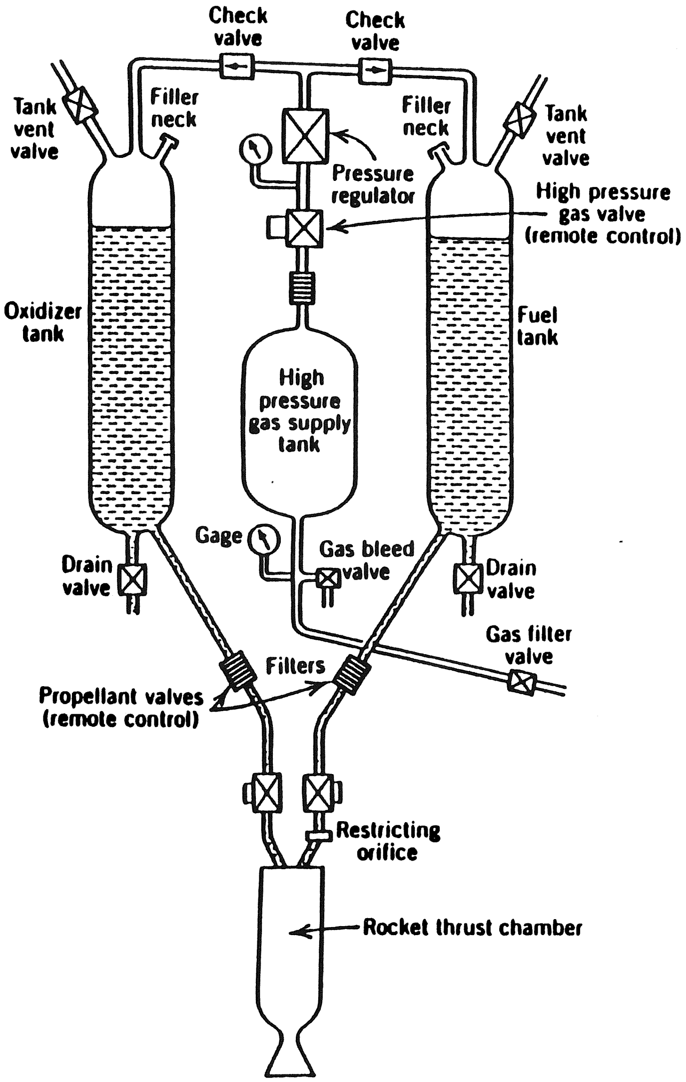
High pressure gas

PUMP FED ROCKET ENGINE



Simplified schematic diagram of a liquid propellant rocket engine with turbopump feed system and a separate gas generator.

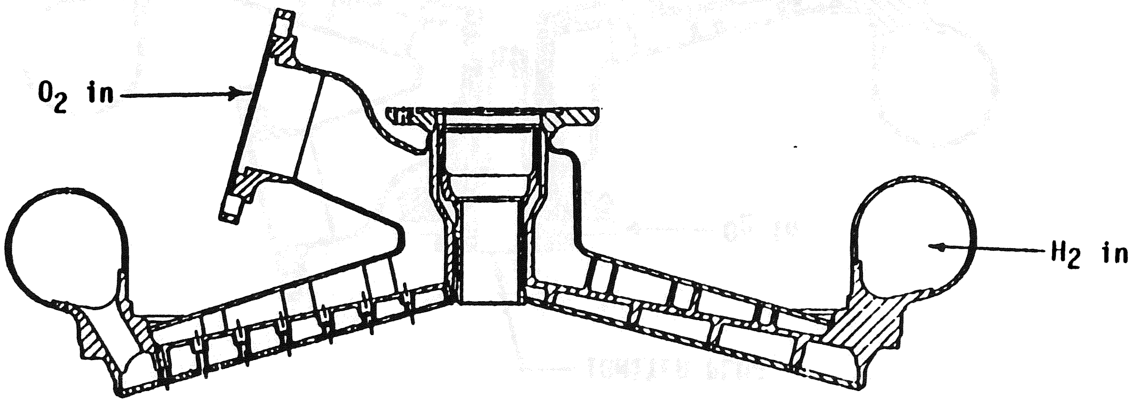
PRESSURE FED ROCKET ENGINE



Schematic diagram of a liquid propellant rocket engine with a gas pressure feed system.

INJECTION

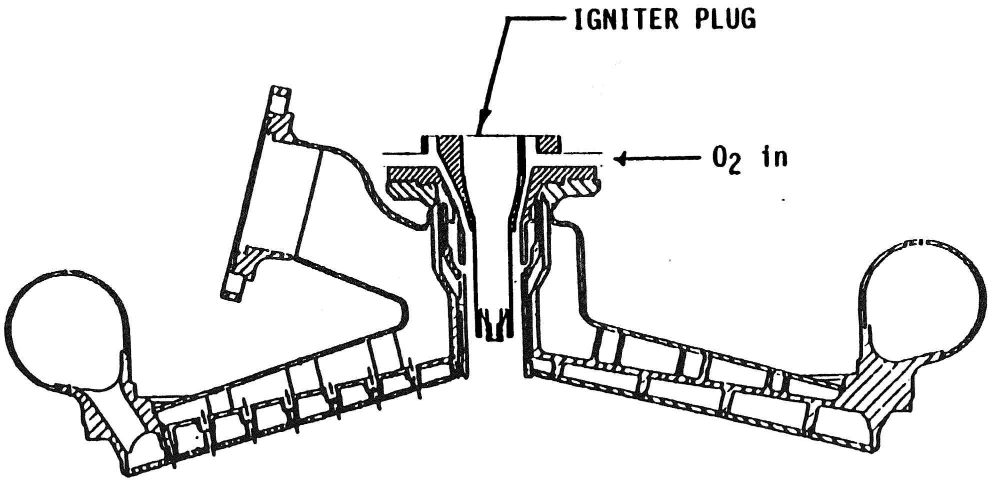
Injection distributes the oxidizer & fuel into a flow pattern that results in thorough mixing when the propellents enter the combustion chamber. Oxidizer & fuel flows are controlled so that the required mixture ratio of oxidizer to fuel is maintained.



TYPICAL RL10 INJECTOR ASSEMBLY

IGNITION

Ignition occurs near the face of the injector as the propellants enter the thrust chamber. An ignition device is used to initiate combustion; the heat from the combustion gases provides continous ignition.



TYPICAL RL10 IGNITER / INJECTOR ASSEMBLY

COMBUSTION

- Energy conversion process in which chemical potential energy is converted into thermal gas potential energy. The process is rapid but not instantaneous; burning takes place throughout the combustion chamber and nozzle, with some residual burning in the exhaust gas jet. The bulk of the combustion process, however, is completed before the gases enter the nozzle.
- The combustion process consists of the following steps:

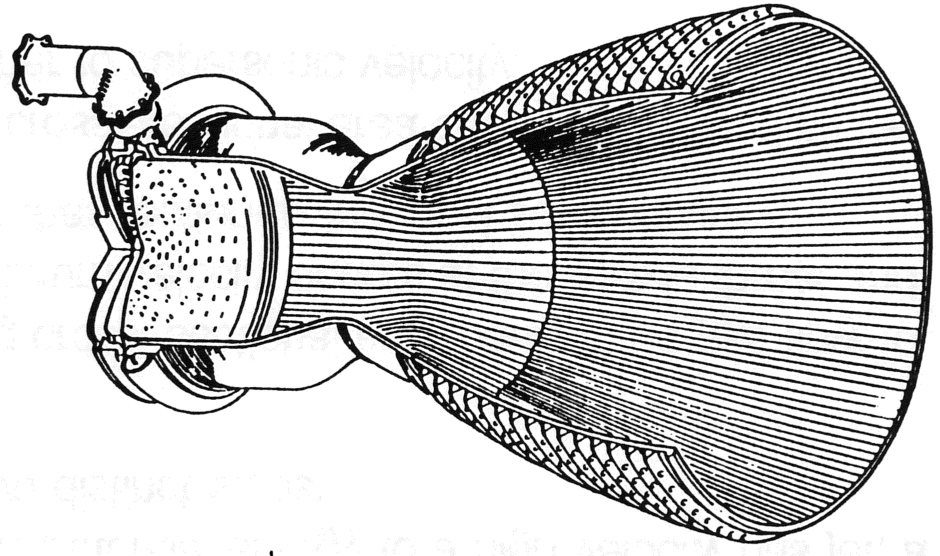
injection

impingement and mixing

atomization and mixing

vaporization

gas reaction



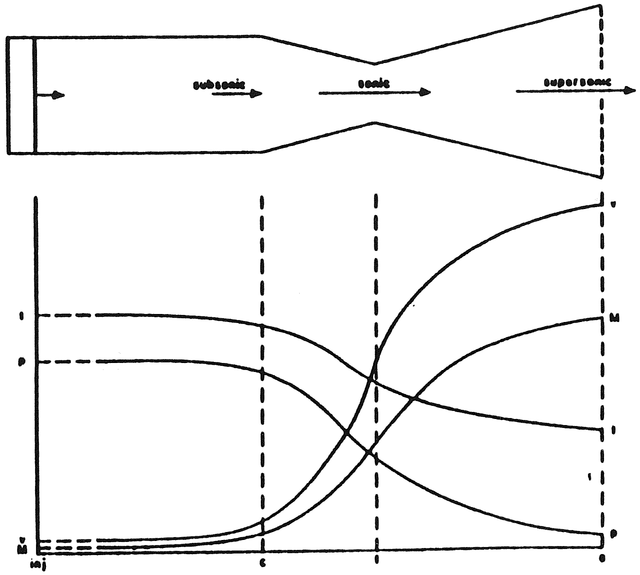
TYPICAL THRUST CHAMBER ASSEMBLY

EXPANSION

In the nozzle the gas undergoes an expansion process. The nozzle is employed to assist in the change from gas potential energy to a high velocity gas jet, a process which is divided into two distinct steps:

Converging – The decreasing cross–sectional area causes the gas flow to speed up. Maximum velocity occurs at the nozzle throat and corresponds at that section to the local sonic velocity.

Diverging – The increasing cross–sectional area caused the gas flow to accelerate further to supersonic velocity.



Variation of Parameters in Thrust Chamber

FUNDAMENTAL THRUST EQUATION

The following exhaust jet characteristics remain constant at the nozzle exit:

V_e "Theoretical Exhaust Velocity"

$\dot{m}$ "Propellant Mass Flow Rate"

P_e "Exit pressure"

The product of $\dot{m}$ and V_e is called the "momentum thrust"

$$F_1 = \dot{m} V_e$$

F_1 ~ pounds

in ~ slugs/sec.

V_e ~ ft./sec.

The product of the difference between exit and ambient pressure and the exit area is called the "pressure thrust"

$$F_2 = (P_e - P_a) A_e$$

F_2 ~ pounds

P_e ~ psia

P_a ~ psia

A_e ~ in^2

FUNDAMENTAL THRUST EQUATION

The total thrust of the rocket engine

F = momentum thrust + pressure thrust

$$\boxed{F = \dot{m} V_e + (P_e - P_a) A_e}$$

ROCKETS vs GAS TURBINES

BASIC DIFFERENCE ARE EVIDENT IN MANY AREAS

- Operating Conditions
- Environment
- Fluids
- Performance Criticality
- Failure Consequences
- Fabrication & Assembly
- Testing
- Maintenance

OPERATING CONDITIONS

	ROCKETS	PW F100–220
Turbine Inlet Temperature (°F)	–100 to 1500	2500° F
Inlet Temperature (°F)	–320°	600
Shaft Speed (RPM)	ATD 35000 Adv. Engine 100000	13400
Accel/Decell Time (10%–95%)	0.3/0.1 sec	4 sec
Time Between Overhaul (Hrs)	7.5 (ATD)	2000
Run/Mission Time (min.)	8	80
Lubrication	Dry Lube/propellants	Oil
Combustion Temp(°F)	6000	2700
Pressure Rise	6500 psi	580
Power Density (lb thrust / lbm)	70	7.4
H.P. Generated	76,000	29,000

ENVIRONMENT

OPERATING ENVIRONMENT VARIES WIDELY FOR ROCKET ENGINES

	ROCKETS	**GAS TURBINES**
Altitude	No Altitude Ceiling	75000 ft
Ambient Pressure	14.7 to Vacuum	14.7 to 0.5
Temperature (°F)	–300 to 150	–60 to 140
Purges	H_C, N_2	N/R
Desiccants	Dryness Imperative	N/R

FLUIDS

MANY PROPELLANT COMBINATIONS USED IN ROCKETS

ROCKETS		GAS TURBINES	
Oxidizer	**Fuel**	**Oxidizer**	**Fuel**
LOX	Hydrogen	Air	JP
N_2O_4	RP-1		
	Methane		
	Propane		
	N_2H_4 (Hydrazine)		
	UDMH		

SPECIAL CONSIDERATIONS

LOX Compatibility – Engine Material Ignition possible in presence of LOX

Hydrogen Embrittlement: Material Degradation due to H_2

Phase Change: large volume change for liquid → vapor transition and violent boiling.

Cooldown: Pumps and propellant ducts require conditioning.

Material Properties at Cryogenic Temperatures.

Difficult to Seal.

ENGINE PERFORMANCE

CHARACTERISTICS CRITICAL TO MISSION

- Limited fuel Capacity (No Refuel Capability)
- Payload Penalty for Excess Fuel Carried
- Cost Per Pound to Orbit Critical ($10,000 /lbm to GEO)
- O/F Match Critical to Avoid Depletion

SSME: 1 /sec Isp (≈ 0.22%) = 1100 lbs Delivered to LEO
= $2,200,000 ($2000 / lbm to LEO)

RL10: 1/sec ISP (≈ 0.23%) = 50 lbs Delivered LEO→GEO
= $500,000

FAILURES

IMPACT OF ROCKET FAILURES WIDE REACHING

ROCKETS	GAS TURBINES
• Few Vehicles – "National Asset"	Large Fleets
• "One of a kind" Payloads	Large Inventories
• Strategic Payload Deliveries	Alternate Weaponry
• Vehicle/Payload cost > $1B	Aircraft Cost (F–15) = $20M
• Loss of National Launch Capability	Multiple Weapon Delivery systems

Noteworthy:

Hubble Space Telescope: $1.4B Payload
$7M/Month Storage

Total Cost to Space Science Alone Due to Challenger Failure = $1.5B

FABRICATION & ASSEMBLY

ROCKET PROGRAMS DIFFICULT TO FIT IN P&W SYSTEMS

- Materials (Aluminum Housings, LOX Cleaning)
- Limited buys
- Only Production Work at GPD
- Most Components Built & Tested Here
- Controls Fabricated here
- Schedule Criticality
  - Test Stand Availability
  - Delays may cause mission slip greater than a year

TESTING

ROCKET TESTING PROVIDES CHALLENGES FOR PROGRAM PLANNING

	ROCKETS	GAS TURBINES
Test Stands Engines/Pumps	RL10: E–6 (only capability) in existence) ATD: E–8 (unique T/P Stand)	A–Area: 11 C–Area: 7 Others: E. Hartford, Gov't, etc.
Components	E Area: 7	D–Area: 55+
Run Duration	RL10: 650 sec ATD: 6–20 sec	No Limit
Propellant Cost	RL10: $12.00/sec SSME: $312/sec	$0.20 / sec
Steam Cost (RL10)	$2500/run	N/A
Stand Crew	12 / shift	5 / shift
Propellant handling	Cryogenics	Ambient

OTHER CONDITIONS

- Zero "G" Simulation
- No "Fleet Pacer"
- Altitude Simulation
- No idle Mode – No "Check Run"
- Rapid Transient – Aborts Required

MAINTENANCE

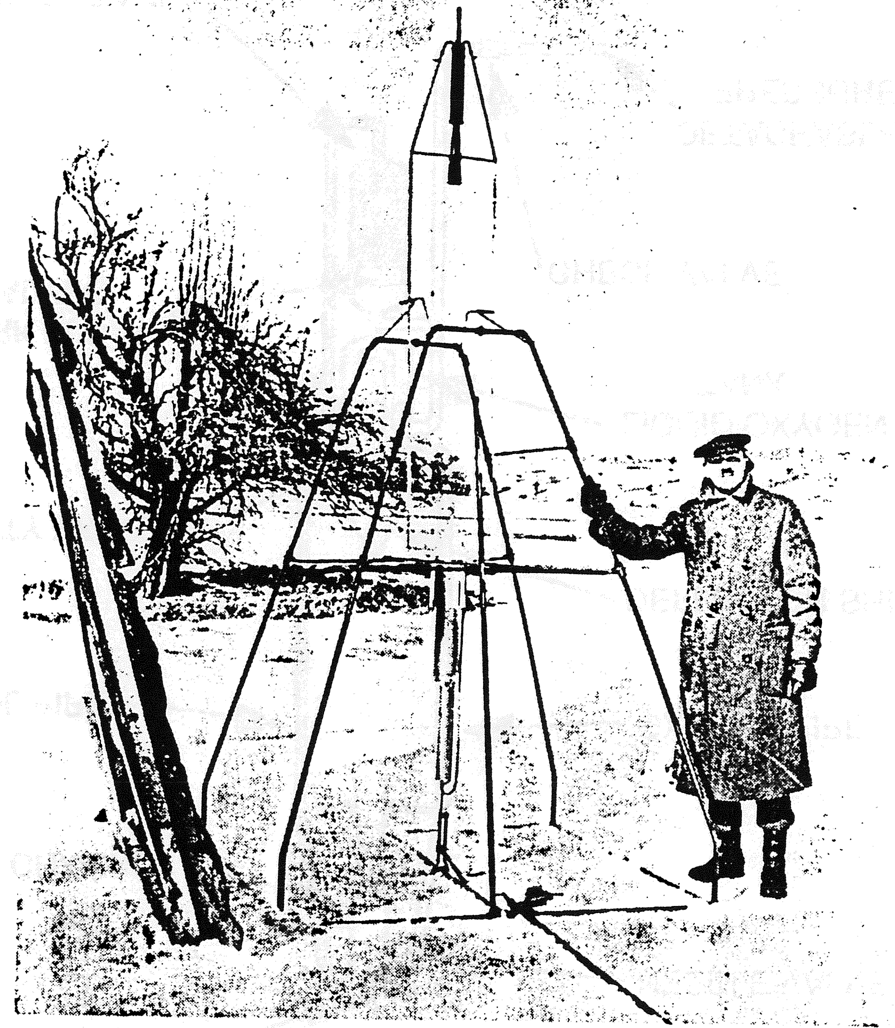
ROCKETS NOT ALWAYS EASILY MAINTAINED

	Rockets	**Gas Turbines**
GROUND BASED	Access Problems	Easy Access
	Engine Removal Difficult	Pull Engine
SPACE BASED	$50,000 +/hr	N/A
VERIFICATION	Not Always Possible	Start/Run Engine

HISTORY OF ROCKETS

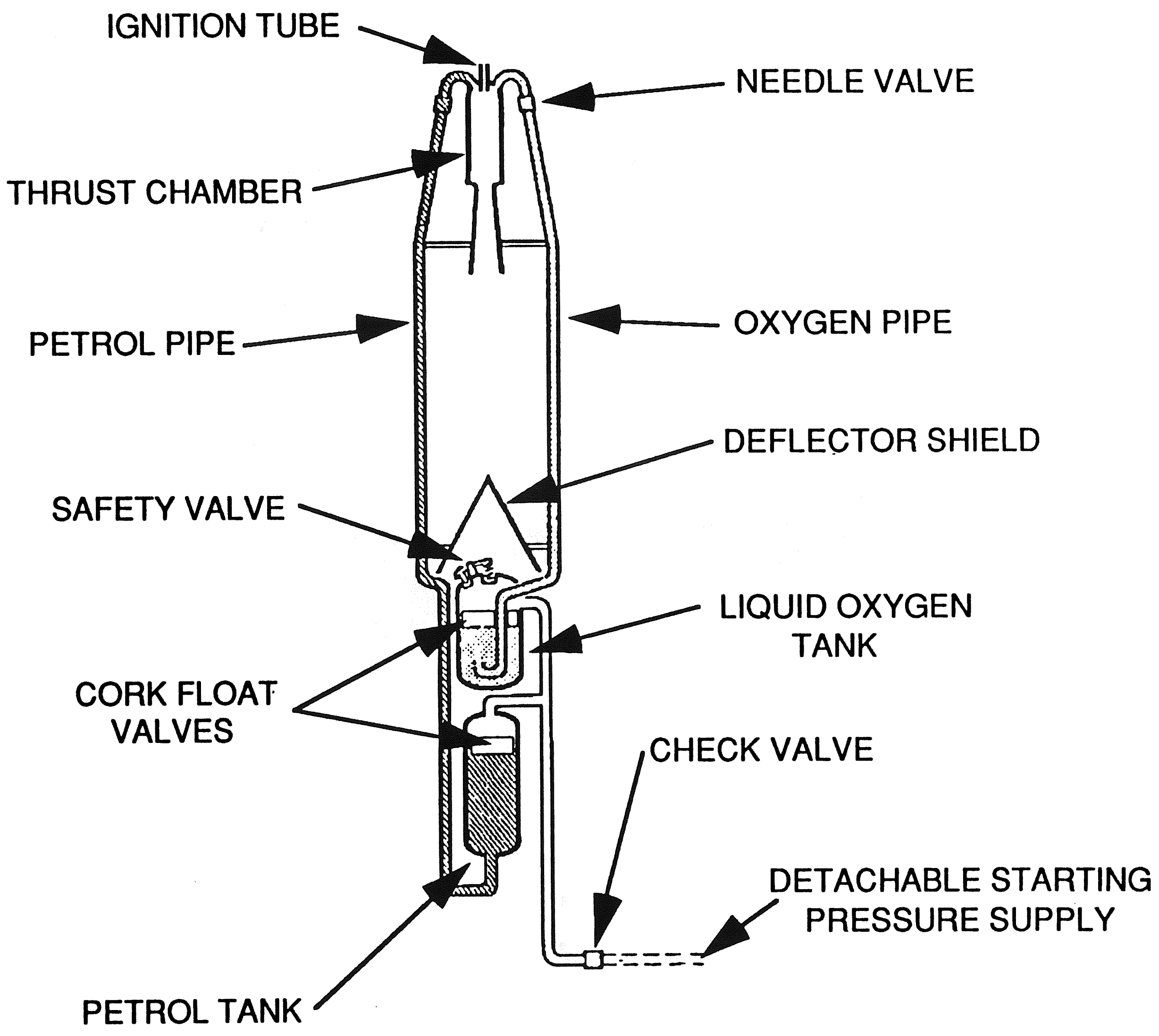
1232	• Chinese use "fire arrows" to defend city of Kai– feny– Fu against invading Mongols.
1895	• First application of liquid propellants to a rocket is claimed to have been made by P.E. Paulet in Peru.
1909	• Robert H. Goddard developed idea of multiple or step rockets and worked out a plan for using hydrogen and oxygen as fuels for interplanetary flight.
1920	• Smithsonian Institution publishes Goddard's paper "A Method of Reaching Extreme Altitudes", Newspapers have field day. • Goddard begins experimenting with liquid fuel rockets.
1923	• "The Rocket into Interplanetary Space" by Hermann Oberth was published – first mathematical considerations of rocket flight.
1926	• Worlds first liquid–propellant rocket is fired by Goddard on March 16, 1926, at Auburn, Massachusetts.

GODDARD AND HIS ROCKET



Before developing his relatively large 1941 rocket, Goddard experimented with many smaller models such as this liquid oxygen–gasoline burner which was fired on 16 March 1926 at Auburn, Massachusetts.

GODDARDS LIQUID PROPELLANT ROCKET - 1926



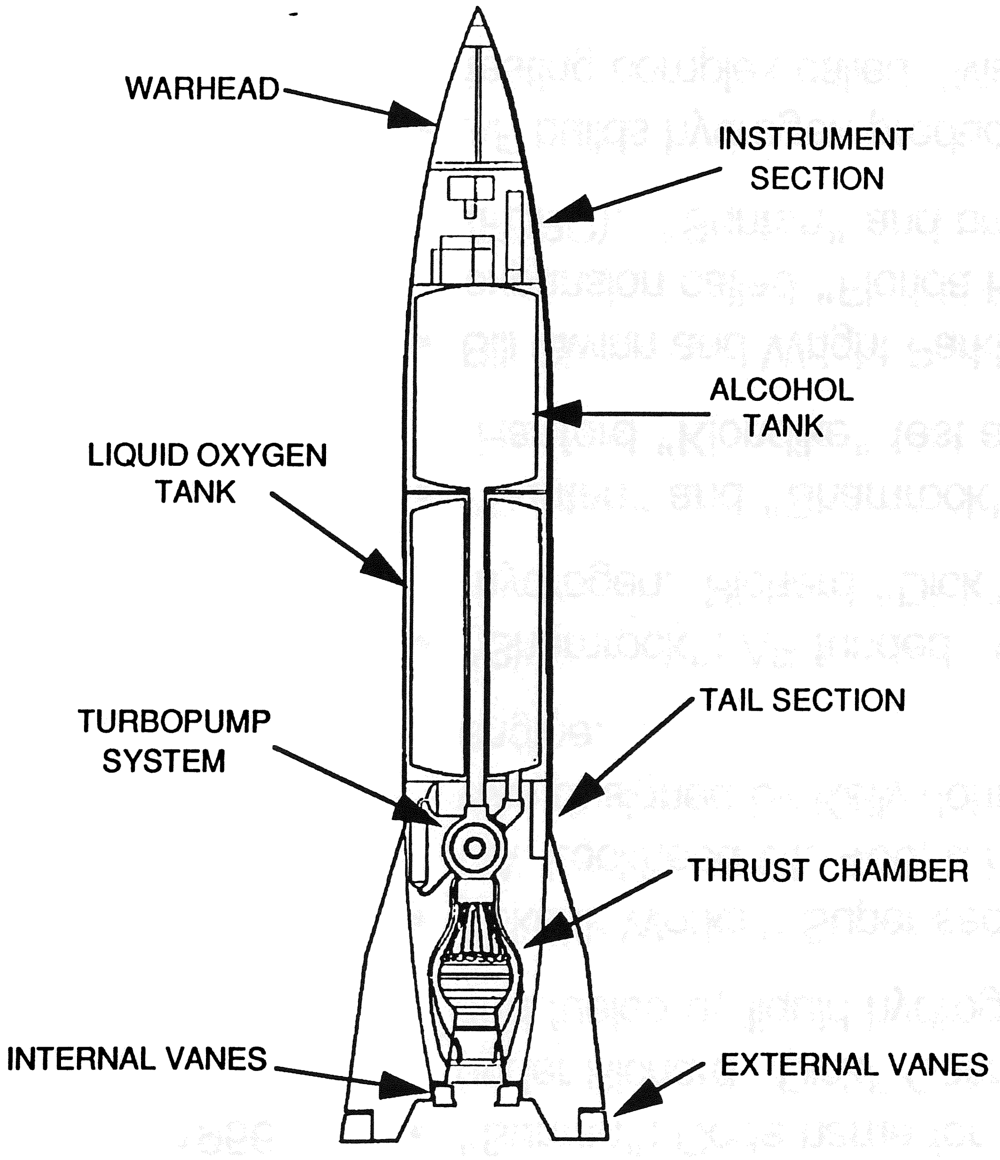
HISTORY OF ROCKETS

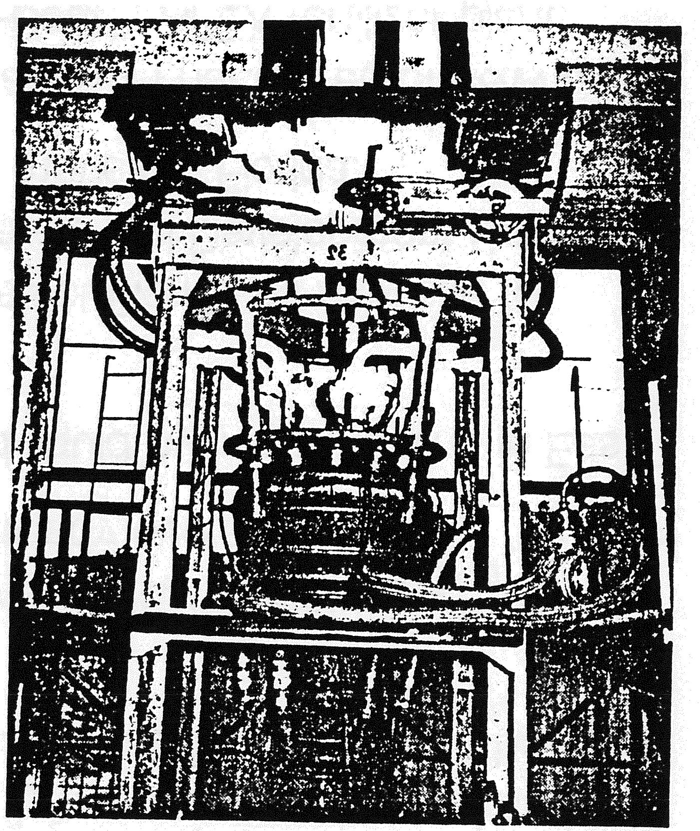
1929	• Goddard launchs fourth liquid–propellant rocket on July 17, 1929 which carried instruments in its nose. State Fire Marshall orders Goddard to discontinue his test in the State.
1930	• With the support of Charles P. Lindbergh and the backing of the Guggenheim Foundation, Goddards moves to Roswell, N.M. to continue his experiments.
1931	• Goddard successfully tests first remotely controlled rocket.
1932	• April 1932, Goddard developed and shot the first rocket controlled by a gyroscope.
1934	• Goddard launchs “Nell” – this rocket was steered in vertical flight by a gyroscope linked to four moving vanes set in the jet exhaust stream – a device later used by the Germans on their V–2’s.
1936	• Smithsonian Institution publishes Goddard’s work on March 16, 1936 under the title ”Liquid–Propellant Rocket Development”.

HISTORY OF ROCKETS

1937	• Germans establish huge liquid–propellant rocket research center at Peenemunde leading toward an operational V–2.
1940	• Goddard describes his experiment & presents possiblities of long–range liquid–propellant rockets for military use to the Military. • Army – “The next war, will be won with the trench mortar”. • Air Corp & Navy – See no possibilities in the rocket as a missle weapon but is interested in developing rocket–assisted take–off units for aircraft.
1941	• Goddard goes to work for U.S. Navy at Annapolis developing rocket–assisted take–off units until 1945.
1944	• Germans launch first V–2 rockets into London.
1945	• August 10, 1945 – Goddard dies at the age of 63. A total of 214 U.S. patents on rockets are credited to him.

GERMAN V-2 ROCKET WEAPON





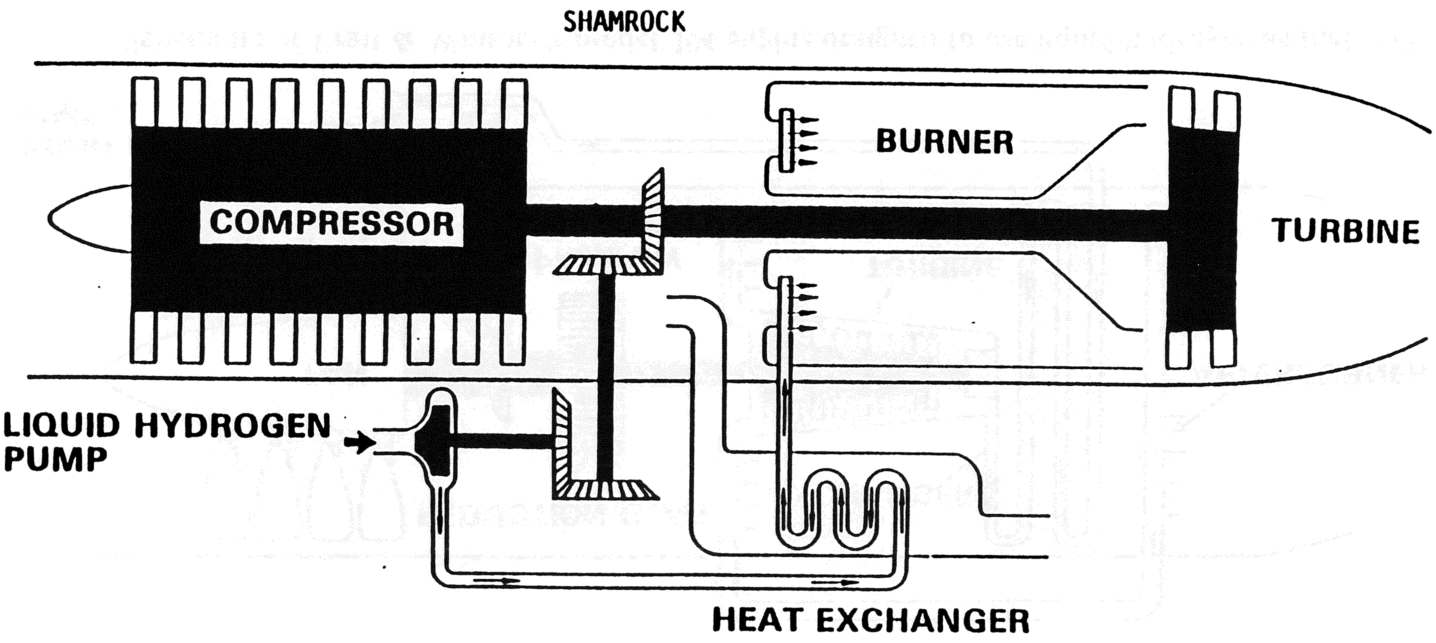
The V-2 rocket motor in a static test stand south of Saalefeld, near Lehesten, Germany.

HISTORY OF P&W ROCKET PROPULSION

1956

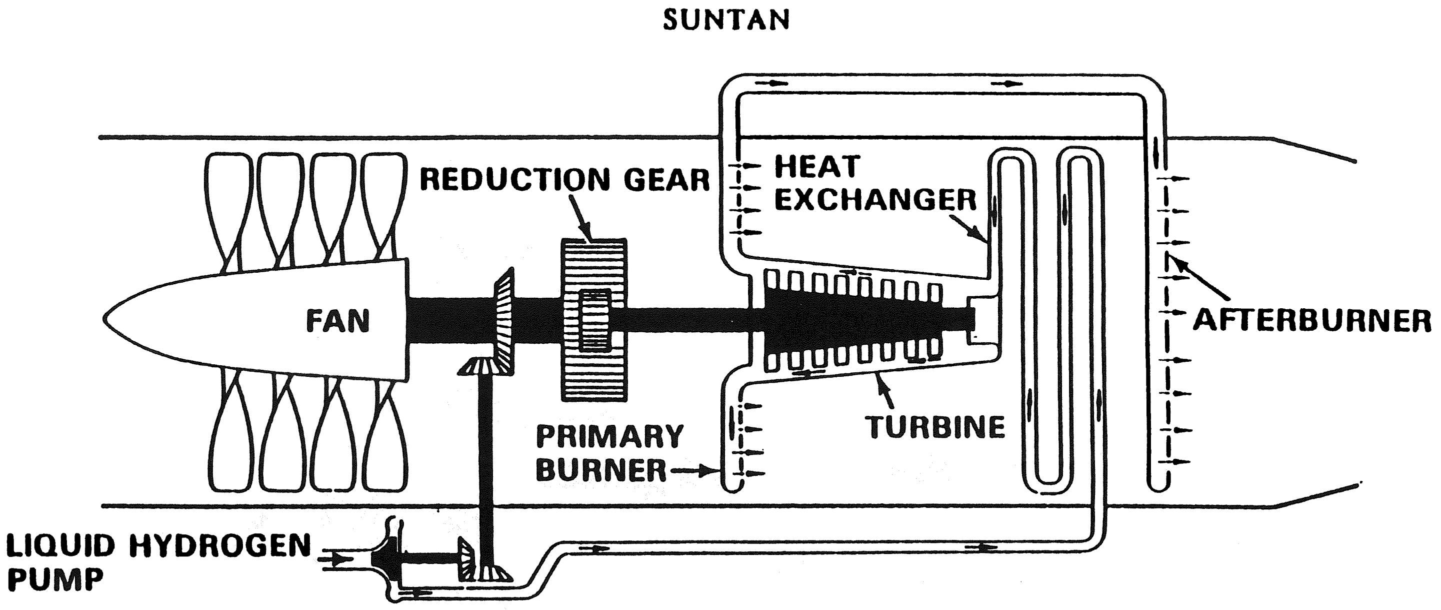
- "Suntan": Code name for a highly classified engine project under Richard "Dick" Coar. Designated 304. Built for AF and fueled by liquid hydrogen (SF–1)
- "Skunk Works": Super secret engineering center in Burbank, CA. Lockheed CL–400, a high–flying reconnaissance aircraft, was designed by Kelly Johnson. "Suntan" 304 was to be the engine.
- "Shamrock": AF funded. Converts P&W J57 to burn liquid hydrogen. Richard "Dick" Mulready overseer.
- "Suntan" and "Shamrock" had liquid hydrogen testing in East Hartford "Klondike" test area.
- Bill Gwinn and Wright Parkins select Florida as new site for expansion called "Florida Research and Development Center" (FDRC). "Suntan" and project moves to Florida.
- AF builds hydrogen producing facility in north area of P&W's testing complex called "Mama–Bear". APEX fertilizer plant.

P&W'S "SHAMROCK" (MODIFIED J-57)



Schematic of the Pratt & Whitney Aircraft J-57 jet engine modified to use liquid hydrogen as fuel,

P&W'S "SUNTAN" (304)



Schematic of Pratt & Whitney's model 304 engine designed to use liquid hydrogen as fuel, 1956.

P&W'S MODEL 304 ENGINE CHARACTERISTICS

	MODEL 304-1	TEST PERFORMANCE		MODEL 304-2	TEST PERFORMANCE
CHARACTERISTIC	SPEC A6600	ENG 1	ENG 2	SPEC A-6600A	
SEA-LEVEL STATIC THRUST NEWTONS (LBS)	55600 (12,500)	55422 (12,460)	53429 (12,012)	60048 (13,500)	35028 (7,875)
THRUST SPECIFIC FUEL CONSUMPTION, KG/N*HR	1.10	1.252	1.220	0.900	0.937
COMPRESSOR SPEED, RPM	3600	3630	3300	3600	2503
PUMP DISCHARGE PRESSURE, ATM	–	54	42	–	34
OVERAL TURBINE EFFICIENCY	–	–	0.475	–	0.507

NOTE: MODEL 304-1 HAD 4 COMPRESSOR STAGES

MODEL 304-2 HAD 5 COMPRESSOR STAGES

HISTORY OF P&W ROCKET PROPULSION (Continued)

1957

- Test Area B completed with large test stand for 304 engine and small component stands. 304 sucessfully started.

1958

- May 27, FRDC officially dedicated
- AF Cutbacks, SR–71 Selectd over CL–400. P&W participation in "Suntan" or 304 is dropped.
- AF still interested in boosting satellites.
- Perry Pratt in E. Hartford wins United Aircraft support in submitting proposal for the development of a hydrogen–fueled rocket engine.
- Contract award for development of a high–energy rocket engine fueled by liquid hydrogen from the Air Research & Development Command. Program designated XLR115. To be used in Centaur stage of Atlas booster.

1960

- NASA/MSFC assumes XLR115 program from AF, uses P&W designation RL10.
- NASA picks 6 RL10's to power Saturn IV stage.
- NASA awards Rocketdyne J–2 engine contract.

HISTORY OF P&W ROCKET PROPULSION (Continued)

	• Regenerative expander (Bootstrap) cycle unique (hydrogen pressure drives turbomachinery). Multiple start–stop cycles practical. • Attempt to test two engines in vehicle simulation explodes E–5 test stand. (Nov. 7, 1960) Operator error.
1961	• E–5 test stand explodes again (Jan 13, 1961), Ignition system modified to provide O_2 directly to the igniter. Automatic abort systems instituted.
1962	• RL10 Preliminary Flight Rating Test completed. • Initial launch of Atlas–Centaur failed due to Centaur stage structural failure (May 8, 1962).
1963	• First successful launch of Atlas–Centaur (AC–2) Nov. 27, 1963. • FRDC builds a Liquid Propellant Research Facility (LPRF). • From 1963 to 1987 the RL10 demonstrates 100% reliability in 282 in space firings on Atlas, Titan and Saturn launch vehicles.
1964	• NASA/MSFC awards 350K Fuel Pump Technology Program to P&W.

HISTORY OF P&W ROCKET PROPULSION (Continued)

	• First successful launch of the Saturn I (S–IV stage w/ RL10s) Jan. 29, 1964.
1965	• NASA/MSFC awards 350K Lox Pump Technology program to P&W.
1968	• AFRPL awards 250K LO_2 /LH_2 Demonstrator Engine Contract to P&W. Program is designated the XLR129–P–1.
1970	• SSME RFP – $6 million contract for 11 month study & proposal.
1971	• SSME Contract awarded to Rocketdyne. • P&W protests NASA decision in vain.
1974	• First successful launch of Titan–Centaur (TC–1) Feb. 11, 1974 Viking Landers, Helios and Voyager Spacecraft are launched on Titan–Centaur Vehicle between 1974 and 1977.
1980	• RL10 Program finished. (Intelsat selects Atlas–Centaur as the launch vehicle for the Intelsat V payload –RL10 Program rejuvenated).

HISTORY OF P&W ROCKET PROPULSION (Continued)

1981	• RL10A–3–3A Qualification completed (Dec. '81). • PW receives RL10 order for Shuttle–Centaur Vehicles.
1983	• NASA requests P&W do study SSME design & recommend improvements.
1984	• NASA issues Study Contract on Space Shuttle Main Engine.
1986	• MSFC awards SSME ATD Contract to P&W. Value $198 Million. • NASA cancels Shuttle–Centaur Program. • P&W receives RL10 order for the Titan IV Program. • AFRPL awards Lox/HC Acoustic Liner Technology Program to P&W. • NASA/MSFC awards STME/STBE Study Contracts to P&W.
1987	• AFRPL awards Tripropellant Thrust Chamber Technology Program to P&W.
1988	• MSFC awards STBE Injector Technology Demonstrator Program to P&W.

HISTORY OF P&W ROCKET PROPULSION (Continued)

1989
- MSFC awards ALS Oxidizer Turbopump Low Cost Demonstrator Program to P&W.
- MSFC awards Space Transportation Engine (STE) Phase B Study to P&W.
- AFAL awards Rocket Engine Condition Monitoring System (RECMS) to P&W.
- NASA/LeRC awards Advanced Expander Test Bed (AETB) to P&W.

1990
- P&W, Rocketdyne, and Aerojet sign Memorandum of Understanding to form STE partnership.
- MSFC begins negotiations for Split Expander Thrust Chamber Assembly Demonstrator.
- P&W begins testing of ATD LOX and Fuel Turbopumps at E-8 stand.

ROCKET ENGINE FUNDAMENTALS

SESSION 2
PART 1

APRIL 8, 1991

JAY R. COBIA/
VINCE GARCIA

OBJECTIVES

1. STUDENTS WILL LEARN ABOUT THE VARIOUS ASPECTS OF PAYLOAD DELIVERY AND BE ABLE TO UNDERSTAND THE REQUIREMENTS FOR VEHICLES WHICH USE P&W PROPULSION
2. STUDENTS WILL LEARN ABOUT THE DIFFERENT ON-GOING ROCKET PROGRAMS AT P&W AND BE ABLE TO UNDERSTAND THE GOAL OF EACH
3. STUDENTS WILL LEARN ABOUT FUTURE PROPULSION SYSTEMS TO BE ABLE TO UNDERSTAND THE POTENTIAL FOR PROPULSION SYSTEM APPLICATIONS
4. STUDENTS WILL LEARN ABOUT UTC INVOLVEMENT IN SPACE AND BE ABLE TO UNDERSTAND WHERE THE CORPORATION IS HEADING IN ROCKET PROPULSION AND RELATED AREAS

SPACE TRANSPORTATION ARCHITECTURE STUDY (STAS)

- JOINT DEPARTMENT OF DEFENSE & NASA STUDY

- PURPOSE:
  - STUDY THE DEVELOPMENT OF A SECOND GENERATION SPACE TRANSPORTATION SYSTEM MAKING USE OF MANNED AND UNMANNED SYSTEMS TO MEET THE REQUIREMENTS OF ALL USERS
  - REVITALIZE THE NATIONS LAUNCH AND LOGISTICS/SUPPORT TECHNOLOGY BASE
  - SATISFY THE FUTURE NEEDS OF USERS
  - SUBSTANTIALLY REDUCE THE COSTS OF SPACE OPERATIONS
  - DEVELOP A FLEXIBLE AND ROBUST SPACE TRANSPORTATION SYSTEM
  - MAINTAIN WORLD LEADERSHIP IN SPACE TRANSPORTATION

- P&W PARTICIPATED AS SUBCONTRACTOR

STAS MISSION REQUIREMENT

REQUIREMENTS	DOD	CIVIL
o ORBIT INCLINATION	MOSTLY POLAR	MOSTLY LOW
o LUNAR OR PLANETARY MISSIONS	NONE	YES
o LARGE CONSTELLATIONS	YES	NO
o MICROGRAVITY	NONE	SIGNIFICANT
o SPACE SERVICES	TBD	YES
o PAYLOAD/USER DIVERSITY	LOW	HIGH
o INTERNATIONAL INVOLVEMENT	NONE	SIGNIFICANT
o CIVIL & COMMERCIAL INVOLVEMENT	NONE	SIGNIFICANT
o MANNED PRESENCE IN SPACE	TBD	SIGNIFICANT
o SECURITY	VERY HIGH	LOW
o RELIABILITY	HIGH	VERY HIGH

STAS ISSUES

MANY ISSUES BEING CONSIDERED TO MEET THE MISSION MODEL

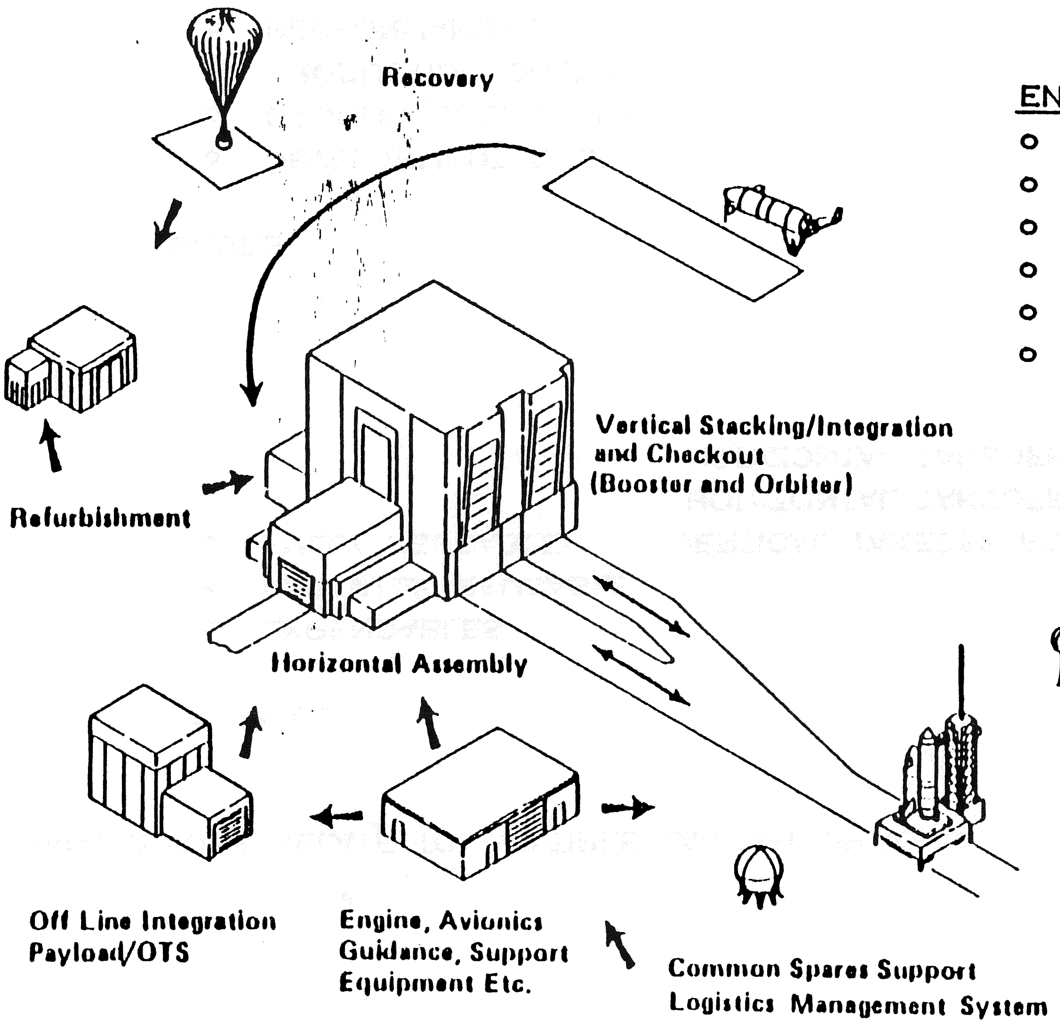
- MANNED VS. UNMANNED
- EXPENDABLE VS. REUSABLE (FULL OR PARTIAL)
- GROUND BASED VS. SPACE BASED ORBIT TRANSFER VEHICLES
- TECHNOLOGY REQUIREMENTS
- SYSTEM FLEXIBILITY
- RELIABILITY
- LIFE CYCLE COST
- MANIFESTING
- MODULARIZATION

STAS ELEMENTS

ARCHITECTURE IDENTIFIES AREAS OF OPERATION

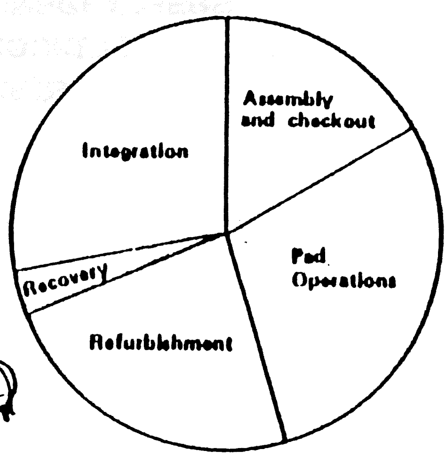
- GROUND OPERATIONS
- BOOST & RECOVERY OPERATIONS
- INTRA ORBIT OPERATIONS
- INTER ORBIT OPERATIONS

GROUND OPERATIONS CONCEPTS



ENGINE ISSUES

- ENGINE LIFE
- DIAGNOSTIC CAPABILITY
- EASE OF MAINTENANCE
- MINIMAL PRE–FLIGHT CHECKOUT
- UTILIZE EXISTING FACILITIES
- COMMON EQUIPMENT



Partially Reusable

STAS ELEMENTS

BOOST & RECOVERY

MANY ISSUES RELATE TO MULTIPLE LAUNCH OPTIONS

OPTIONS

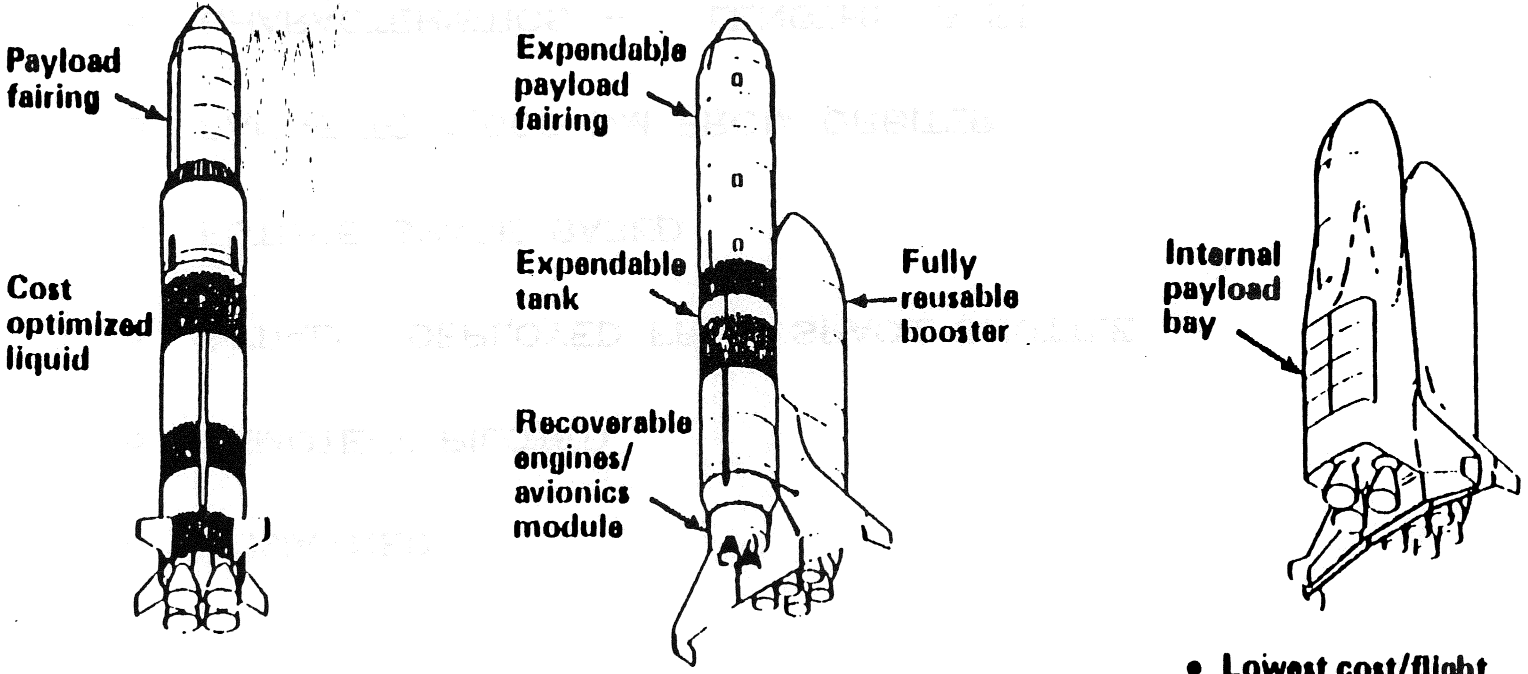
- EXPENDABLES
- PARTIALLY REUSABLE
- FULLY REUSABLE: VERTICAL TAKEOFF ROCKETS
  HORIZONTAL TAKEOFF ROCKETS
  HORIZONTAL TAKEOFF AIRBREATHERS

ISSUES

- DEVELOPMENT COST
- GROUND OPERATIONS
- PRODUCTION COSTS
- PERFORMANCE
- COST/FLIGHT
- CARGO RETURN

REUSABILITY OPTIONS

NO CLEAR–CUT WINNER MEANS MIXED OR GROWTH SYSTEMS



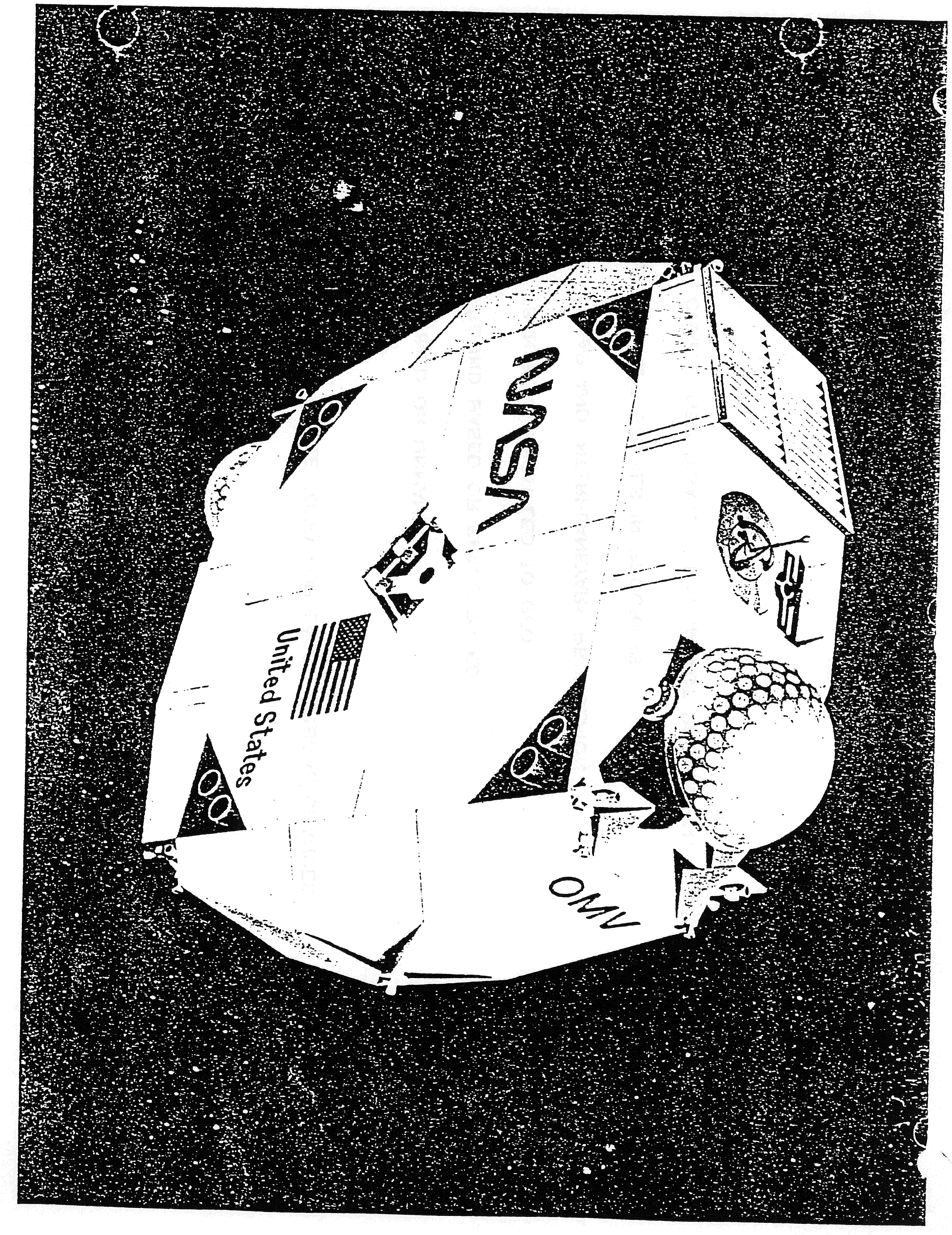
	Expendable	Partially Reusable	Fully Reusable
PRO's	• Lowest DDT&E • Payload integration • Simpler ops	• Highest performance • Increased robustness • Payload integration	• Lowest cost/flight • Abort capability • Return cargo • Increased robustness
CON's	• Production costs • No abort	• Production costs of expendable hdwe • No abort	• High DDT&E • Lower performance • Payload bay limitations

STAS ELEMENTS

INTRA ORBIT OPERATIONS

ORBITAL MANEUVERING VEHICLE (OMV) FOR SMALL ORBITAL CHANGES

- UNMANNED
- REMOTELY PILOTED
- INITIALLY DEPLOYED FROM SPACE SHUTTLE
- FUTURE SPACE BASED
- RANGE TO 1,000 NM FROM ORBITER
- CHARACTERISTICS — LENGTH: 3 FT
  DIAMETER: 15 FT
  WEIGHT: 11,000 LB FULLY FUELED
  MONOPROPELLANT OR BIPROPELLANT


NASA
United States
OMV

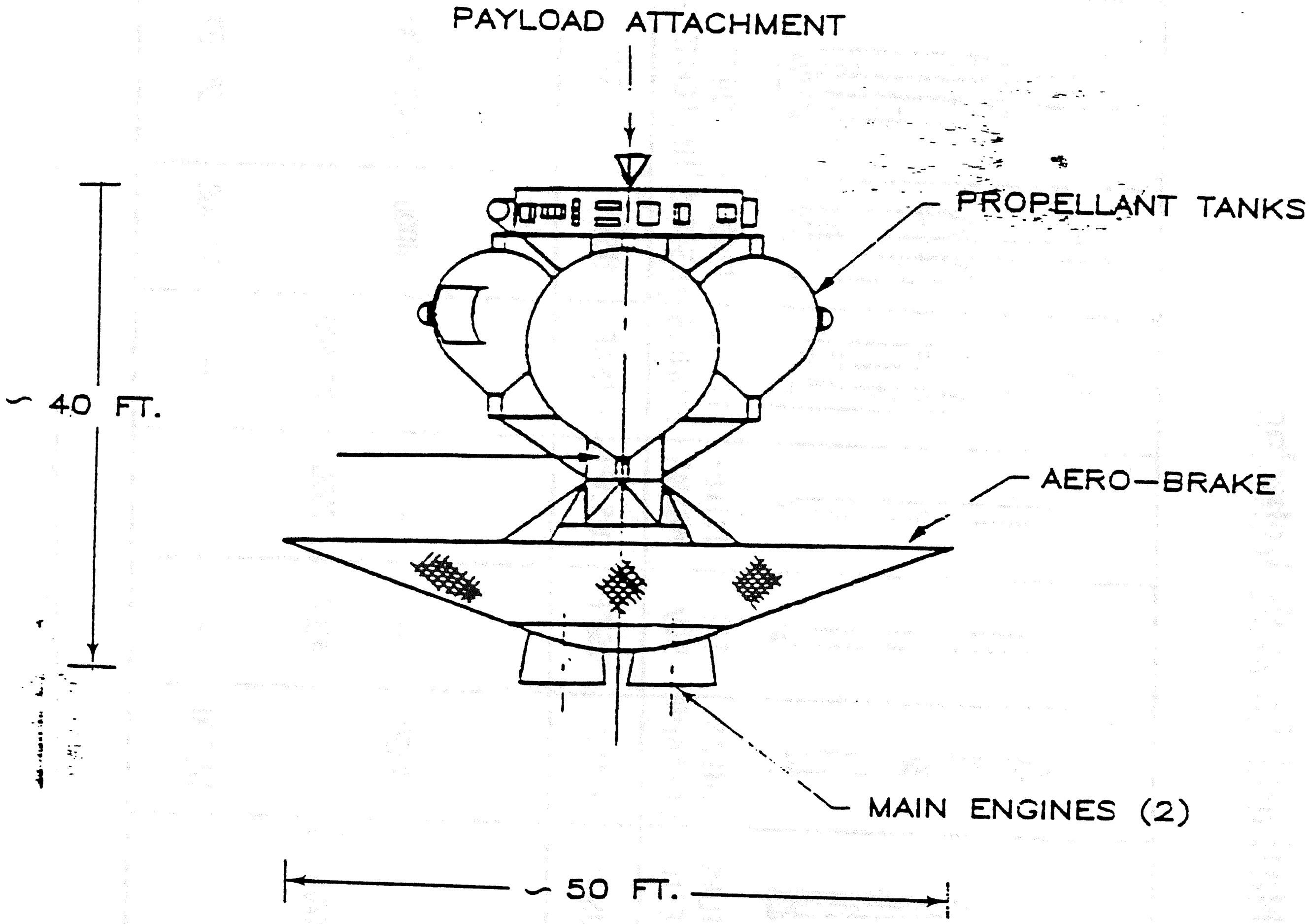
STAS ELEMENTS

INTER ORBIT OPERATIONS

ORBITAL TRANSFER VEHICLE (OTV) FOR LARGE ORBITAL CHANGES

- MANNED OR UNMANNED
- GROUND BASED OR SPACE BASED
- TRANSFERS FROM LEO TO GEO
- LUNAR AND INTERPLANETARY PROPULSION
- CONCEPT STUDIES IN PROGRESS
- CHALLENGER DISASTER MAY IMPACT PROPELLANT SELECTION

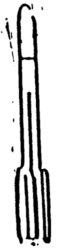
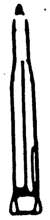
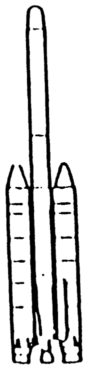
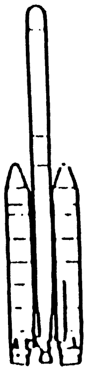
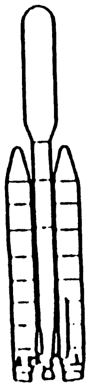
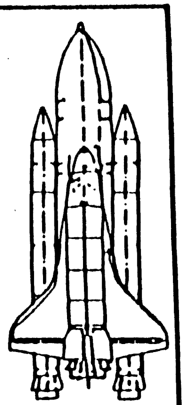
SPACE BASED ORBIT TRANSFER VEHICLE



(TYPICAL CONFIGURATION)

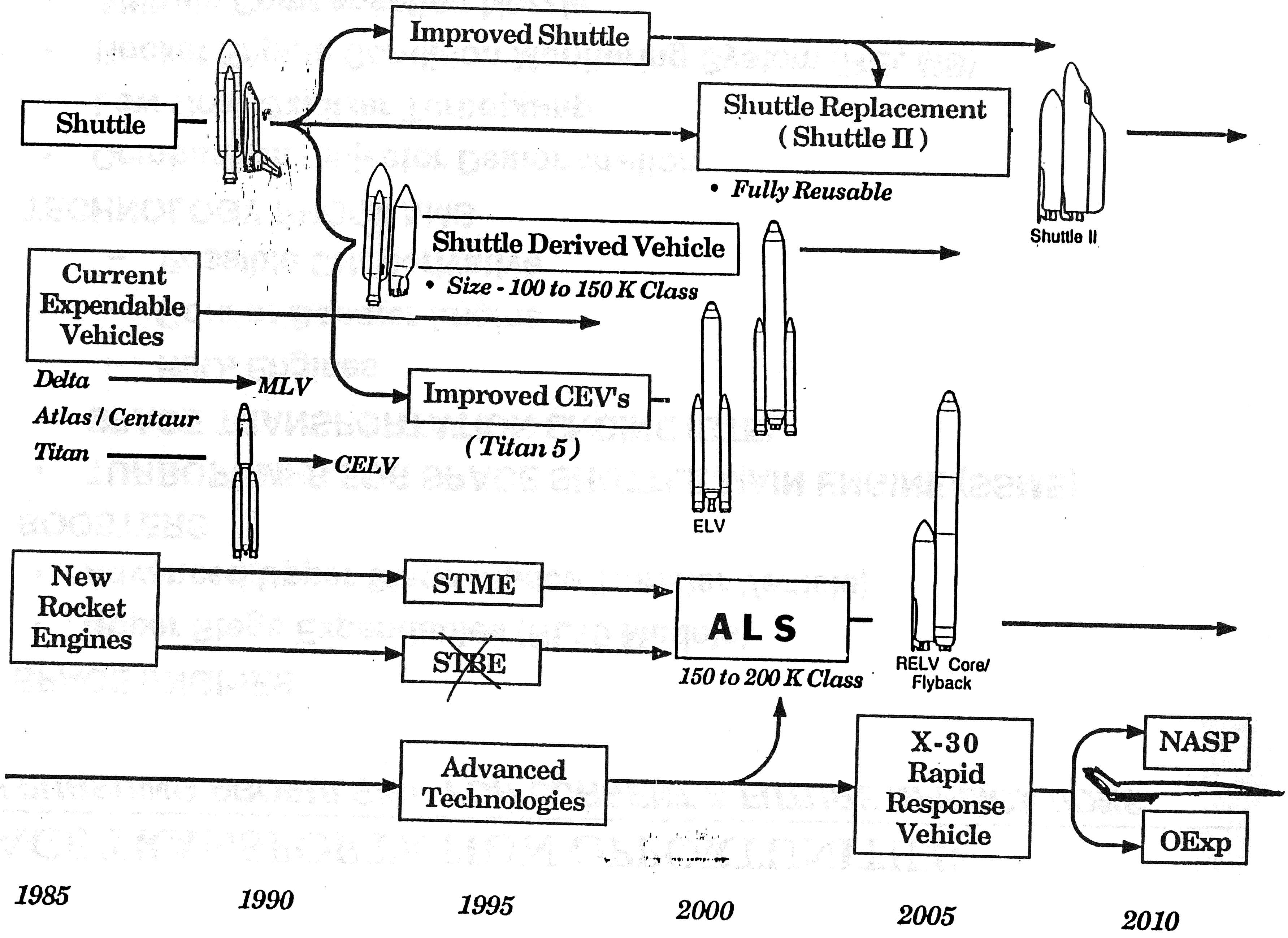
PAYLOAD WEIGHT: 20,000 LB. CLASS TO GEOSYNCHRONOUS ORBIT
PAYLOAD SIZE: UNCONSTRAINED

National Launch Vehicles

	 SCOUT	 DELTA	 ATLAS- E/H	 ATLAS CENTAUR	 TII SLV	 TIIIB- AGENA	 TITAN 34D/RGS	 T34D IUS/T-S	 T34D7 IUS/CENT G'	 STS/IUS/
RESPONSIBLE AGENCY	NASA	NASA	USAF	NASA	USAF	USAF	USAF	USAF	USAF	NASA/USAF
PERFORMANCE CAPABILITY: (lb)										
SYNCH EQ	—	1450		2650	—	—	—	4000	5000/10000	5000/10,000
100 nmi POLAR	460	5500*	3800/5500*		4200	7950	27,600	—		32,000
100 nmi DUE EAST	570	7800		12,300	—	—	—	32,900	39,100	65,000

* AKM REQUIRED

Space Transportation Trends and Development Roadmap



Shuttle

Improved Shuttle

Shuttle Replacement
(Shuttle II)

- *Fully Reusable*

Shuttle Derived Vehicle

- *Size - 100 to 150 K Class*

Current
Expendable
Vehicles

Delta → *MLV*

Atlas / Centaur

Titan → *CELV*

Improved CEV's

(Titan 5)

New
Rocket
Engines

STME

~~STBE~~

ALS

150 to 200 K Class

Advanced
Technologies

X-30
Rapid
Response
Vehicle

NASP

OExp

SPACE TRANSPORTATION OPPORTUNITIES

P&W PURSUING PROPULSION FOR CURRENT & FUTURE APPLICATIONS

¤ SPACE ENGINES
- Upper Stage Expendables (RL10 Models)
- Advanced Upper Stage (Space Transfer Vehicle)

¤ BOOSTERS
- TURBOPUMPS FOR SPACE SHUTTLE MAIN ENGINE (SSME)
- SPACE TRANSPORTATION ENGINE (STE)
  - H_2/O_2 Engines
  - Core or Booster Engine
  - Possible CH_4 Derivative

¤ TECHNOLOGY PROGRAMS
- Combustion - Injector Demonstration
- Low Cost Oxidizer Turbopump
- Rocket Engine Condition Monitoring System (RECMS)
- Altitude Compensating Nozzle
- Split Expander Thrust Chamber Demonstrator

RL10

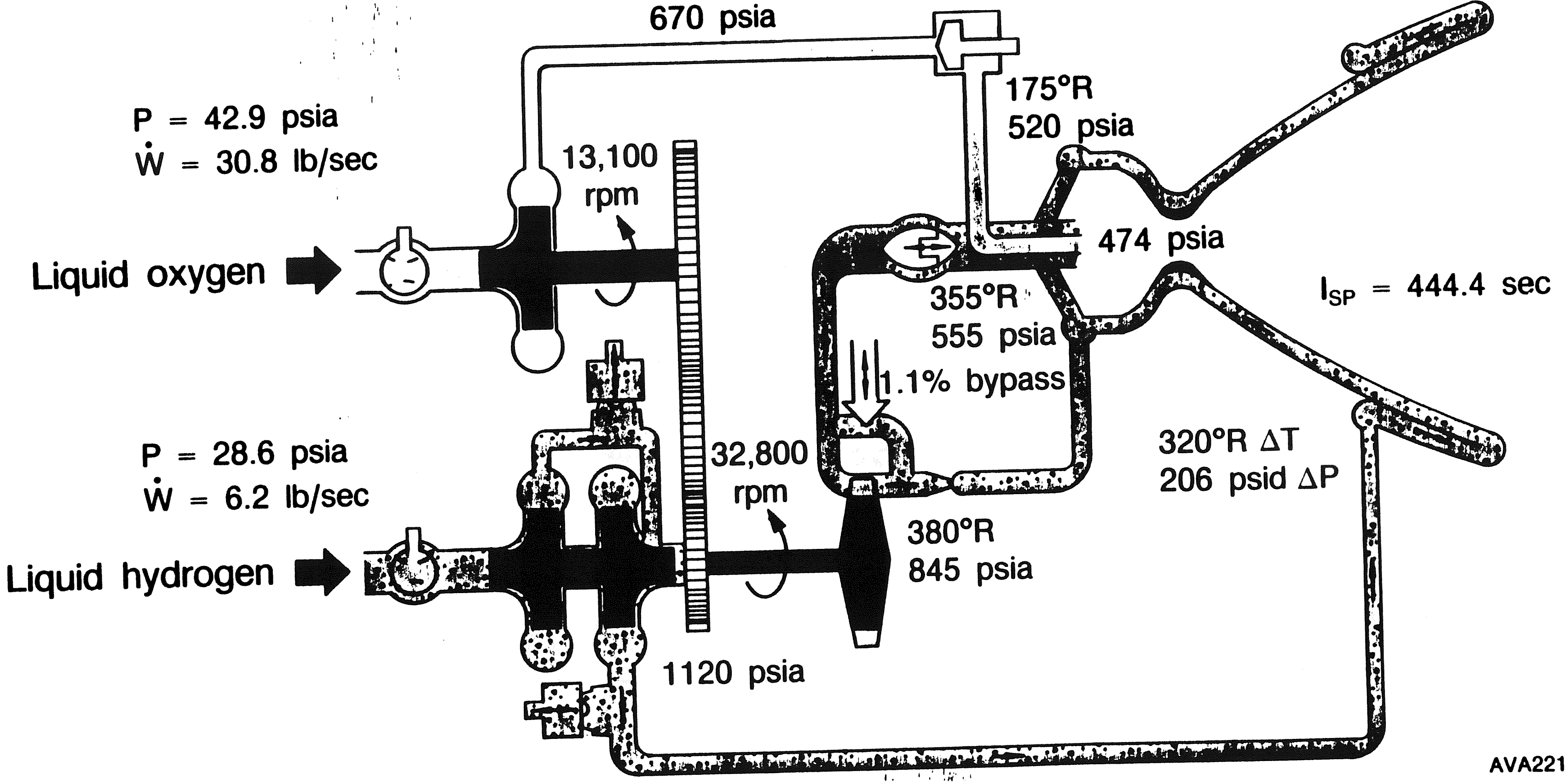
Dependable Propulsion Continues to Provide Opportunities

- TITAN IV - PRODUCTION IN PROGRESS
- ATLAS I/II - PRODUCTION STARTED
- ATLAS IIA - RL10A-4 UPGRADE DEVELOPMENT ACTIVITIES UNDERWAY
- NEW APPLICATION STUDIES
  - Foreign - possible applications (e.g.; Ariane)
  - ELV Stages - possible new/replacement upper stages (Delta, Titan IV)

Engineering Manager: Bob Foust

RL10A-3-3A PROPELLANT FLOW SCHEMATIC

16.5K thrust, 5.0:1 MR



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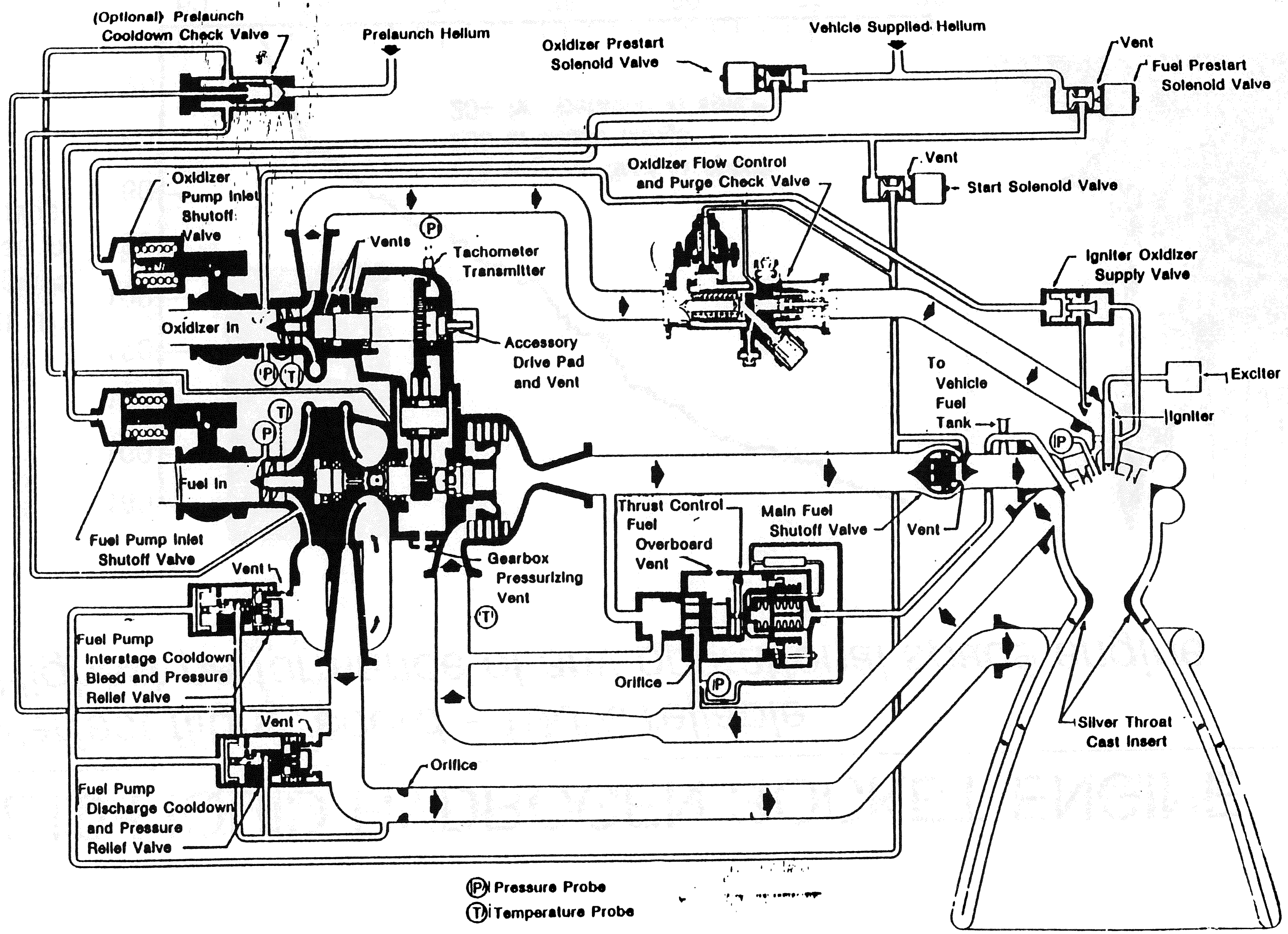
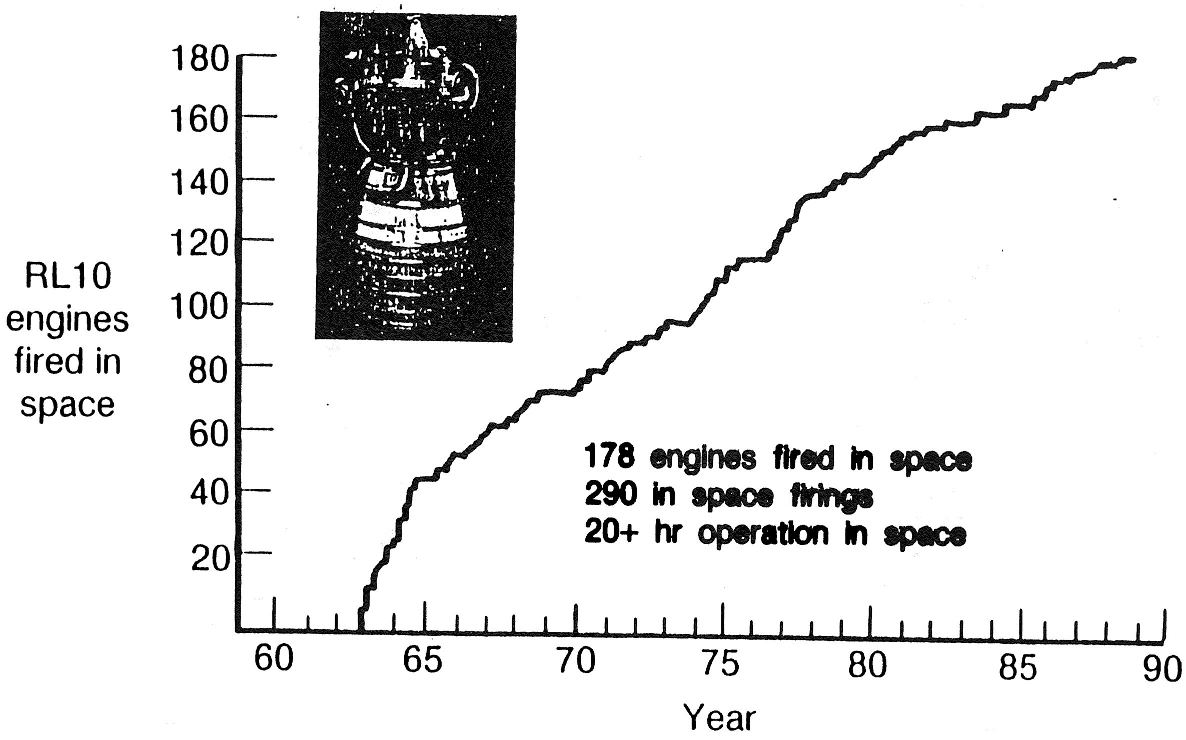
RL10 PROPELLANT FLOW SCHEMATIC

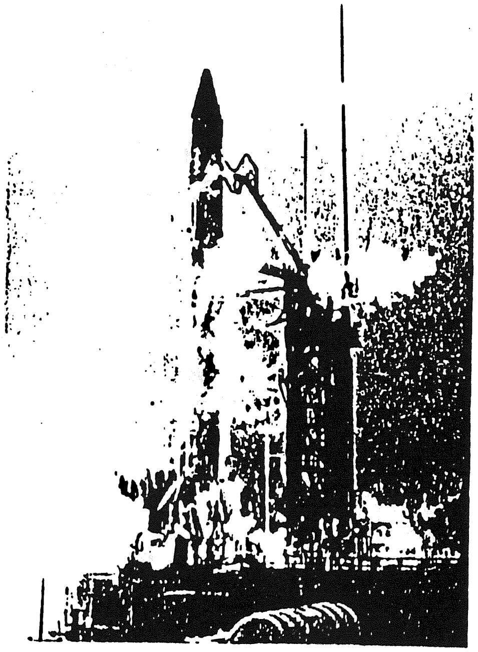


RL10 LIQUID HYDROGEN ROCKET ENGINE

- *Perfect flight record - 100% reliable*
- *Highest performance of any operational space engine*



178 engines fired in space
290 in space firings
20+ hr operation in space

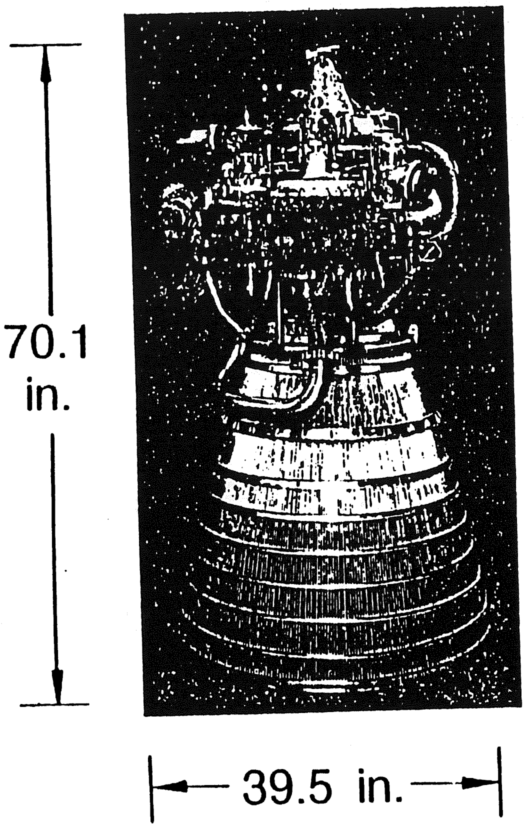


RL10 EVOLUTION

GROWTH THROUGH THE YEARS

MODEL NO.	A-1	A-3	A-3-1	A-3-3	A-3-3A	A-4
VAC. THRUST, LBS	15,000	15,000	15,000	15,000	16,500	20,250
CHAMBER PRESSURE, PSIA	300	300	300	395	475	565
THRUST/WEIGHT	50	50	50	50	54	54
EXPANSION RATIO	40:1	40:1	40:1	57:1	61:1	84:1
ISP, SEC AT 5.0 O/F (5.5)	424	429	433	442.4	444.4	(449.0)
FLIGHT CERTIFICATION DATE	11/61	6/62	9/64	10/66	11/81	~2/91

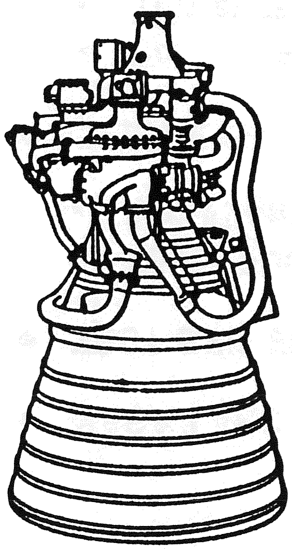
RL10A-3-3A ENGINE



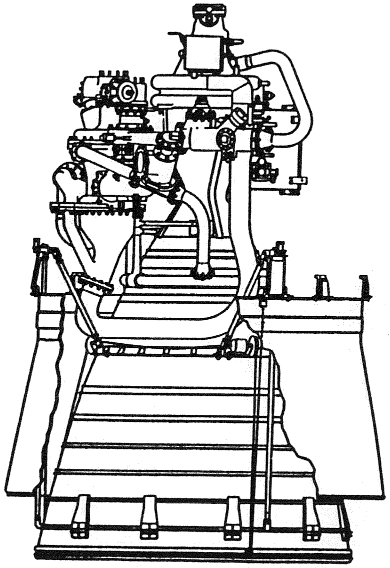
Vacuum thrust, lb	16,500
Specific impulse, sec	444.4
Weight, lb	305
Mixture ratio	5:1
Chamber pressure, psia	475
Vehicle applications	Saturn IV Atlas Centaur Titan Centaur

COMPARISONS

RL10 A-3-3A VS. RL10 A-4



RL10A-3-3A		RL10A-4
16,500	THRUST, LB	20,800
442.5	SPECIFIC IMPULSE SEC @ 5.5:1 O/F	449.5
70	LENGTH, IN	70/90
40	DIAMETER, IN	46
305	WEIGHT, LBS	365
5.0:1/5.5:1	MIXTURE RATIO	5.5:1
4.4:1 - 5.8:1	MIXTURE RATIO - RANGE	4.9:1 - 5.8:1



ADVANCED EXPANDER TEST BED - AETB

Propulsion For Space Transfer Vehicle

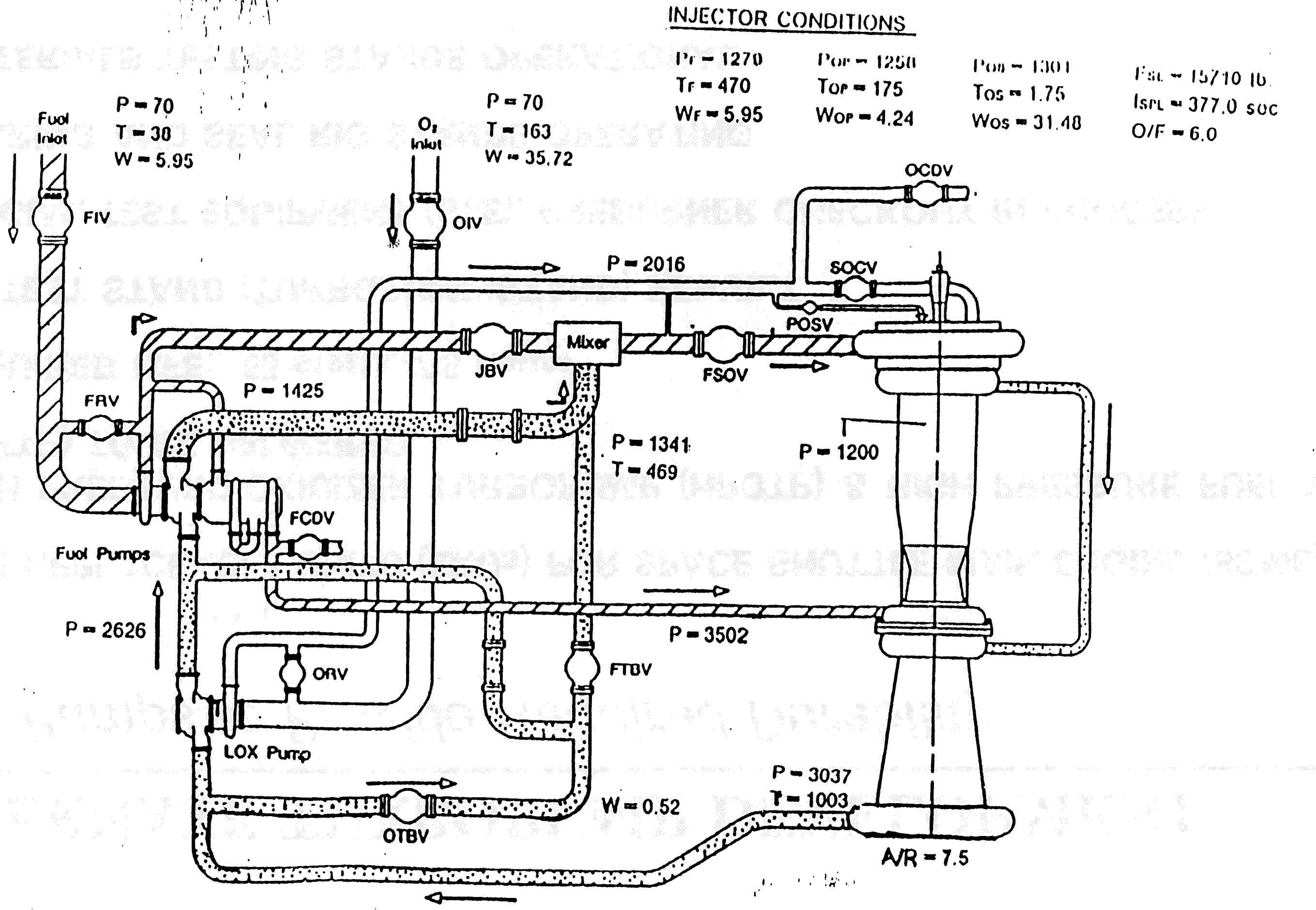
- **NASA LeRC PROGRAM**
- **TEST BED ENGINE**
  - Flight Functional
  - Not Flight Weight
- **SPLIT EXPANDER CYCLE**
  - Pc = 1200
  - Thrust = 20,000
- **PROPOSED PROGRAM:**
  - $30 Million
  - 5 Years
    - 3 years to deliver test article
    - 2 years to support testing at LeRC
  - Target Start - April 1990

Engineering Manager: Art Masters

UNITED TECHNOLOGIES PRATT&WHITNEY

AETB OPERATING CONDITIONS

NORMAL OPERATING POINT



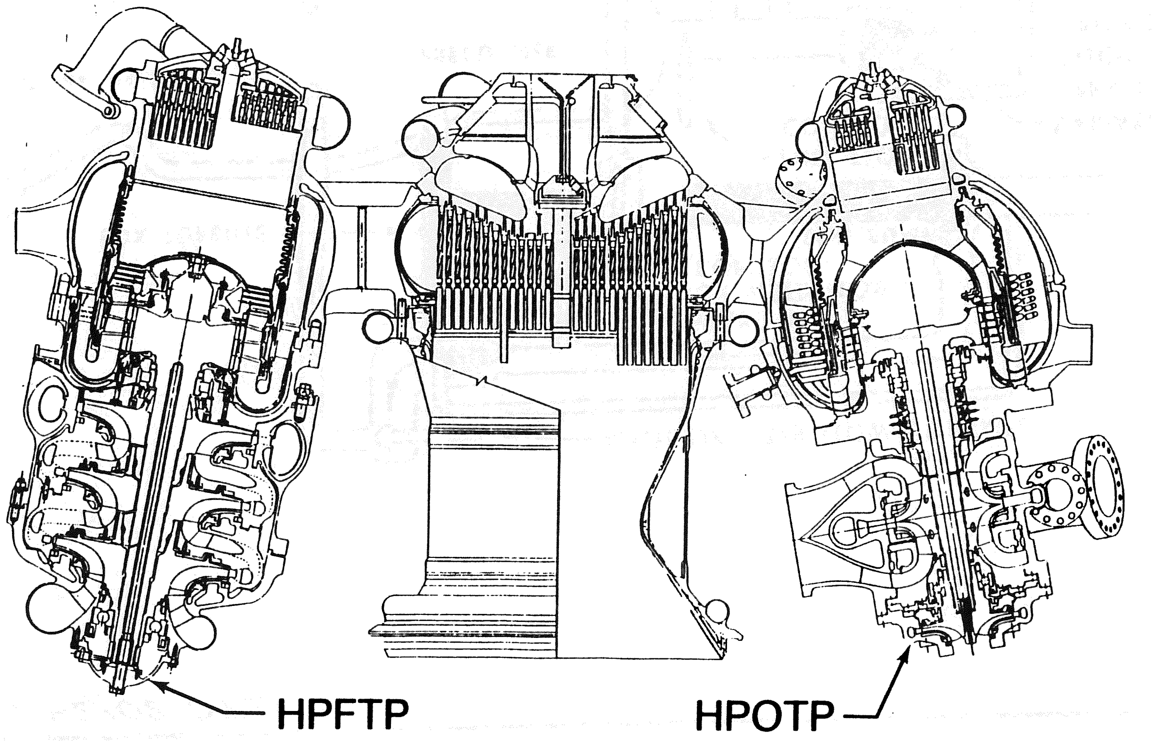
ALTERNATE TURBOPUMP DEVELOPMENT

P&W Pumps to Provide Required Durability

- **LINE REPLACEABLE UNITS (LRUs) FOR SPACE SHUTTLE MAIN ENGINE (SSME)**
- **HIGH PRESSURE OXIDIZER TURBOPUMP (HPOTP) & HIGH PRESSURE FUEL TURBOPUMP (HPFTP) TO BE DELIVERED.**
- **REQUIRED LIFE: 55 starts, 7.5 hours**
- **E-8 TEST STAND (TURBOPUMP STAND) REBUILT**
- **SPECIAL TEST EQUIPMENT (STE), PREBURNER CHECKOUT IN PROCESS**
- **BEARING AND SEAL RIG STANDS OPERATING**
- **MATERIALS TESTING STANDS OPERATIONAL**
- **OTHER SSME COMPONENT OPPORTUNITIES:**
  - **Versatile Powerhead**
  - **Controller**
  - **Thrust Chamber**

Engineering Manager: Hal Gibson

ALTERNATE TURBOPUMP DEVELOPMENT PROGRAM



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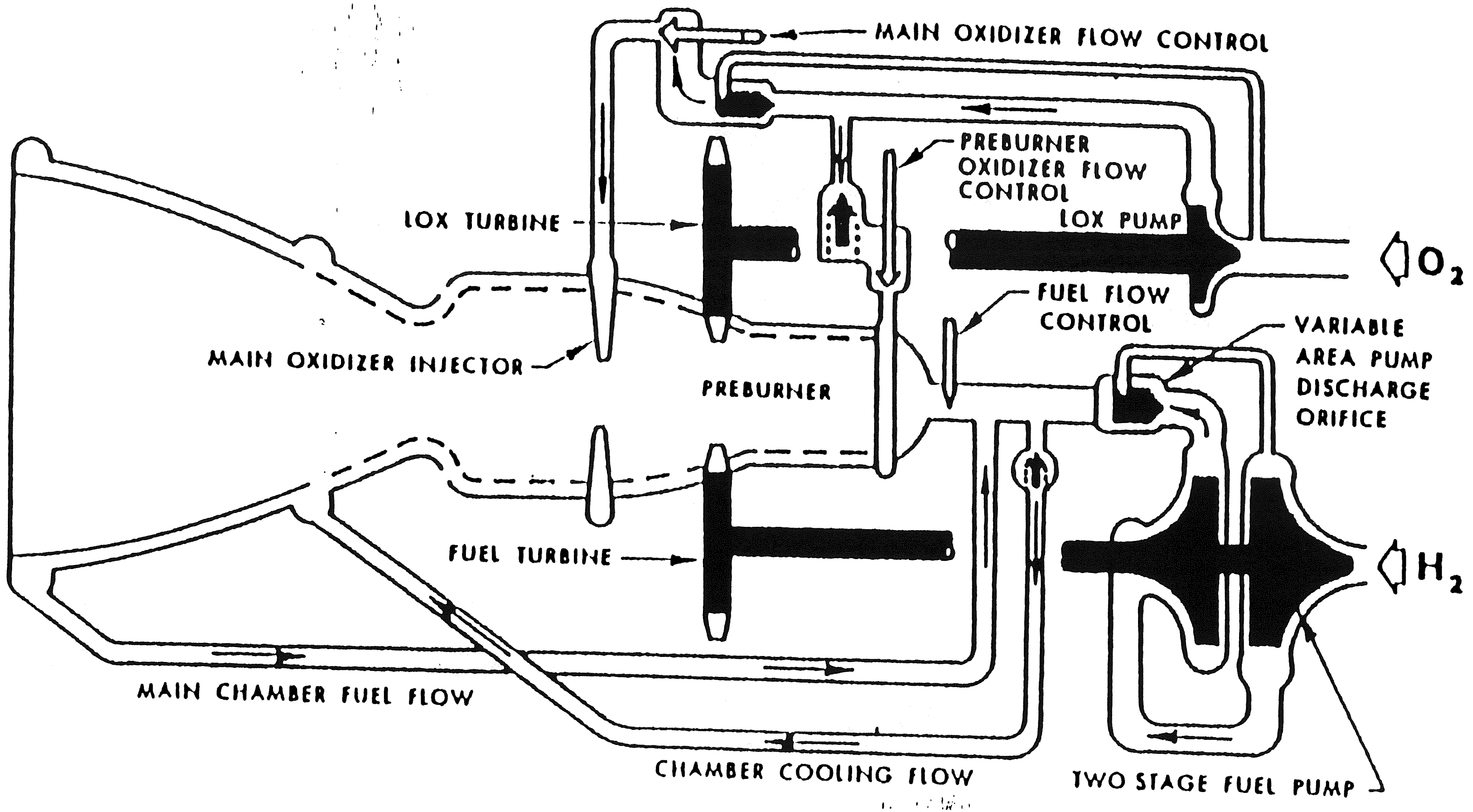
PREBURNER CYCLE

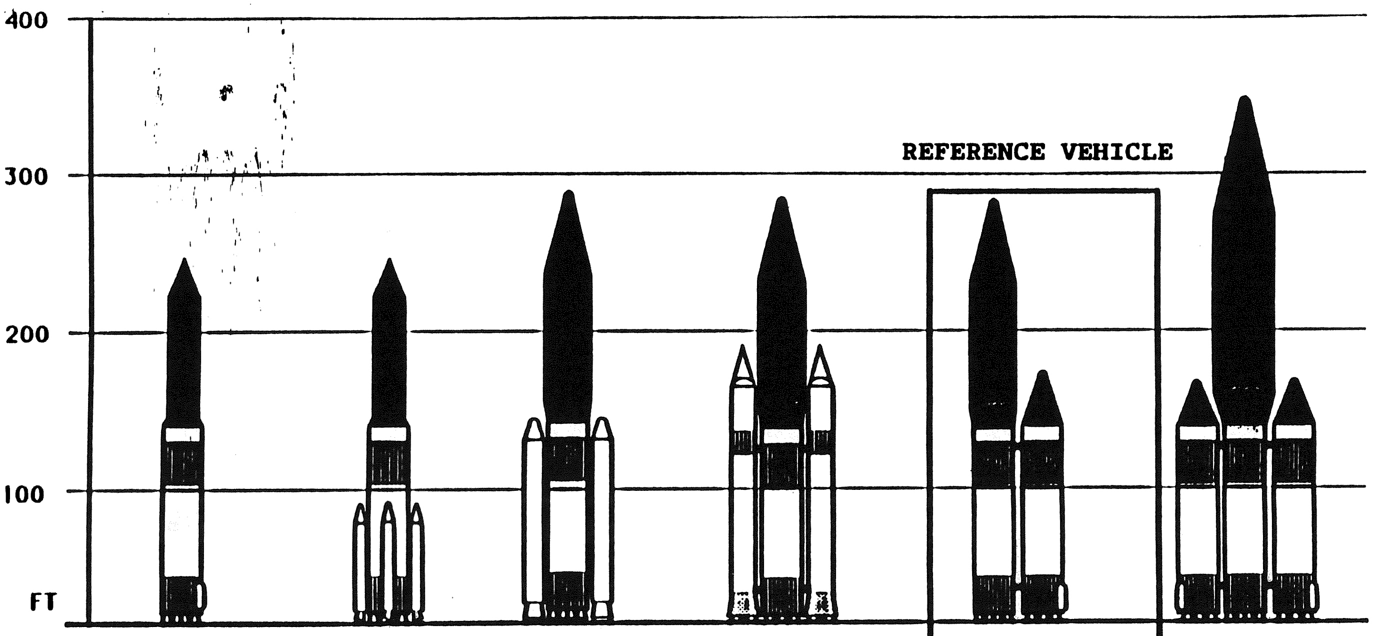
SSME/ADVANCED H2/02 TYPE



ADVANCED LAUNCH SYSTEM

- AIR FORCE SPONSORED PROGRAM
- FORMER HEAVY LIFT LAUNCH VEHICLE
  - Low cost access to space
  - Payloads > 100,000 lbms to LEO
- PHASE B UNDERWAY
  - Three study contracts
    - General Dynamics
    - Boeing
    - Martin Marietta/McDonnell Douglas
- P&W CONTRACT FOR SPACE TRANSPORTATION ENGINE (STE) STUDY

THE ALS FAMILY



RATED PAYLOAD	40-80 KLB	40-80 KLB	80-110 KLB	100-251 KLB	80-120 KLB	120-300 KLB
BOOSTER TYPE	STAGE & HALF	ALS-SRM	SRMU	STS LRB	ALS LRB	ALS LRB (2)
BOOSTER PROP	LO2/LH2	SOLID	SOLID	LO2/LH2	LO2/LH2	LO2/LH2
# OF BOOSTERS	N/A	3-8	2-4	2-6	1-4	2-4

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ALS BUILDING BLOCKS

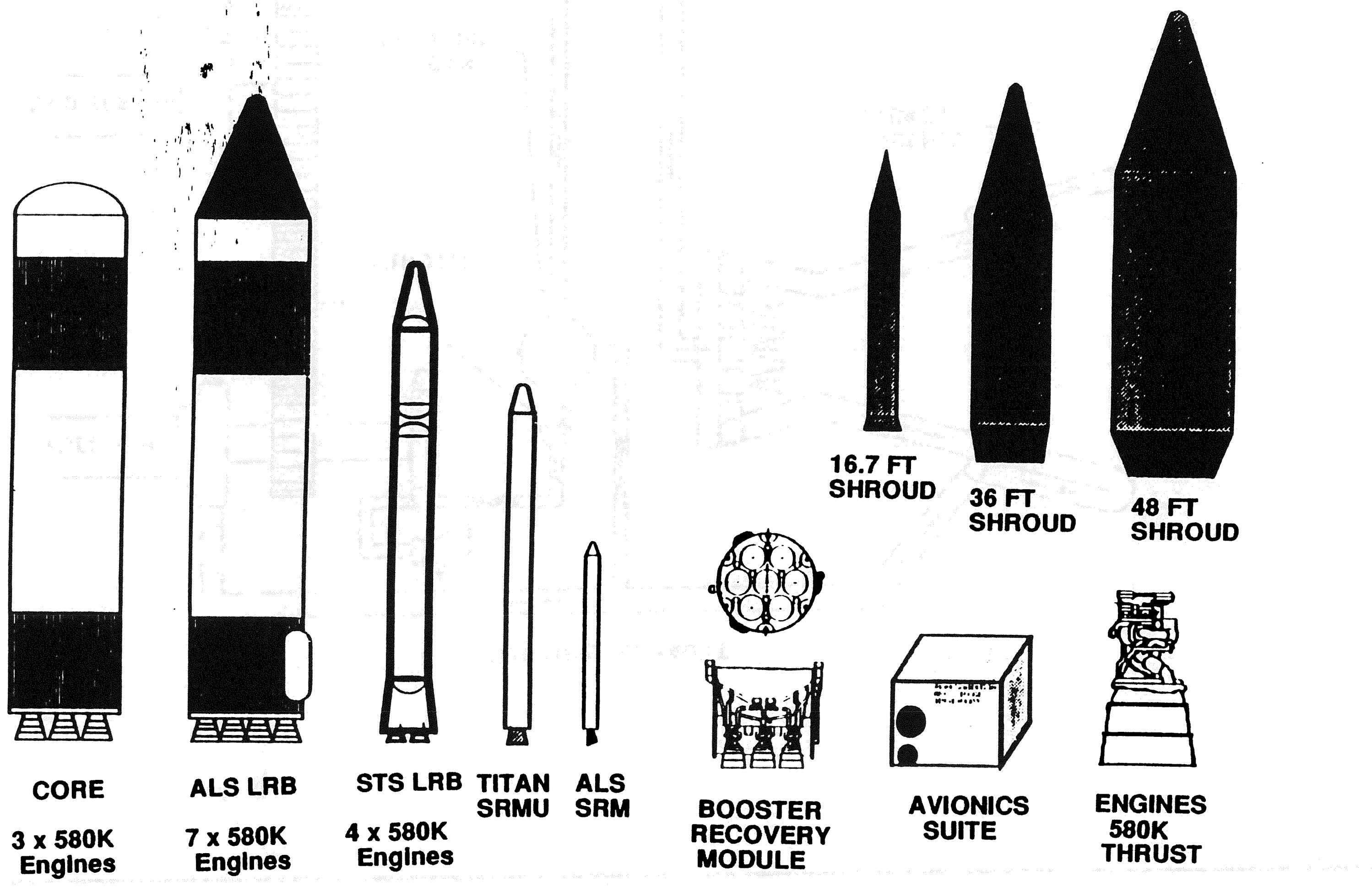

Fig 3-1

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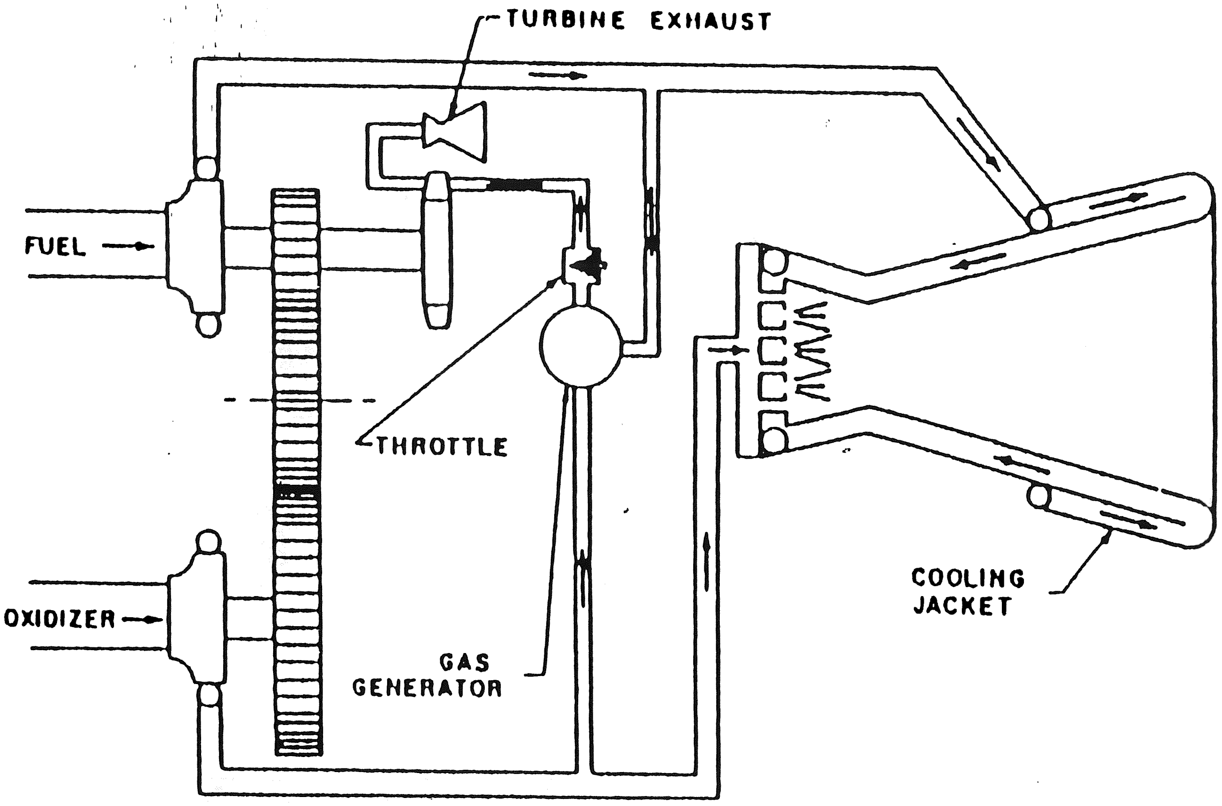
GAS GENERATOR CYCLE

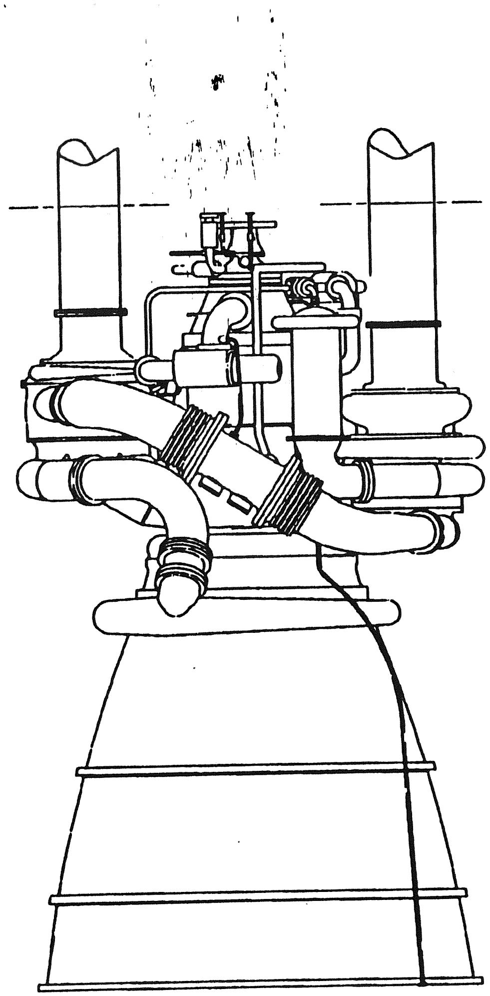


SPACE TRANSPORATION ENGINE STUDY PROGRAM

- H_2/O_2 ENGINE FOR ADVANCED LAUNCH SYSTEM (ALS)
- PHASE B PRELIMINARY DESIGN IN PROCESS
- OTHER POSSIBLE APPLICATIONS:
    - Unmanned Cargo Vehicle
    - Shuttle
    - SSME Replacement Engine

Engineering Manager: Don Connell

STE CHARACTERISTICS



ENGINE CHARACTERISTICS	
VACUUM THRUST	580,000 LBS
SEA LEVEL THRUST	502,040 LBS
CHAMBER PRESSURE	2250 PSIA
INLET MIXTURE RATIO	6.0
CHAMBER MIXTURE RATIO	7.06
VACUUM Isp	432.2 SEC
SEA LEVEL Isp	374.1 SEC
DRY WEIGHT	7936 LBS
LENGTH	150 IN
DIAMETER	86.4 IN
AREA RATIO	40:1
COST (TFU)	$9.348 M
TOTAL SYSTEM LCC	
EXPENDABLE	$38542 M
REUSABLE	$32681 M

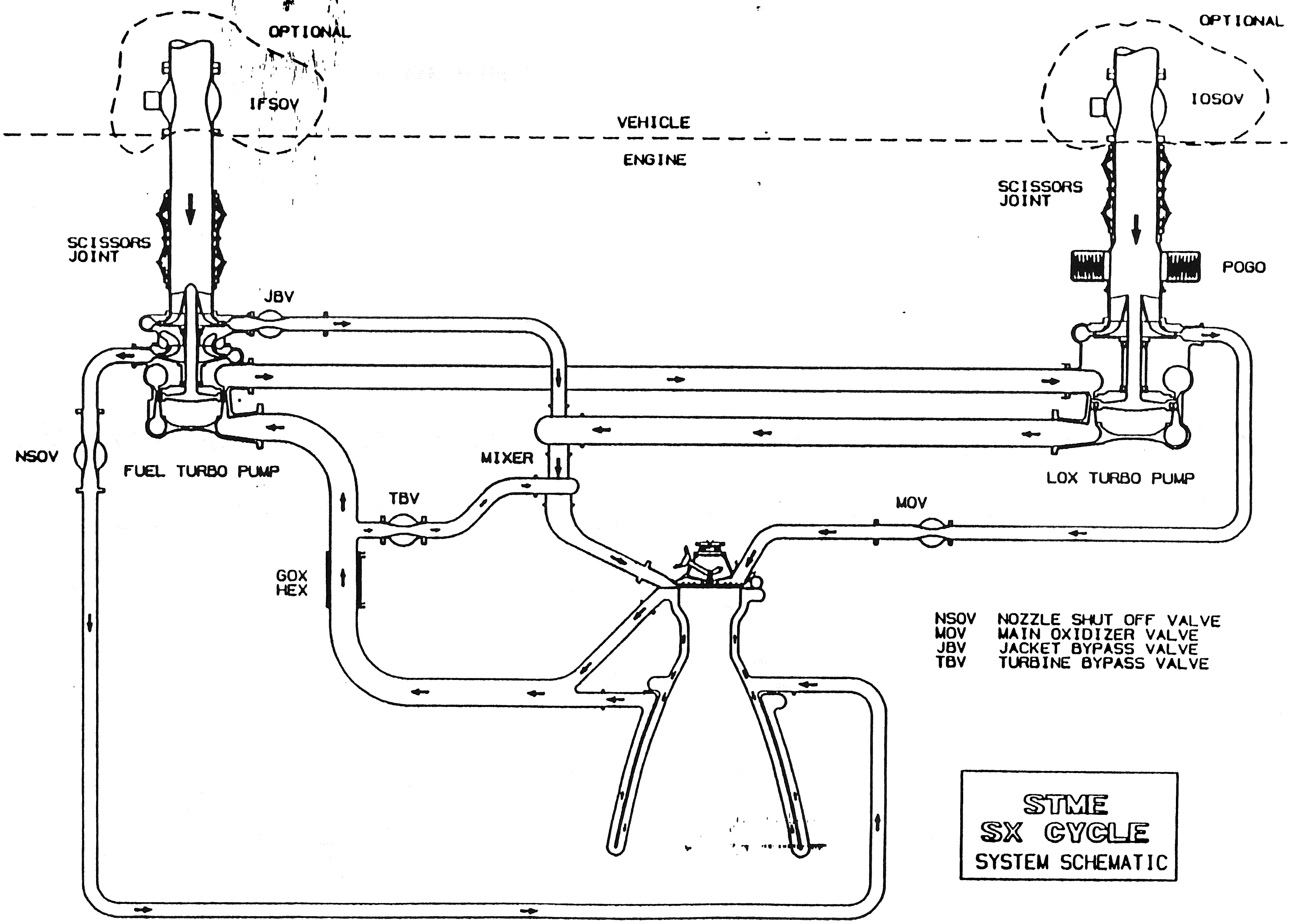
SPLIT EXPANDER ENGINE

P&W CONCEPT FOR FUTURE LARGE ROCKET ENGINES

- ¤ BASED ON RL10 TYPE EXPANDER CYCLE
- ¤ LOWER PRESSURE THAN GAS GENERATOR
- ¤ LARGE DIAMETER ENGINE NEEDED TO PRODUCE REQUIRED THRUST
- ¤ P&W IS NEGOTIATING WITH NASA/MSFC FOR DEMONSTRATOR THRUST CHAMBER CONTRACT
  - Build One Set of Hardware
    - Chamber
    - Injector
  - Demonstrate Performance
    - Stability
    - Heat Pick-Up for Turbine Power

SPLIT EXPANDER CYCLE

Fuel System Flow Split Gives Higher Chamber Pressure



NSOV NOZZLE SHUT OFF VALVE
MOV MAIN OXIDIZER VALVE
JBV JACKET BYPASS VALVE
TBV TURBINE BYPASS VALVE

STME
SX CYCLE
SYSTEM SCHEMATIC

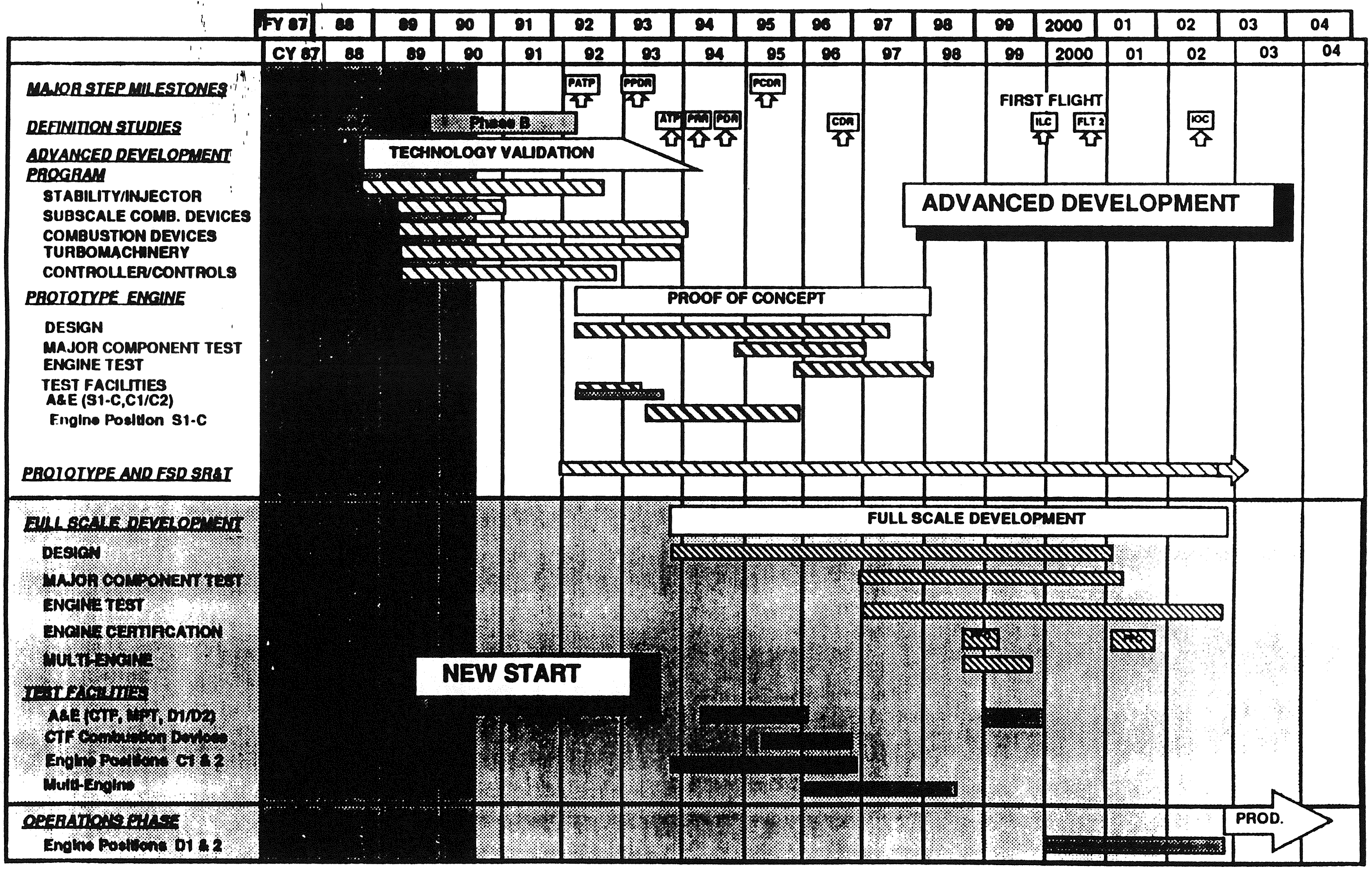
ADVANCED LAUNCH SYSTEM

Advanced Development Programs (ADPs)

AGENCY/PROGRAM	START DATE	VALUE
Astronautics Laboratory - Air Force		
Rocket Engine Condition Monitoring System	3/1/89	16.9 M
NASA		
Engine Design Phase B (STEP) NASA	3 Q 89	20 M
Split Expander Thrust Chamber Assembly Demonstration NASA	2 Q 90	10 M
O_2 Turbopump	3/1/89	21.7 M

P&W IR&D PROGRAM FOR INJECTOR/THRUST CHAMBER TECHNOLOGY

ALS MASTER SCHEDULE



6/26/90

WRAP–UP

- Rocket Programs Provide Unique Challenges at P&W
- P&W Rocket Experience Provides Strong Base for Current/Future Programs
- P&W Rocket Programs Provide Many Opportunities for Growth

ROCKET ENGINE FUNDAMENTALS

SPACE PROPULSION
PRATT & WHITNEY

SESSION 2
PART 2

APRIL 8, 1991

JAY R. COBIA/
VINCE GARCIA

FUTURE ROCKET PROPULSION SYSTEMS

CHEMICAL

- SOLID - HYBRID
- LIQUID - STME / STBE

NUCLEAR

ELECTRICAL

- ARC–HEATING–ELECTROTHERMAL
- MAGNETOPLASMA
- ION–ELECTROSTATIC

SOLAR

HYBRID ROCKET ENGINES

- Hybrid rocket engines make use of various combinations of solid and liquid propellants.

$$\left.\begin{array}{l}\text{Storable or cryogenic oxidizer}\\ \text{Solid Fuel}\end{array}\right\}\text{ Most Common}$$

- Applications to date have been for target missles and low–cost tactical missiles.
- Advantages

    Low cost for applications where economy is essential and low performance acceptable.

    Simplicity of solid grain fuel

    A liquid for nozzle cooling and thrust modulation

    Start–stop–restart capabilities

    Safety during storage or operation

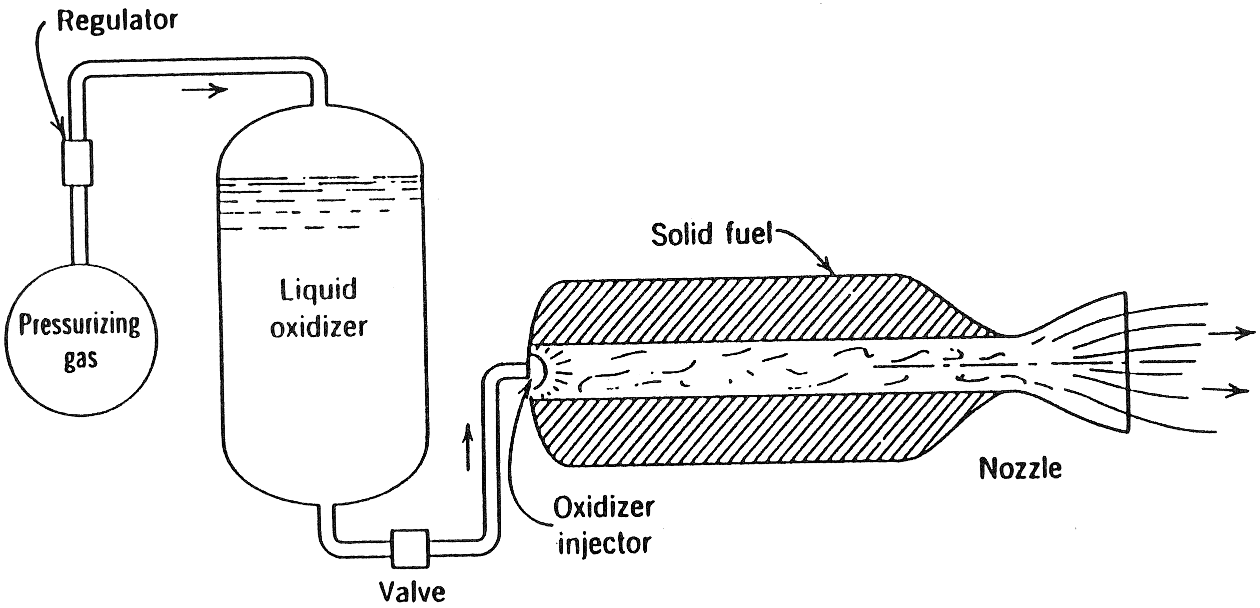
- Disadvantages

    Low performance

    Prediction of grain regression rate (burning rate) is difficult.

    Efficient combustion is difficult to attain – mixture ratio along the grain will not be uniform with a single oxidizer spray injection location.

HYBRID ROCKET ENGINE



Schematic diagram of typical hybrid rocket engine.

LIQUID ROCKET ENGINES

- Liquid propellant rocket engines make use of liquid propellants (oxygen and hydrogen or hydrocarbons) that are fed under pressure from tanks iinto a thrust chamber.
- In the thrust chamber the propellants react to form hot gases, which are accelerated and ejected at high velocities through a supersonic nozzle.
- Programs presently under study at P&W

    Space Transportation Engine (STE)
    Advanced Expander Test Bed (AETB)

- Advantages

    Repetitive Operation

    Numerous start-stop-resstart cycles

    Throttle capability

    Increased run time - dependent on propellant supply

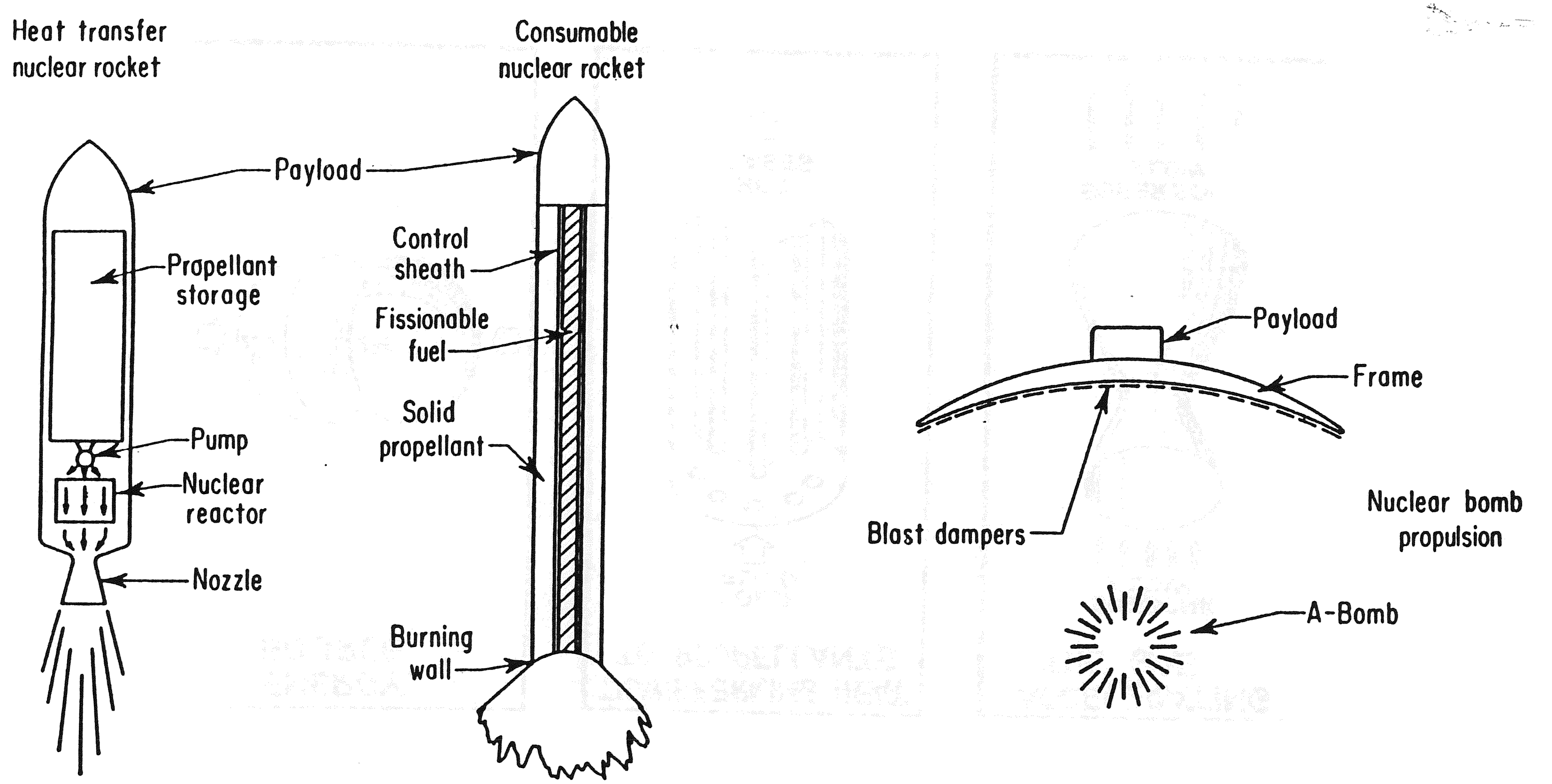
- Disadvantages

    Engine is complicated - requires precision valves and complex feed mechanism and an intricate combustion or thust chamber

NUCLEAR ROCKET ENGINES

- Utilizes a nuclear energy source to heat a working fluid–usually LH2, (Power source is separate from the propellant)
    - FissionReactor (Proven Technology)
    - Radioactive Isotope Decay (Proven Technology)
    - Fusion Reactor (Unproven Technology)
- The hydrogen gases are expanded in a nozzle and accelerated to high ejection velocities (6,000 to 10,000 m/sec).
- Advantages
    - High specific impulse
    - Much lower initial fully fueled weight required for a given desired velocity
- Disadvantages
    - High reactor temperatures (above 2200 °C)
    - Radiation effects on humans, materials and environment
    - Increased vehicle weight due to shielding.

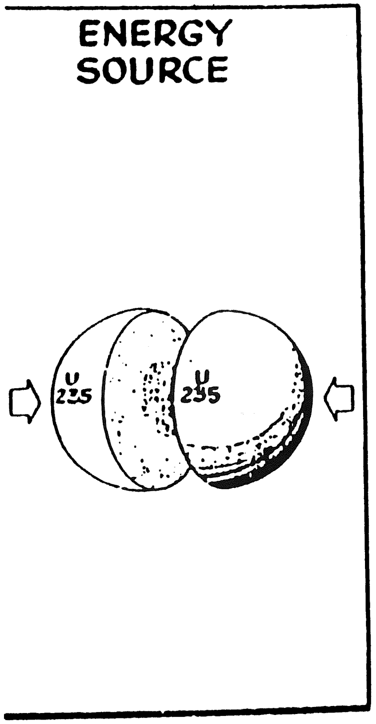
NUCLEAR PROPULSION SYSTEMS

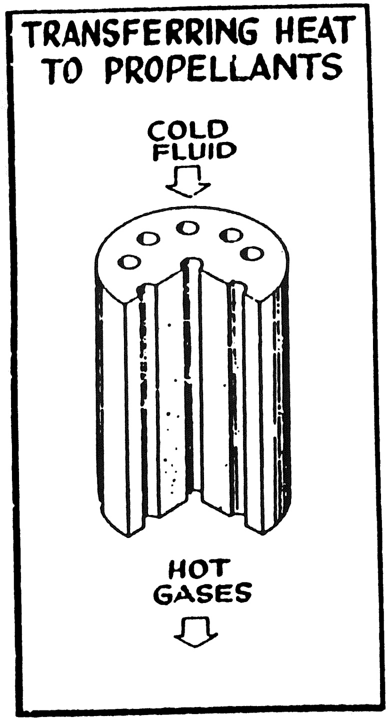


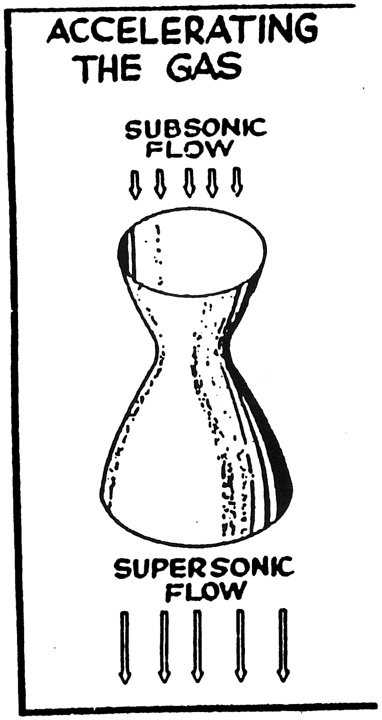
Three types of nuclear propulsion systems. These are all thermal engines.

NUCLEAR PROPULSION SYSTEMS

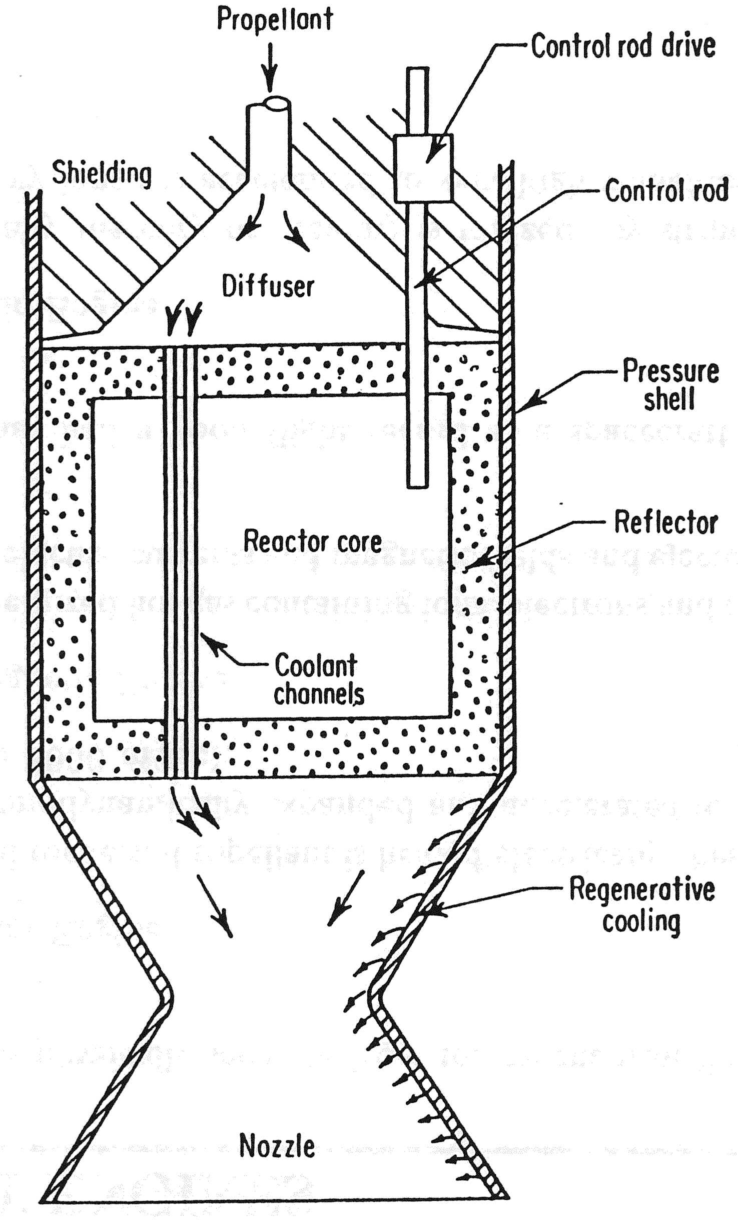
HOW NUCLEAR PROPULSION WORKS







NUCLEAR PROPULSION SYSTEM SCHEMATIC



ELECTRICAL ROCKET ENGINES

- Source of the electric power is physically separate from the mechanism that produces thrust.

- Arc–Heating or Electrothermal Engine

  Most resembles chemical rockets. Propellant is heated electrically (heated resistors or electric arcs) and then the hot gas is thermodynamically expanded and accelerated to supersonic velocities through an exhaust nozzle (1000 to 3000 m/sec).

- Magnetoplasma or Electromagnetic Engine

  Electrical plasma (an energized hot gas containing ions, electrons and neutral particles) is accelerated by the interaction between electric currents and magnetic fields and ejected at very high velocities (4000 to 50,000 m/sec).

  A simple pulsed unit has had a good flight record as a spacecraft attitude control and flight path correction engine.

- Ion Propulsion or Electrostatic Engine

  A working fluid (typically mercury or cesium) is ionized (by stripping off electrons) and then the electrically charged heavy ions are accelerated to very high velocities (4000 to 60,000 m/sec).

ELECTRICAL ROCKET ENGINES (Continued)

- Advantages

  Relatively high performance

  Require small amounts of propellants

- Disadvanatages

    Power source is heavy and usually inefficient

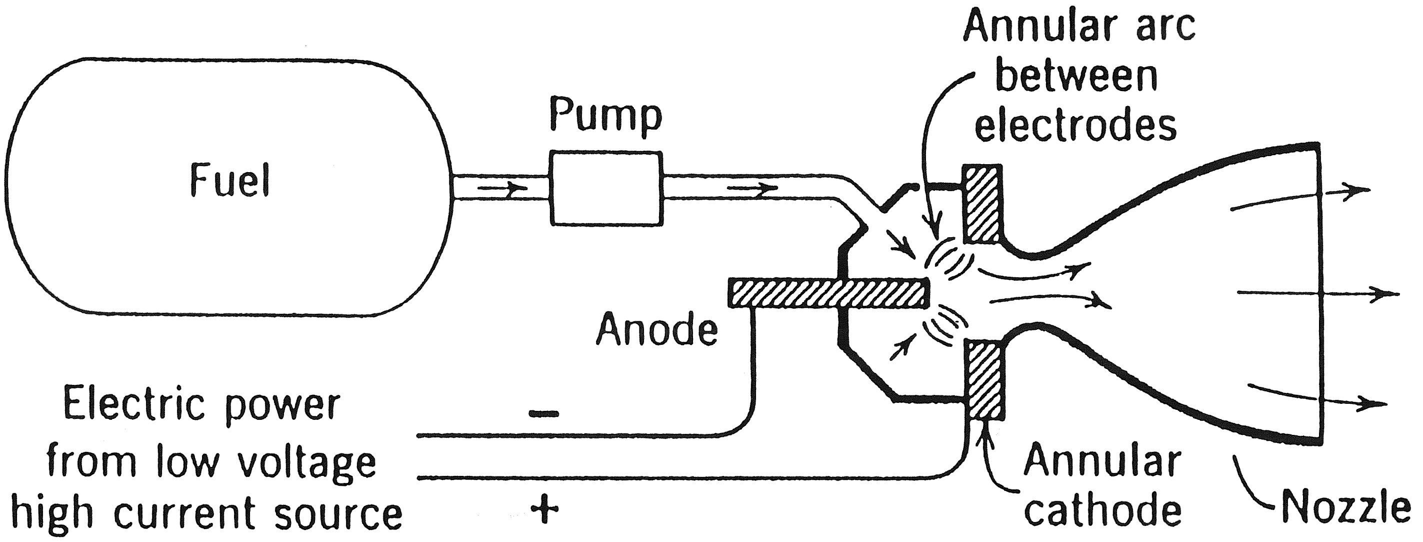
    Thrust is usually low (they have to accelerate large vehicle–based powerplants)

    Thrust must be applied for weeks and months to allow significant increases in vehicle velocity.

    Applications limited to spacecraft control

ELECTRIC ROCKET ENGINES

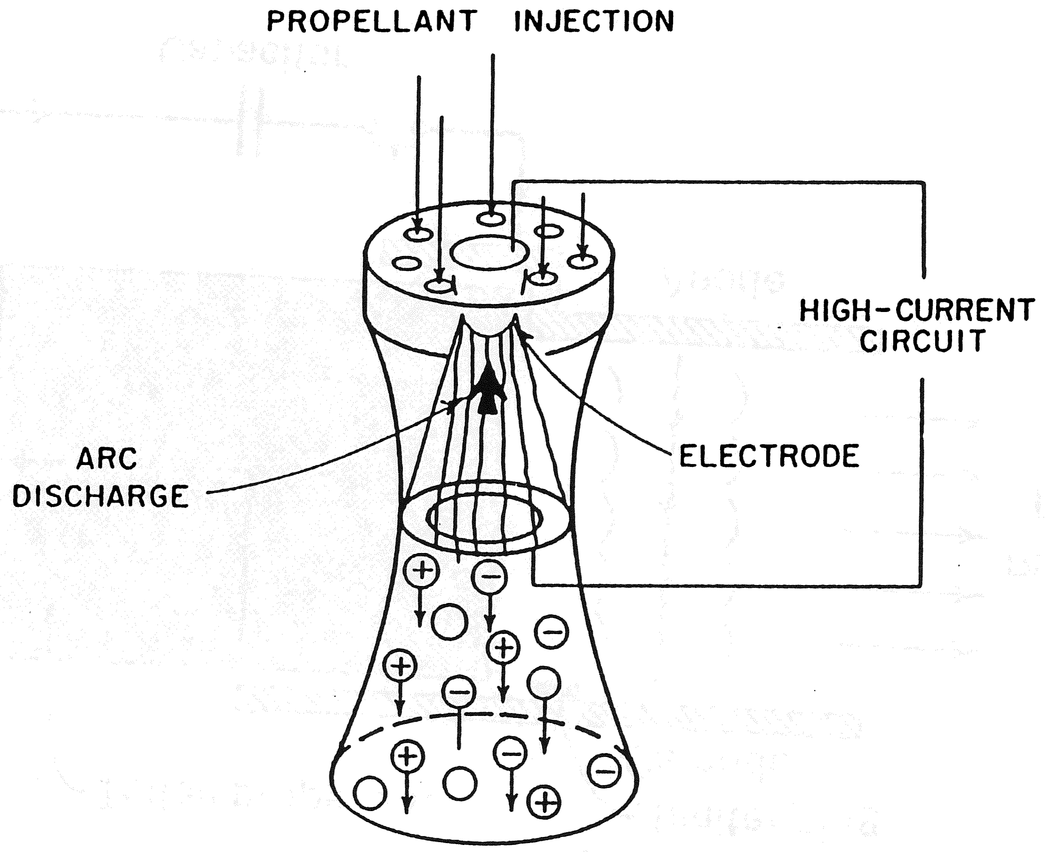
ARC-HEATING ROCKET ENGINE SYSTEM



Schematic diagram of arc-heating rocket engine.

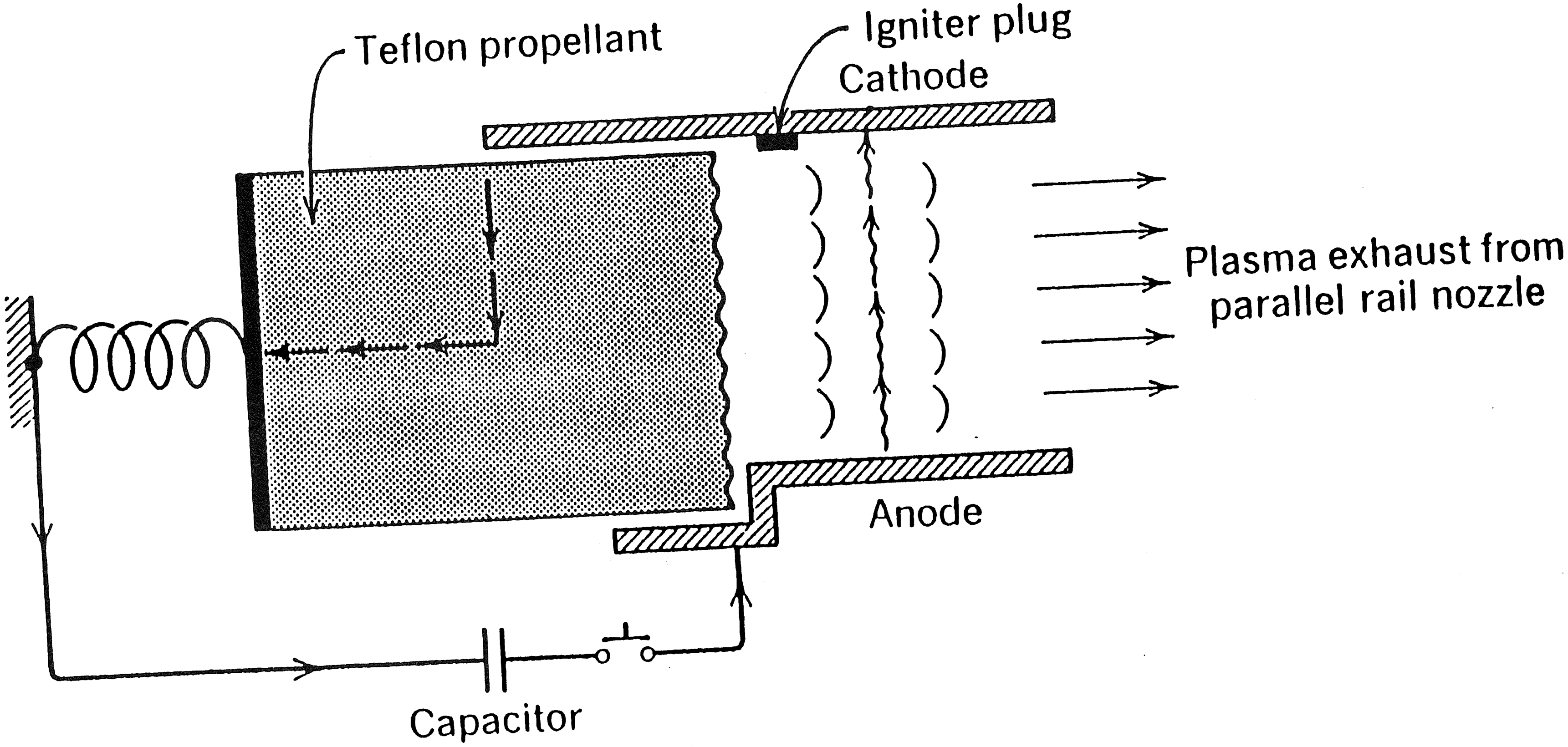
ELECTRIC ROCKET ENGINES

PLASMA ROCKET ENGINE



ELECTRIC ROCKET ENGINES

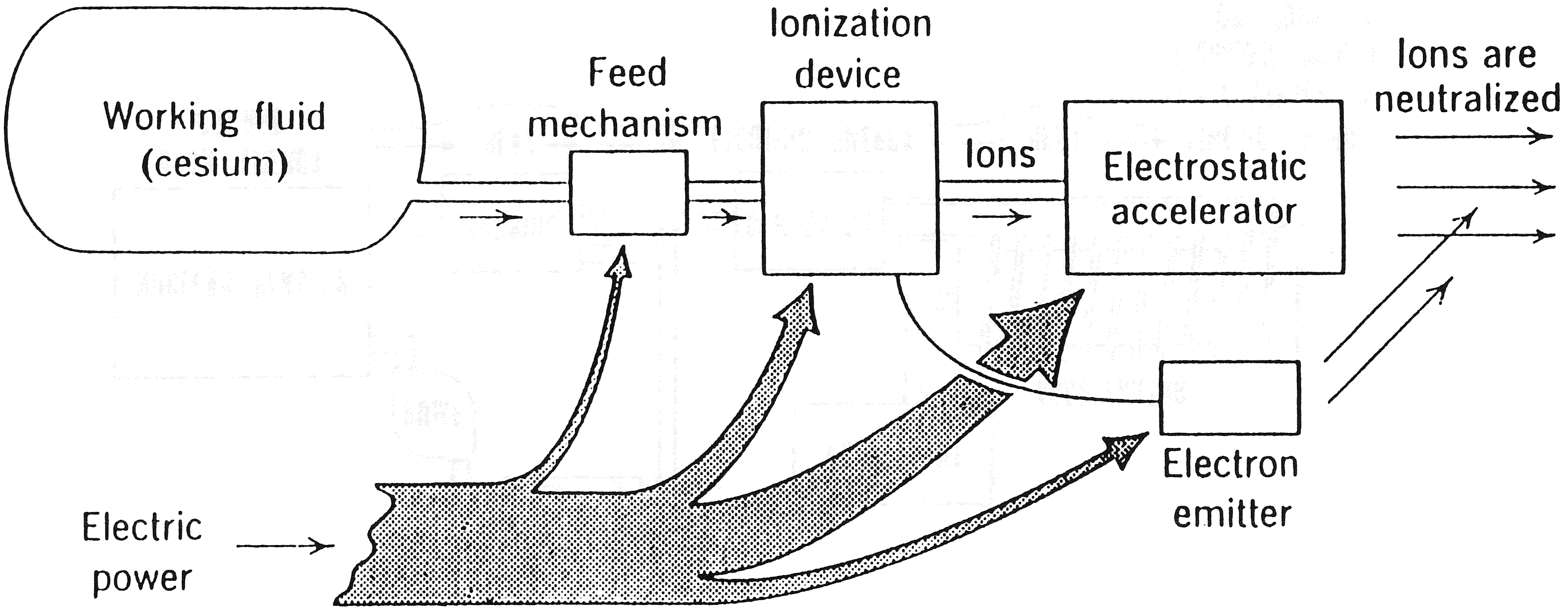
RAIL ACCELERATOR



Simplified diagram of a rail accelerator for a self-induced magnetic acceleration of a current carrying plasma. When the capacitor is discharged an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of current and magnetic field causes the plasma to be accelerated at right angles to both the current and magnetic field, namely, in the direction along the rails. With each small pulse of thrust a small amount of the solid propellant (teflon) is vaporized and converted to a plasma cloud. Actual units can operate with many pulses per second.

ELECTRIC ROCKET ENGINES

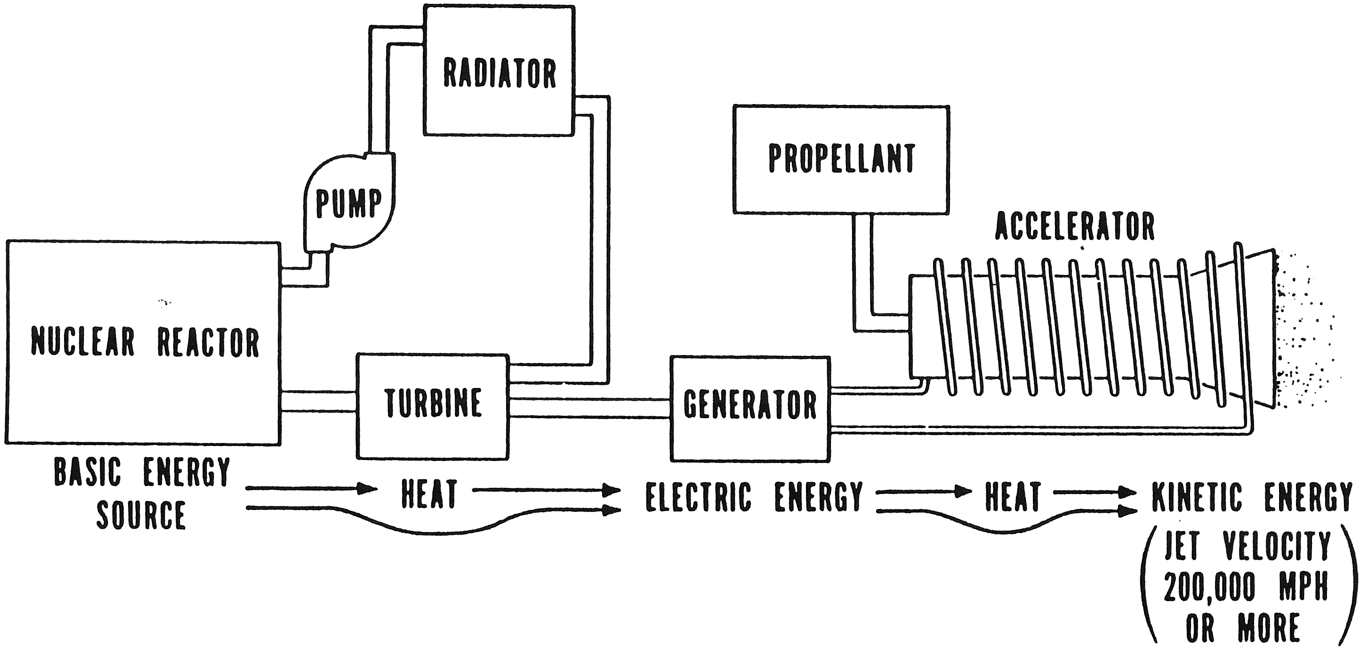
ION ROCKET



Schematic diagram of a typical ion rocket.

ELECTRIC ROCKET ENGINES

NUCLEAR POWERED ION ROCKET



Ion rocket utilizing a nuclear turbo-electric energy source.

SOLAR ROCKET ENGINE

- The power source is the sun and it is external to the vehicle.
- Several methods exist for harnessing solar energy to propel spacecraft.

  Solar Cells – Their use to generate electricity for electric rocket engines is the most highly developed.

  Solar Heating – Method of heating the propellants by optically concentrating infrared radiation to heat a hydrogen working fluid.

  Solar Sail – Method uses the radiation pressure of sunlight and the photon rocket to propel the vehicle.

- Advantages

  Solar cells eliminate large, heavy power plants necessary for electric engines

  Solar radiation is abundant

  Infinite specific impulse obtainable for those solar engines not requiring a working fluid.

- Disadvantages

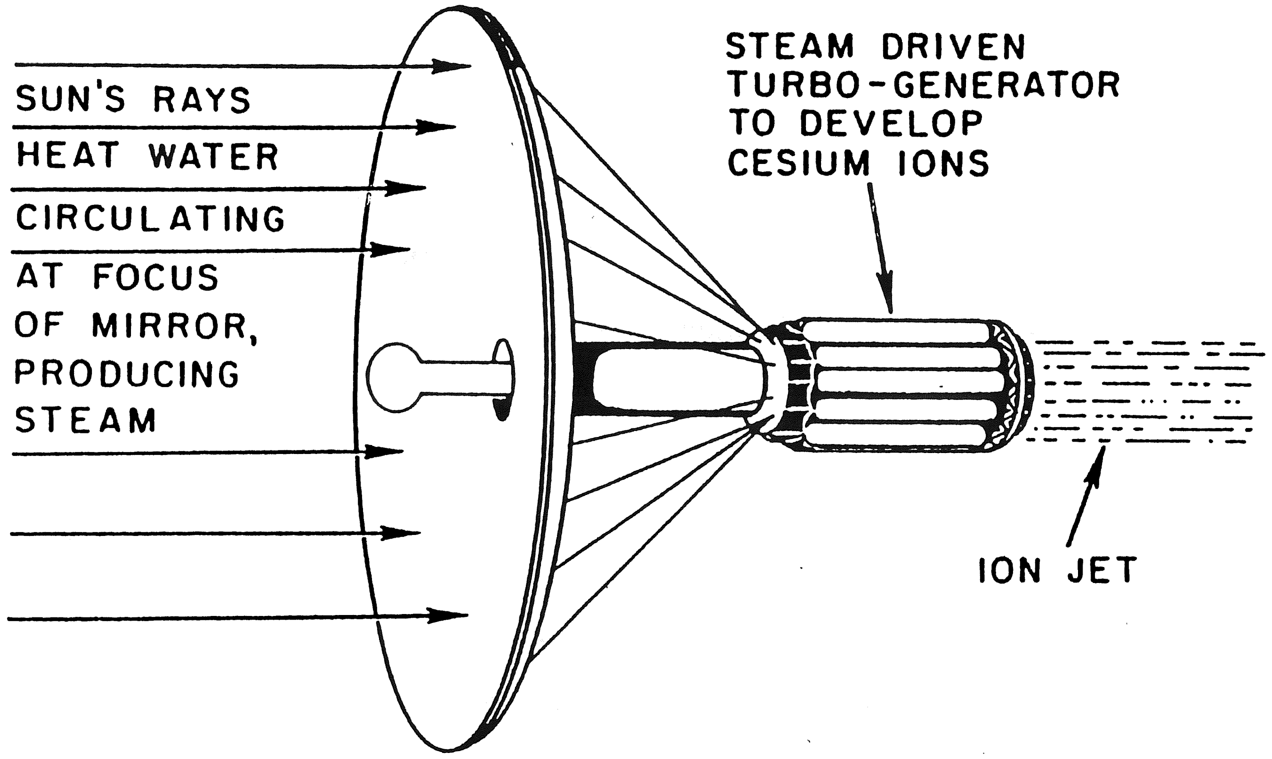
  Very low thrust produced

  Solar driven vehicles must be launched into orbit by some other kind of propulsion system.

  Solar radiation collectors tend to be large and susceptible to environment.

SOLAR POWERED ROCKET ENGINES

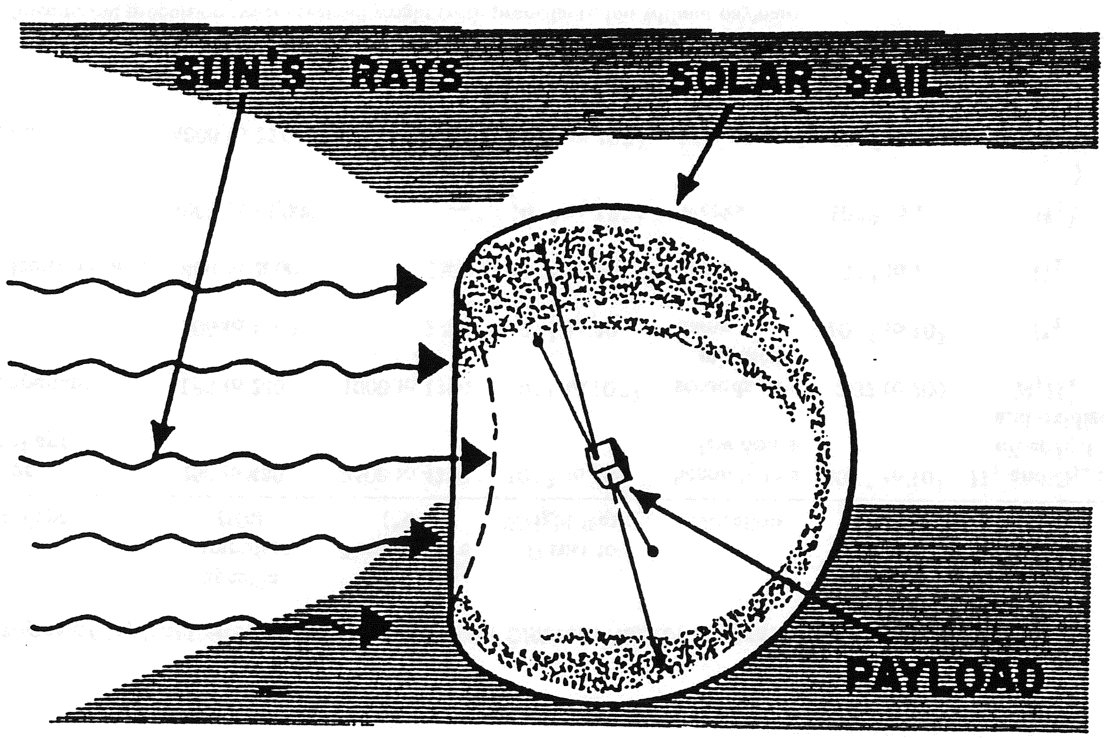
SOLAR POWERED ION DRIVE



SOLAR PROPULSION: POWER CONVERSION BY PRODUCING STEAM TO DRIVE ELECTRIC ION GENERATOR.

SOLAR POWERED ROCKET ENGINES

SOLAR SAIL



SOLAR PROPULSION: DIRECT PHOTON UTILIZATION

ROCKET ENGINE PERFORMANCE

Ranges of Typical Performance Parameters for Different Rocket Engines

Engine Type	Specific Impulse (sec)	Maximum Temperature (°C)	Thrust-to-Weight Ratio[a]	Duration	Specific Power[b] (kw/kg)	Typical Working Fluid	Status of Technology
Chemical-solid or liquid bipropellant	200 to 480	2500 to 4300	10^{-2} to 100	Seconds to a few hours	10^{-1} to 10^3	H_2 and O_2, or other fuel and oxidizer	Flight proven
Liquid monopropellant	180 to 240	1000 to 1300	10^{-1} to 10^{-2}	seconds to minutes	0.02 to 200	N_2H_4	Flight proven
Nuclear fission	500 to 1100	2700	10^{-2} to 30	Same	10^{-1} to 10^3	H_2	Development was stopped
Arc heating—electrothermal	400 to 2000	5500	10^{-4} to 10^{-2}	Days	10^{-3} to 1	H_2	Development continues
Magnetoplasma	3000 to 15,000	—	10^{-5} to 10^{-3}	Weeks	10^{-3} to 1	H_2	Several have flown
Ion—electrostatic	4000 to 25,000	—	10^{-5} to 10^{-3}	Months	10^{-3} to 1	Cs	
Solar heating	400 to 700	1300	10^{-3} to 10^{-2}	Days	10^{-2} to 1	H_2	Not yet flight tested

[a] Ratio of thrust force to full propulsion system sealevel weight (with propellants, but without payload).
[b] Kinetic energy per unit exhaust mass flow.

APPENDIX

ATLAS OF SPACE VEHICLES
AND
PROPUSION SYSTEMS

UNITED
TECHNOLOGIES
PRATT&WHITNEY

LIQUID PROPULSION SYSTEMS APPLICATIONS

- **RL10** The RL10, currently powering the Centaur stage of the Atlas and Titan Centaur vehicles, has compiled 100% reliability in 282 in space firings. A multiple-start power plant, it is gimbal-mounted and operates on a combination of liquid hydrogen and liquid oxygen. The thrust chamber is regeneratively cooled with the hydrogen fuel circulating through 360 tubes that comprise the wall of the chamber. Manufacturer: Pratt & Whitney.
- **SSME** The Space Shuttle Main Engine (SSME) was developed expressly for use on America's Space Shuttle. Using a mixture of liquid oxygen and liquid hydrogen, the SSME can attain a maximum thrust level (in vacuum) of 512,300 pounds at 109% power level. The regeneratively cooled engine also features high performance turbopumps for fuel and oxidizer that protellants develop 77,310 horsepower and 29,430 horsepower, respectively. Ultra-high-pressure operation of the pumps and combustion chamber allows expansion of all hot gases through a high-area-ratio exhaust nozzle to achieve efficiencies never prevously attained in a production rocket engine. These advantages allow a heavier payload to be carried without increasing launch vehicle size. Manufacturer: Rocketdyne.
- **MA-5** The MA-5 propulsion system consists of a booster engine with two thrust chambers and a sustainer engine, all using a bipropellant scheme that combines RP-1 and liquid oxygen. The regneratively cooled engines can be gimbaled during flight to effect vehicle attitude control. Both bell-shaped booster thrusters are connected to a central power package consisting of a turbopump for each chamber, a single gas generator, a pneumatic control package, and the liquid oxygen regulator. The sustainer engine turbopump is thrust chamber mounted. The MA-5 currently powers the Atlas booster. Manufacturer: Rocketdyne.

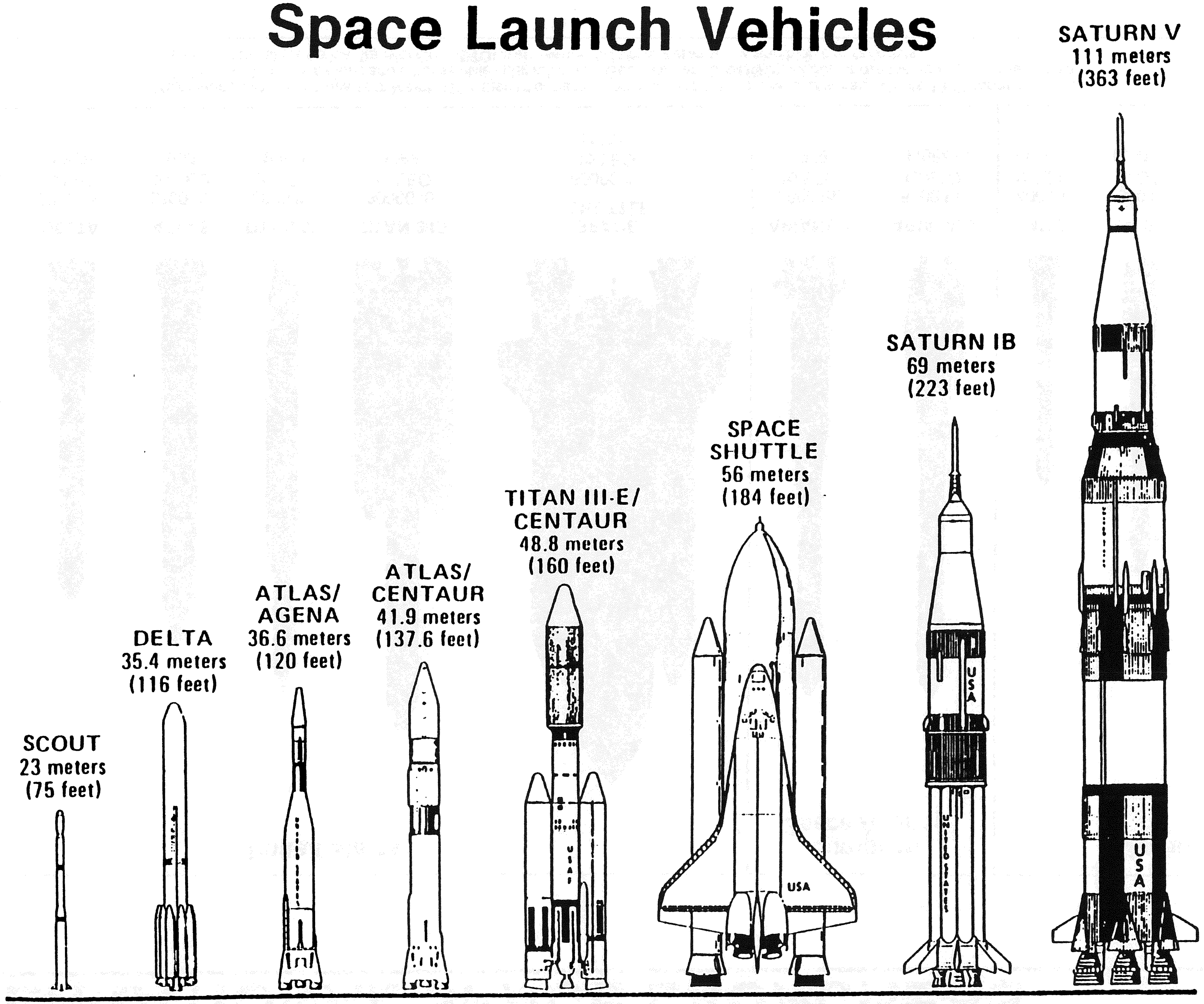
LIQUID PROPULSION SYSTEMS APPLICATIONS (Con't)

- RS-27 The RS-27, currently powering the Delta vehicle, has compiled one of the most consistent and successful launch records in the history of rocketry, with a 100% reliability factor. A single-start power plant, it is gimbal-mounted and operates on a combination of liquid oxygen and RP-1 (kerosene). The thrust chamber is regeneratively cooled with the fuel circulating through 292 tubes that comprise the inner wall of the chamber. Manufacturer: Rocketdyne.
- YLR87 The YLR87 rocket engine assembly is a storable-liquid-bipropellant (Aerozine 50 and N_2O_4) turbopump-fed, dry-jacket-start rocket engine. It is comprised of two independently operating subassemblies which are mounted on a single engine frame. The engine assembly currently powers the Titan booster. Manufacturer: Aerojet TechSystems Co.
- YLR91 The YLR91 rocket engine assembly is smaller but similar to the YLR87. The engine powers the second stage of the Titan vehicle. Manufacturer: Aerojet TechSystems Co.
- J-2 The J-2 provided highly reliable performance for both second and third stages of the Saturn vehicle in the Apollo program. The J-2 features independently driven pumps for both liquid oxygen and liquid hydrogen, a gas generator to supply hot gas to two turbines functioning in series, pneumatic and electrical control interlocks, altitude restart capability, and a propellant utilization system. Manufacturer: Rocketdyne
- F-1 Still the most powerful rocket engine ever built, the F-1's million-and-a-half pounds of thrust (quintupled in a cluster of five) lifted men to the moon atop the 363-foot Saturn vehicle. A single-start, fixed thrust engine, the F-1 is gimbaled and uses liquid oxygen as the oxidizer, while RP-1 (kerosene) is used as the fuel, the turbopump lubricant, and the control system fluid. A gas generator utilizing the same propellants drives the turbine, which is direct-coupled to the turbopump. Manufacturer: Rocketdyne.

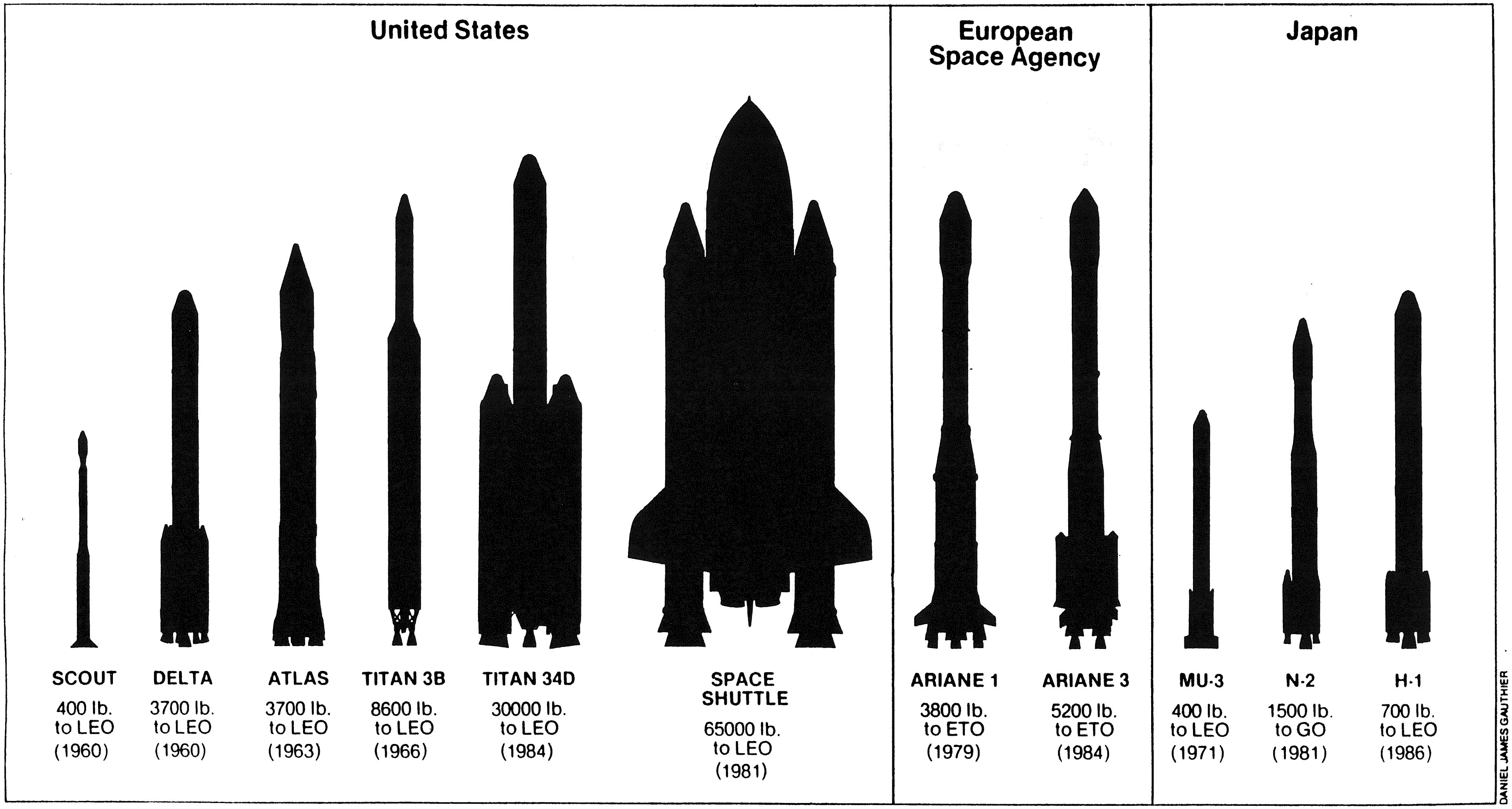
LIQUID PROPULSION SYSTEMS APPLICATIONS (Con't)

- H-1 Initially designed for Apollo program application, the H-1 is a fixed-thrust, single-start gimbaled engine that employs a propellant system of RP-1 (kerosene) and liquid oxygen. Advances include a turbopump with a one-piece gearbox and fuel-additive lubrication, a solid propellant gas generator for start-up, propellant valve sequencing and hypergolic start-up in the thrust chamber. A cluster of eight H-1 engines powered the Saturn 1B first stage. Manufacturer: Rocketdyne.
- Viking V A cluster of four Viking V engines currently power the first stage of the Ariane 2 & 3 vehicles. The engine is a storable-liquid-bipropellant (UH and N_2O_4) which is gas generator fed. Manufacturer: SEP (Societe Europienne de Propulsion)
- Viking IV The Viking IV currently powers the second stage of the Ariane 2 & 3 vehicles. The engine, similar to the Viking V, is a storable-liquid-bipropellant (UH and N_2O_4) gas generator fed which is gimbal-mounted. Manufacturer: SEP
- HM7-B The HM7-B currently powers the third stage of the Ariane 3 vehicle. Designed as a single-start engine, liquid hydrogen and liquid oxygen are delivered to the thrust chamber by a turbopump fed with gases supplied by a generator. Manufacturer: SEP

Space Launch Vehicles



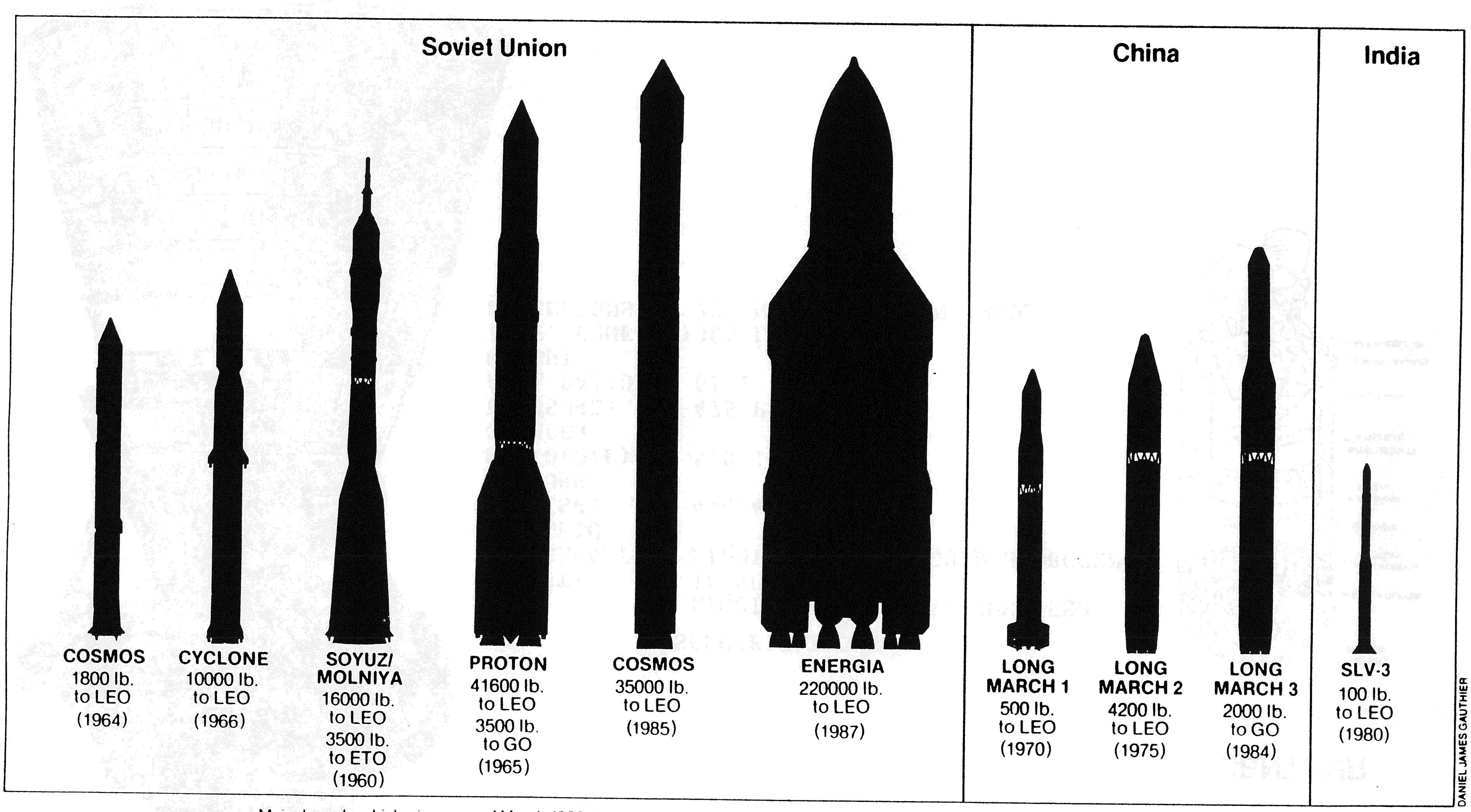
INTERNATIONAL SPACE LAUNCHERS



Major launch vehicles in use as of March 1988, their lift capacity, and date of first launch. Most Low Earth Orbits (**LEO**) range from 150 to 600 miles. Communications and other satellites placed in high Geostationary Orbit (**GO**), 22,300 above the equator, appear fixed at one point in the sky. Elliptical Transfer Orbit (**ETO**) is an intermediate orbit between low and high circular orbits.

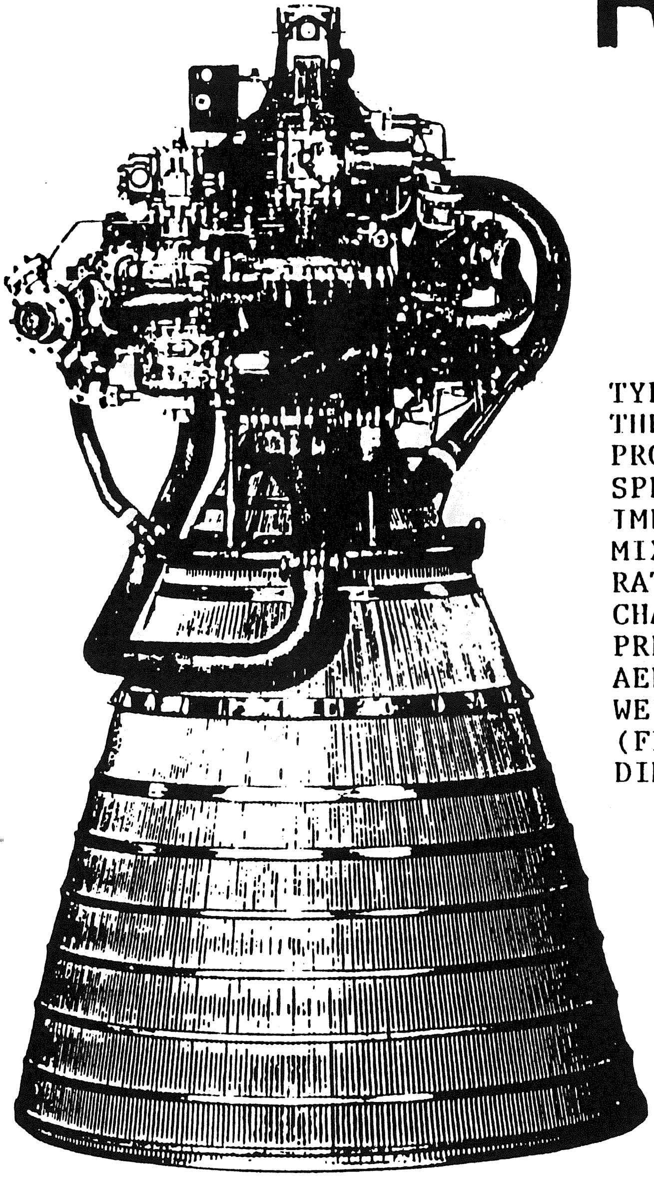
0 ft. 20

INTERNATIONAL SPACE LAUNCHERS CONT.)



Major launch vehicles in use as of March 1988, their lift capacity, and date of first launch. Most Low Earth Orbits (**LEO**) range from 150 to 600 miles. Communications and other satellites placed in high Geostationary Orbit (**GO**), 22,300 above the equator, appear fixed at one point in the sky. Elliptical Transfer Orbit (**ETO**) is an intermediate orbit between low and high circular orbits.

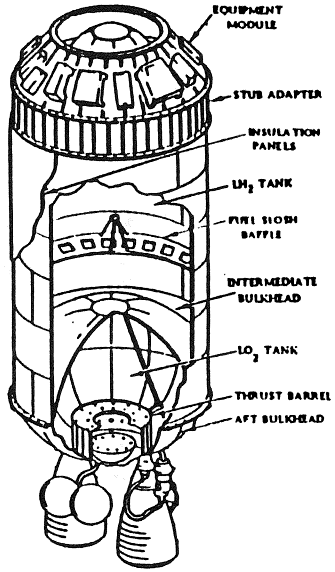
RL10A-3-3A

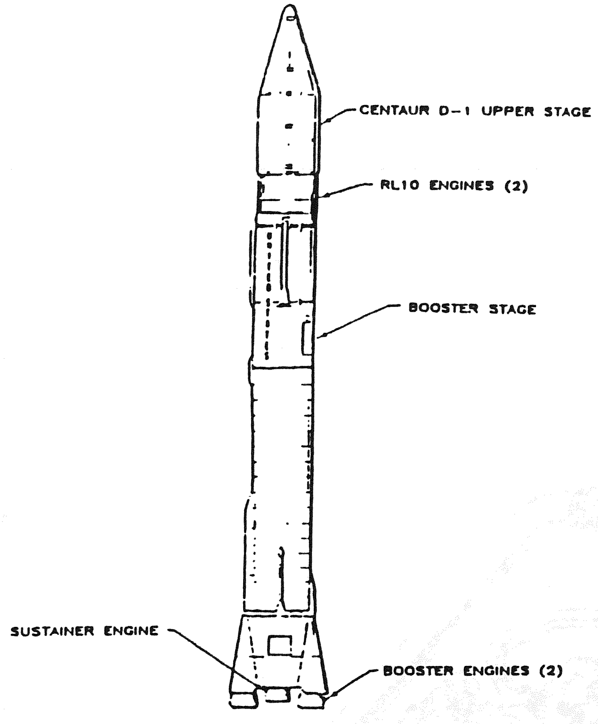


SPECIFICATIONS

TYPE:	LIQUID-PROPELLANT, PUMP-FED
THRUST:	16,500 LB
PROPELLANTS:	LIQUID OXYGEN/LIQUID HYDROGEN
SPECIFIC IMPULSE:	444.4 SEC
MIXTURE RATIO(O/F):	5.0:1
CHAMBER PRESSURE:	475 PSIA
AERA RATIO:	61:1
WEIGHT (FLT. CONFIG.)	305 LB
DIMENSIONS:	70 IN. LONG/40 IN. WIDE

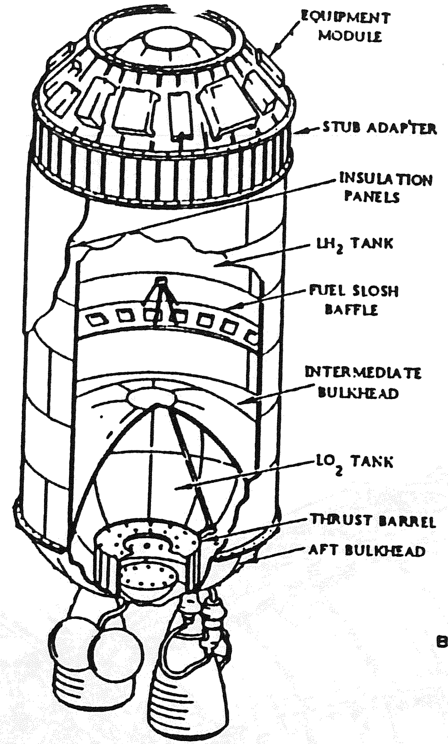
CENTAUR

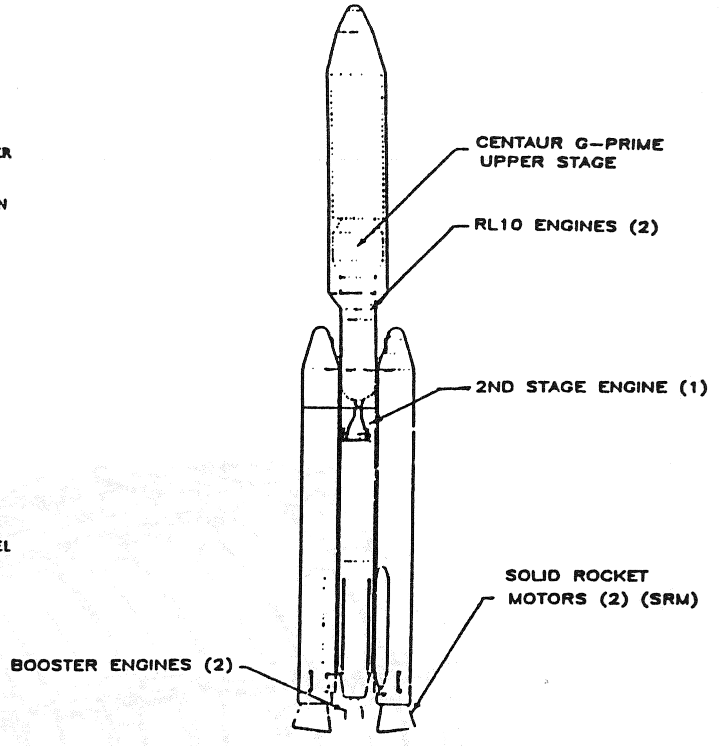




<u>LIFTOFF WEIGHT</u>: 360K LBS.
<u>PAYLOAD WEIGHT</u>: 10,000 LBS. TO LOW EARTH ORBIT
<u>PAYLOAD SIZE</u>: 10 x 25 FT.

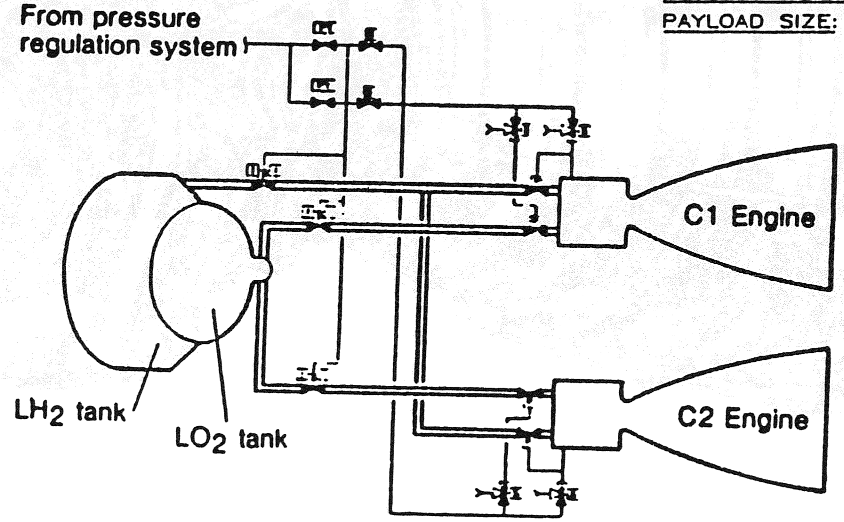
ATLAS CENTAUR





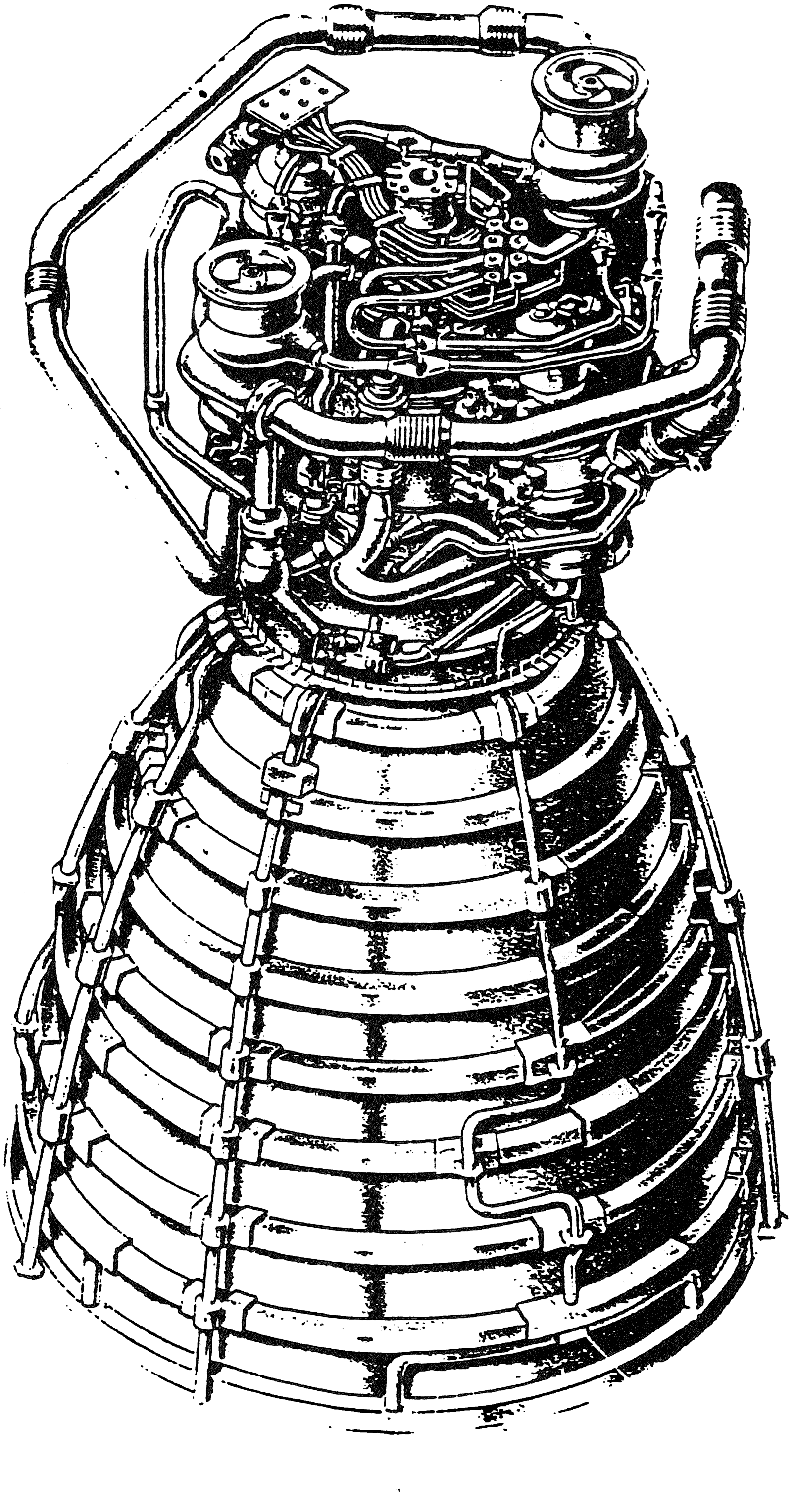
<u>LIFTOFF WEIGHT</u>: 1.9 MILLION LBS.
<u>PAYLOAD WEIGHT</u>: 10,000 LBS. TO GEOSYNCHRONOUS ORBIT
<u>PAYLOAD SIZE</u>: 16.7 x 40 FT.

TITAN 34D7/CENTAUR



CENTAUR

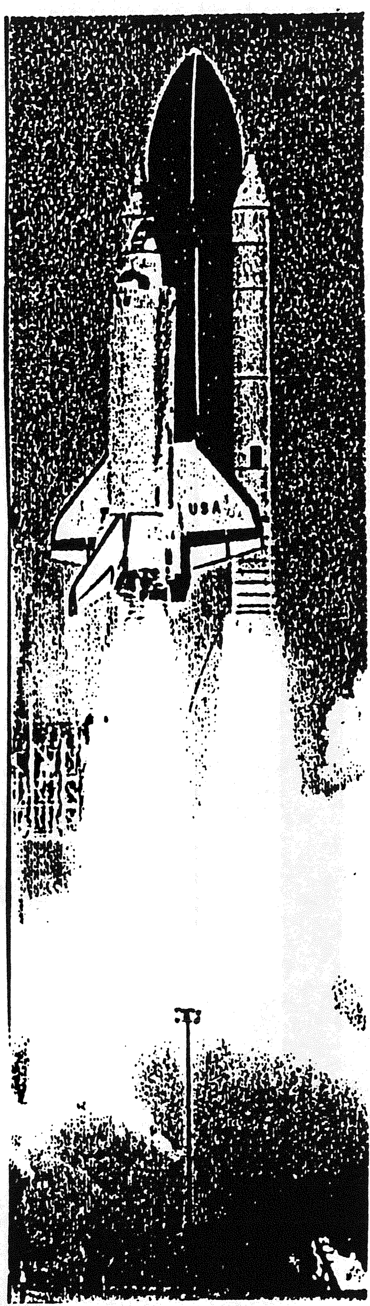
SSME



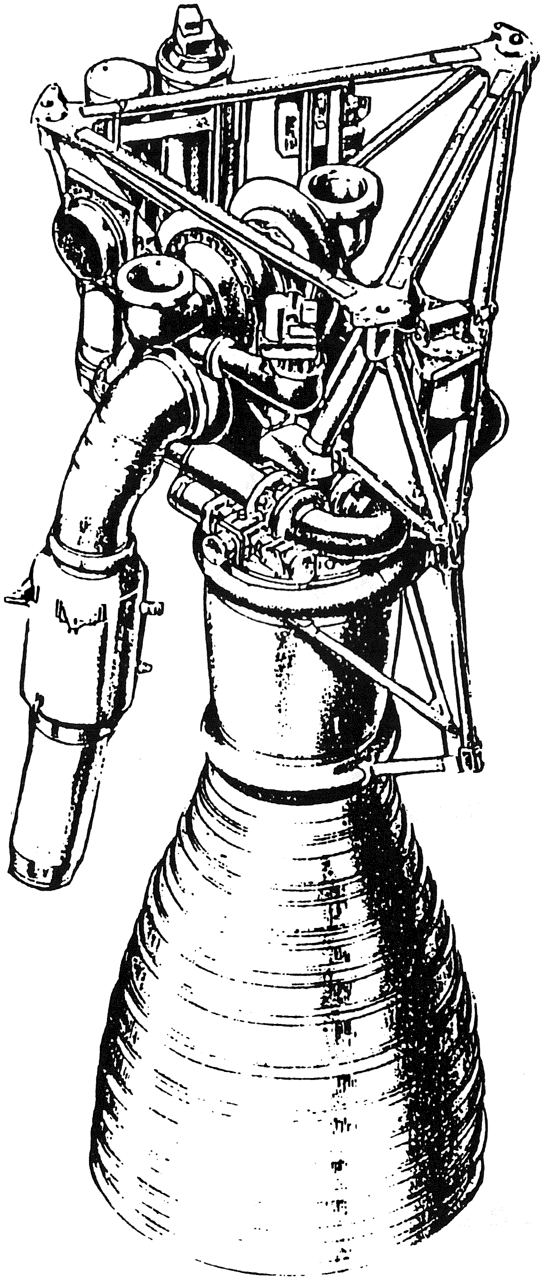
SSME

SPACE SHUTTLE MAIN ENGINE PERFORMANCE (FULL POWER LEVEL)

Maximum Thrust: (109% Power Level)	
At Sea Level	408,750 pounds
In Vacuum	512,300 pounds
Throttle Range	65%~109%
Pressures:	
Hydrogen Pump Discharge	7,040 psia
Oxygen Pump Discharge	8,070 psia
Chamber Pressure	3,260 psia
Specific Impulse (In Vacuum)	453.5 seconds
Power:	
High Pressure Pumps	
Hydrogen	77,310 horsepower
Oxygen	29,430 horsepower
Area Ratio	77.5:1
Weight:	6,990 pounds
Mixture Ratio (O/F)	6.0:1
Dimensions:	168 in. long/96 in. wide
Propellants:	
Fuel	Liquid Hydrogen
Oxidizer	Liquid Oxygen



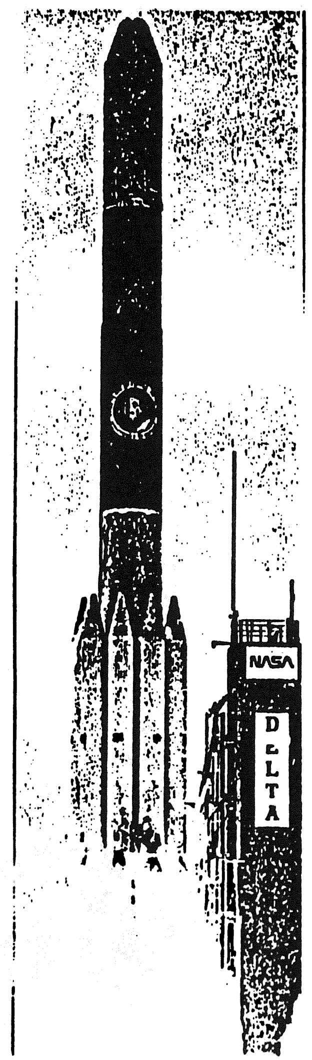
SPACE SHUTTLE



RS-27

Specifications

Type:	Liquid-Propellant/Pump-Fed
Thrust:	207,000 lb
Propellants:	RP-1 (Kerosene)/Liquid Oxygen
Specific Impulse:	262.2 sec
Mixture Ratio (O/F):	2.24:1
Chamber Pressure:	702 psia
Area Ratio:	8:1
Weight:	2261 lb
Dimensions:	142 in. long/76 in. wide

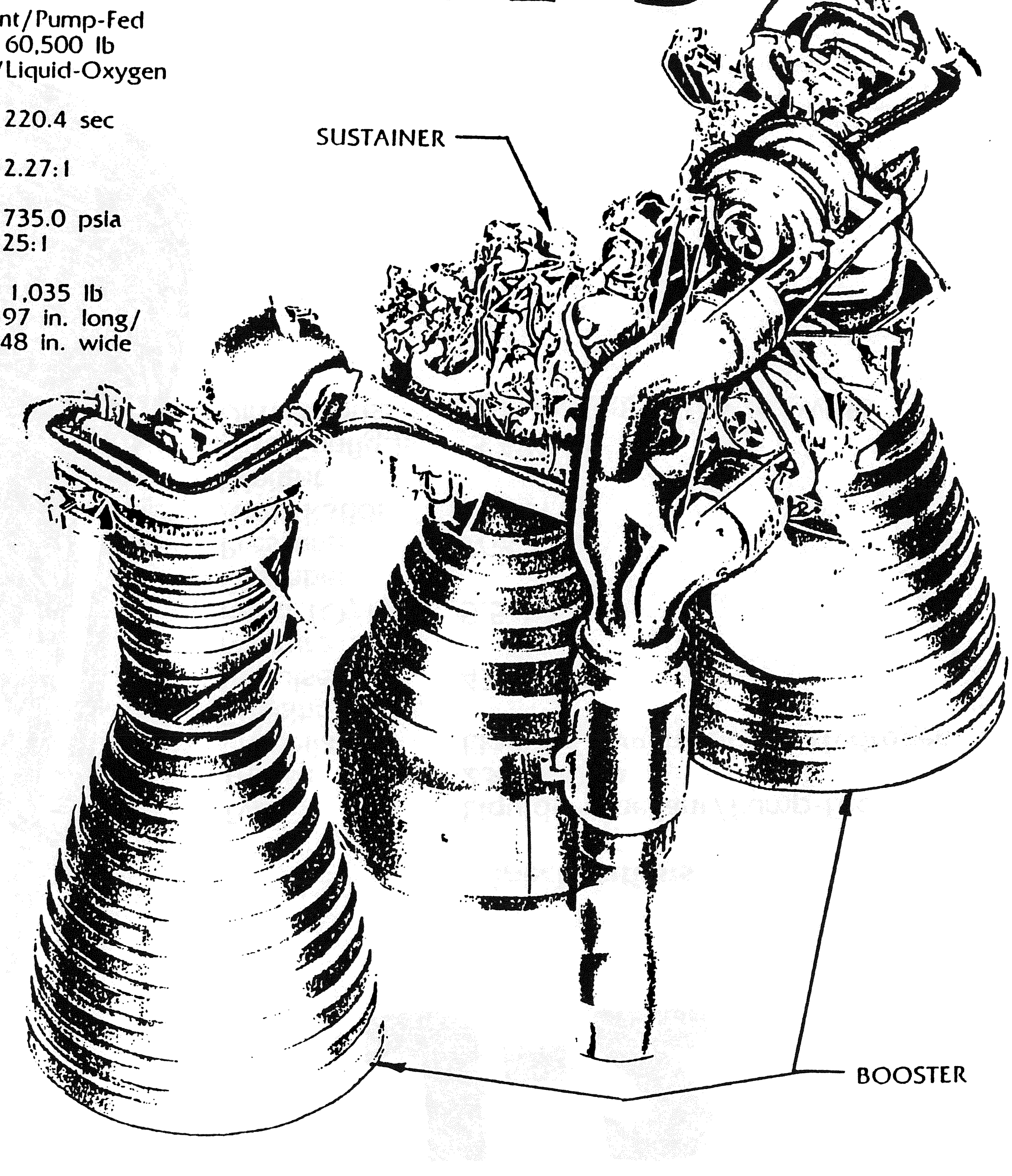


DELTA

MA-5

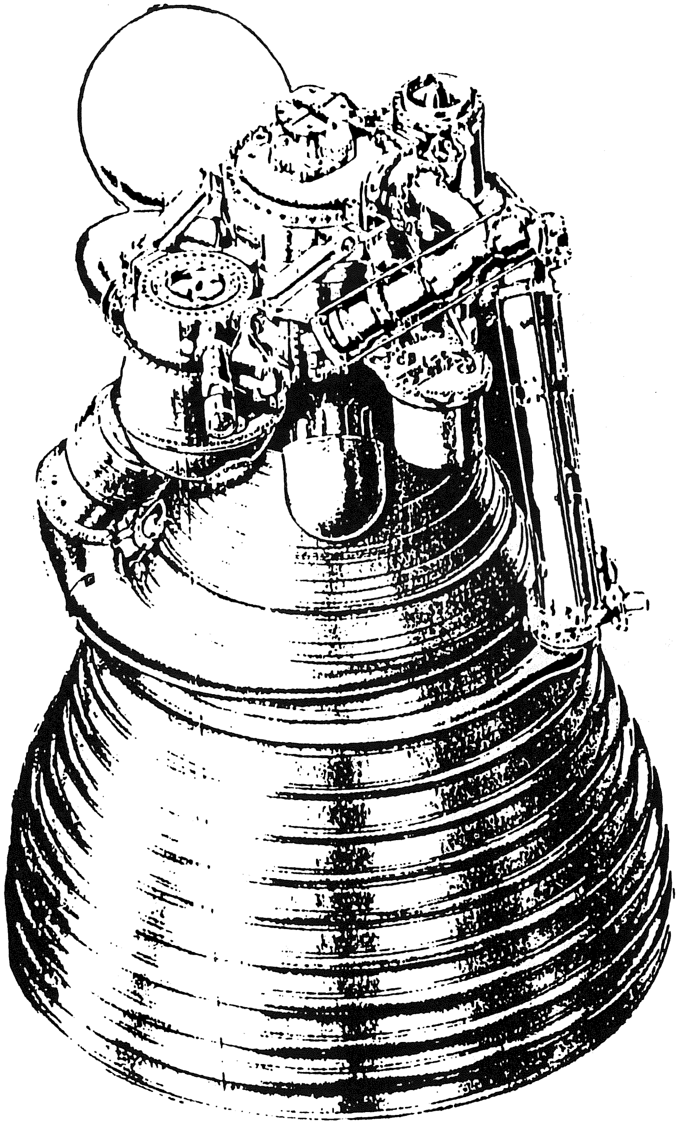
Specifications

	BOOSTER	SUSTAINER
Type:	Liquid-Propellant/Pump-Fed	
Thrust:	377,500 lb	60,500 lb
Propellants:	RP-1 (Kerosene)/Liquid-Oxygen	
Specific Impulse:	259.1 sec	220.4 sec
Mixture Ratio (O/F):	2.25:1	2.27:1
Chamber Pressure:	639.0 psia	735.0 psia
Area Ratio:	8:1	25:1
Weight (Flt. Config.):	3,140 lb	1,035 lb
Dimensions:	97 in. long/ 47 in. wide	97 in. long/ 48 in. wide



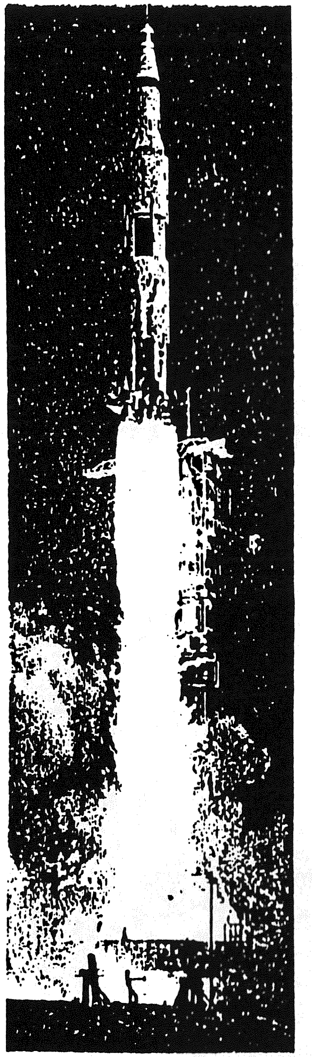

ATLAS CENTAUR

J-2

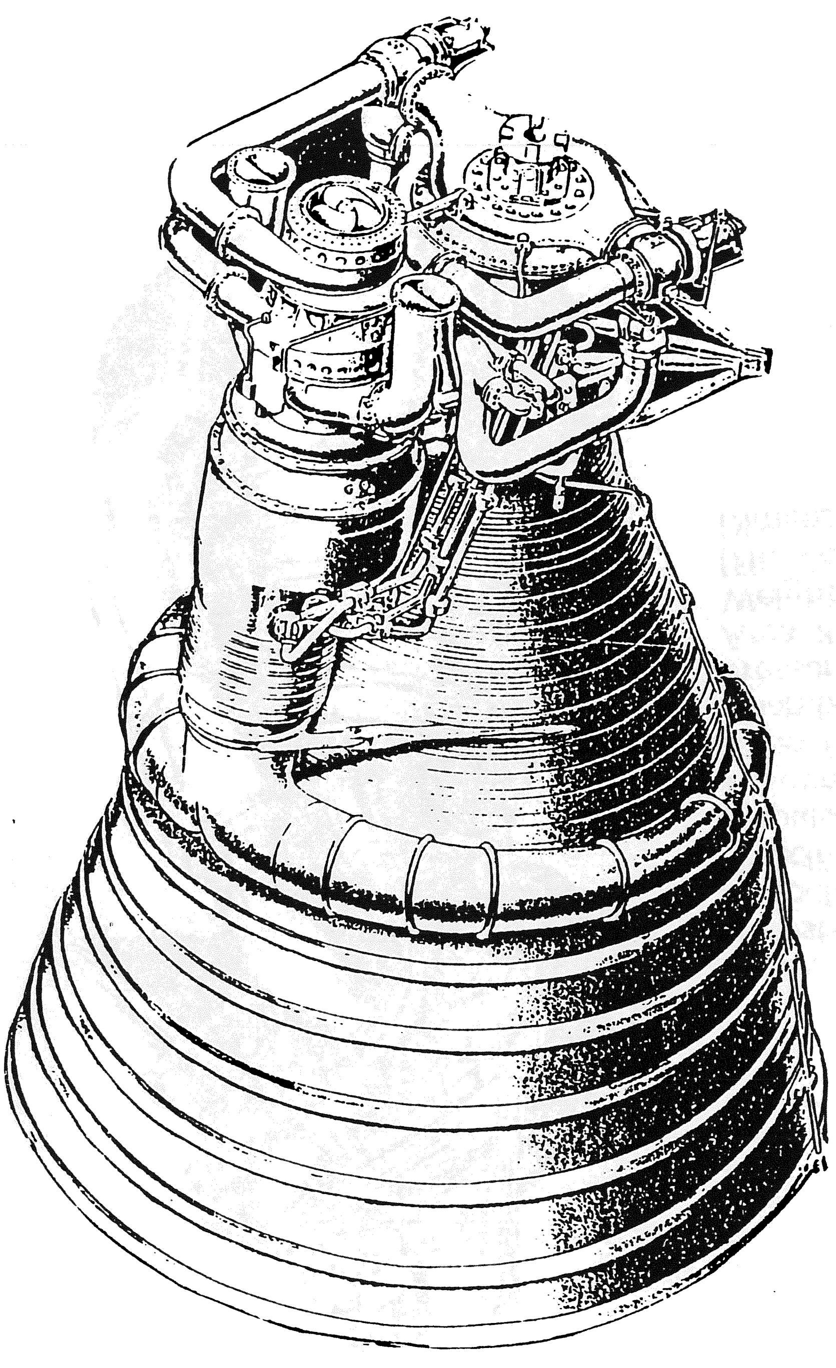


Specifications

Type:	Liquid-Propellant/Pump-Fed
Thrust:	230,000 lb
Propellants:	Liquid Oxygen/Liquid Hydrogen
Specific Impulse:	425 sec
Mixture Ratio (O/F):	5.5:1
Chamber Pressure:	763 psia
Area Ratio:	27.5:1
Weight (Flt. Config.):	3,480 lb
Dimensions:	133 in. long/80.5 in. wide



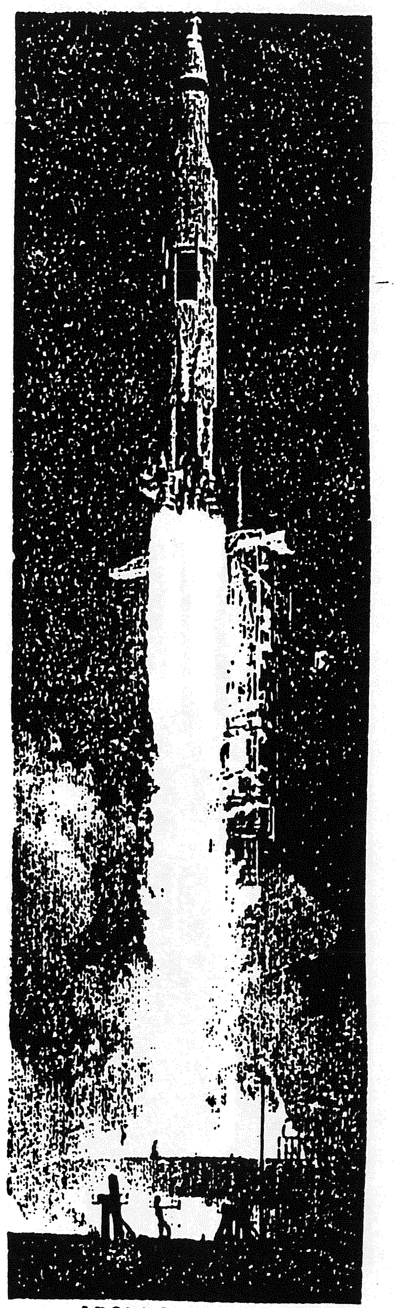
APOLLO SATURN V



F-1

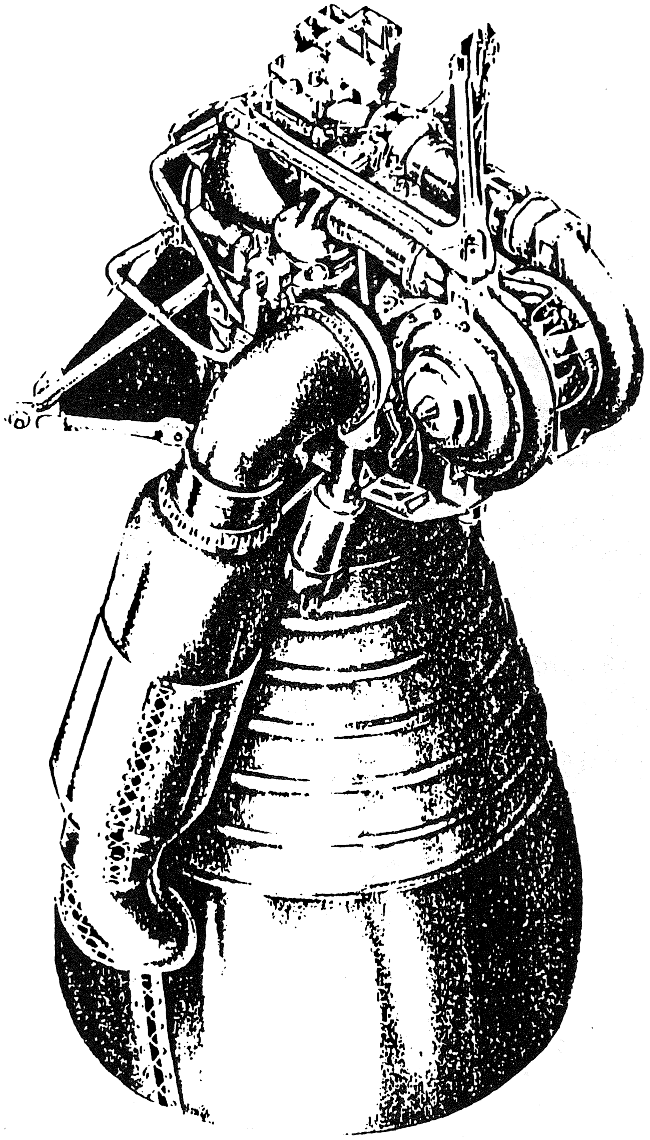
Specifications

Type:	Liquid-Propellant/Pump-Fed
Thrust:	1,522,000 lb
Propellants:	RP-1 (Kerosene)/Liquid Oxygen
Specific Impulse:	265 sec
Mixture Ratio (O/F):	2.27:1
Chamber Pressure:	982 psia
Area Ratio:	16:1
Weight (Flt. Config.):	18,616 lb
Dimensions:	220 in. long/144 in. wide



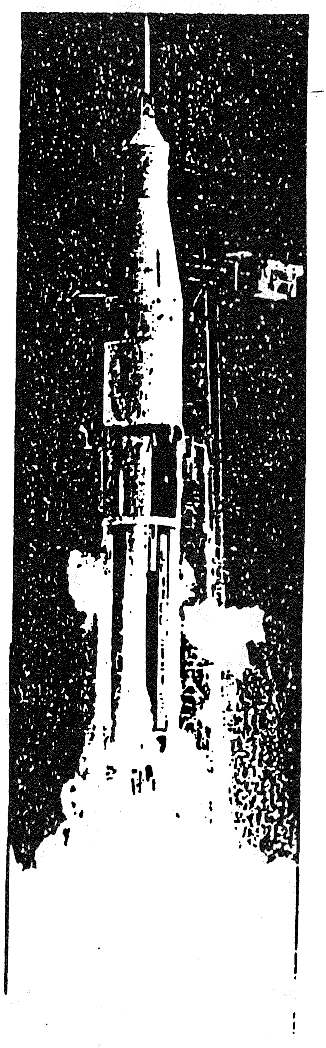
APOLLO SATURN V

H-1



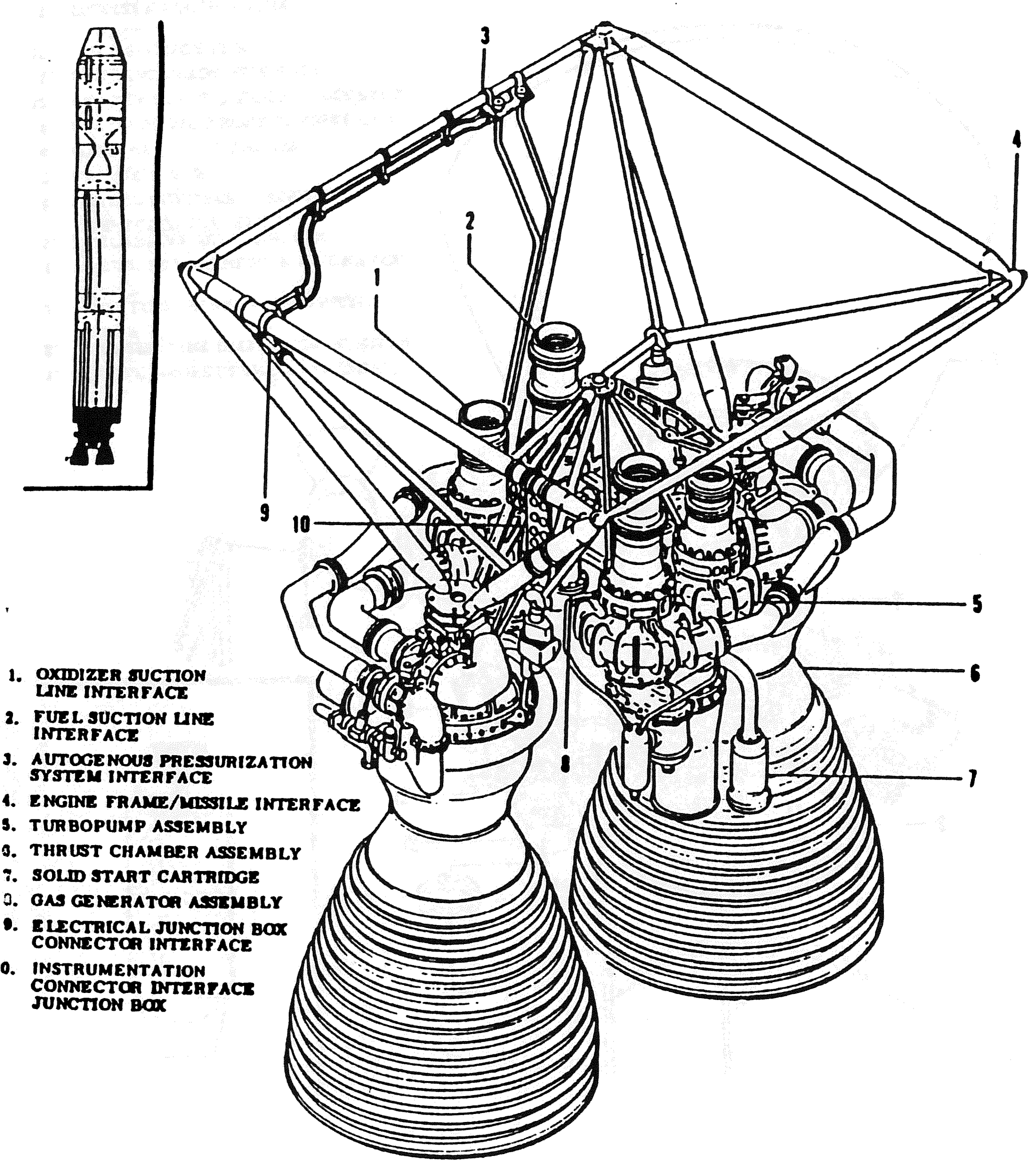
Specifications

Type:	Liquid-Propellant, Pump-Fed
Thrust:	205,000 lb
Propellants:	RP-1 (Kerosene)/Liquid Oxygen
Specific Impulse:	263 sec
Mixture Ratio (O/F):	2.23:1
Chamber Pressure:	700 psia
Area Ratio:	8:1
Weight (Flt. Config.):	2,009 lb
Dimensions:	134 in. long/66 in. wide



APOLLO SATURN 1B

YLR87–AJ–7 ENGINE ASSEMBLY – STAGE I



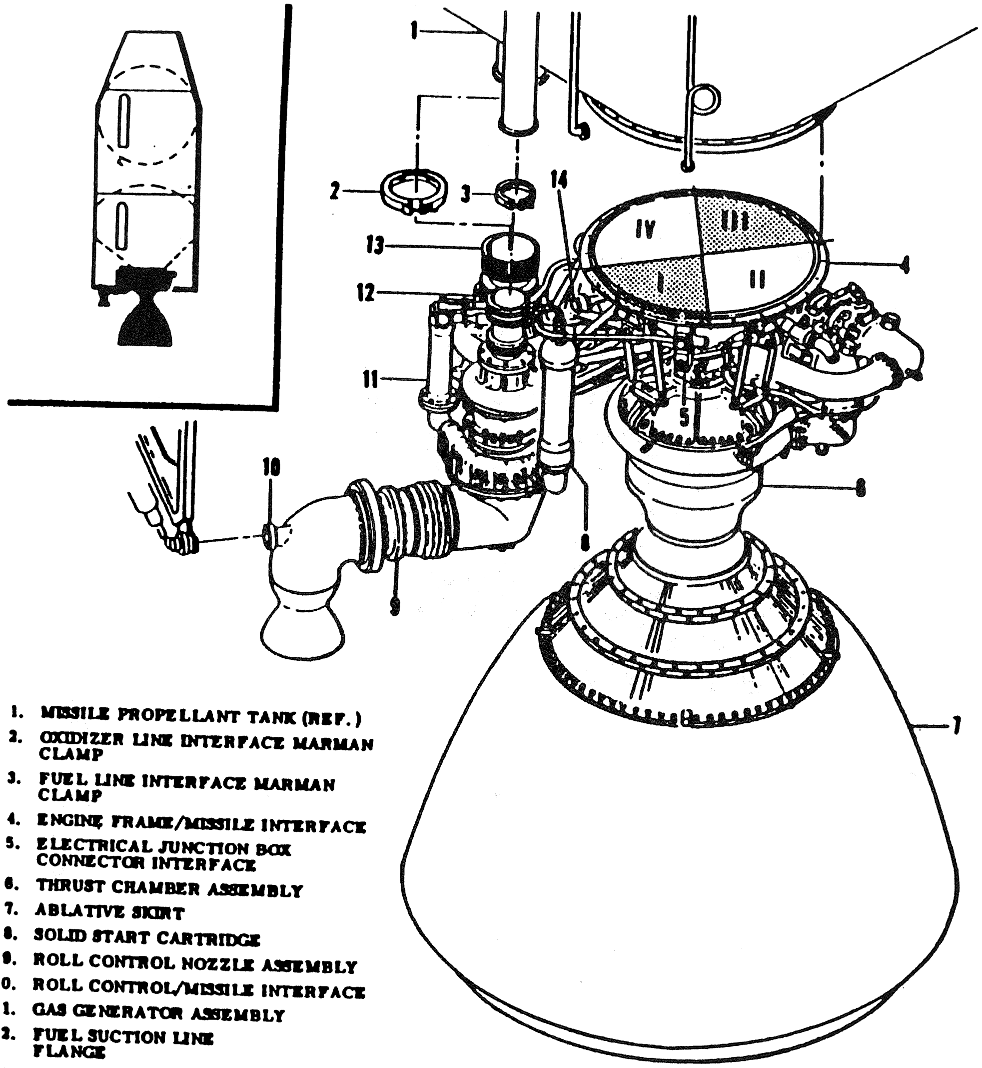
1. OXIDIZER SUCTION LINE INTERFACE
2. FUEL SUCTION LINE INTERFACE
3. AUTOGENOUS PRESSURIZATION SYSTEM INTERFACE
4. ENGINE FRAME/MISSILE INTERFACE
5. TURBOPUMP ASSEMBLY
6. THRUST CHAMBER ASSEMBLY
7. SOLID START CARTRIDGE
8. GAS GENERATOR ASSEMBLY
9. ELECTRICAL JUNCTION BOX CONNECTOR INTERFACE
10. INSTRUMENTATION CONNECTOR INTERFACE JUNCTION BOX

*SPECIFICATIONS

Type:	Liquid–Propellant/ Pump–Fed
Thrust:	546,000 lb
Propellants:	Aerozine–50/N_2O_4
Specific impulse:	301.6 sec
Nominal burn duration:	190 sec

*Titan IV (Vacuum)

YLR91—AJ—7 ENGINE ASSEMBLY — STAGE II



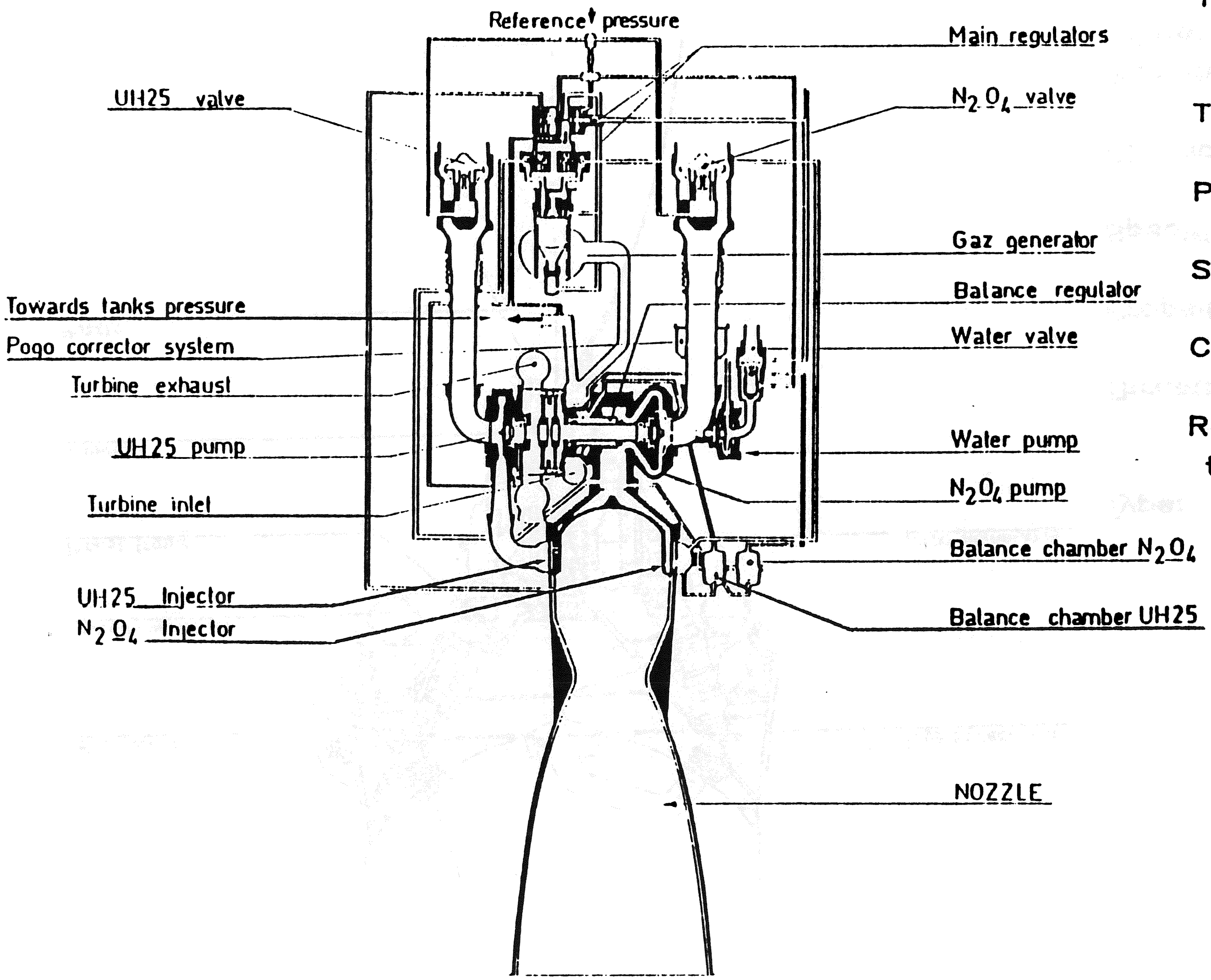
1. MISSILE PROPELLANT TANK (REF.)
2. OXIDIZER LINE INTERFACE MARMAN CLAMP
3. FUEL LINE INTERFACE MARMAN CLAMP
4. ENGINE FRAME/MISSILE INTERFACE
5. ELECTRICAL JUNCTION BOX CONNECTOR INTERFACE
6. THRUST CHAMBER ASSEMBLY
7. ABLATIVE SKIRT
8. SOLID START CARTRIDGE
9. ROLL CONTROL NOZZLE ASSEMBLY
10. ROLL CONTROL/MISSILE INTERFACE
11. GAS GENERATOR ASSEMBLY
12. FUEL SUCTION LINE FLANGE
13. OXIDIZER SUCTION LINE FLANGE
14. AUTOGENOUS PRESSURIZATION SYSTEM INTERFACE FLANGE

*SPECIFICATIONS

Type:	Liquid—Propellant/ Pump—Fed
Thrust:	103,320 lb
Propellants:	Aerozine—50/N_2O_4
Specific Impulse:	316.6 sec
Nominal burn duration:	232 sec

*Titan IV (Vacuum)

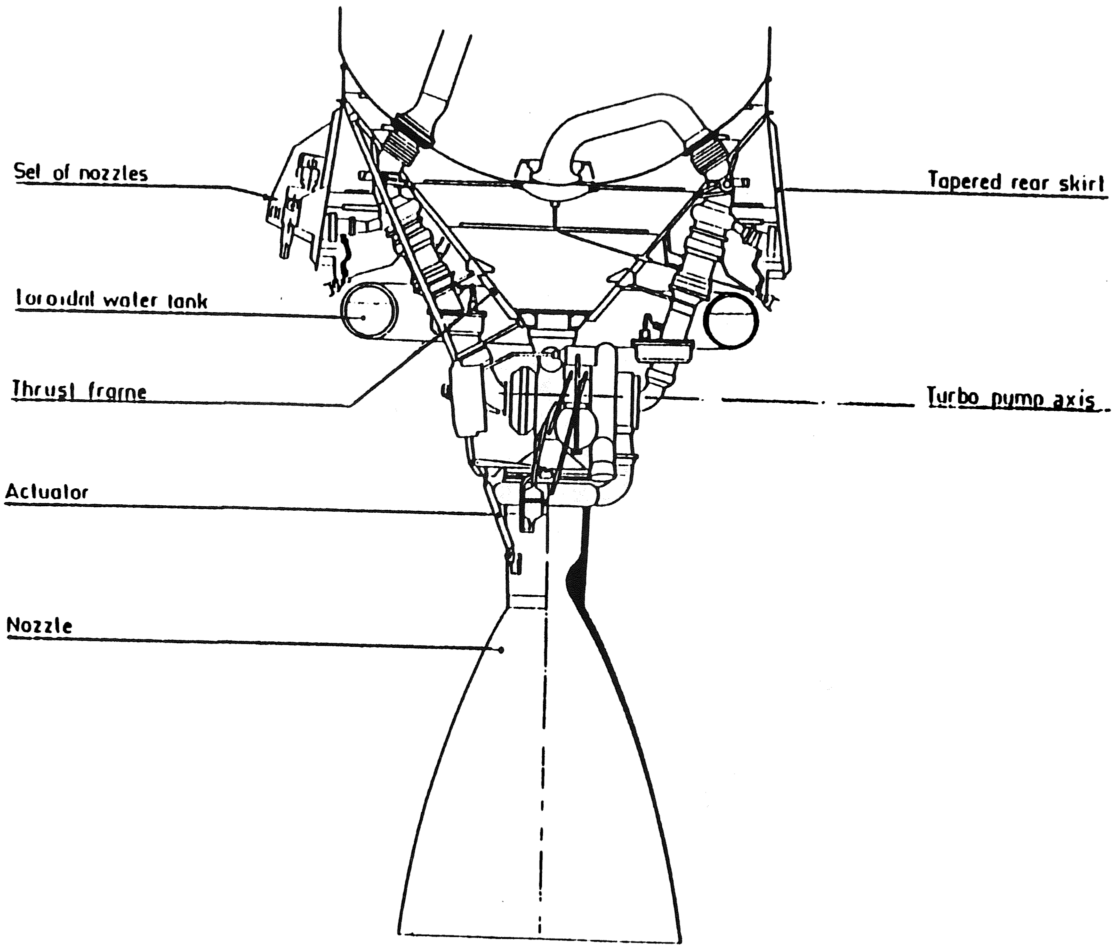
VIKING V ENGINE



SPECIFICATIONS

Type:	Liquid–Propellant, Pump–Fed
Thrust:	149,500 lb
Propellants:	UH25/N_2O_4
Specific impulse:	277 sec
Chamber pressure:	848 psia
Run duration time:	135 sec

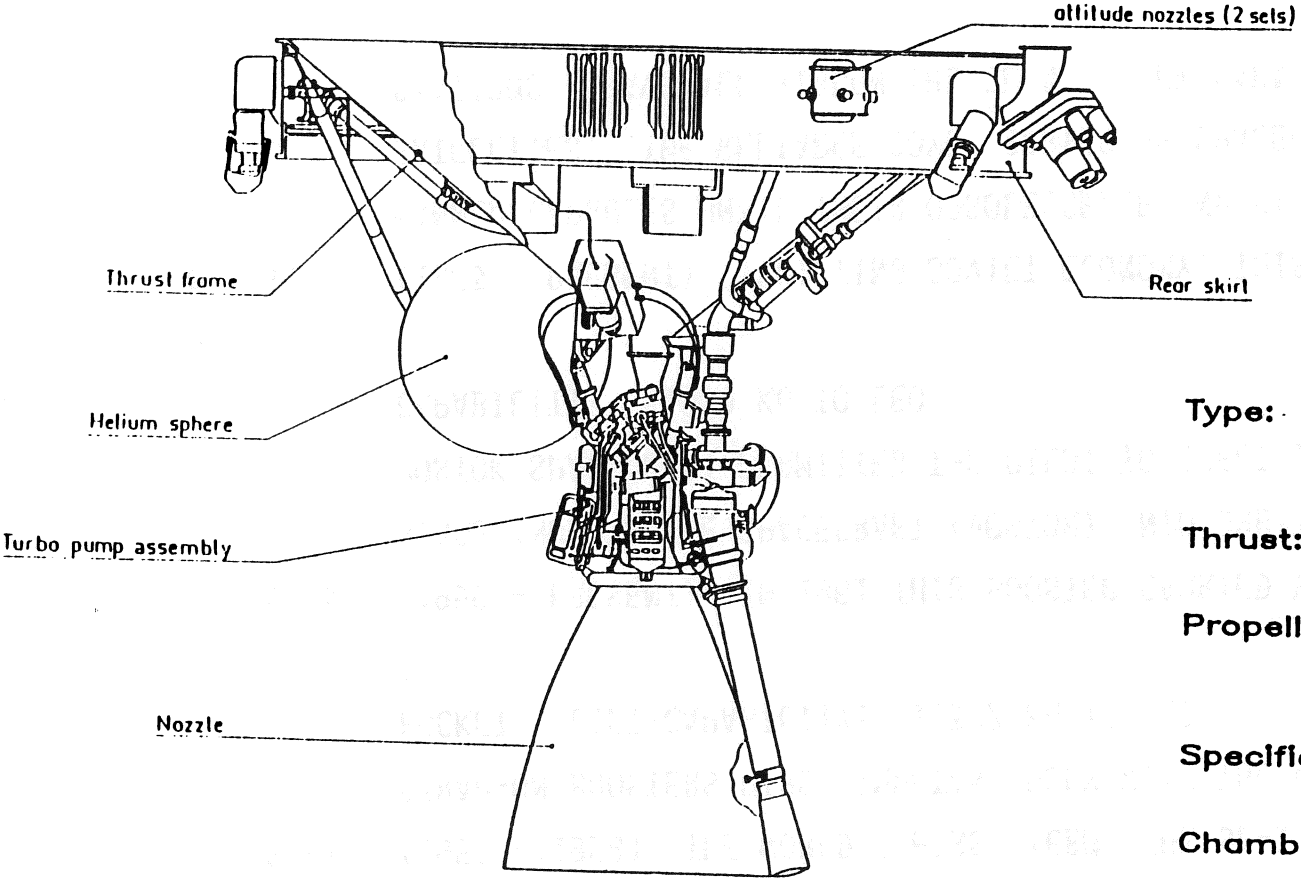
VIKING IV ENGINE



SPECIFICATIONS

Type:	Liquid–Propellant/ Pump–Fed
Thrust:	176,500 lb
Propellants:	UH25/N_2O_4
Specific impulse:	291 sec
Chamber pressure:	848 psia
Run duration time:	130 sec

HM7-B



SPECIFICATIONS

Type:	Liquid–Propellant/ Pump–Fed
Thrust:	14,160 lb
Propellants:	Liquid Hydrogen/ Liquid Oxygen
Specific impulse:	444 sec
Chamber pressure:	507 psia
Run duration time:	720 sec

SOVIET LAUNCH VEHICLES

TO ORBIT MORE THAN A HUNDRED PAYLOADS A YEAR, THE SOVIETS EMPLOY PROVED BOOSTERS. ALL BUT TWO, THE PROTONS, DERIVE FROM BALLISTIC MISSILES; ALL BURN LIQUIDS.

- SL-1 (1957 - 1958) THE WORLD'S FIRST ICBM, THE SL-1 LAUNCHED SPUTNIK IN 1957. FOUR STRAP-ON BOOSTERS FIRE SIMULTANEOUSLY WITH THE CORE ENGINE IN THIS SINGLE-STAGE ROCKET. LIFT CAPABILITY: 1,327 KG TO LEO

- SL-3 (1959 - PRESENT) IN 1961 THIS BOOSTER CARRIED YURI GARGARIN ALOFT. THE SECOND STAGE INJECTS THE SPACECRAFT (VOSTOK) INTO ORBIT. THE CIRCULAR HATCH IN THE VOSTOK SPACECRAFT PERMITTED THE PILOT TO EJECT IN AN EMERGENCY. LIFT CAPABILITY: 6,000 KG TO LEO

- SL-4 (1963 - PRESENT) TYPIFYING SOVIET ECONOMY, THIS MODIFIED SL-1 LAUNCHED VOSKHOD MANNED CAPSULES UNTIL THEIR OBSOLESCENCE AND STILL CARRIES RECONNAISSANCE SATELLITES. THE RELIABLE SOYUZ SERIES OF SPACECRAFT - TAXIS TO THE SPACE STATIONS - HAVE ALL RIDDEN THE SL-4. LIFT CAPABILITY: 7,500 KG TO LEO

- SL-6 (1960 - PRESENT) A THIRD STAGE GIVES EXTRA LIFT FOR LAUNCHING COMMUNICATIONS, EARLY WARNING, AND PLANETARY PAYLOADS. SIMILAR TO SL-3 VEHICLE. LIFT CAPABILITY: 2,100 KG TO ELLIPTICAL ORBIT

- SL-8 1964 - PRESENT) ADAPTED FROM INTERMEDIATE BALLISTIC MISSILE, THE SL-8 IS COMPARABLE TO THE U.S. THOR-DELTA. A SPECIAL PAYLOAD DESIGN ENABLES THE SL-8 TO LAUNCH EIGHT COMMUNICATIONS SATELLITES AT ONCE. LIFT CAPABILITY: 1,700 TO LEO

- SL-11 (1966 - PRESENT) ADAPTED FROM LONG-RANGE BALLISTIC MISSILE, THE SL-11 IS COMPARABLE TO THE ATLAS-CENTAUR. THE SL-11 LOFTS SURVEILLANCE SATELLITES AND OPERATIONAL ANTI-SATELLITE WEAPONS. LIFT CAPABILITY: 4,000 KG TO LEO

- SL-14 (1977 - PRESENT) ADAPTED FROM LONG-RANGE BALLISTIC MISSILE, THE SL-14 IS ALSO COMPARABLE TO THE ATLAS-CENTAUR. SATELLITES FOR METEOROLOGY, GEODESY, COMMUNICATIONS AND ELECTRONIC INTELLIGENCE GATHERING RIDE THE SL-14 INTO ORBIT. LIFT CAPABILITY: 5,500 KG TO LEO

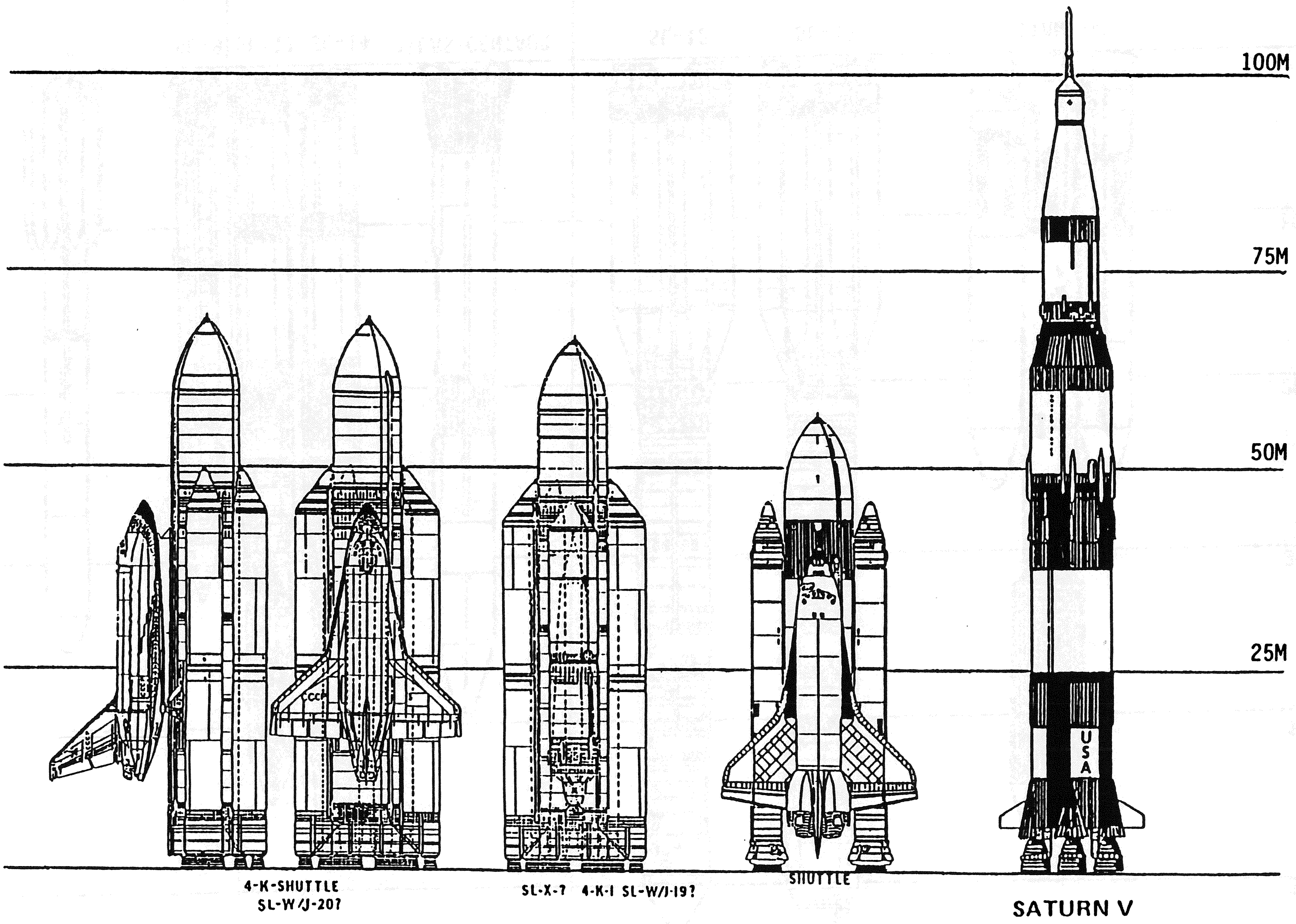
- SL-12 (1967 - PRESENT) MIGHTIEST OF OPERATIONAL SOVIET ROCKETS, THE PROTONS HAVE BEEN ADVERTISED FOR COMMERCIAL LAUNCHES OF SATELLITES. THE SL-12 HAS FOUR STAGES. LIFT CAPABILITY: 2,100 KG TO GEOSTATIONARY ORBIT

- SL-13 (1968 - PRESENT) WITH ABOUT A SIXTH THE LIFT OF AMERICA'S DISCONTINUED SATURN V MOON ROCKET THE THREE-STAGE PROTON OUTLIFTS THE U.S. TITAN 34-D. THE SALYUT AND MIR SPACE STATIONS RODE THIS MODIFIED PROTON INTO ORBIT. LIFT CAPABILITY: 19,500 KG TO LEO

- SL-16 (PRESENTLY IN DEVELOPMENT) A MEDIUM LIFT BOOSTER NOW BEING TESTED. THE SL-16 FIRST STAGE HAS LOX/KEROSENE STRAP-ON RECOVERABLE BOOSTERS.

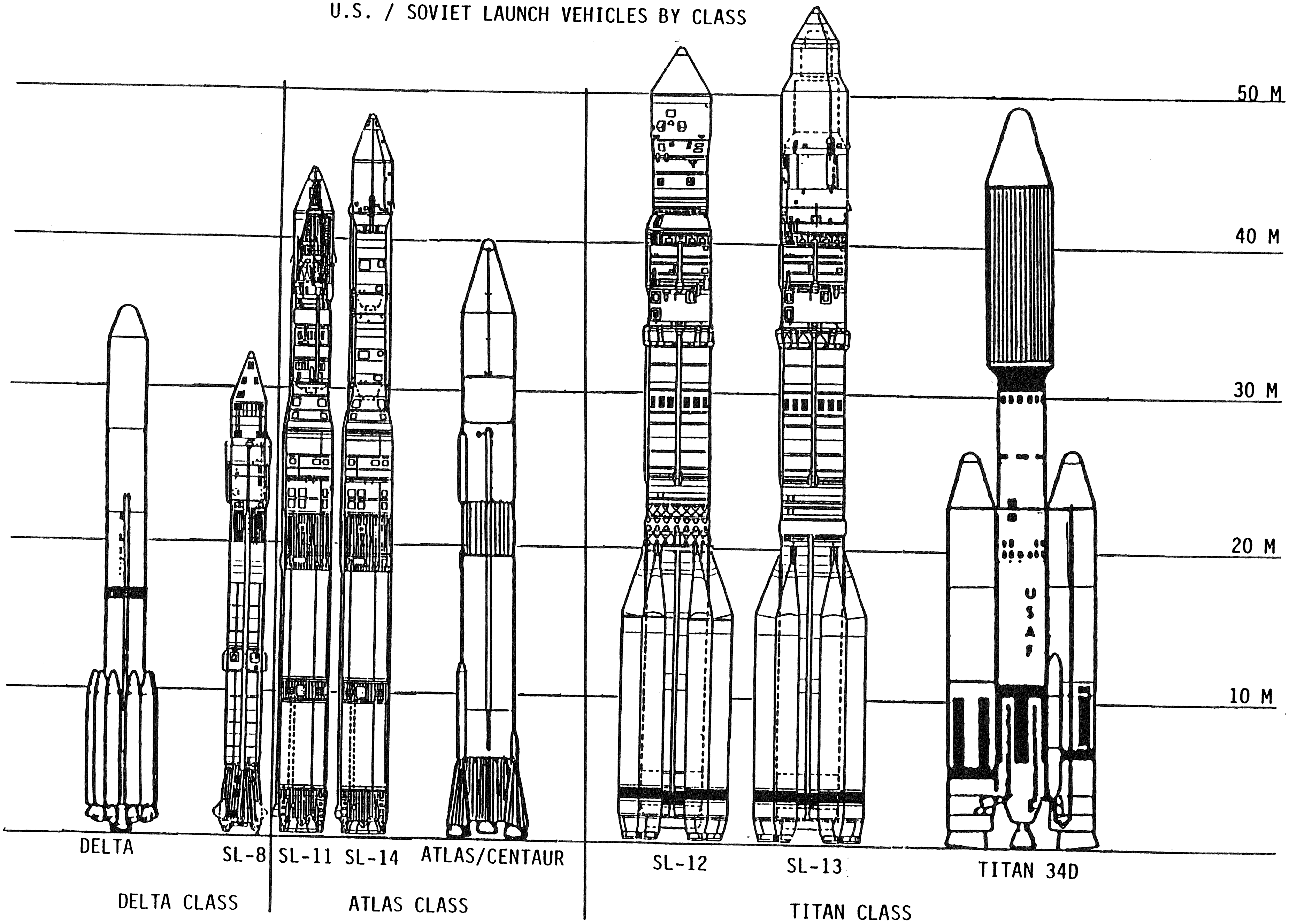
- SLW-1 (PRESENTLY IN DEVELOPMENT) THE SLW-1 ENERGIA HEAVY-LIFT LAUNCH VEHICLE WAS FIRST TESTED ON MAY 15, 1987. COMPARABLE TO THE SATURN V, ONE ENERGIA LAUNCH IS EQUIVALENT TO FIVE PROTON SL-13 OR 15 SL-4 LAUNCHES. THE FIRST STAGE CAN ACCOMMODATE 4-8 LOX/KEROSENE STRAP-ON RECOVERABLE BOOSTERS OF THE SL-16 TYPE. CORE STAGE FUELS THE SOVIETS FIRST LIQUID OXYGEN/LIQUID HYDROGEN ENGINE. THE ENERGIA HAS THE CAPABILITY TO LAUNCH HEAVY MILITARY PAYLOADS, LARGE SPACE STATION ELEMENTS AND THE SOVIET SPACE SHUTTLE. LIFT CAPABILITY: 100,000 KG TO LEO

U.S. / SOVIET LAUNCH VEHICLES BY CLASS (CONT'D)



HEAVY LIFT CLASS

U.S. / SOVIET LAUNCH VEHICLES BY CLASS



ROCKET ENGINE CYCLES
LIQUID & SOLID ROCKET MOTORS

J. Paul Spinn

Propulsion Systems Analysis

OVERVIEW

- **PERFORMANCE CONSIDERATIONS**
  - **NOMENCLATURE**
  - **CHAMBER NOZZLE**
  - **ROCKET CONVENTION**
  - **TYPES OF ROCKET ENGINES**
- CHEMICAL ROCKET SYSTEMS
- ROCKET ENGINE CYCLES

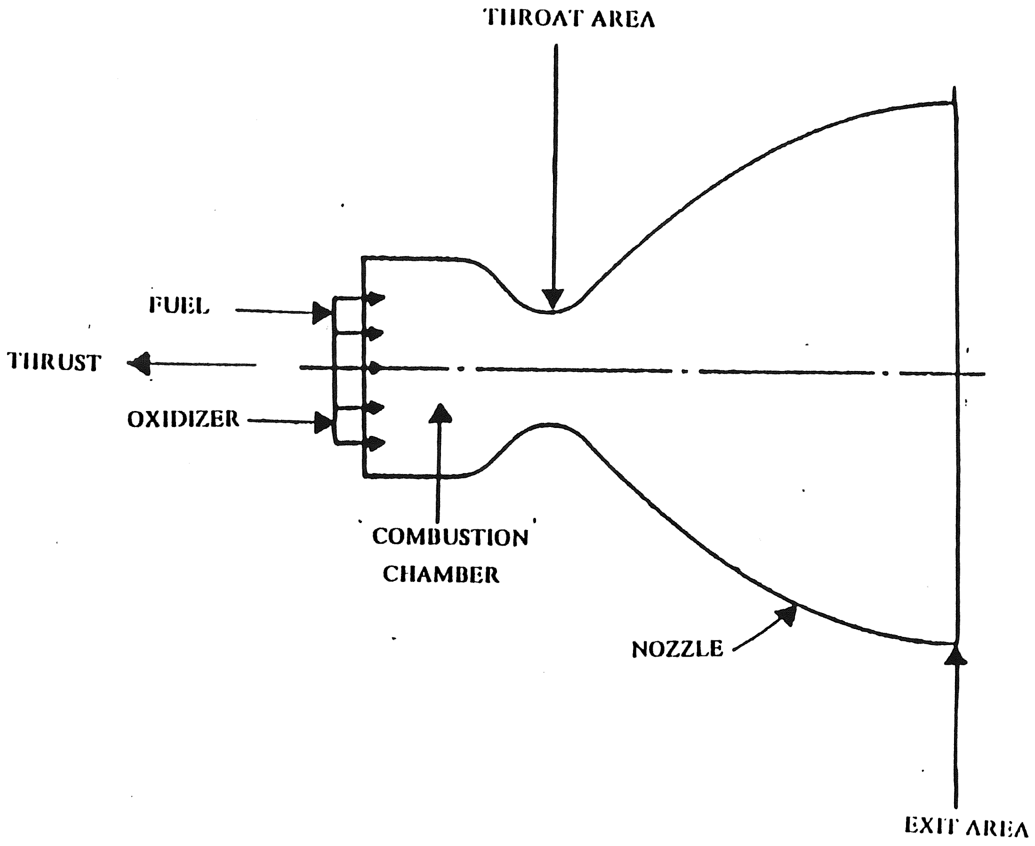
PERFORMANCE CONSIDERATIONS

NOMENCLATURE

- SPECIFIC IMPULSE
- THRUST
- MIXTURE RATIO
- OXIDIZER FRACTION
- BULK DENSITY
- CHAMBER PRESSURE
- AREA RATIO

PERFORMANCE CONSIDERATIONS

Typical Chamber Nozzle Configuration



PERFORMANCE CONSIDERATIONS

NOMENCLATURE

- **SPECIFIC IMPULSE, Isp**
  - Defined as $\dfrac{\text{THRUST}}{\text{TOTAL PROPELLANT FLOW}}$
  - With units of $\dfrac{\text{LBF}}{\text{LBM / SEC}}$
  - Often quoted in seconds
  - Specific impulse is the inverse of specific fuel consumption (SFC). However, SFC does not include the oxidizer flow. Generally, the higher the specific impulse the better.

PERFORMANCE CONSIDERATIONS

NOMENCLATURE

- THRUST, F
  - Defined as MASS FLOW * VELOCITY + PRESSURE * AREA
  - With units of LBF
- MIXTURE RATIO, MR
  - Defined as $\dfrac{\text{OXIDIZER FLOW}}{\text{FUEL FLOW}}$
  - Since this is ratio of masses, it is dimensionless.
  - Mixture ratio is the inverse of the fuel-to-air ratio.

PERFORMANCE CONSIDERATIONS

NOMENCLATURE

- **OXIDIZER FRACTION, OFR**
  - Defined as $\dfrac{\text{OXIDIZER FLOW}}{\text{TOTAL PROPELLANT FLOW}}$
  - Since this is ratio of masses, it is dimensionless.
  - Oxidizer fraction is useful numerically since it inherently stable since it range is from 0 to 1.
- **BULK DENSITY**
  - Overall Bulk Density at the engine inlet. The higher the bulk density is the better.

PERFORMANCE CONSIDERATIONS

NOMENCLATURE

- **CHAMBER PRESSURE, Pc**
  - This is the pressure in the "main" combustion chamber. It is often referred to as "P_c" and is synonymous with turbine exit pressure in the jet motor world.

PERFORMANCE CONSIDERATIONS

NOMENCLATURE

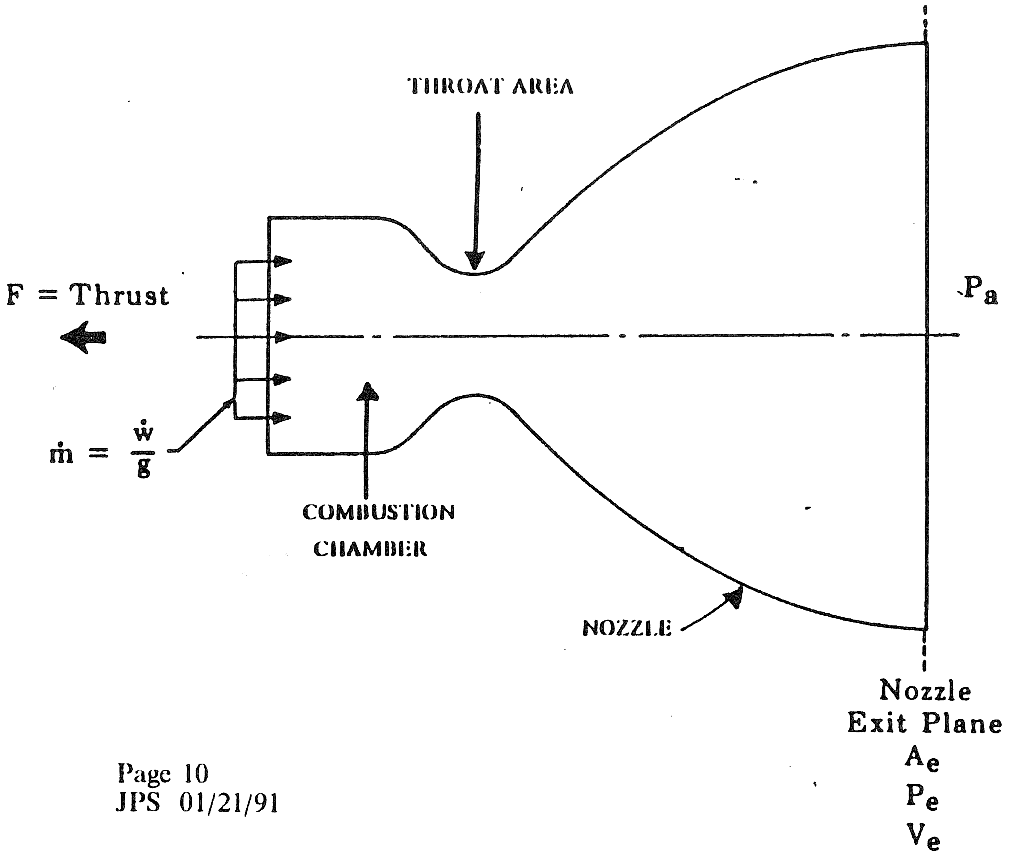
- **AREA RATIO**
  - Define as: $\frac{\text{NOZZLE EXIT AREA}}{\text{NOZZLE THROAT AREA}}$
  - Often referred to as "ε"
  - "ε" is generally greater than that used on jet engines due to the higher characteristic chamber pressures found in rockets (ie. SSME ATD has Pc of 3000 psia at 100 % RPL)

PERFORMANCE CONSIDERATIONS

Chamber Nozzle

$$F = \dot{m} V_e + (P_e - P_a)A_e$$

Total thrust = momentum thrust + pressure thrust



$\dot{m}$ = mass rate of flow = $\frac{\dot{w}}{g}$

V_e = nozzle exit velocity, ft/sec

P_e = nozzle exit pressure, psi

P_a = ambient pressure, psi

A_e = area at the nozzle exit plane, ft^2

$\dot{w}$ = weight flowrate of propellants, lb/sec

g = gravitational constant, ft/sec^2

Using the law of conservation of energy,

$$I_s = \sqrt{2\frac{J}{g}(h_c - h_e)} = \sqrt{\frac{2J\Delta H}{\overline{M}}}$$

Assuming isentropic expansion of ideal gases

$$I_s = \sqrt{\frac{2\gamma}{\gamma - 1}\left(\frac{RT_c}{g\overline{M}}\right)\left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma - 1}{\gamma}}\right]}$$

where:

h_c = specific enthalpy of propellant gases within the rocket chamber, Btu/lb

h_e = specific enthalpy of gases after discharge from the rocket nozzle, Btu/lb

J = mechanical equivalent of heat, 778 ft-lb/Btu

H = heat per mole released during expansion

R = universal gas constant, 1544 ft-lb/mole°R

g = gravitational constant, 32.2 ft/sec^2

T_c = chamber temperature, °R

P_e = exhaust pressure, psi

P_c = chamber pressure, psi

$\overline{M}$ = mean molecular weight, lb/mole.

γ = ratio of the specific heats

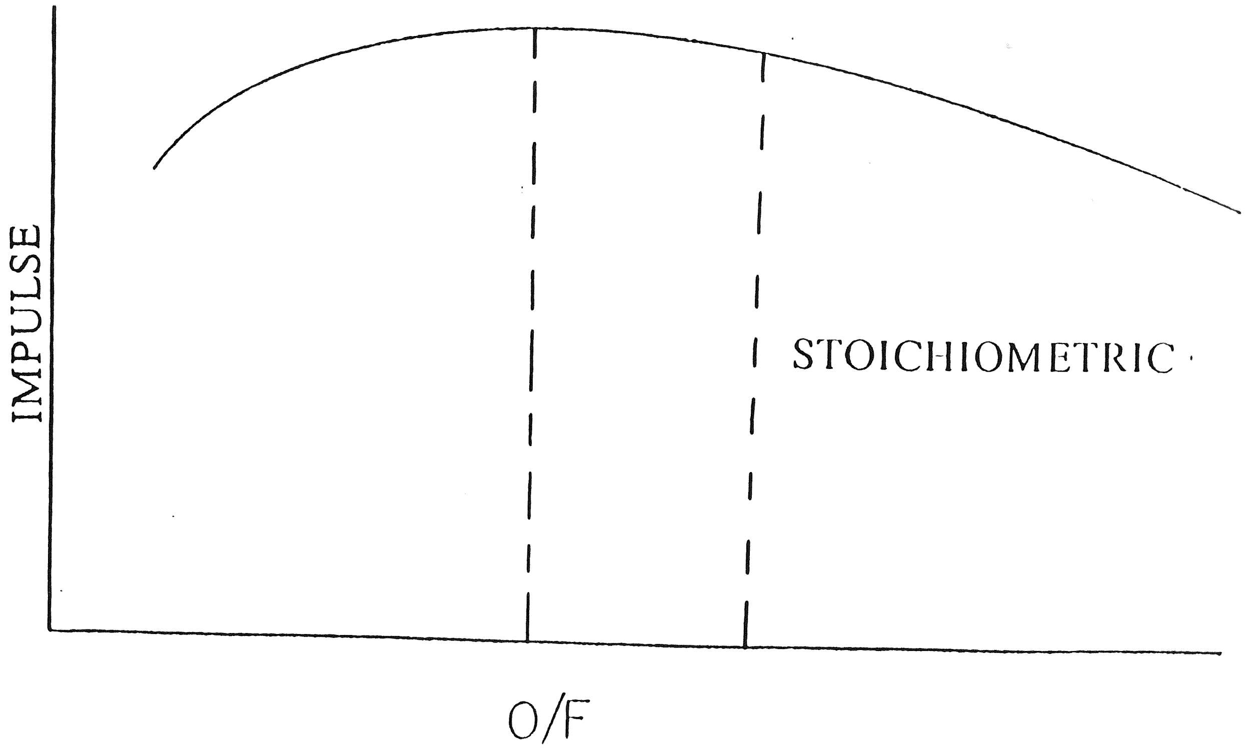
PERFORMANCE CONSIDERATIONS

Rocket Conventions

- ROCKET ENGINES NOMINALLY OPERATE FUEL RICH (O/F'S LESS THAN STOICHIOMETRIC)
- HIGHER IMPULSES SEEN RELATIVE TO JETS
- HIGHER GAS VELOCITIES WITH LIGHTER MOLECULES

$$VEL = \sqrt{\gamma g \frac{1545}{M} T}$$

M = Molecular Weight



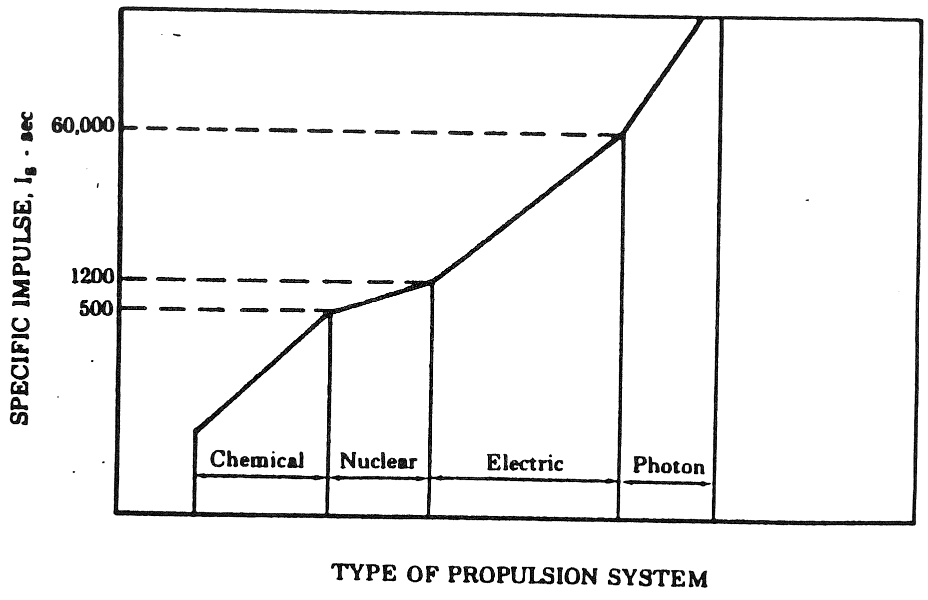
PERFORMANCE CONSIDERATIONS

Types of Rocket Engines

- **ELECTRIC**
- **NUCLEAR**
- **CHEMICAL**

PERFORMANCE CONSIDERATIONS

Types of Rocket Engines - Impulse Comparison



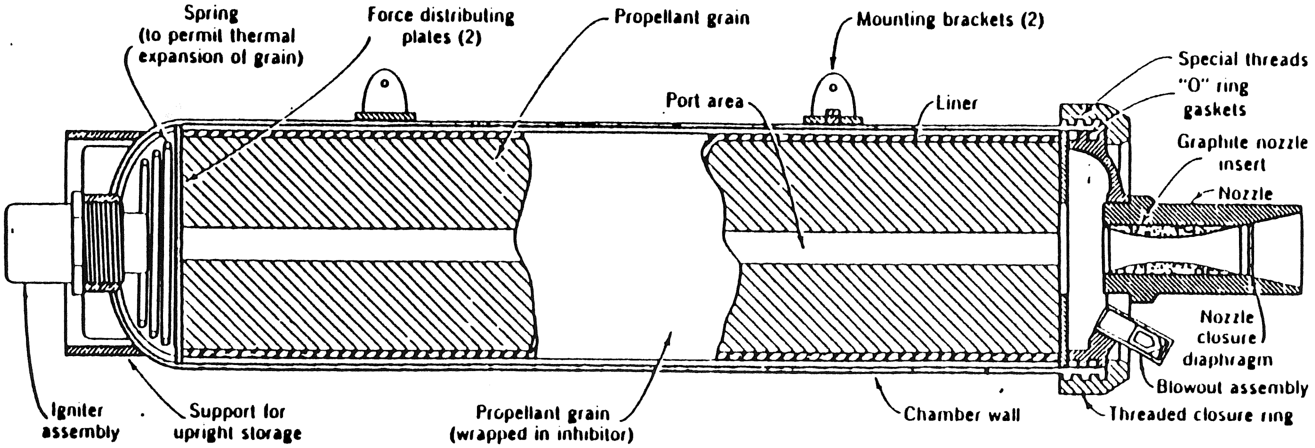
OVERVIEW

- PERFORMANCE CONSIDERATIONS
- **CHEMICAL ROCKET SYSTEMS**
  - **SOLID PROPELLANT ROCKETS**
  - **LIQUID PROPELLANT ROCKETS**
- ROCKET ENGINE CYCLES

CHEMICAL ROCKET SYSTEMS

Solid Propellant Rockets - Five Main Components

- CHAMBER
- NOZZLE
- PROPELLANT GRAIN
- PROPELLANT CORE OR PORT GEOMETRY
- IGNITOR



CHEMICAL ROCKET SYSTEMS

Solid Propellant Rockets - Desirable Propellant Properties

- HIGH CHEMICAL RELEASE PROMOTING HIGH PERFORMANCE (ie. SPECIFIC IMPULSE)
- LOW MOLECULAR WEIGHT FOR HIGHER SPECIFIC IMPULSE
- STABILITY AND STORAGE CONSIDERATIONS
- SHOULD BE UNAFFECTED BY ATMOSPHERIC CONDITIONS
- MILITARY CONSIDERATIONS : SMOKELESS AND NONLUMINOUS
- RESISTANT TO AUTO-IGNITION FROM OUTSIDE HEAT SOURCES OR IMPACT
- FABRICATION AND RAW MATERIAL CONSIDERATIONS

CHEMICAL ROCKET SYSTEMS

Solid Propellant Rockets - Burning Rate

- **DEFINITION OF BURNING RATE : VELOCITY AT WHICH PROPELLANT IS CONSUMED NORMAL TO BURNING SURFACE**
- **APPEARS IN THE FLOWRATE CALCULATION :**

$$W \equiv A_b(R_b)(\rho_p)$$

WHERE :

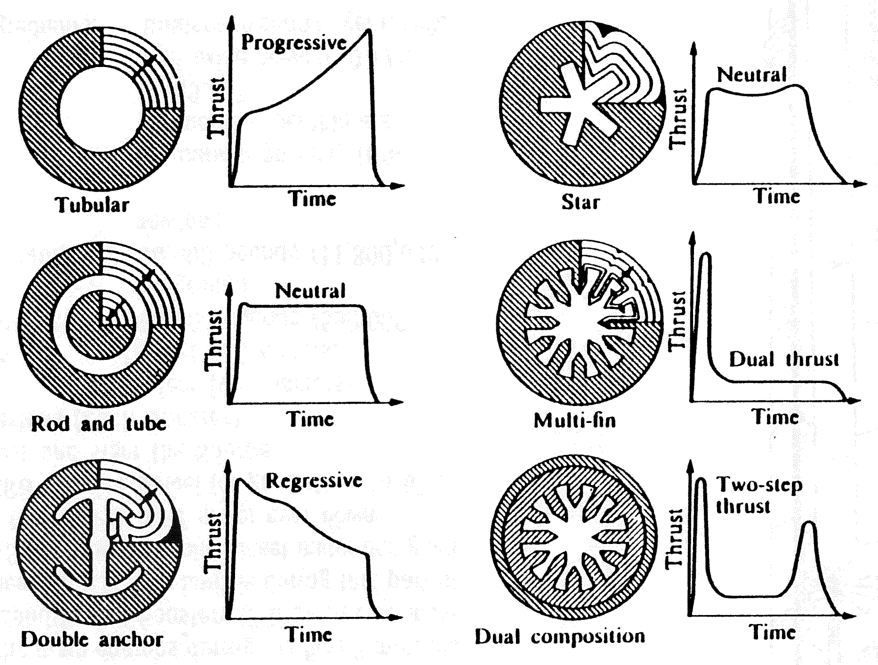
A_b IS THE BURNING AREA

R_b IS THE BURNING RATE

ρ_p IS THE PROPELLANT DENSITY

CHEMICAL ROCKET SYSTEMS

Solid Propellant Rockets - Typical Grain-Port Configurations



CHEMICAL ROCKET SYSTEMS

Solid Propellant Rockets - Space Shuttle SRB's

Solid Rocket Boosters

Two large solid-propellant rockets operate in parallel with the main engines during the first 2 minutes of flight. The Solid Rocket Boosters (SRBs) provide most of the power needed to lift the Shuttle during this part of the flight. The SRB is the first solid rocket motor designed for reuse and the largest solid motor ever flown.

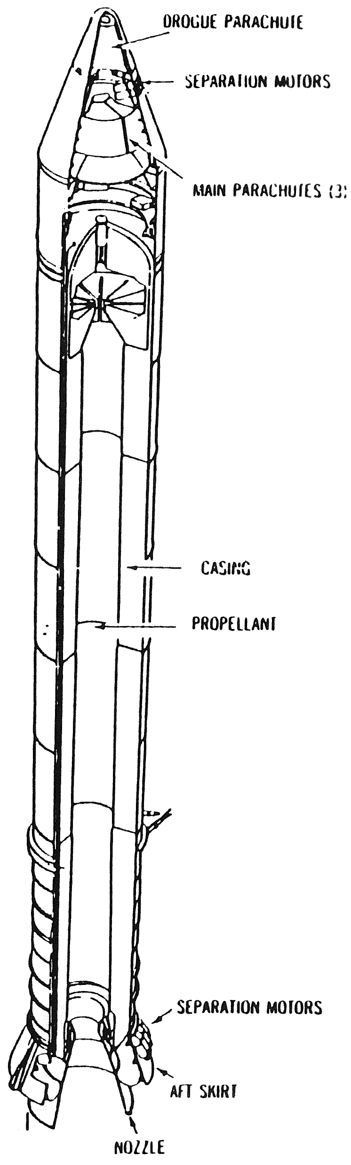
The SRB **nozzles** swivel (or **gimbal**) up to 6° to direct the thrust and steer the Shuttle.

Overall Dimensions (Each Booster)

Length:	149 feet (45.5 meters)
Diameter:	12 feet (3.6 meters)
Weight at launch:	1,300,000 pounds (590,000 kilograms)
Thrust at launch:	2,650,000 pounds (11,800,000 newtons)

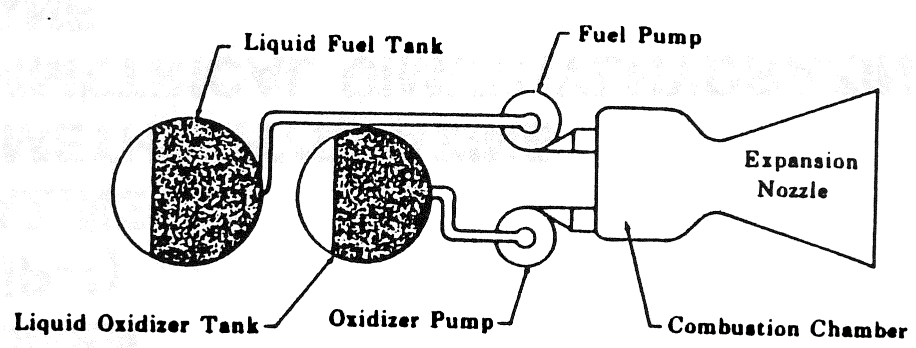
Propellants

Fuel:	Aluminum powder, 16%
Oxidizer:	Ammonium perchlorate, 69.83%
Catalyst:	Iron oxide powder, 0.17%
Other ingredients:	Binder and curing agent, 14%



CHEMICAL ROCKET SYSTEMS

Liquid Propellant Rockets



THE LIQUID ROCKET ENGINE USES TWO OR MORE PROPELLANTS. THESE PROPELLANTS ARE PUMPED FROM THEIR RESPECTIVE TANKS TO HIGH PRESSURE AND THEN INJECTED INTO A COMBUSTION CHAMBER WHERE THEY ARE BURNED AND EXPANDED TO HIGH VELOCITIES THROUGH A NOZZLE TO PROVIDE THRUST.

CHEMICAL ROCKET SYSTEMS

Liquid Propellant Rockets - Common Oxidizers and Fuels

- **OXIDIZERS**

  OXYGEN
  FLUORINE
  NITROGEN-TETROXIDE

- **FUELS**

  HYDROGEN
  RP-1 (JP-7)
  HYDRAZINE
  MONOMETHYLHYDRAZINE
  UNSYMMETRICAL DIMETHYLHYDRAZINE
  METHANE

Liquid Propellant Rockets - Liquid Propellant Properties

Fuel	Oxidizer	Best oxidizer–fuel mixture ratio (by weight)	Theoretical combustion temperature, °F	Ratio of specific heats	Average molecular weight of combustion products	Bulk density, 80°F propellant combination temperature, gm/cm³	Specific impulse, 400 psia, sea-level expansion, sec	Storability²	
								Fuel	Oxidizer
Ammonia	Oxygen	1.3	4940	1.23	19	0.88	255	F-P	F-P
Ammonia	RFNA (22% NO_2)	2.15	4220	1.24	21	1.12	230	F-P	G
Diborane	Fluorine	5	7880	1.3	21	1.07	270	G	F-P
Diethylenetriamine	Oxygen	1.5	6500	1.24	21	1.06	245	G	F-P
Diethylenetriamine and hydrazine	Oxygen	2.5	6000	1.22	22		245	G	F-P
Hydrazine (anhydrous)	Chlorine trifluoride	2.4	6000	1.33	23	1.46	255	G	F-P
Hydrazine (anhydrous)	Fluorine	2	7740	1.33	19	1.3	290	G	F-P
Hydrazine (anhydrous)	H_2O_2 (90%)	1.5	4170	1.25	18	1.2	245	G	G
Hydrazine (anhydrous)	H_2O_2 (99.6%)	1.7	4690	1.22	19	1.24	255	G	F-P
Hydrazine (anhydrous)	Nitrogen tetroxide	1.2	5000	1.26	19	1.2	250	G	F-P
Hydrazine (anhydrous)	Oxygen	0.75	5370	1.25	18	1.06	265	G	F-P
Hydrazine (anhydrous)	RFNA (15% NO_2)	1.3	4980	1.25	20	1.26	247	G	G
Hydrogen (max. I_{sp})	Fluorine	9.42³	8100	1.31	10	0.46	390	F-P	F-P
		3.8	4600	1.30	7.8	0.27	360	F-P	F-P
Hydrogen (max. I_{sp})	Oxygen	8	5870	1.22	16	0.43	360	F-P	F-P
		3.5	4500	1.26	9	0.26	348	F-P	F-P
JP-4	H_2O_2 (99.6%)	6.5	4830	1.2	22	1.28	238	G	F-P
JP-4	Oxygen	2.3	5770	1.24	22	0.98	247	G	F-P
Kerosene	Oxygen	2.2-2.5	5200	1.24	22	0.99	240	G	F-P
Unsymmetrical dimethyl hydrazine	Oxygen	1.4	5650	1.24	20	0.96	249	G	F-P
Unsymmetrical dimethyl hydrazine	RFNA (22% NO_2)	2.6	5200	1.23	22	1.23	241	G	G
Unsymmetrical dimethyl hydrazine	WFNA	2.7	5100	1.23	22	1.22	240	G	G

[1] Courtesy *Space/Aeronautics* [17]. [2] G = good; F = fair; P = poor. "Good" storability means liquid can be stored in ordinary vessels or tanks over long periods and at many temperatures without decomposition or change of state. [3] Mixture ratio yielding the highest loading density, or mass ratio.

CHEMICAL ROCKET SYSTEMS

Liquid Propellant Rockets - Liquid Vs. Solid

LIQUID	SOLID
GENERALLY LESS SENSITIVE TO AMBIENT TEMP AFFECTS	SIMPLE DESIGN AND CONSTRUCTION
LIGHT - RELIGHT CAPABILITY	FEW SERVICE PROBLEMS
OFTEN LIGHTER UNITS FOR LONGER HIGH THRUST APPLICATIONS	OFTEN LIGHTER UNITS FOR SHORT HIGH THRUST APPLICATIONS
GENERALLY HIGHER SPECIFIC IMPULSE	MORE RELIABLE
THRUST VARIED READILY	SELF-CONTAINED
CAPABLE OF PRETEST	

CHEMICAL ROCKET SYSTEMS

Liquid Propellant Rockets - The Benefits of Hydrogen

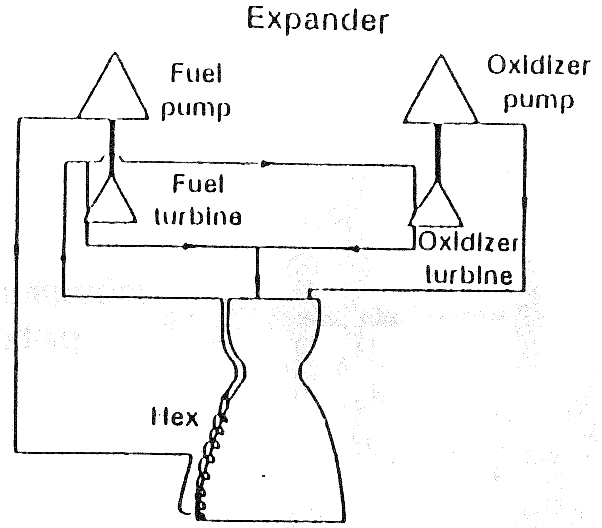
- HIGH HEAT CAPACITY
  - PROVIDES COOLING FOR COMBUSTION CHAMBER AND NOZZLE
  - PROVIDES ENERGY FOR TURBINE TO POWER PUMPS
- LOW MOLECULAR WEIGHT
  - YIELDS HIGHER COMBUSTION VELOCITIES AND THUS PROVIDES HIGHER SPECIFIC IMPULSE
- HIGH HEAT OF COMBUSTION
  - MORE ENERGY PER POUND THAN ANY OTHER FUEL
- WHEN BURNED WITH OXYGEN, RESULTS IN POLLUTION FREE EXHAUST

OVERVIEW

- PERFORMANCE CONSIDERATIONS
- CHEMICAL ROCKET SYSTEMS
- **ROCKET ENGINE CYCLES**
  - **CYCLES**
  - **RL10**
  - **SSME ATD/ALS**
  - **ROCETS**
  - **HISTORICAL**

ROCKET ENGINE CYCLES

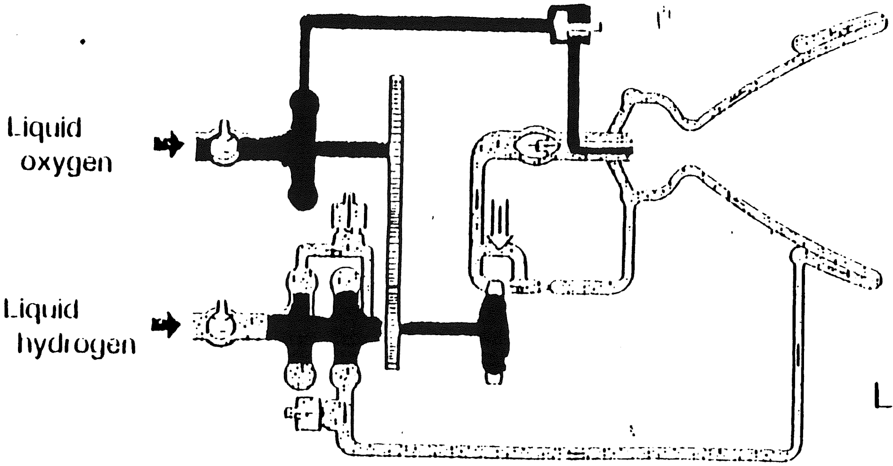
Cycles - Expander Cycle

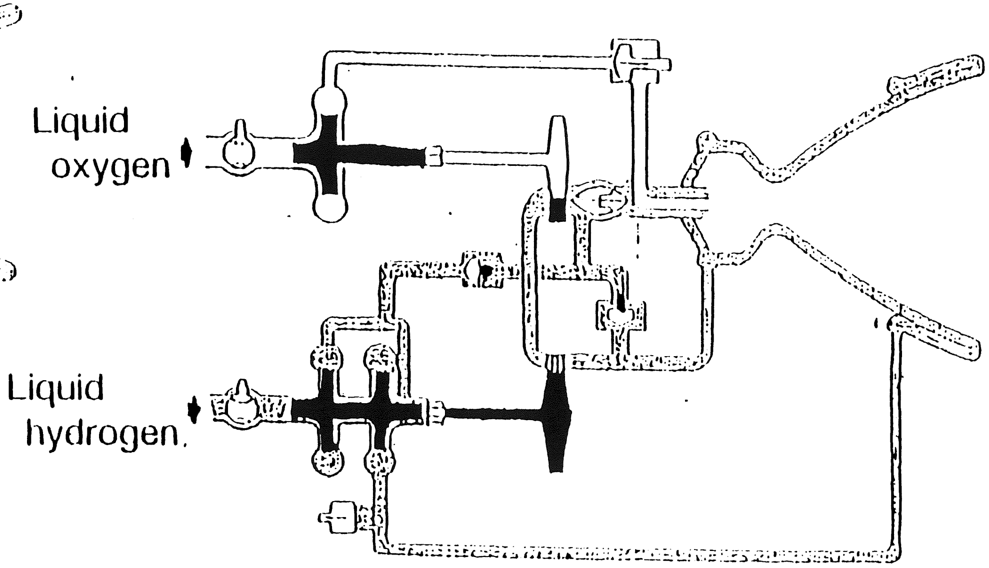


- **EXPANDER CYCLE USES HEAT PICK-UP FROM THE THRUST CHAMBER/NOZZLE TO PROVIDE POWER TO THE TURBINES**
  - **SIMPLICITY EQUALS RELIABILITY (NO FIRE EXCEPT IN CHAMBER)**
  - **HIGH IN PERFORMANCE**
  - **HISTORICALLY, LIMITED IN THRUST TO LESS THAN 1,000,000 LBF**
  - **BENIGN ENVIRONMENT FOR TURBINE (LESS THAN 800 DEG R)**
  - **CURRENTLY USED IN THE RL10 CONFIGURATION**

ROCKET ENGINE CYCLES

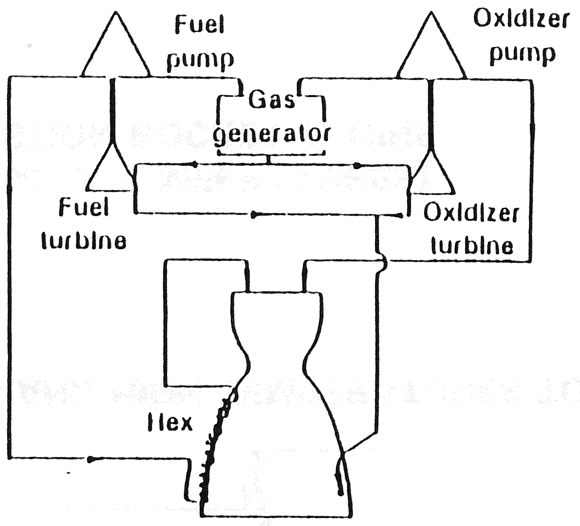
Cycles - Expander Cycle vs. Split-Expander





ROCKET ENGINE CYCLES

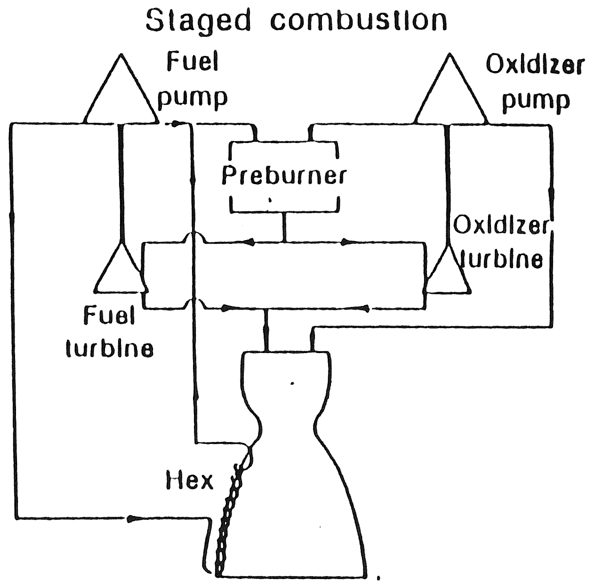
Cycles - GAS GENERATOR CYCLE


Gas generator

- **GAS GENERATOR CYCLE USES MODERATE TEMPERATURES (1600 DEG R) AND LARGE TURBINE PRESSURE RATIOS TO PROVIDE POWER TO THE TURBINES**
  - **LOW IN PERFORMANCE**
  - **CAPABLE OF HIGH THRUST**
  - **EARLIEST CYCLE DEVELOPED, USED IN MOST BOOSTER APPLICATIONS**

ROCKET ENGINE CYCLES

Cycles - Staged Combustion Cycle



- STAGED COMBUSTION CYCLE USES HIGH PRESSURES AND HIGH TEMPERATURES TO PROVIDE POWER TO THE TURBINES
  - HIGH IN PERFORMANCE
  - CAPABLE OF HIGH THRUST
  - HARSH ENVIRONMENT FOR TURBINE (5000 PSIA, 2000 °R TEMPERATURES)
  - SSME IS THE ONLY OPERATIONAL STAGED COMBUSTION ROCKET ENGINE

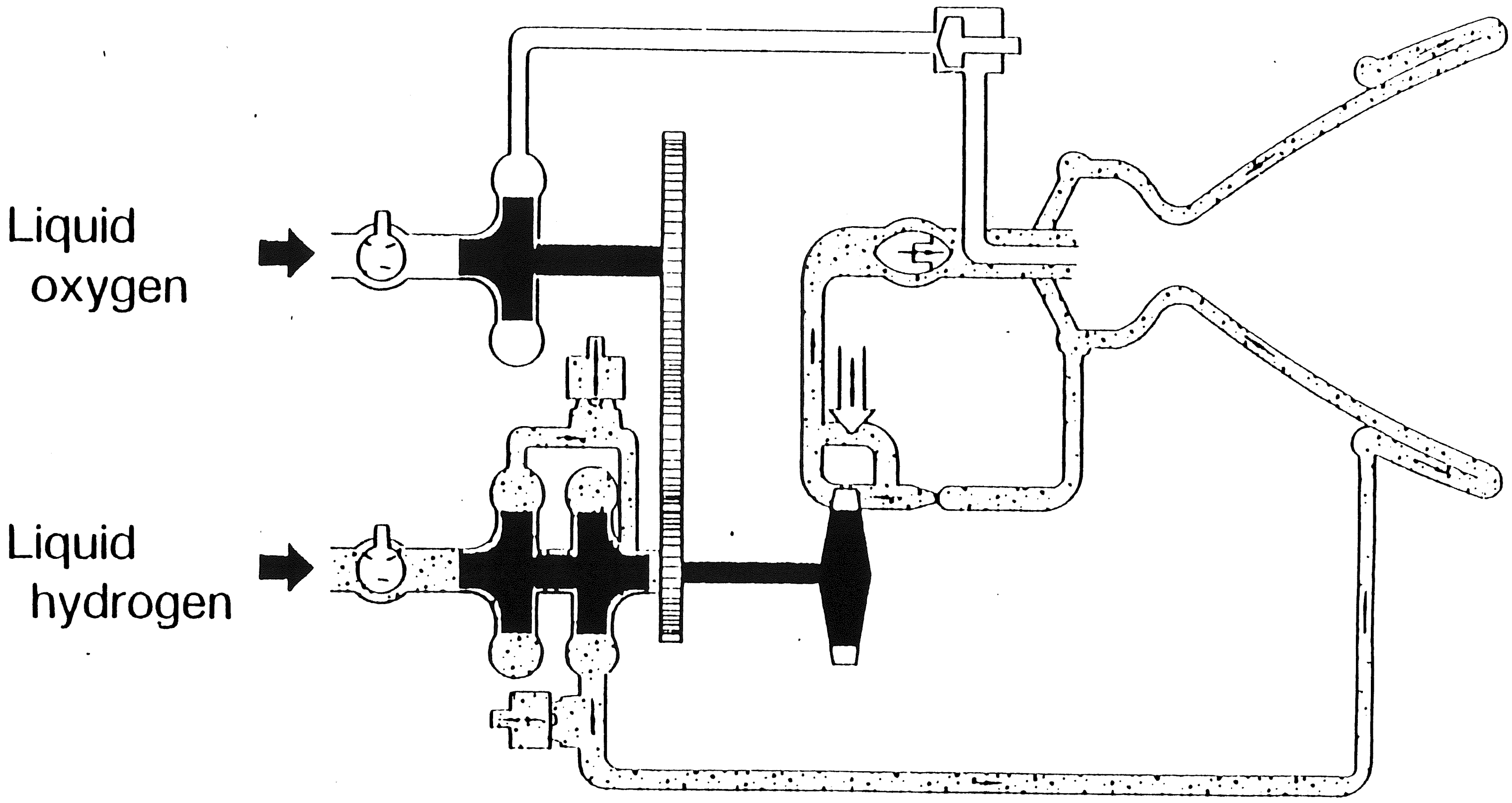
ROCKET ENGINE CYCLES

Cycles - Liquid Engine Cycle Comparisons

	EXPANDER	GAS GENERATOR	STAGED COMBUSTION
TURBINE INLET PRESSURE - PSIA	LESS THAN 3000	LESS THAN 3000	5000
TURBINE INLET TEMP DEG R	LESS THAN 800	1600	1900
THRUST - LBF	LESS THAN 1,000,000	HIGH THRUST	HIGH THRUST
IMPULSE	HIGH	MODERATE	HIGH
TECHNOLOGY COST	LOW	LOW	HIGH

ROCKET ENGINE CYCLES

RL10 - Prestart (Pump Cooldown)



Liquid oxygen

Liquid hydrogen

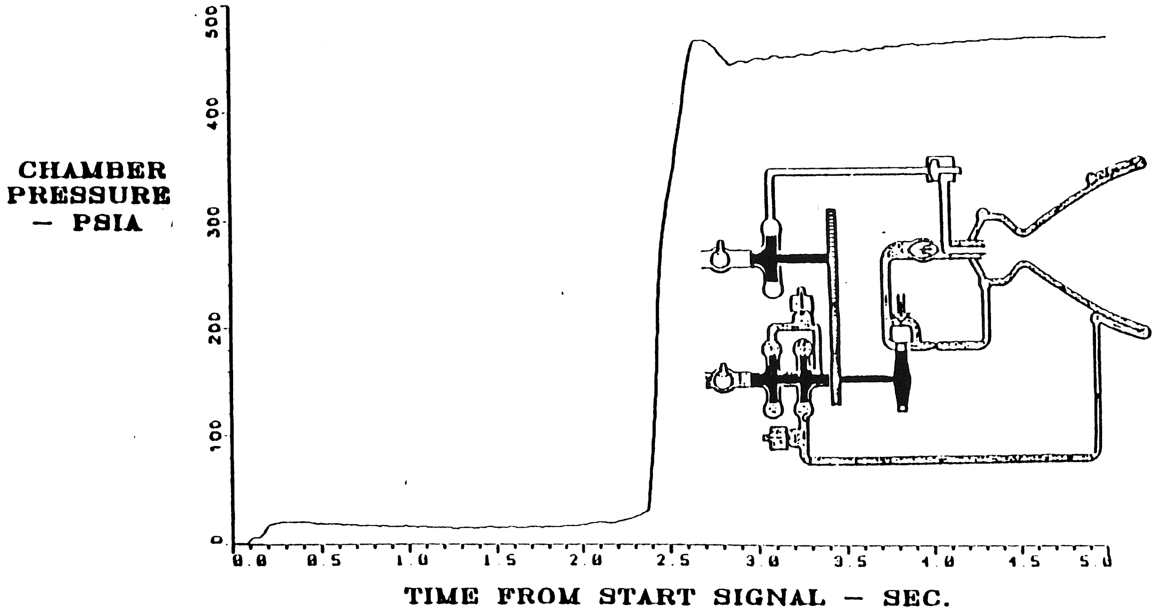
ROCKET ENGINE CYCLES

RL10 - Prestart (Pump Cooldown)

- PUMPS NEED TO BE AT CRYOGENIC TEMPERATURE TO AVOID CAVITATION DURING START SEQUENCE
- INLET VALVES ARE OPENED 5 TO 9 SECONDS PRIOR TO START, ALLOWING PROPELLANTS TO COOL PUMPS
- LIQUID OXYGEN FLOWS THROUGH PUMP, MIXTURE RATIO CONTROL VALVE, AND IS EXPELLED OVERBOARD THROUGH THE COMBUSTION CHAMBER/NOZZLE
- LIQUID HYDROGEN FLOWS THROUGH THE FIRST STAGE, HALF OF THE FLOW IS VENTED OVERBOARD THROUGH THE INTERSTAGE VALVE, THE OTHER HALF PASSES THROUGH THE SECOND STAGE AND IS THEN VENTED OVERBOARD VIA THE COOLDOWN VALVE
- MAIN FUEL SHUTOFF VALVE REMAINS CLOSED, DEADHEADING THE COMBUSTION CHAMBER/NOZZLE LEG

ROCKET ENGINE CYCLES

RL10 - Start Transient



ROCKET ENGINE CYCLES

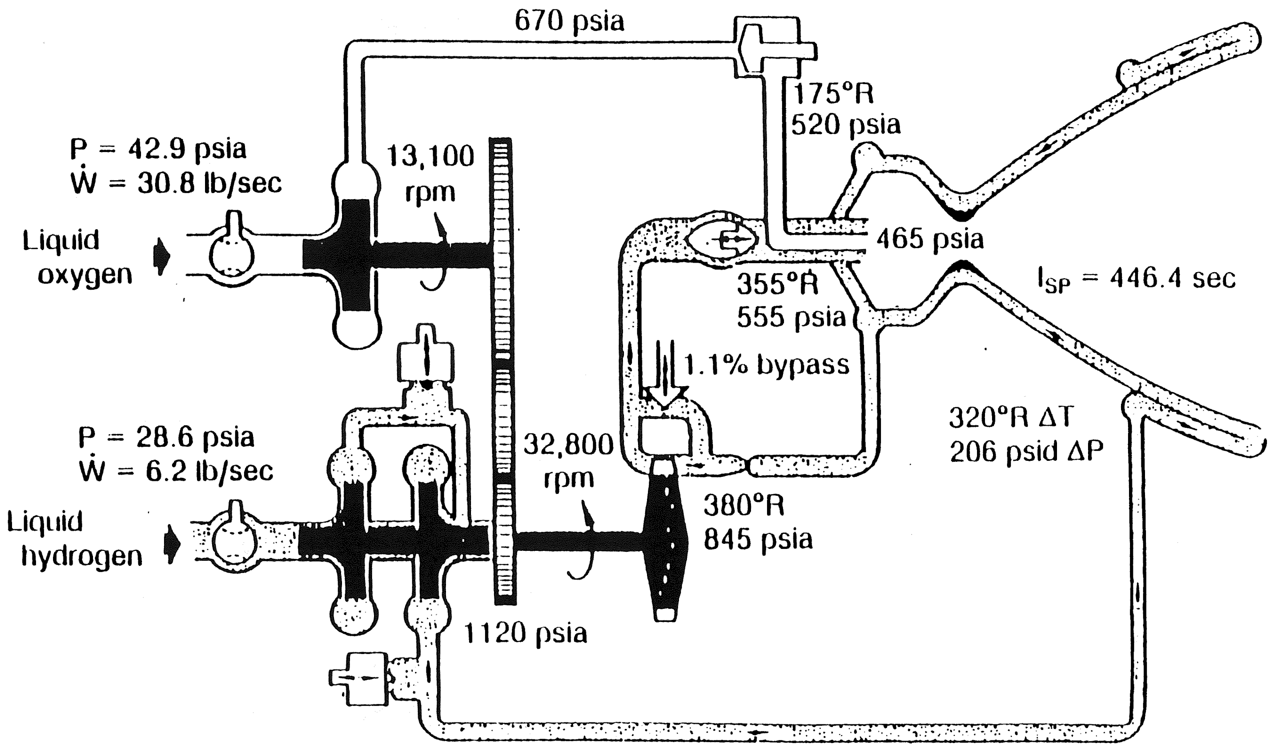
RL10 - Start Transient

- AT START SIGNAL, MAIN FUEL VALVE OPENS AND PUMP DISCHARGE COOLDOWN VALVE CLOSES ALLOWING HYDROGEN TO FLOW THROUGH CHAMBER/NOZZLE COOLING TUBES. IGNITION OCCURS AT 0.15 SEC USING A SPARK IGNITER.
- RL10 STARTS ITSELF EMPLOYING A 'BOOTSTRAP' CYCLE. WITH THE THRUST CONTROL VALVE CLOSED, HYDROGEN, WARMED BY THE AMBIENT CHAMBER/NOZZLE, FLOWS THROUGH THE TURBINE AND STARTS PUMP ROTATION.
- AS PUMP'S SPEED INCREASES, MIXTURE RATIO CONTROL VALVE OPENS COMPLETELY, ALLOWING LIQUID OXYGEN TO FILL INJECTOR MANIFOLD. FUEL PUMP INTERSTAGE VALVE ALSO CLOSES SO ALL FUEL GOES THROUGH TURBINE . AS LIQUID OXYGEN REACHES COMBUSTION CHAMBER, CHAMBER PRESSURE RISES RAPIDLY.
- WHEN OXYGEN INJECTOR MANIFOLD FILLS WITH LIQUID, CHAMBER PRESSURE INCREASES ALONG THE 'LOX-LINE'. AT AN APPROXIMATE CHAMBER PRESSURE OF 300 PSIA THE THRUST CONTROL VALVE OPENS WIDE SLOWING TURBINE ACCELERATION AND PREVENTING EXCESSIVE THRUST OVERSHOOT. THRUST CONTROL STARTS TO CONTROL CHAMBER PRESSURE SETTLING INTO STEADY-STATE.

ROCKET ENGINE CYCLES

RL10 - Steady State

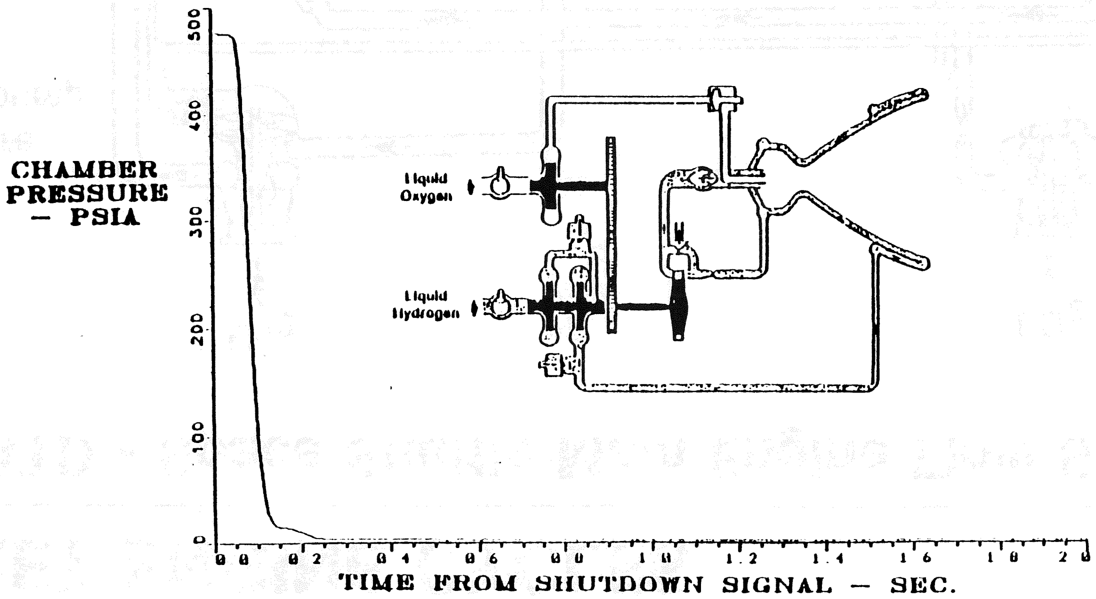
16.5K thrust, 5.0:1 MR



ROCKET ENGINE CYCLES

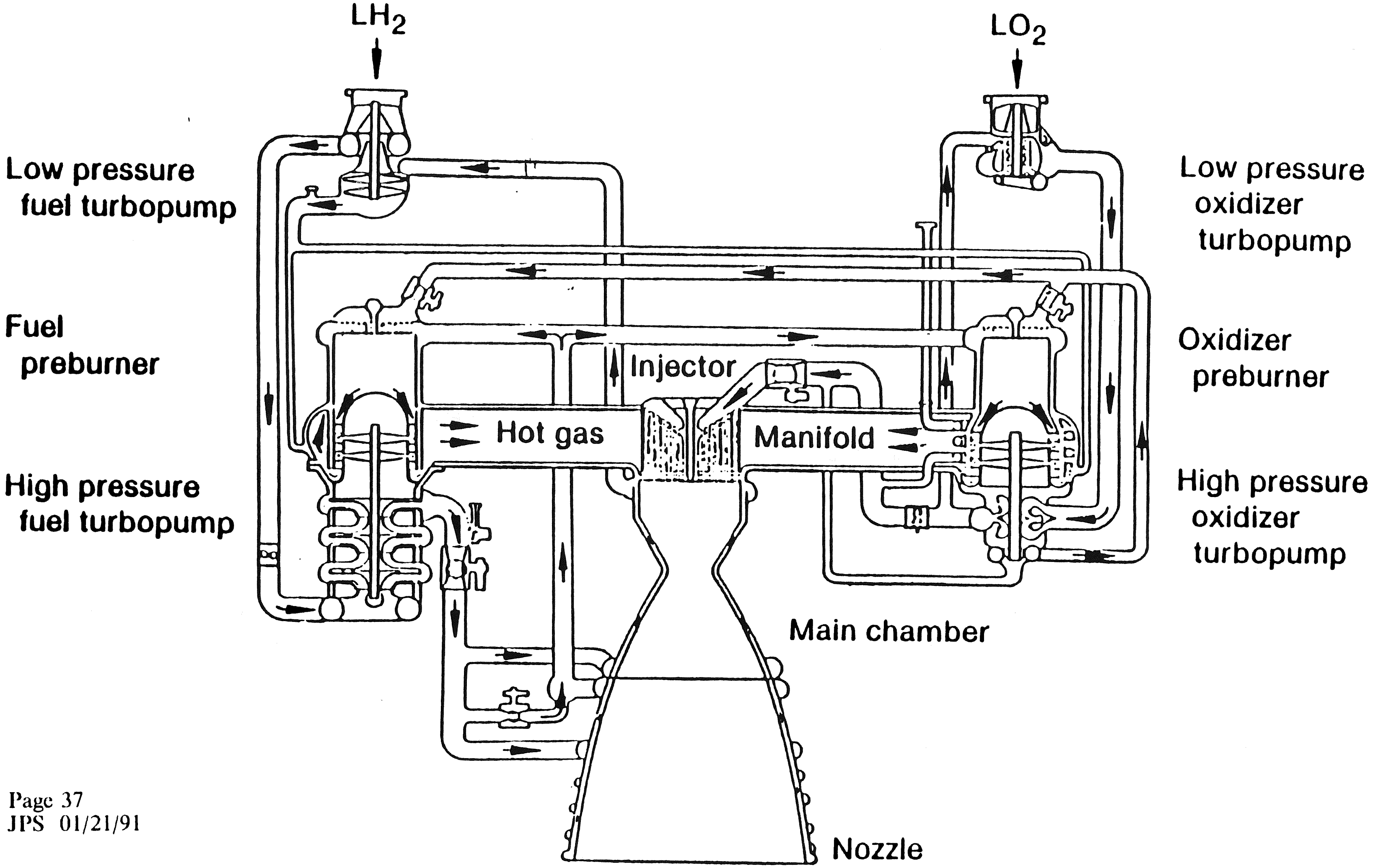
RL10 - Shutdown Transient

- AT THE SHUTDOWN SIGNAL THE MAIN SHUTOFF AND BOTH INLET VALVES CLOSE, BOTH COOLDOWN VALVES OPEN TO VENT TRAPPED FUEL.
- ENGINE SHUTS DOWN IN LESS THAN 0.15 SECS.
- OXYGEN IN PROPELLANT LINES BLEED OUT THROUGH THRUST CHAMBER PRODUCING SLIGHT THRUST.
- VALVES ARE IN POSITION FOR ENGINE RESTART.



ROCKET ENGINE CYCLES

SSME ATD - Space Shuttle Main Engine Flow Schematic



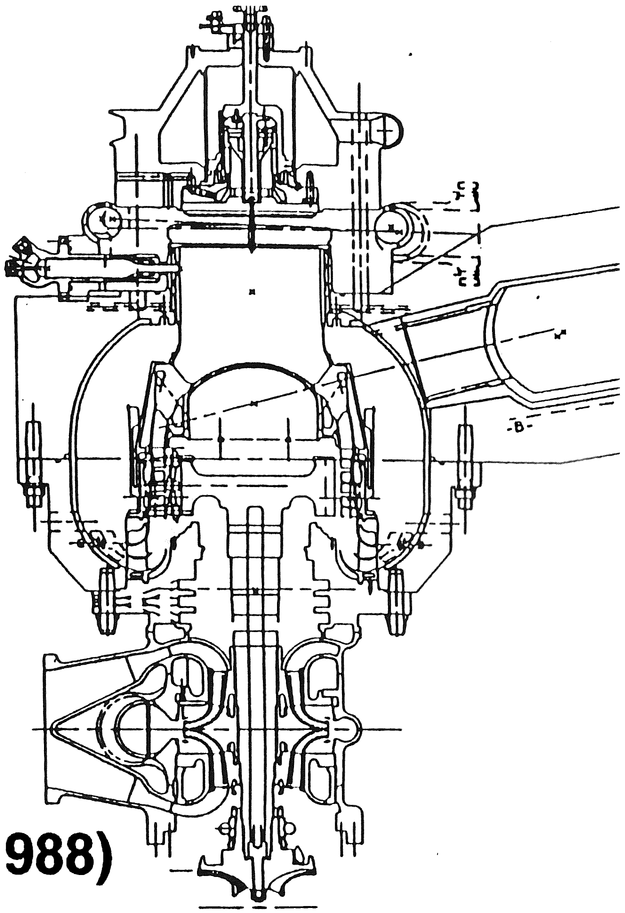
ROCKET ENGINE CYCLES

SSME ATD - Space Shuttle Vs. RL10

	RL10	SSME
CHAMBER PRESSURE (PSIA)	465.	3272.
THRUST (LBF)	16500.	513836.
TURBINE INLET TEMPERATURE (DEG R)	380.	1561. (OXID) 1893. (FUEL)
SPECIFIC IMPULSE (SEC)	446.4	452.8
TOTAL PROPELLANT FLOW (LBM/SEC)	37.	1137.
FUEL PUMP SPEED (RPM)	32800.	36298.
OXIDIZER PUMP SPEED (RPM)	13100.	29256.
ENGINE WEIGHT (LBF)	305.	6668.

ROCKET ENGINE CYCLES

SSME ATD - E-8 TEST PROGRAM



- GOALS
  - Design Verification
  - Life Demonstration
- TESTING SEQUENCE
  - Facility Check-out (November-December 1988)
  - Check-out Chamber (August 1989)
  - Turbine Simulator (January 1990)
  - HPFTP - UNIT 1-1, 2-1,1-2 and HPOTP 1-1, 2-1a, 2-1b (January,1990-January,1991)

ROCKET ENGINE CYCLES

ALS - SUB-SCALE TEST PROGRAM

- GOALS
  - High Reliability
  - Low Cost
  - High Operability Potential
  - Development of Validated Flight Engine Design Data Base
- TESTING SEQUENCE
  - Sub-Scale 40K Injector Test, September 1990, Huntsville
  - Currently Converting Test Stand to Full Scale 580K Injector Test

ROCKET ENGINE CYCLES

ROCETS - Rocket Engine Transient Simulations

- GOAL
  - To create an generic in-house modeling system that can handle a variety of tasks and can be utilized by the general engineering population.
  - To advance the state-of-the-art of closed loop implicit solvers. This particular solver system can handle both steady state and transient solutions for rocket engine systems.
  - To create a standardized library of software for rocket engine modules and sub-modules.

ROCKET ENGINE CYCLES

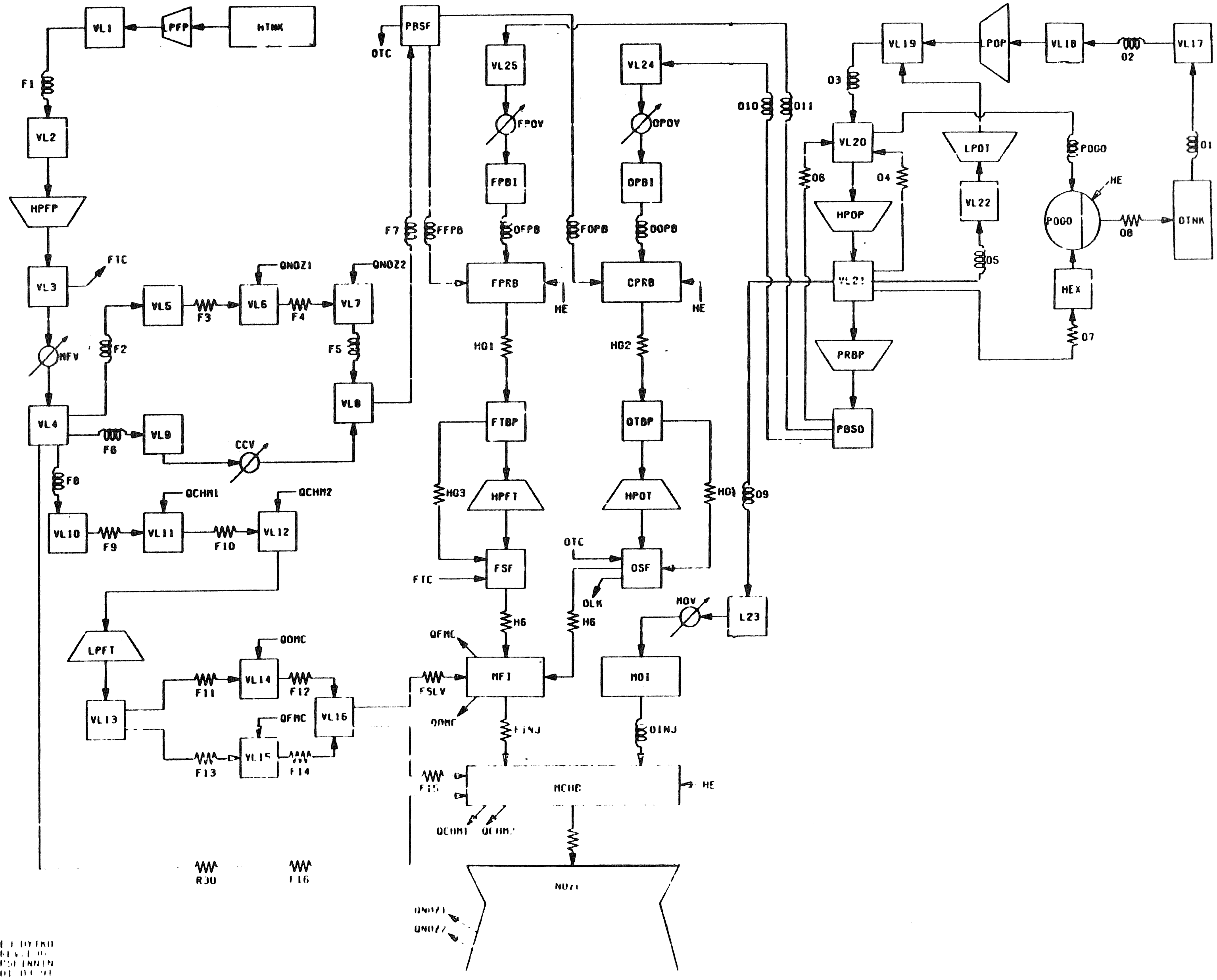
ROCETS - Rocket Engine Transient Simulations

COMPONENTS

1.	LIBRARY	CENTRAL SOURCE OF ALL SOFTWARE CODE TO ALLOW MULTIPLE-USERS
2.	SIMULATION INPUT/OUTPUT	PROCESSORS THAT CONDUCT SYSTEM FUNCTIONS
3.	EXECUTIVE PROGRAMS	USER CONFIGURATION INPUT, RUN INPUT, AND OUTPUTS
4.	DOCUMENTATION	SYSTEM STANDARDS, ENGINEERING & COMPUTING DESCRIPTIONS, USER'S MANUAL, AND QUALIFICATION TEST PLANS
5.	MAINTENANCE PROCEDURES	INSTRUCTIONS FOR SYSTEM UPKEEP

ROCKET ENGINE CYCLES

ROCETS - Rocket Engine Transient Simulations



ROCKET ENGINE CYCLES

Historical - Pratt & Whitney Rocket Engine Experience

- RAN FIRST LIQUID HYDROGEN ROCKET ENGINE - 1959
- ESTABLISHED FEASIBILITY OF HIGH PRESSURE STAGED COMBUSTION ENGINE UNDER AIR FORCE SPONSORSHIP (AFRPL OR XLR129 PROGRAM)
- MORE FLIGHT EXPERIENCE THAN ALL OTHER HYDROGEN ROCKET ENGINES - 1963 TO PRESENT
- PERFECT FLIGHT RECORD - 100 % RELIABLE

ROCKET ENGINE CYCLES

Historical - RL10 Overview

- DESIGNED AND BUILT IN LESS THAN A YEAR
- THE FIRST LIQUID HYDROGEN ROCKET ENGINE
- NEVER FAILED IN FLIGHT
- NUMBER OF RL10'S PRODUCED EACH YEAR HAS HELD STEADY SINCE MID 1970'S (8)
- AIR FORCE TO BUY 10 TITAN CENTAUR VEHICLES TO SUPPLEMENT SHUTTLE LAUNCHES INTO THE 1990'S
- GENERAL DYNAMICS COMMERCIAL VENTURE FOR 19 INITIAL VEHICLES WITH AN ADD ON OF 38 MORE VEHICLES THROUGH 1997

ROCKET ENGINE CYCLES

Historical - RL10 History

- CREATED IN 1959 AND FLOWN FOR THE FIRST TIME IN 1963 ABOARD AN ATLAS-CENTAUR
- SURVEYOR, 1966 - SOFT LUNAR LANDINGS IN PREPARATION FOR THE APOLLO LUNAR LANDINGS
- MARINER, 1969 - MARS FLY-BY
- VOYAGER, 1977 - JUPITER (1979), SATURN (1980), URANUS (1986), NEPTUNE (1989)
- VIKING, 1978 - MARS SOFT LANDING
- PIONEER, 1978 - VENUS ORBITER
- 178 ENGINES FIRED IN SPACE
- 290 IN SPACE FIRINGS
- 76,246 SECONDS RUN TIME IN SPACE

LIQUID ROCKET ENGINE CONTROL
AND
ENGINE MONITORING SYSTEMS

E. A. Petrino

Propulsion System Analysis

OVERVIEW

- **REQUIREMENTS**
- CONTROL MODE
- START & SHUTDOWN CONSIDERATIONS
- SENSORS
- CONTROL COMPONENTS
- ENGINE MONITORING

ENGINE CONTROL DESIGNED TO MEET SYSTEM REQUIREMENTS

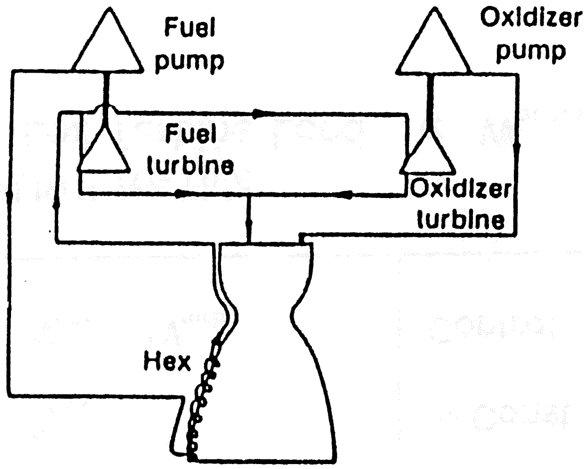
SYSTEM REQUIREMENTS	ENGINE CONTROL REQUIREMENTS
VEHICLE LIMITS	THROTTLE CAPABILITY
MISSION	THRUST ACCURACY SAFETY REDUNDANCY
PAYLOAD	MIXTURE RATIO CONTROL
LIFE CYCLE COSTS	LOW COST DESIGN AUTOMATIC CHECKOUT REDUCED MAINTENANCE REUSABLE REDUCED PAD CHECKOUT TIME

OVERVIEW

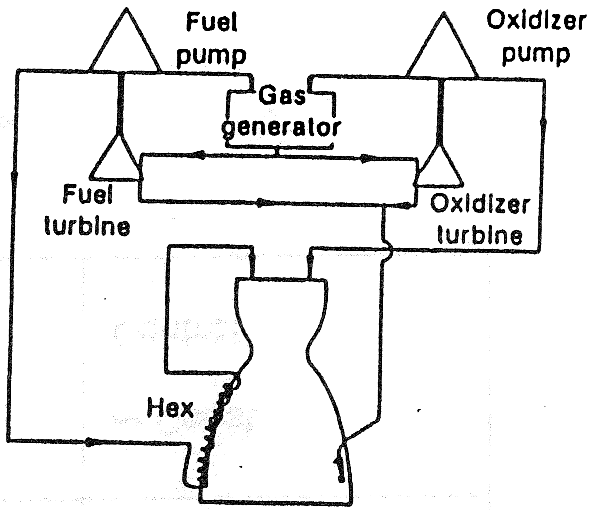
- REQUIREMENTS
- **CONTROL MODE**
- START & SHUTDOWN CONSIDERATIONS
- SENSORS
- CONTROL COMPONENTS
- ENGINE MONITORING

LIQUID ROCKET ENGINE CYCLES

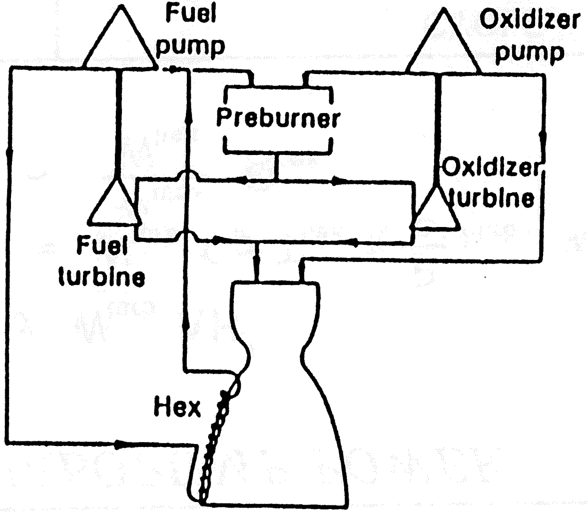
Expander



Gas generator



Staged combustion



EXPANDER CYCLE CONTROL FUNDAMENTALS

KEY: CONTROL TURBOPUMP POWER

- THRUST:

  Turbine Power $\alpha\ W_{turb}\ \Delta H'$

$$= W_{turb}\ Cp\ T_{hex}\ f(\frac{P}{P})_{turb} \quad \text{where} \quad T_{hex} = \frac{Q_{hex}}{Cp\ W_{hex}} + T_{fuel}$$

$$\sim \frac{W_{turb}}{W_{hex}}\ Q_{hex}$$

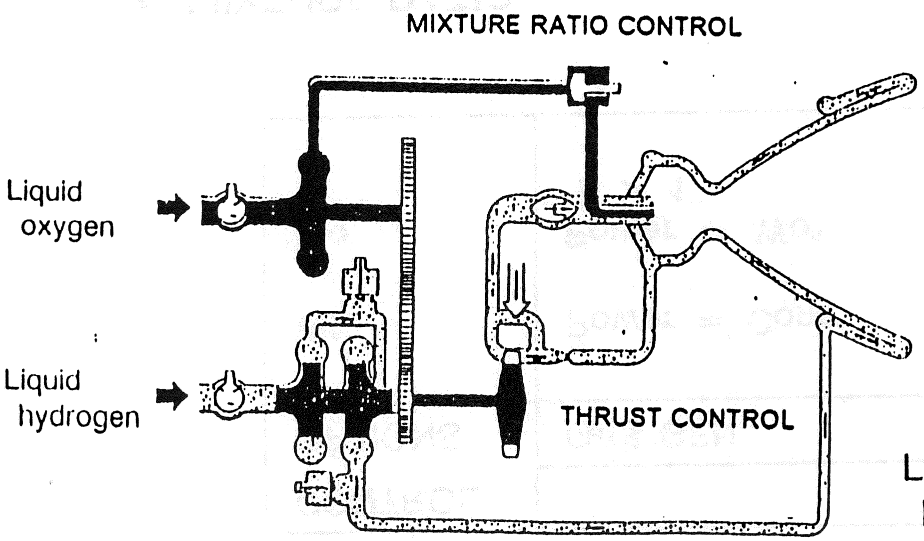
CONTROL OPTIONS	CYCLE	
	EXPANDER	SPLIT EXPANDER
Q_{hex}	~ Const	~ Const
$W_{hex} - W_{turb}$	Control	Control

- MIXTURE RATIO:

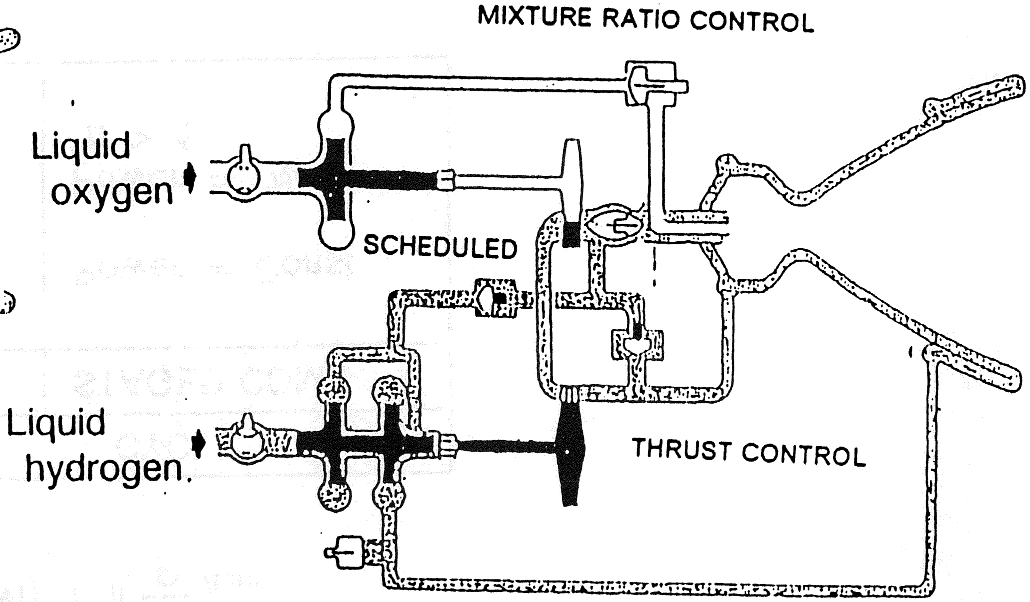
  Lox Turbine Load $\alpha\ W_{pump}\ \Delta P_{pump}$

EXPANDER CYCLE CONTROL SYSTEMS

EXPANDER CYCLE



SPLIT EXPANDER CYCLE



GAS GENERATOR & STAGED COMBUSTION CONTROL FUNDAMENTALS

KEY: CONTROL AVAILABLE TURBOPUMP POWER

- THRUST:

  Turbine Power α $W_{turb}\ \Delta H'$

  $= (Wo + Wf)\ T\ f(\frac{P}{P})_{turb}$

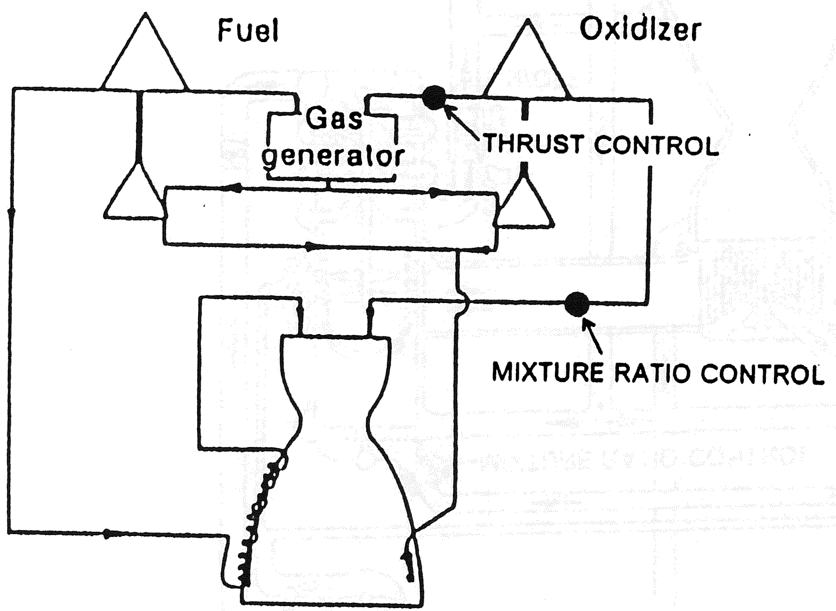
CONTROL OPTIONS	CYCLE	
	GAS GEN	STAGED COMB
Wf	Power = Const	Power = Const
Wo	**Power = Wo^n** **$n > 1$**	**Power = Wo^n** **$n > 1$**

- MIXTURE RATIO:

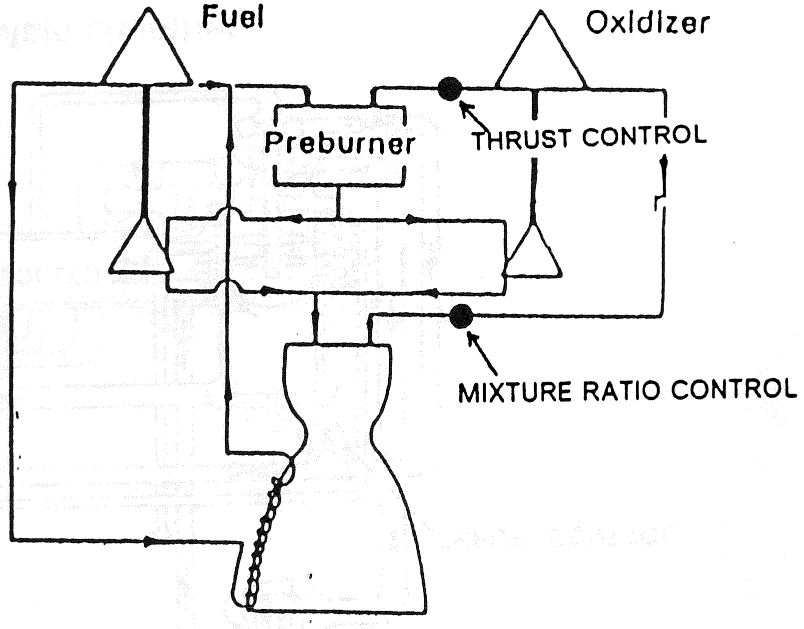
  Lox Turbine Load α $W_{pump}\ \Delta P_{pump}$

GAS GENERATOR & STAGED COMBUSTION CONTROL SYSTEMS

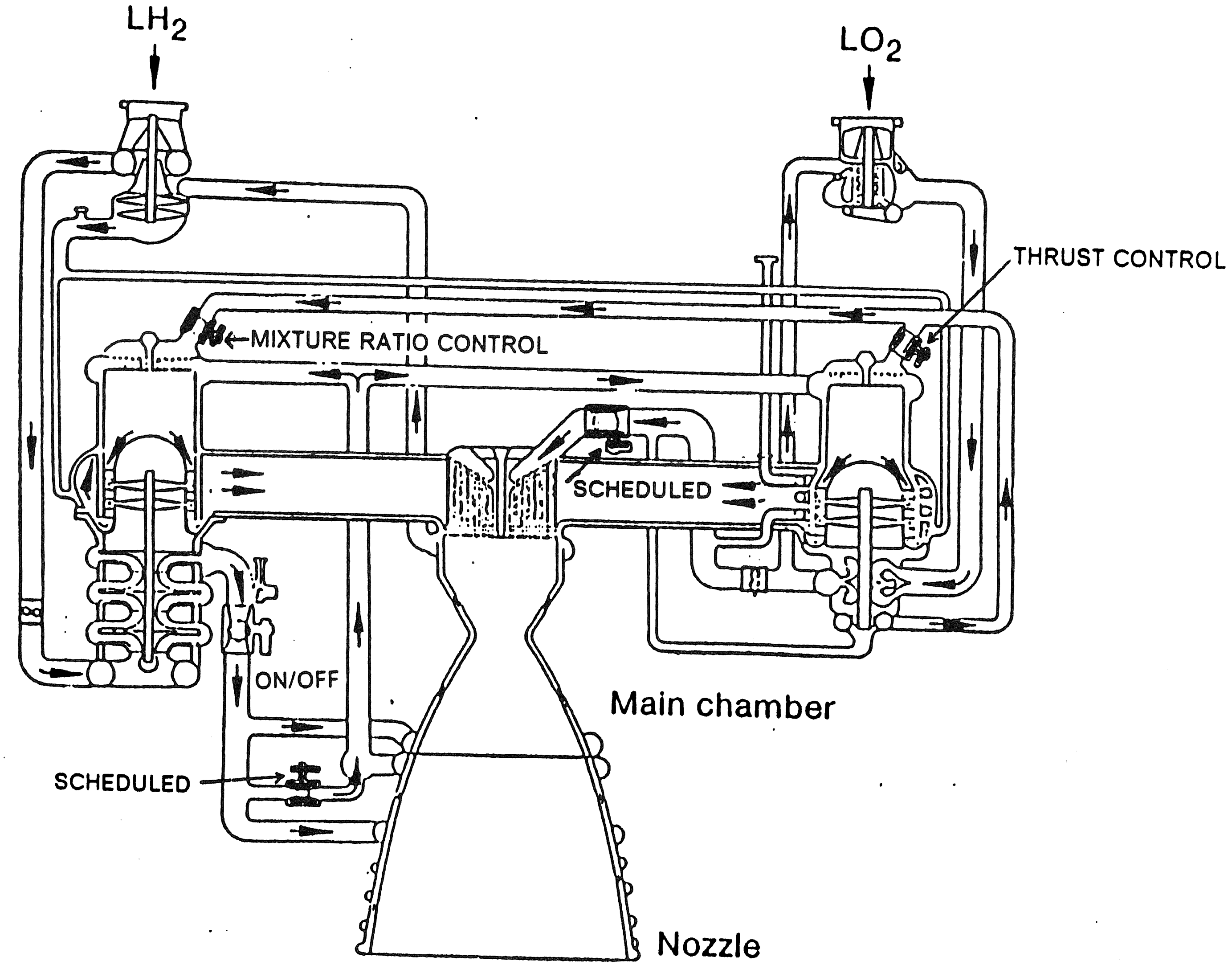
Gas generator



Staged combustion



SSME CONTROL SYSTEM



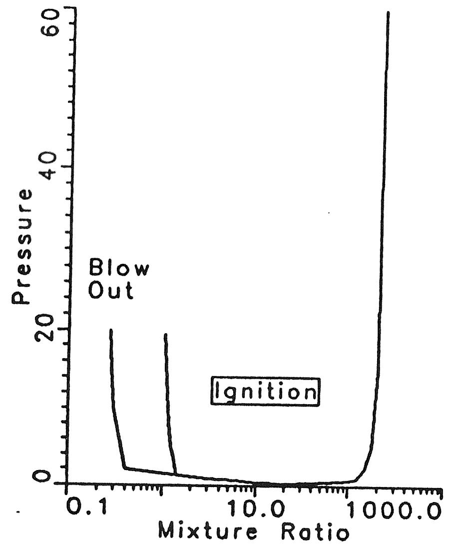
OVERVIEW

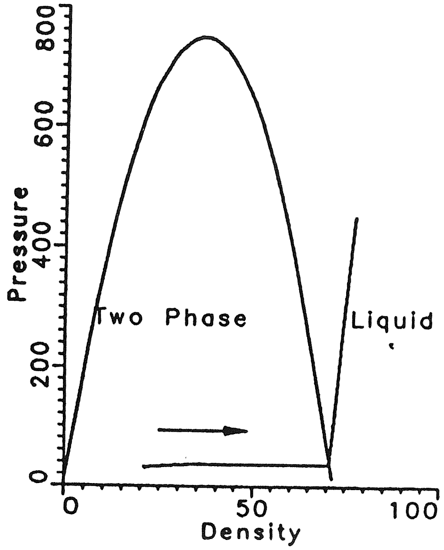
- REQUIREMENTS
- CONTROL MODE
- **START & SHUTDOWN CONSIDERATIONS**
- SENSORS
- CONTROL COMPONENTS
- ENGINE MONITORING

START PHILOSOPHIES

FUEL VS LOX LEAD

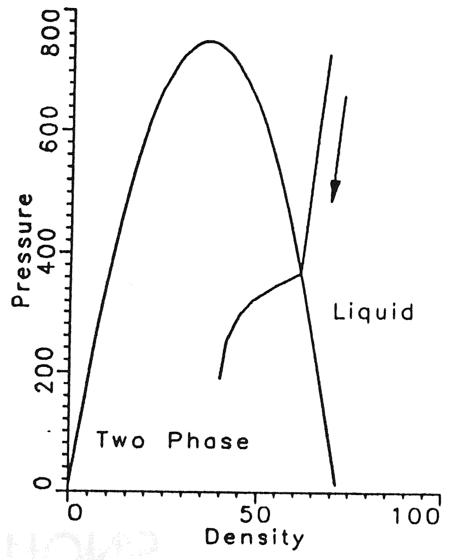
	R/D (FUEL LEAD)	PW (LOX LEAD)
FUEL	GAS	GAS
LOX	2 PHASE	LIQUID
IGNITOR ENERGY	HIGH	LOW





SHUTDOWN PHILOSOPHY

- REQUIREMENTS
  - PREVENT TEMPERATURE SPIKES
  - INERT THE SYSTEM



- IMPLEMENTATION
  - THROTTLE TO MINIMUM THRUST (SSME)
  - CLOSE LOX BURNER VALVE(S) FIRST
  - PURGE LOX INJECTORS AT RATE TO PREVENT TEMPERATURE SPIKE
  - PURGE LOX INJECTORS EARLY ENOUGH TO PREVENT BOILING
  - PURGE FUEL LINES

OVERVIEW

- REQUIREMENTS
- CONTROL MODE
- START & SHUTDOWN CONSIDERATIONS
- **SENSORS**
- CONTROL COMPONENTS
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THRUST AND MIXTURE RATIO COMPUTATION

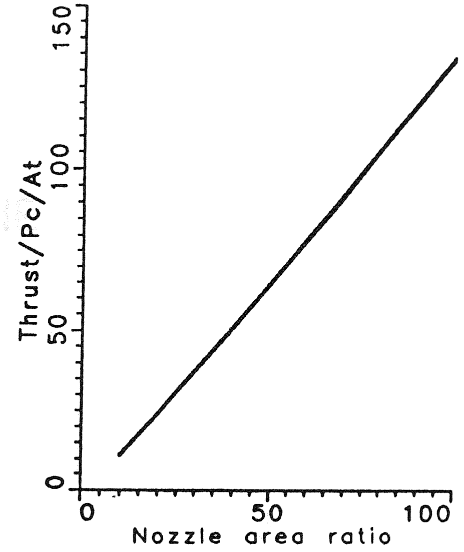
REQUIRES Pc, Wf or Wo MEASUREMENTS

- THRUST

$$\text{Thrust} = \frac{W\ V}{g_c}$$

It can be shown that:

$$\frac{\text{Thrust}}{\text{At Pc}} = f(Ar)\ f(\gamma)$$

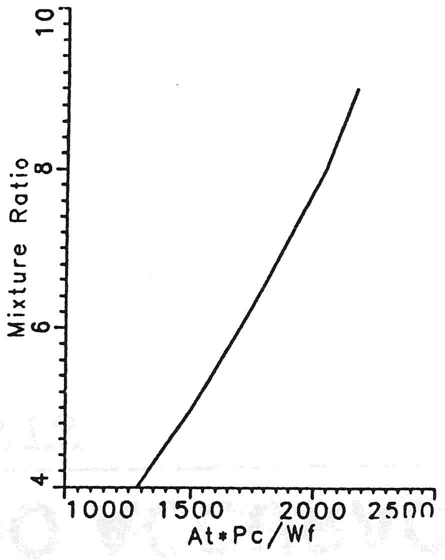


- MIXTURE RATIO

$$O/F = \frac{Wo}{Wf}$$

It can be shown that:

$$O/F = f\left(\frac{Pc\ At}{Wf}\right)$$



THRUST AND MIXTURE RATIO ACCURACY

DICTATED BY VEHICLE REQUIREMENTS

- THRUST

$$\frac{\Delta \mathrm{Thrust}}{\mathrm{Thrust}} = \frac{\Delta \mathrm{Pc}}{\mathrm{Pc}}$$

- MIXTURE RATIO

$$\frac{\Delta \mathrm{O/F}}{\mathrm{O/F}} = \sqrt{\left(\frac{\Delta \mathrm{Wf}}{\mathrm{Wf}}\right)^2 + \left(\frac{\Delta \mathrm{Wo}}{\mathrm{Wo}}\right)^2}$$

$$\frac{\Delta \mathrm{O/F}}{\mathrm{O/F}} = 1.6\sqrt{\left(\frac{\Delta \mathrm{Wf}}{\mathrm{Wf}}\right)^2 + \left(\frac{\Delta \mathrm{Pc}}{\mathrm{Pc}}\right)^2} \quad \text{, SSME}$$

COMPONENT ACCURACY REQUIREMENTS

$O/F = \pm 1\%$, $Thrust = \pm 1\%$

$$\text{ACCURACY} = \pm \sqrt{(\text{SENSORS})^2 + (\text{COMPUTER I/O})^2 + (\text{VALVE HYSTERESIS})^2}$$

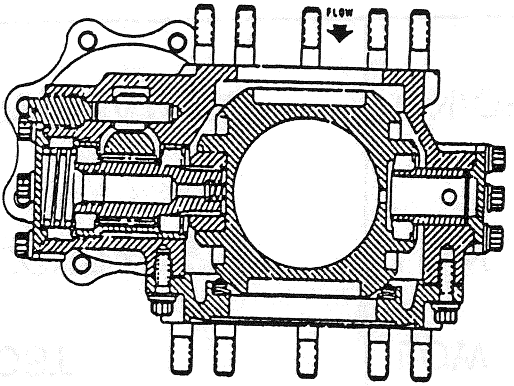
- SENSORS
  - PRESSURE ± 0.25% FULLSCALE
  - TEMPERATURE ± 0.5°
  - FLOW ± 0.5% FULLSCALE
- COMPUTER I/O ± 0.1%
- VALVE HYSTERESIS < ± 0.5%

OVERVIEW

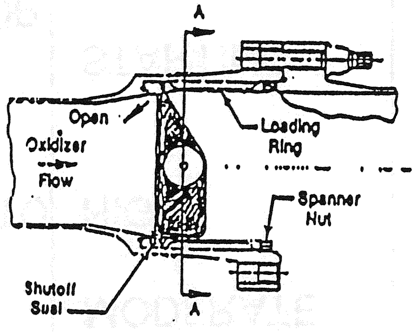
- REQUIREMENTS
- CONTROL MODE
- START & SHUTDOWN CONSIDERATIONS
- SENSORS
- **CONTROL COMPONENTS**
- ENGINE MONITORING

POTENTIAL ROCKET CONTROL VALVES

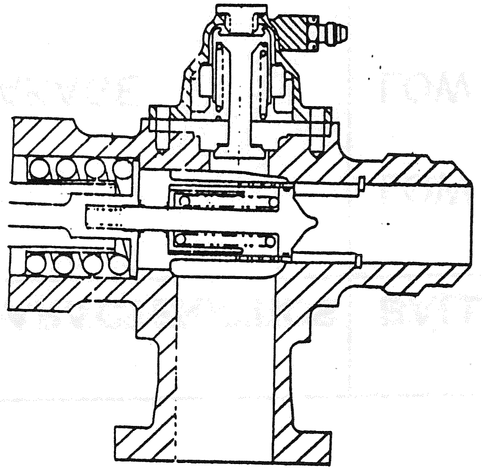
BALL VALVE



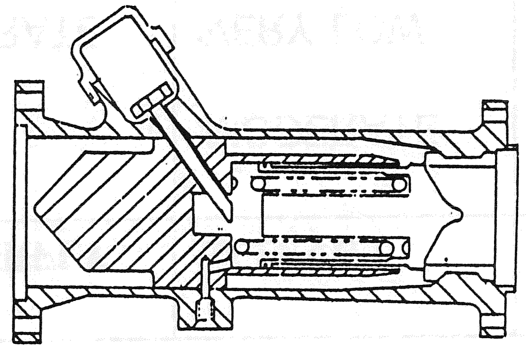
BUTTERFLY VALVE



SLEEVE VALVE



POPPET VALVE



CONTROL VALVE CHARACTERISTICS

CHARACTERISTICS	**BALL**	**BUTTERFLY**	**SLEEVE**	**POPPET**
ΔP	LOW	LOW	MODERATE	LOW
LEAKAGE	LOW	MODERATE TO HIGH	VERY LOW	VERY LOW
COMPLEXITY	LOW	LOW	MODERATE	LOW
COST	LOW	MODERATE	MODERATE	LOW
ACCURACY	LOW	MODERATE TO HIGH	HIGH	MODERATE
APPLICATION	ON/OFF	SCHEDULED CLOSED LOOP	STARTING	ON/OFF

POTENTIAL VALVE ACTUATORS

CHARACTERISTICS	HYDRAULIC	ELECTRO-MECHANICAL	PROPELLANT
ACCURACY	HIGH	MODERATE	LOW
WEIGHT	MODERATE	HIGH	LOW
COST	LOW	MODERATE	HIGH
COMPLEXITY	LOW	MODERATE	HIGH
RELIABILITY	HIGH	MODERATE	HIGH
MAINTENANCE	HIGH	MODERATE	LOW

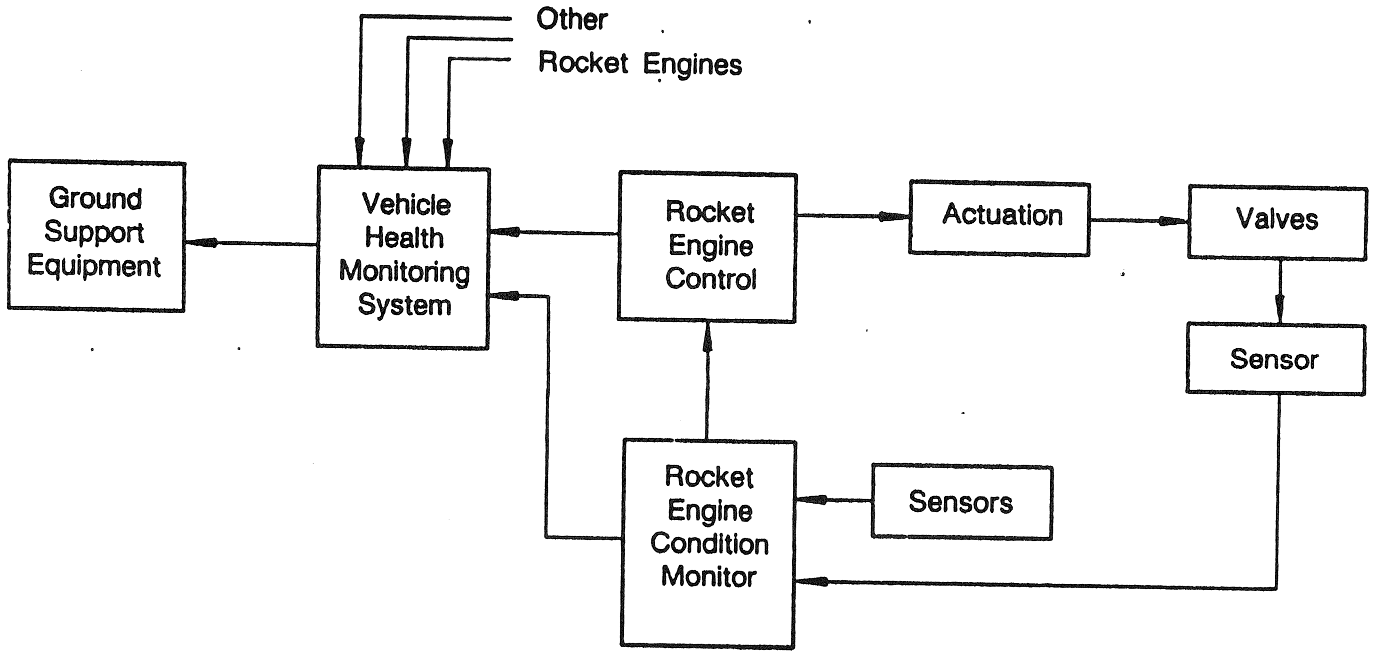
CONTROL PROCESSOR DESIGN PARAMETERS

PROCESSOR	ARCHITECTURE	INTERFACES
LOGIC	RELIABILITY	VEHICLE
NUMERIC	REDUNDANCY	SENSORS
THROUGH PUT	FAULT DETECTION	ACTUATORS
MEMORY	FAULT ACCOMMODATION	SOLENOIDS
LANGUAGE		

OVERVIEW

- REQUIREMENTS
- CONTROL MODE
- START & SHUTDOWN CONSIDERATIONS
- SENSORS
- CONTROL COMPONENTS
- **ENGINE MONITORING**

SYSTEM APPROACH TOWARDS ENGINE MONITORING



MONITORING SYSTEM FUNCTIONS

- **PREFLIGHT**
  - AUTOMATED VEHICLE SYSTEM CHECKOUT
  - TRACE ANOMALIES TO LRU
- **FLIGHT SAFETY**
  - PREDICT/DETECT FLIGHT CRITICAL FAILURES
  - TAKE ACTION TO PREVENT CATASTROPHIC FAILURES
- **POST FLIGHT MAINTENANCE**
  - TREND COMPONENT PERFORMANCE
  - TRACK COMPONENT LIFE
  - IDENTIFY LRU MAINTENANCE
  - TRACK ENGINE COMPONENTS
  - MAINTENANCE DATA BASE

MONITORING SYSTEM DESIGN CONSIDERATIONS

DESIGN PARAMETER	DESIGN OPTIONS
FAULT RESPONSES	FAIL OPERATIONAL FAIL DEGRADED FAIL SAFE
FAULT DETECTION	RANGE AND RATE CHECKS ANALYTICAL REDUNDANCY DUPLICATE COMPONENTS
FAULT RESOLUTION	DIRECT COMPARISON ESTIMATE
FAULT ACCOMMODATION	CHANGE CONTROL MODE AND RECONFIGURE THROUGH REDUNDANCY
HARDWARE REDUNDANCY	ACTIVE/STAND BY AVERAGE VOTING

DIAGNOSTIC/PROGNOSTIC ALGORITHMS

SENSOR FAILURE DETECTION ALGORITHMS

ALGORITHM	**DEFINITION**
LIMIT CHECK	MEASURED PARAMETER VS ACCEPTABLE RANGE
SENSOR SYNTHESIS	ESTIMATE SENSOR VALUE FROM OTHER MEASUREMENTS
MODEL BASED DETECTION FILTER	DEVIATION FROM SYSTEM MODEL

MAINTENANCE ALGORITHMS

ALGORITHM	**DEFINITION**
TRENDING	COMPARISON OF PREVIOUS DATA FOR SIGNIFICANT DEVIATIONS
LCF LIFE DEBIT	REAL TIME CALCULATION OF LIFE USED DUE TO THERMAL CYCLING
TIME LIFE DEBIT	NON CYCLIC LIFE USAGE

INTEGRATED CONTROL SYSTEM BENEFITS

- REDUCED PRE-LAUNCH MAINTENANCE COSTS
- IMPROVED FLIGHT RELIABILITY AND SAFETY
- REDUCED POST FLIGHT REFURBURISHMENT COST

ROCKET ENGINE PERFORMANCE I

PUMPS

- DEFINITIONS & TERMINOLOGY
- HYDRODYNAMIC DESIGN
- PERFORMANCE
- AXIAL LOADS
- P&W HISTORY
- SSME/ATD
- PRELIMINARY PUMP DESIGN
- REFERENCES

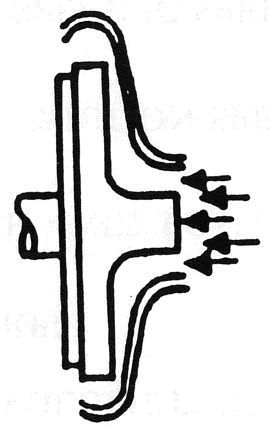
Danny Lawing X 3116

PUMPS

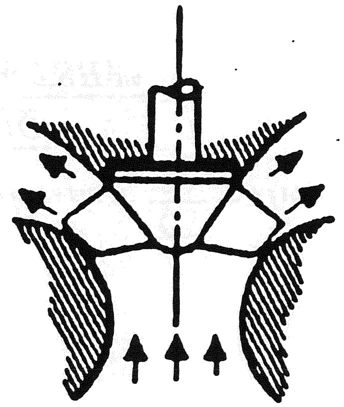
- TRANSFORMS MECHANICAL WORK TO FLUID ENERGY
- A MACHINE THAT HANDLES A LIQUID

- TWO TYPES:
  - 1.) STATIC OR POSITIVE DISPLACEMENT - PISTON (VOLUMETRIC CHANGE)
  - 2.) DYNAMIC OR KINETIC (ΔVELOCITY) - FOR CONSTANT SPEED CAN SUPPLY A WIDE RANGE OF FLOWS AT CONSTANT PRESSURE
- DYNAMIC
  - 1.) AXIAL
  - 2.) MIXED FLOW
  - 3.) CENTRIFUGAL

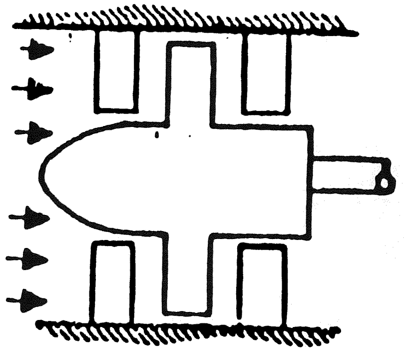
DYNAMIC PUMP TYPES



RADIAL FLOW
(CENTRIFUGAL)



MIXED-FLOW



AXIAL-FLOW

ROCKET PUMP TERMINOLOGY

H - HEAD $= \frac{P}{\rho}$

U - DISK SPEED

C_U - TANGENTIAL FLUID VELOCITY

C_M - MERIDIONAL FLUID VELOCITY

Q - VOLUMETRIC FLOW

N - RPM

NPSH - NET POSITIVE SUCTION HEAD $= H_{STATIC} + \frac{C_m^2}{2g} - H_{VAP}$

N_{SS} - SUCTION SPECIFIC SPEED $= \frac{N\sqrt{Q}}{(NPSH + TSH)^{3/4}}$

N_S - SPECIFIC SPEED $= \frac{N\sqrt{Q}}{(\Delta H)^{3/4}}$

NPSP - NET POSITIVE SUCTION PRESSURE $= P_{STATIC} + \frac{C_M^2}{2g}\rho - P_{VAP}$

D_S - SPECIFIC DIAMETER $= \frac{D_2(\Delta H)^{1/4}}{\sqrt{Q}}$

CAVITATION - FORMATION OF VAPOR BUBBLES IN LIQUID AS STATIC PRESSURE FALLS BELOW VAPOR PRESSURE

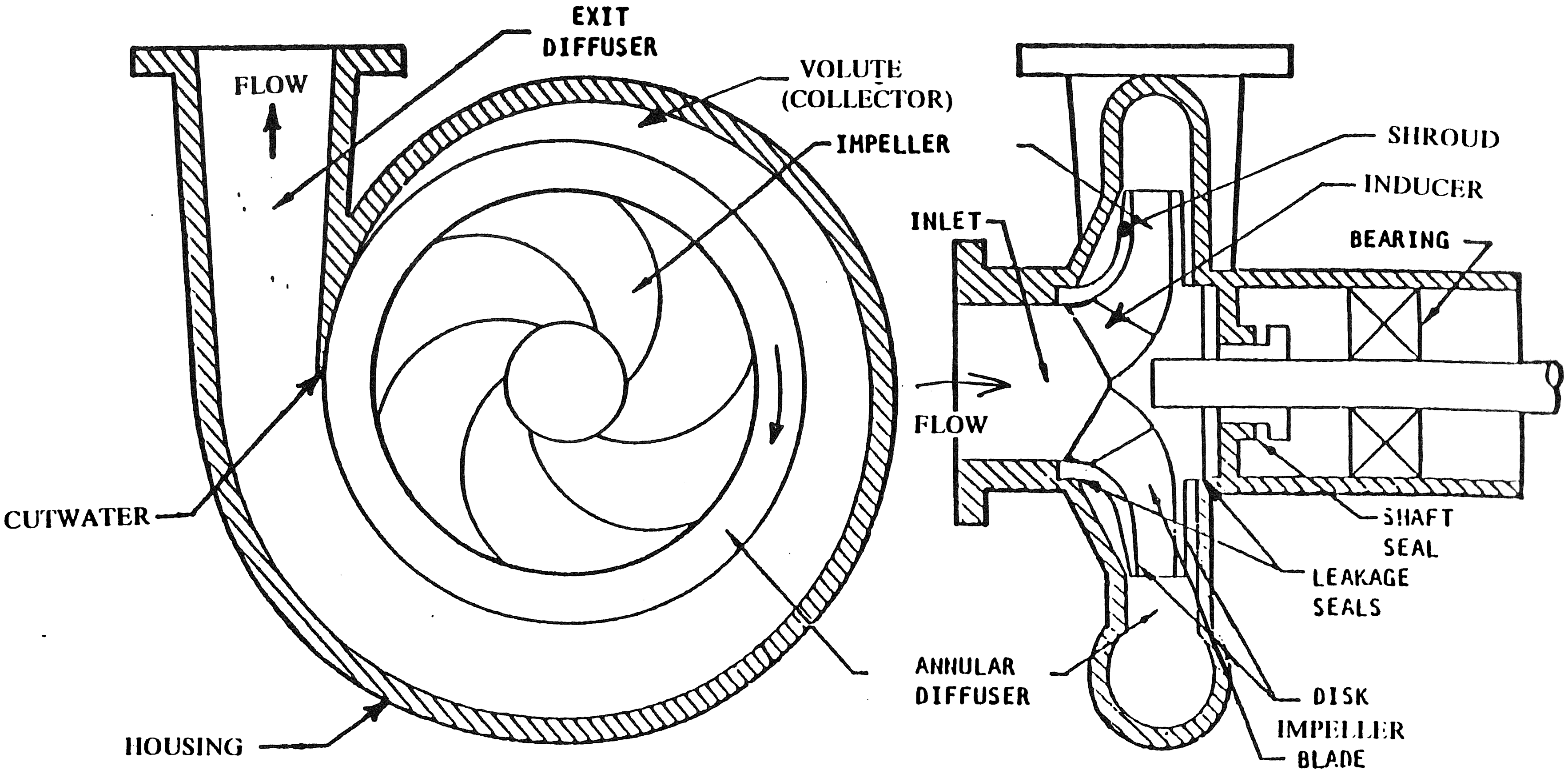
ROCKET PUMP TERMINOLOGY (Cont.)

TSH - THERMODYNAMIC SUPPRESSION HEAD

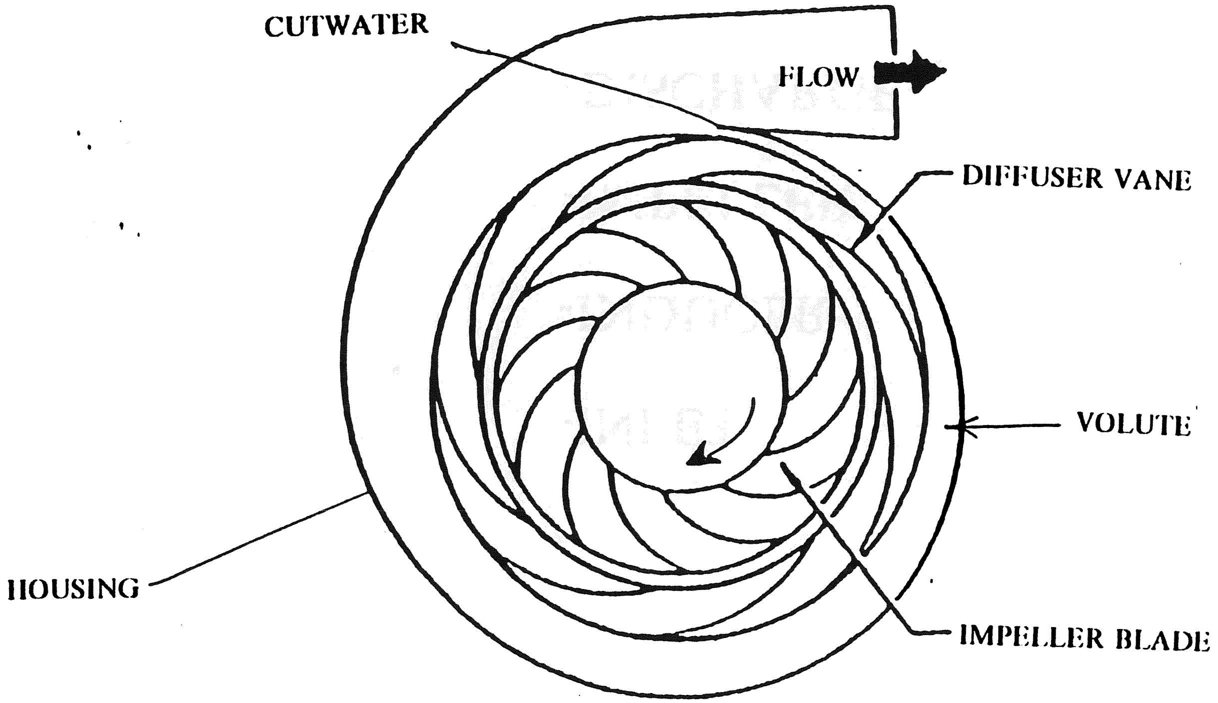
Ψ - HEAD COEFFICIENT $= \dfrac{C_U}{U}$

ϕ - FLOW COEFFICIENT $= \dfrac{C_M}{U}$

VOLUTE-TYPE CENTRIFUGAL PUMP



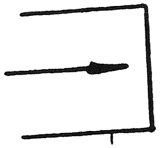
DIFFUSER-TYPE CENTRIFUGAL PUMP



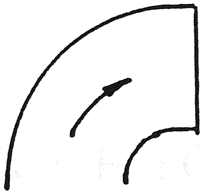
FUNDAMENTALS OF PUMP HYDRODYNAMIC DESIGN

- -INLET
- -INDUCER
- -IMPELLER
- -DISCHARGE

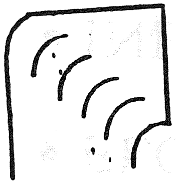
INLET CONFIGURATIONS



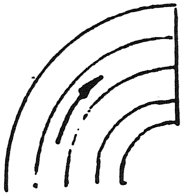
STRAIGHT



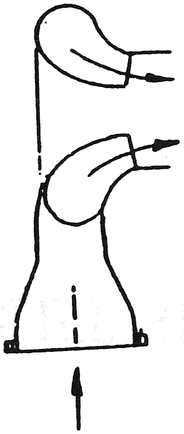
BEND



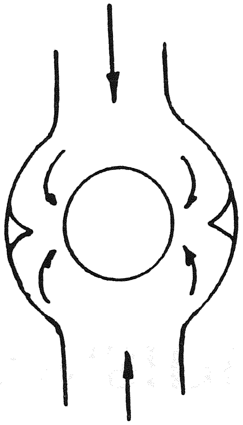
TURNING VANES



SPLITTER



SINGLE ENTRY
VOLUTE



DOUBLE ENTRY
VOLUTE

INLET SECTION - DESIGN CONSIDERATIONS

- LOW ΔP
- FLOW UNIFORMITY
- INLET GUIDE VANES
  - STRUCTURAL STRENGTHENING
  - FLOW INCIDENCE TO INDUCER

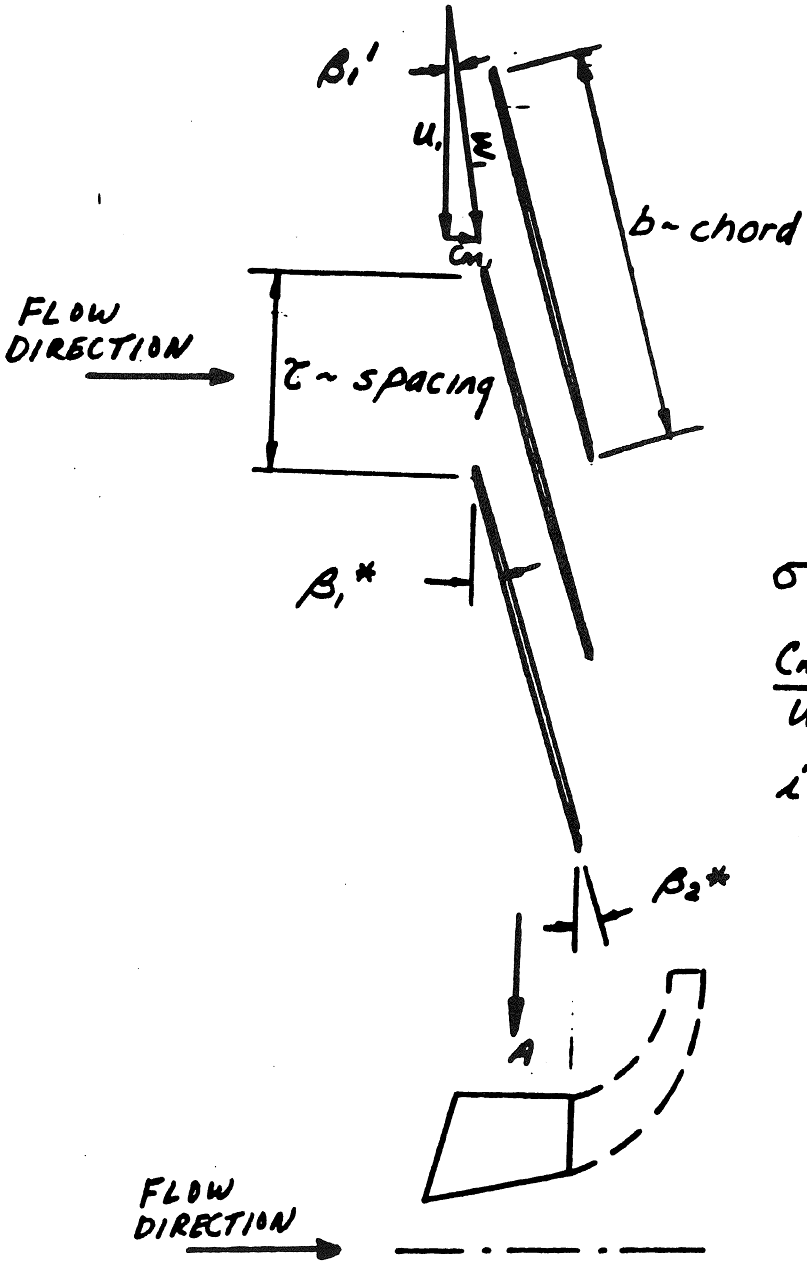
INDUCER

- SMALL PUMP TO INCREASE NPSH AT MAIN IMPELLER INLET BY BOOSTING FLUID PRESSURE TO ELIMINATE CAVITATION

INDUCER - DESIGN CONSIDERATIONS

- CAPABLE OF OPERATING UNDER MORE SEVERE INLET CONDITIONS THAN IMPELLER

INDUCER GEOMETRY



VIEW A

σ = SOLIDITY = $\frac{b}{\tau}$

$\frac{C_{m_1}}{U_1} = \phi_{1t} = \tan \beta_1'$

i = incidence angle = $\beta_{1t}^* - \beta_1'$

CAVITATION

- SPONTANEOUS FORMATION OF VAPOR BUBBLES IN THE LIQUID AS THE STATIC PRESSURE FALLS BELOW THE VAPOR PRESSURE
- EFFECTS
  - MARKED DROP IN EFFICIENCY
    - FORMATION OF THE VAPOR ALTERS THE LIQUID FLOW PATH THEREFORE PUMP PERFORMANCE
  - PITTING OR EROSION OF METAL PARTS
    - COLLAPSE OF VAPOR OR BUBBLES CREATES LOCAL PRESSURE FORCES THAT MAY RESULT IN MATERIAL DAMAGE
- TO ELIMINATE CAVITATION
  - INCREASE INLET FLOW HEAD (NPSH) TO IMPELLER BY USING INDUCER

SUCTION PERFORMANCE

- $NPSH$ = NET POSITIVE SUCTION HEAD

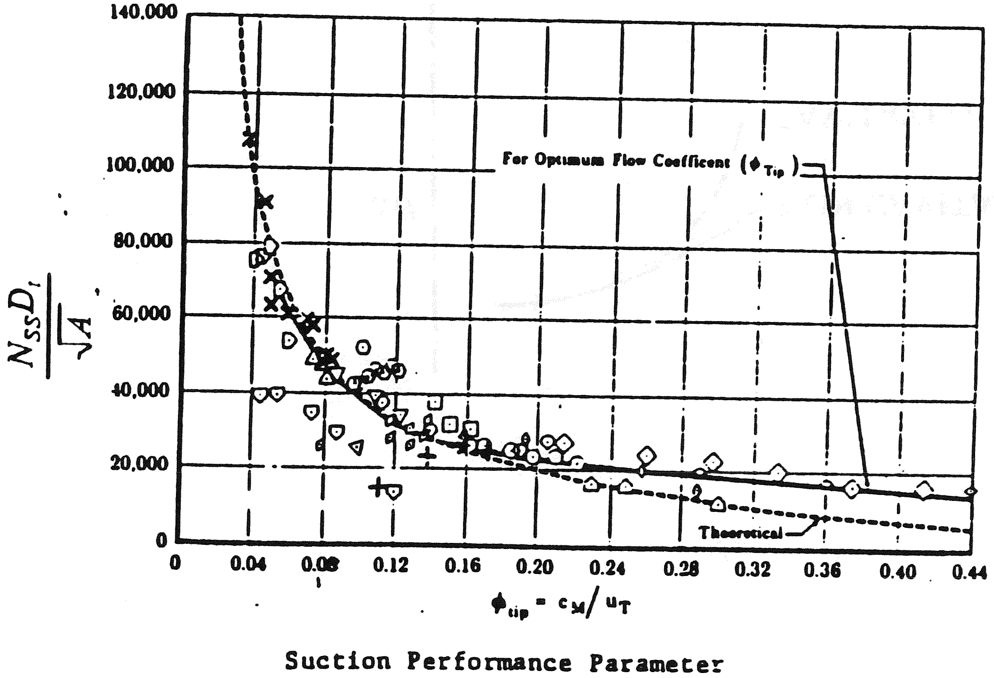
$$= H_{TOT} - H_{VAPOR}$$

$$= H_{STATIC} + \frac{Cm^2}{2g} - H_{VAP}$$

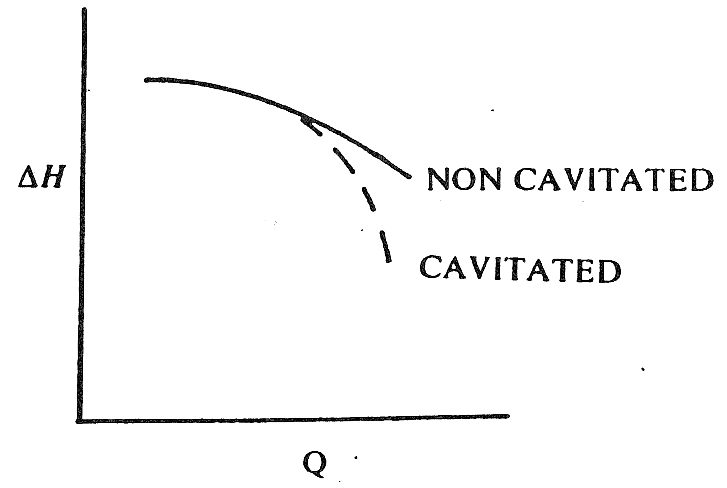
- TSH = THERMODYNAMIC SUPPRESSION HEAD
  - ALLOWS A FLUID TO BE PUMPED AT LOWER VALUES OF NPSH
  - $TSH_{H20} \doteq 0$
  - $TSH_{LOX} = 5\text{-}15ft$
  - $TSH_{LH2} = 90ft +$
- N_{SSH20} = SUCTION SPECIFIC SPEED
  - PARAMETERS TO EVALUATE THE CAPABILITY OF A PUMP TO OPERATE UNDER ADVERSE INLET CONDITIONS
- $N_{SSH20} = \dfrac{N\sqrt{Q}}{(NPSH_{FLUID} + TSH)^{3/4}}$

INDUCER N_{SS} CAPABILITY EVALUATION

- EMPIRICAL CURVE
- OPERATION IN H_20



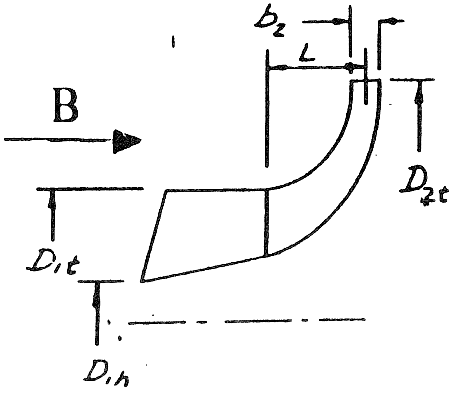
SUCTION PERFORMANCE CHARACTERISTICS

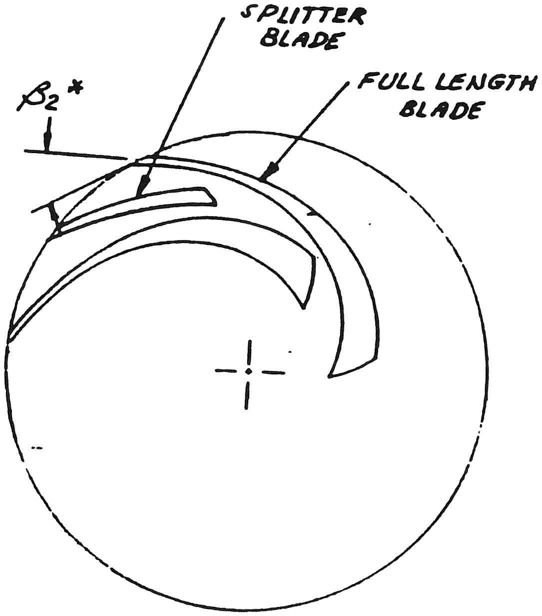


IMPELLER

- MAJOR CONTRIBUTOR TO HEAD RISE IN PUMP
- HEAD RISE CALCULATIONS
  - EULER -H_e
  - INPUT -H_I
  - ACTUAL - H

IMPELLER GEOMETRY





VIEW B

HEAD RISE (ΔH)

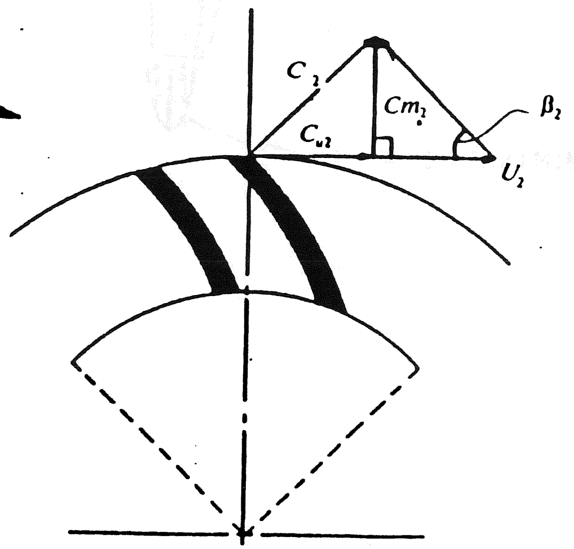
EULERS (H_e) - THEORETICAL

- $H_e = \dfrac{U_2 C_{u2} - U_1 C_{u1}}{g}$

- ASSUMING FLUID ENTERING PUMP HAS $C_{U1} = 0$

- $H_e = \dfrac{U_2 C_{U2}}{g}$

-SUBSTITUTING VELOCITIES FROM DISCHARGE VELOCITY TRIANGLE



- $H_e = \dfrac{U_2^2}{g} - \dfrac{U_2 Cm_2}{g \tan \beta_2}$

  OR

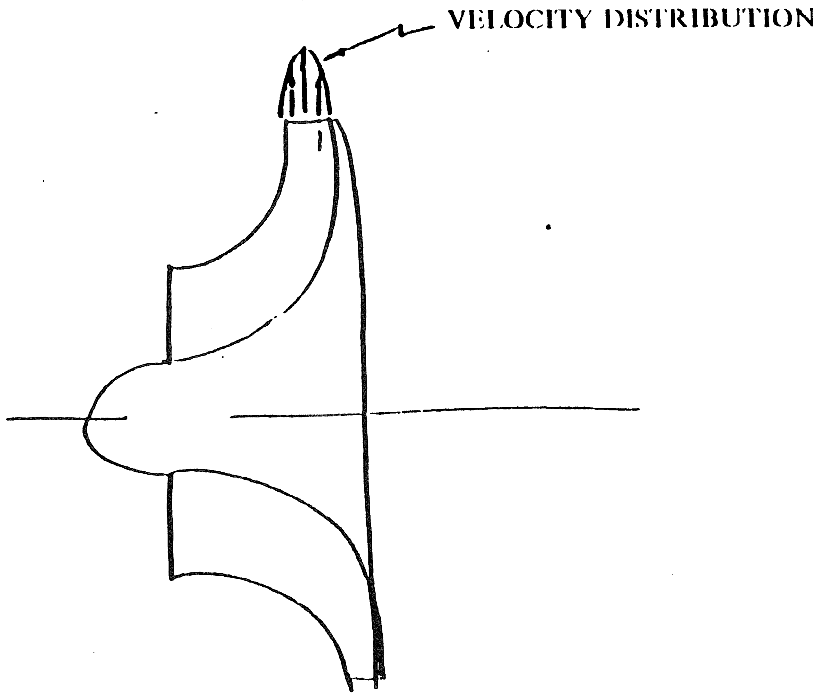
- $\dfrac{H_e g}{U_2^2} = \Psi_e = 1 - \dfrac{\varphi}{\tan \beta_2}$
- $\Psi_e = \dfrac{\Delta H_e g}{U_2^2} = \dfrac{C_{U_2}}{U_2}$

- 1.) INLET
- 2.) EXIT

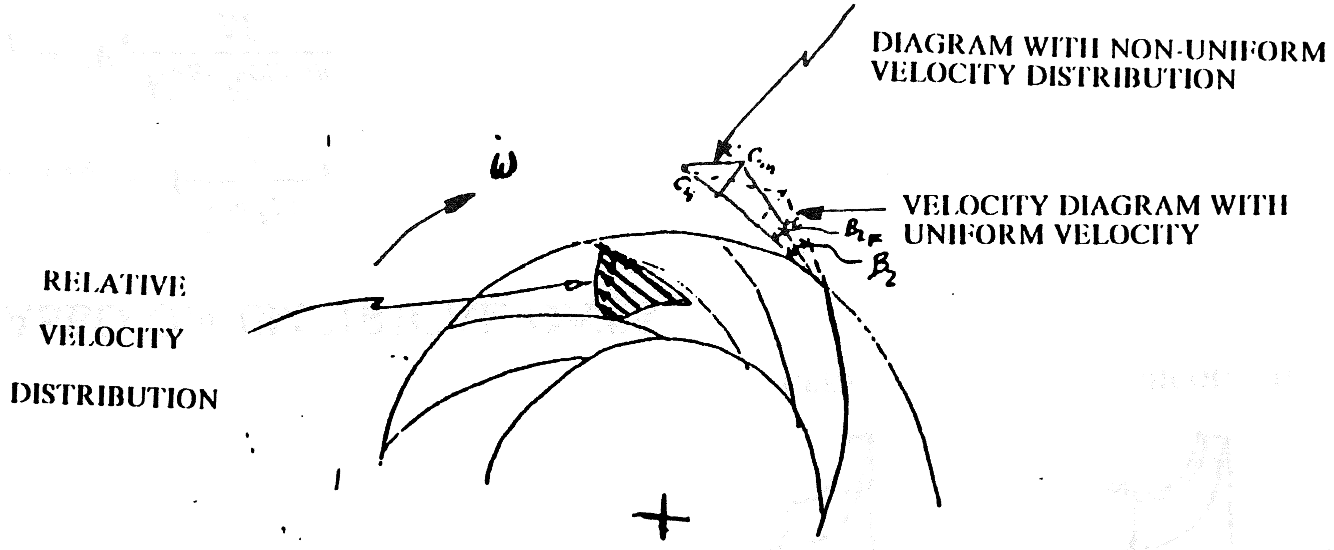
INPUT (H_I) = EULERS (H_e) - INABILITY OF A REAL IMPELLER TO IMPART FULL THEORETICAL ENERGY INPUT TO FLUID

$$H_I = H_E \times K_{VEL} K_{CLEAR} / M$$

K_{VEL} -NON UNIFORM VELOCITY DISTRIBUTION ACROSS THE IMPELLER CHANNEL



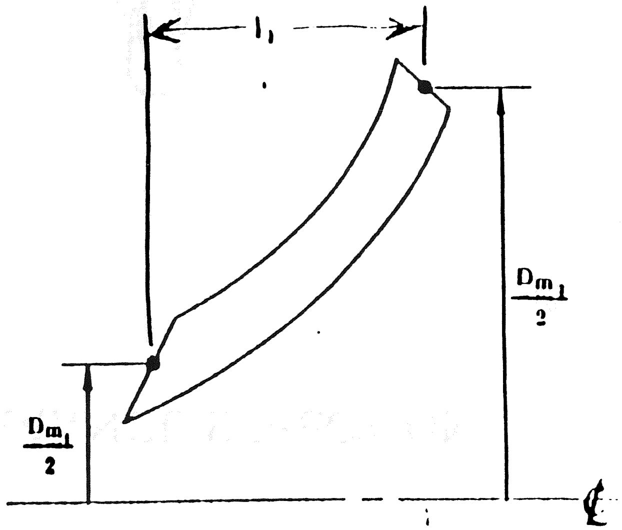
M - SLIP FACTOR - NON-UNIFORM VELOCITY DISTRIBUTION FROM BLADE TO BLADE



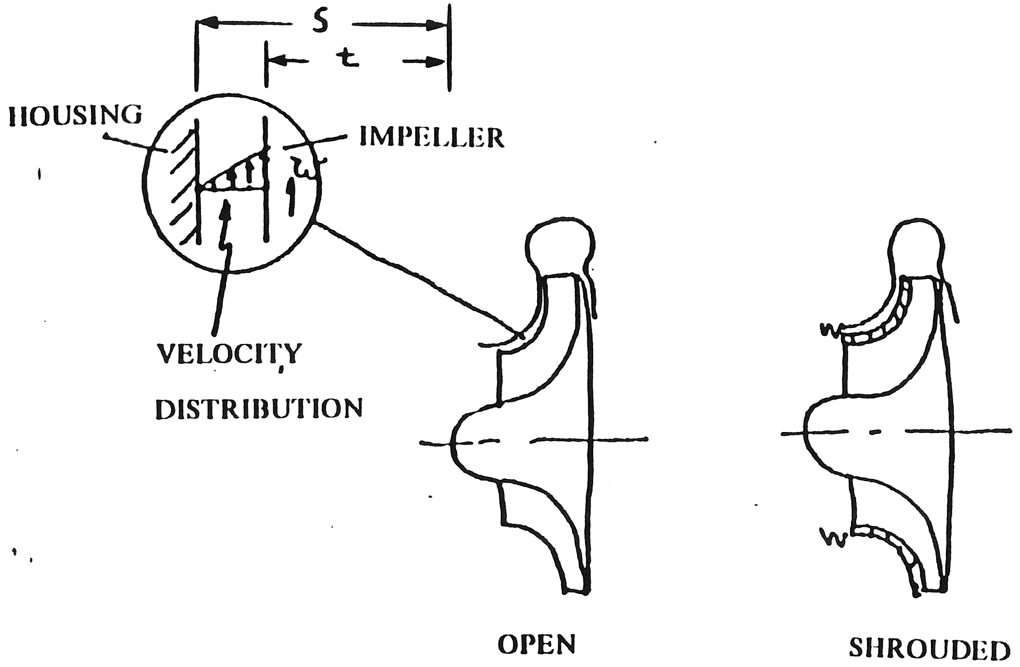
- EMPIRICAL DATA CORRELATION (PFIEDERER'S)
- $M = 1 + \dfrac{1 + .06 \sin \beta_2}{BLADE\#(1 + \delta)[x^2 + 0.25(1 - \delta)^2]^{1/2}}$

$\delta = \dfrac{Dm_1}{Dm_2}$,

$X = \dfrac{h}{Dm_2}$



K_{CLEAR} - OPEN, UNSHROUDED IMPELLER CLEARANCE EFFECT ON ANGULAR VELOCITY



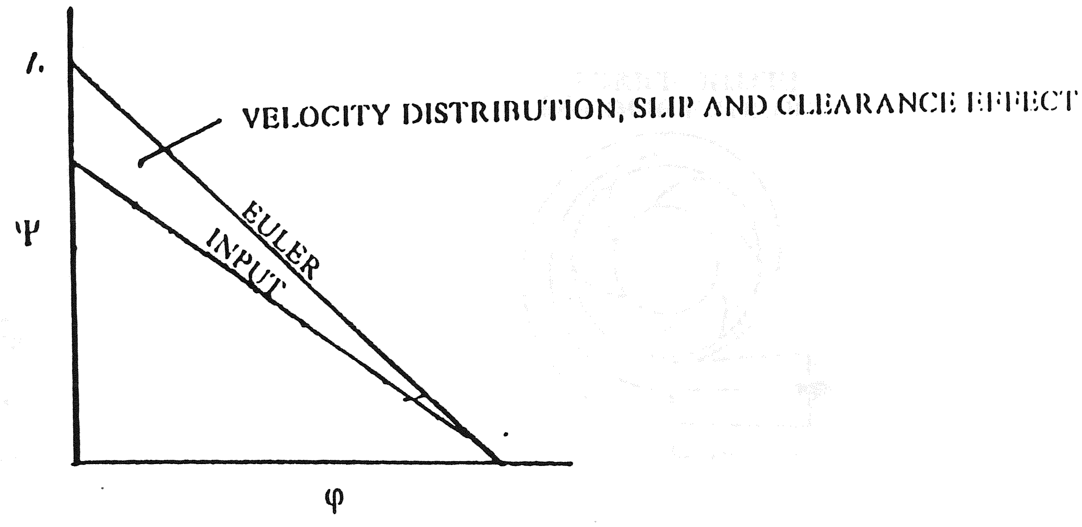
- BASED ON EMPIRICAL DATA

$$K_{CLEAR} = \left(\frac{1 + t/s}{2}\right)^2$$

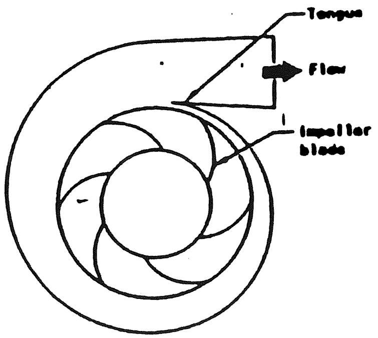
$$\Psi_I = \Psi_e \frac{K_{VEL} K_{CLEAR}}{M}$$

$$\Psi_A = \Psi_I \, \eta_{hd}$$

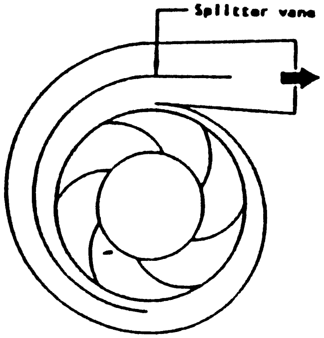
OVERALL EFFECT OF IMPELLERS INABILITY TO IMPART FULL THEORETICAL ENERGY TO FLUID



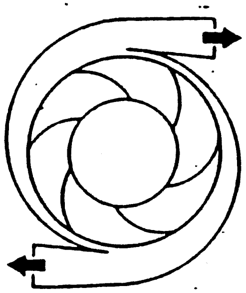
DISCHARGE CONFIGURATIONS



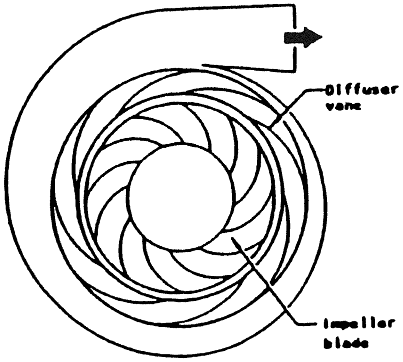
(a) SINGLE-TONGUE SINGLE-OUTLET



(b) DOUBLE TONGUE SINGLE OUTLET



(c) DOUBLE-TONGUE DOUBLE-OUTLET



(d) VANED DIFFUSER

DISCHARGE

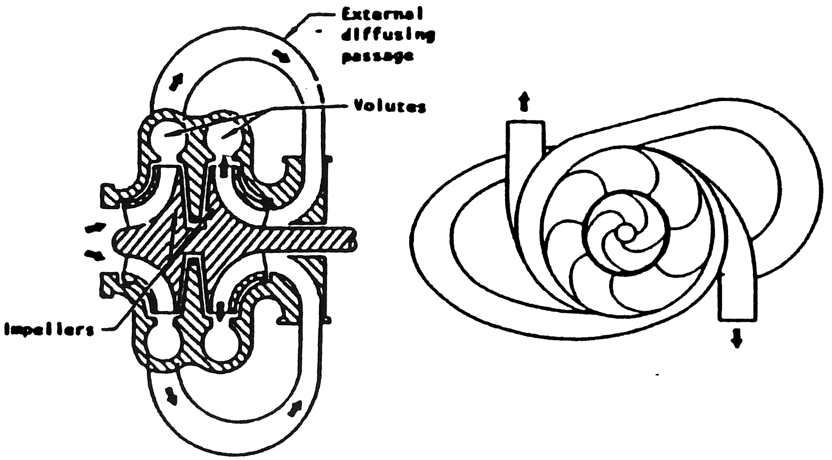
DIFFUSER

- REDUCES THE HIGH ABSOLUTE VELOCITY AT THE IMPELLER TIP TO CONVERT THE KINETIC ENERGY TO STATIC HEAD OR PRESSURE
- DESIGN CONSIDERATIONS
  - FLOW STABILITY
  - LOSSES

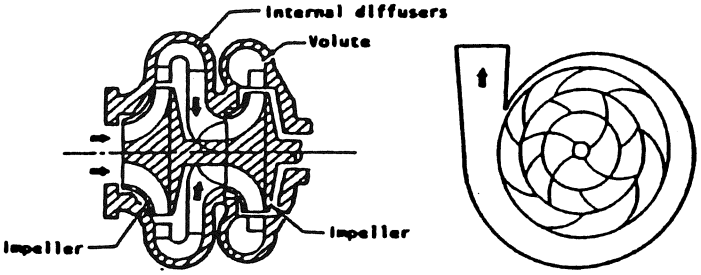
COLLECTOR

- RECEIVES FLUID AND GUIDES IT TO OUTLET PORT
- DESIGN CONSIDERATIONS
  - FLOW STABILITY
  - MINIMUM ΔP

INTERSTAGE CONFIGURATIONS



(a) EXTERNAL DIFFUSING PASSAGE



(b) INTERNAL CROSSOVER PASSAGE

PUMP CLASSIFICATION

N_S = SPECIFIC SPEED

- CLASSIFIES TYPE OF PUMP NEEDED WITH RESPECT TO COMBINATIONS OF N, Q, ΔH

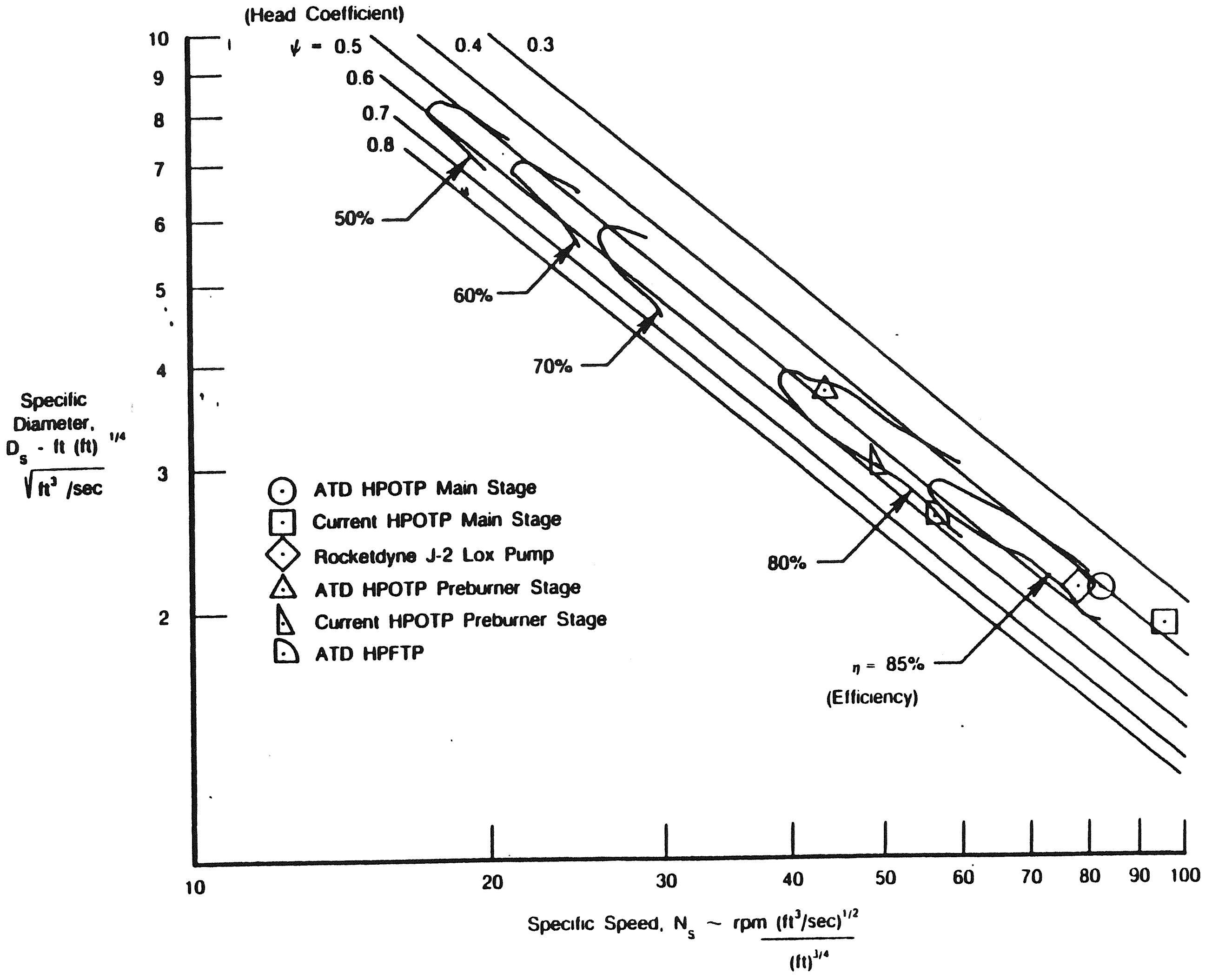
$$N_S = \frac{N\sqrt{Q}}{(\Delta H)^{3/4}}$$

D_S SPECIFIC DIAMETER

- PARAMETER TO COMPARE PUMP LOADING AND BLADE ANGLE

$$D_S = \frac{D_2(\Delta H)^{1/4}}{\sqrt{Q}}$$

Specific Diameter Versus Specific Speed Correlation for Optimized Designs



PERFORMANCE

- HEAD VS FLOW
  - PARAMETERS
  - CHARACTERISTICS
- SIMILARITY RELATIONSHIPS

- EFFICIENCY
  - PARAMETERS
  - CHARACTERISTICS

HEAD - FLOW PARAMETERS

Ψ = HEAD COEFFICIENT

$$= \Delta \cdot H \frac{g}{U^2} = \frac{Cu}{U}$$

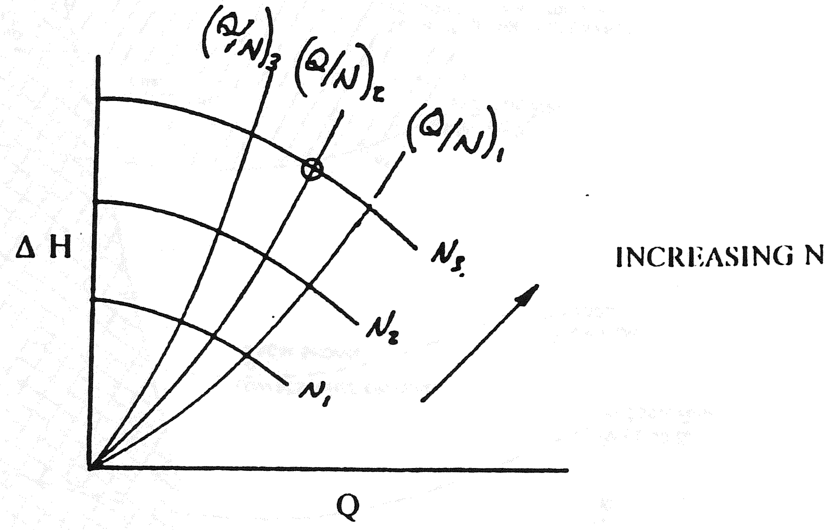
$$\propto \frac{\Delta H}{N^2 D^2}$$

φ = FLOW COEFFICIENT

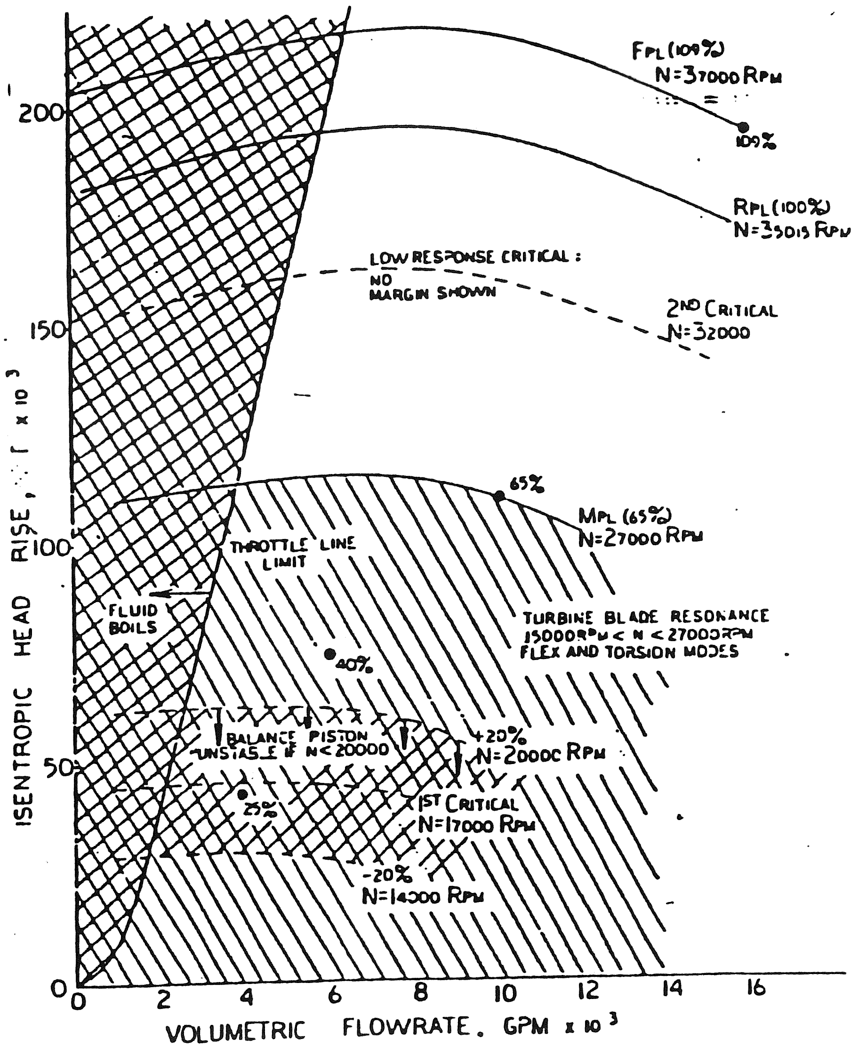
$$= \frac{Cm}{U}$$

$$\propto \frac{Q}{ND^3}$$

HEAD - FLOW CHARACTERISTIC

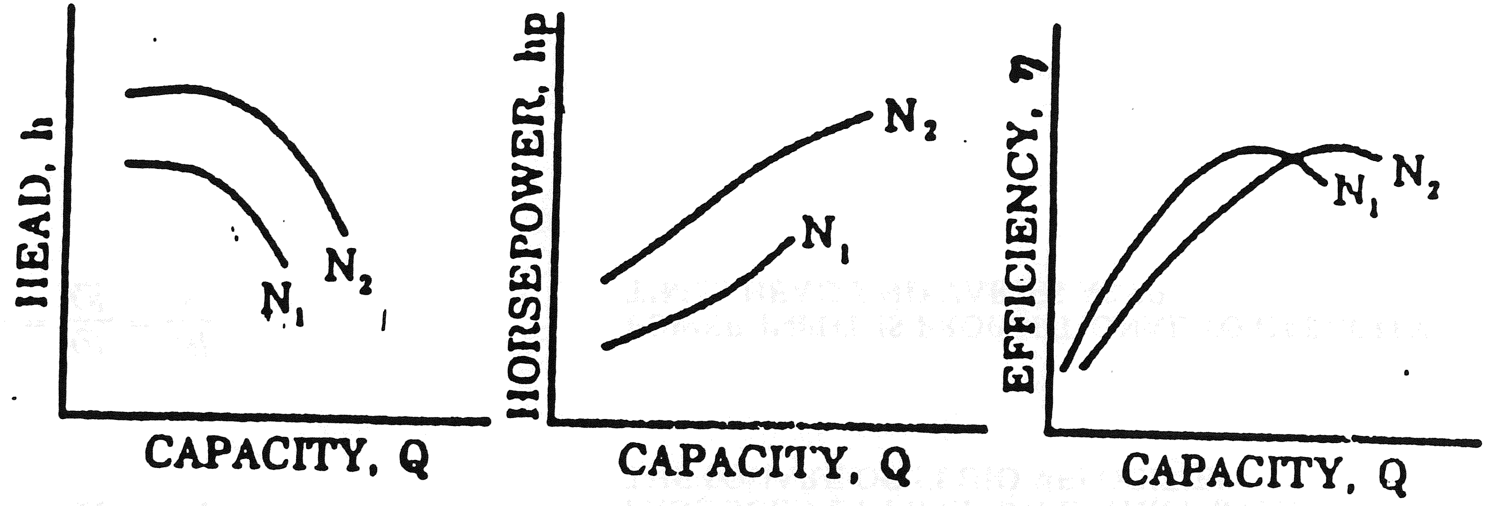


CURRENT SSME HPFTP HEAD VS FLOW CURVES WITH DESIGN LIMITATIONS



CHARACTERISTICS PERFORMANCE

$N_2 > N_1$



Variations in Efficiency, Power,
and Head with Capacity

SIMILARITY RELATIONS

$$\frac{Q_1}{Q_2} = \frac{N_1}{N_2}$$

FLOW CONDITIONS WILL BE SIMILAR IF CAPACITY IS CHANGED PROPORTIONALLY WITH RPM

$$\frac{H_1}{H_2} = \frac{Q_1^2}{Q_2^2} = \frac{N_1^2}{N_2^2}$$

IF VISCOSITY IS NEGLECTED, THE HEAD PRODUCED BY PUMP WILL VARY WITH THE SQUARE OF FLUID VELOCITIES

$$\frac{P_1}{P_2} = \frac{Q_1^3}{Q_2^3} = \frac{N_1^3}{N_2^3}$$

POWER INPUT IS PROPORTIONAL TO CAPACITY TIMES HEAD AND VARIES AS N^3

EFFICIENCY PARAMETER

η_{hd} = HYDRAULIC EFFICIENCY

$$\eta_{hd} = \frac{\Delta H_{OUT}}{\Delta H_{IN}}$$

$$\eta_{hd} = \frac{\Delta H_P}{\Delta H} = \frac{\text{PUMP HEAD RISE}}{\text{INPUT HEAD RISE}} = \frac{\dfrac{\Psi_A U_1^2}{g} - (1 - \eta_{REC}) \dfrac{C_2^2}{2g}}{\dfrac{\Psi i U_1^2}{g}}$$

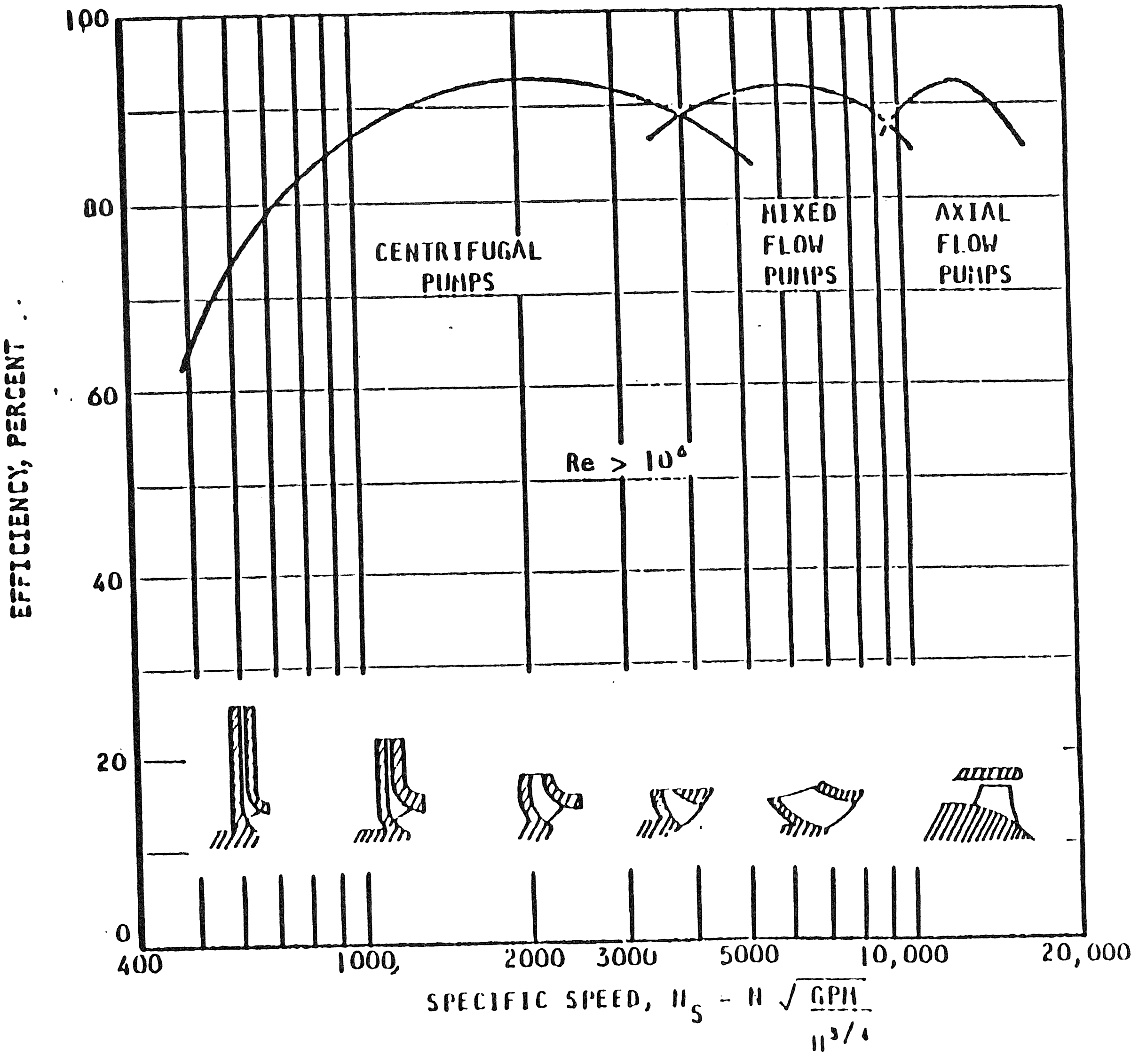
η_m = MECHANICAL EFFICIENCY

$$\eta_m = \frac{SHP - \text{MECH LOSS}}{SHP}$$

η_v = VOLUMETRIC EFFICIENCY

$$\eta_v = \frac{Q_{OUT}}{Q_{OUT} + Q_{LEAKAGE}}$$

RELATIVE MERITS OF VARIOUS TYPES OF PUMPS

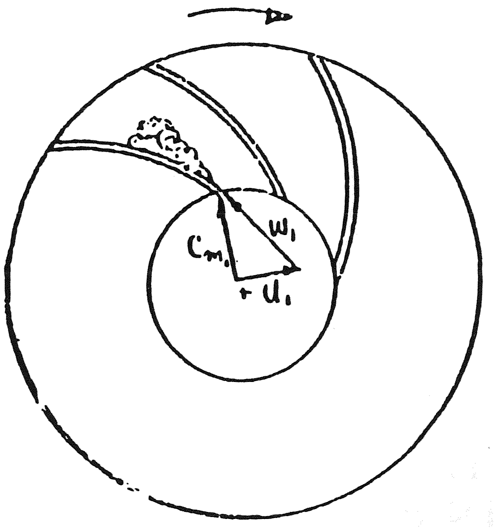


HYDRAULIC EFFICIENCY (η_{hd})

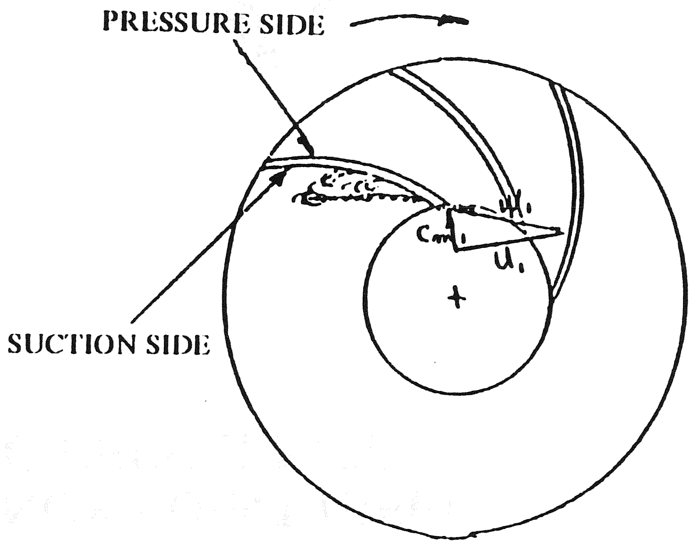
HYDRAULIC LOSSES

- SKIN FRICTION
- EDDY AND SEPARATION DUE TO CHANGES IN DIRECTION AND VELOCITY OF FLOW

SHOCK LOSSES - FLUID APPROACHES AT HIGH ANGLE OF ATTACK

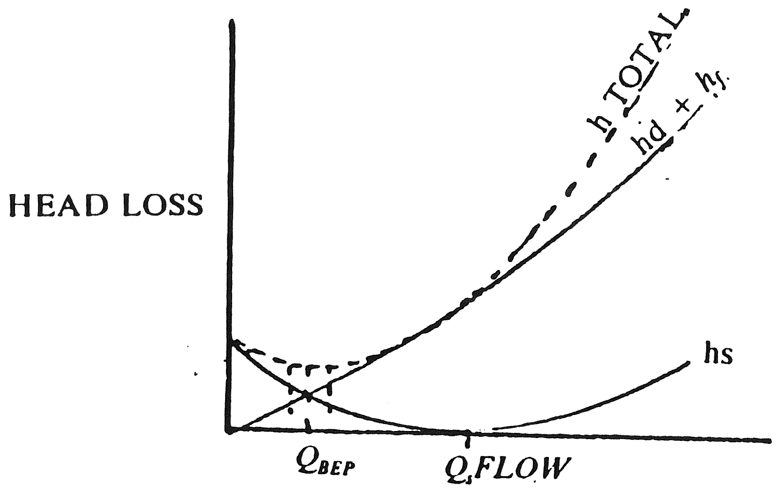


HIGH FLOW



LOW FLOW

BEST EFFICIENCY POINT (BEP)
HYDRAULIC EFFICIENCY

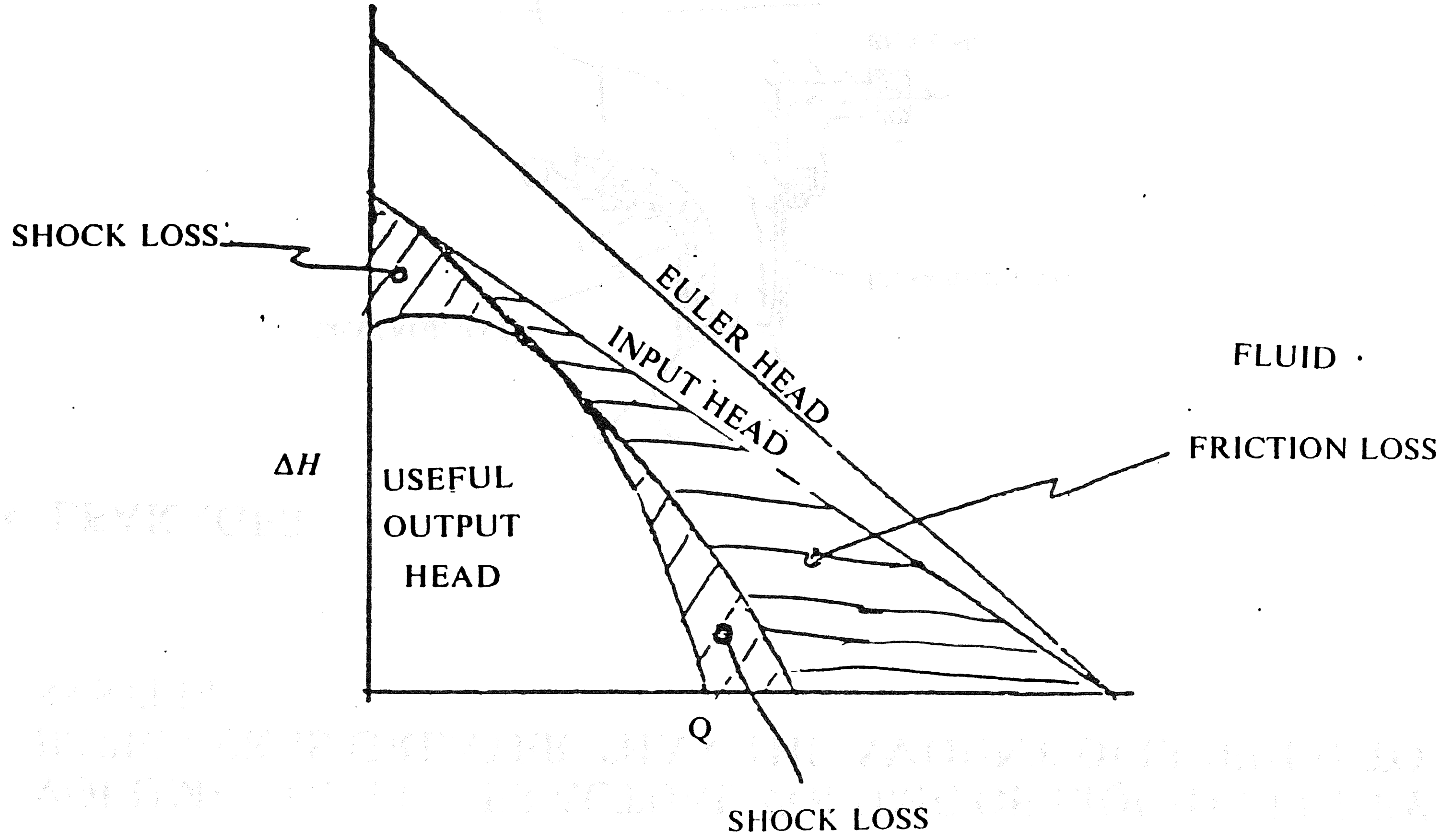


$h_{FRICTION}\,(h_f) \propto Q^2$

$h_{DIFFUSER}\,(h_d) \propto Q^2$

$h_{SHOCKLOSS}\,(h_s) \propto (Q - Q_s)^2$

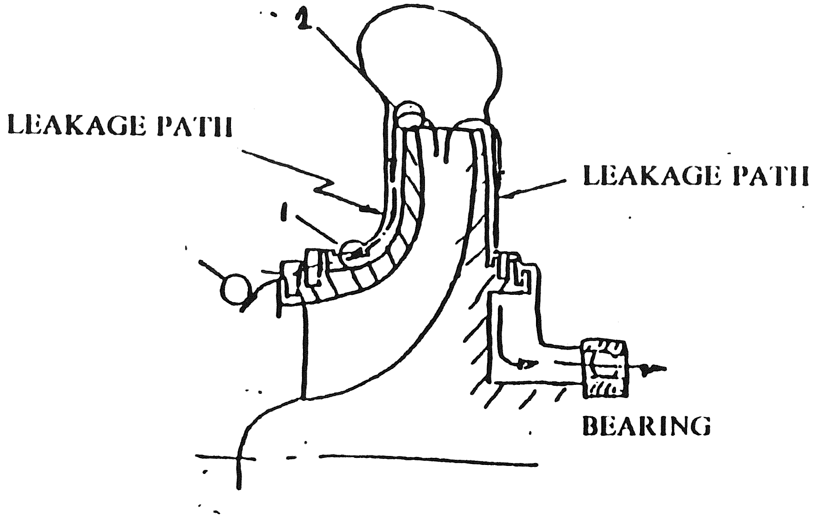
OVERALL EFFECT OF HYDRAULIC LOSSES ON HEAD OUTPUT



VOLUMETRIC EFFICIENCY (η_v)

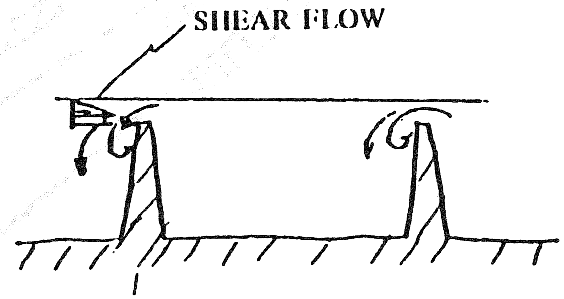
- VOLUMETRIC LOSSES-ACTUAL VOLUME OF LIQUID FELT BY THE IMPELLER IS GREATER THAN THE AMOUNT DELIVERED TO THE SYSTEM

- LEAKAGES



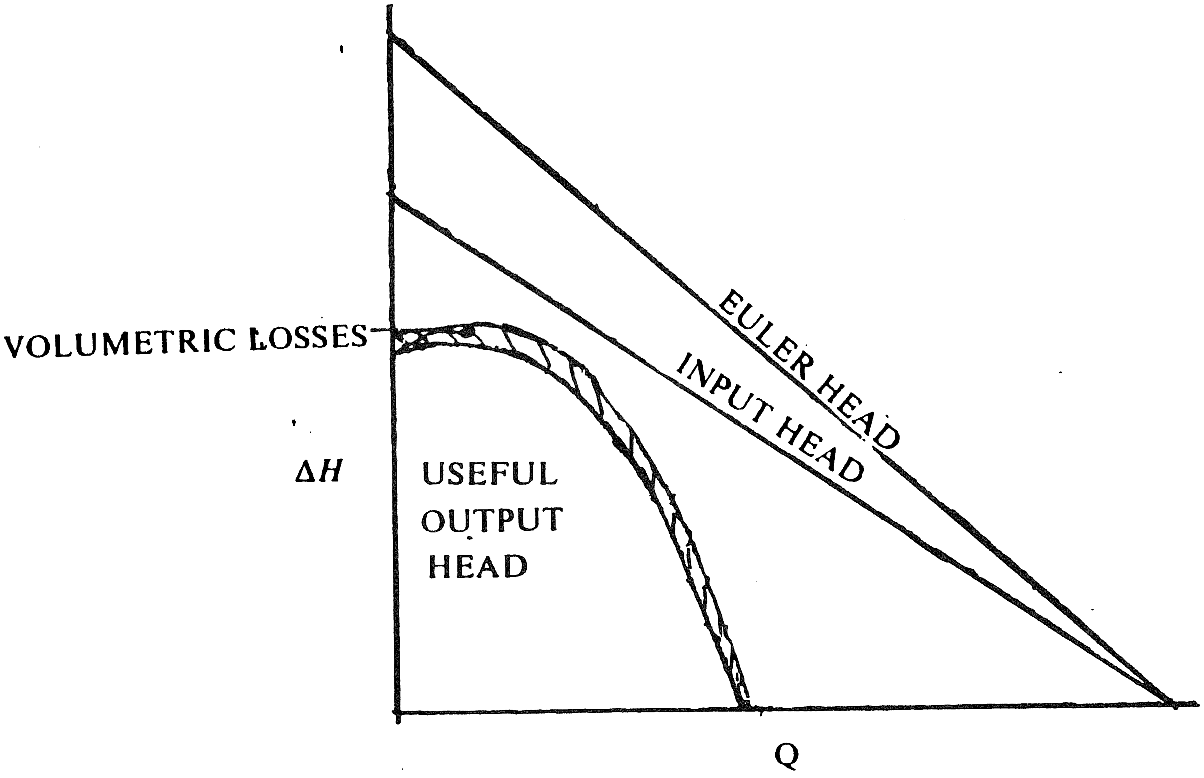
SHROUDED IMPELLER

- NOT CLEARLY DEFINABLE
- LEAKAGE OCCURS ACROSS THE BLADE TIPS FROM PRESSURE SIDE TO SUCTION SIDE
- EFFECTS ARE TWO FOLD
  - CLEARANCE EFFECTS (K_{CLEAR})
  - EFFICIENCY



OPEN IMPELLER

OVERALL EFFECT OF VOLUMETRIC LOSSES ON HEAD OUTPUT



MECHANICAL EFFICIENCY (η_M)

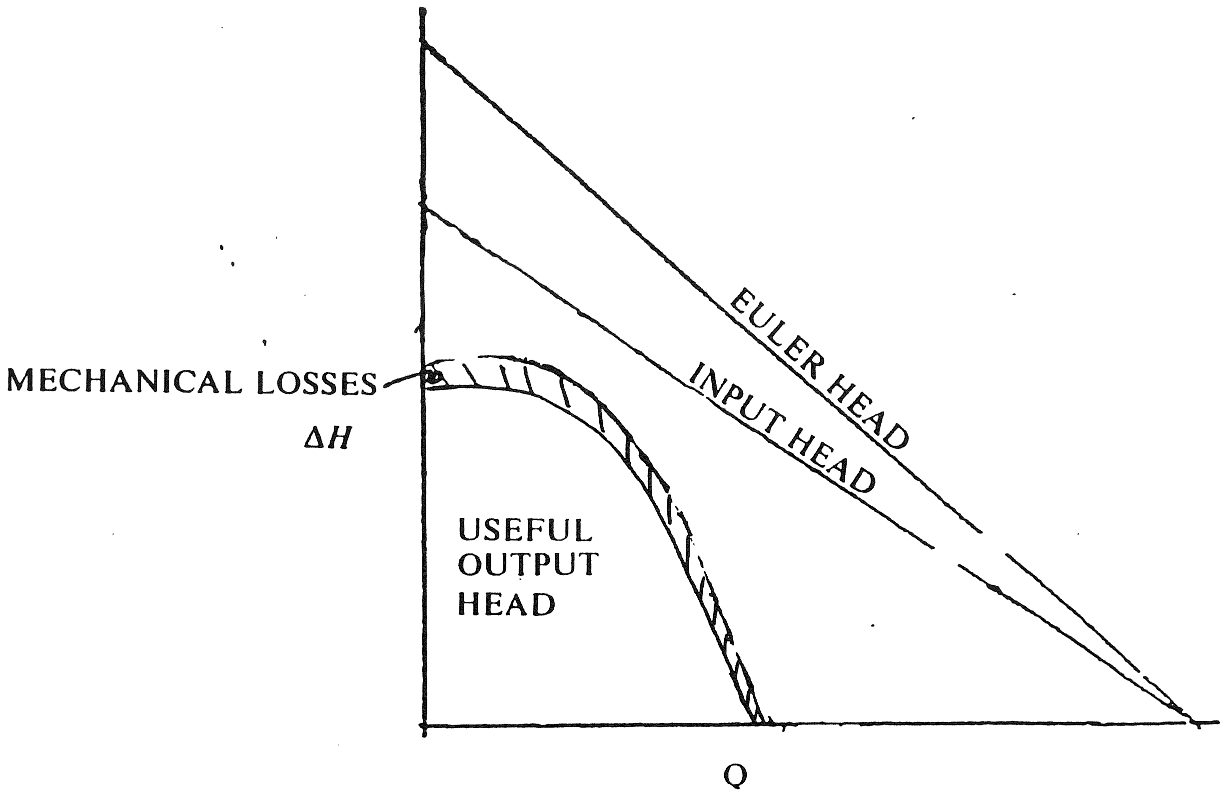
MECHANICAL LOSSES

- BEARING FRICTION AND SEAL RUBBING ARE USUALLY A SMALL PERCENTAGE OF MECHANICAL LOSSES DEPENDING ON PUMP SIZE

- DISK FRICTION
  - MAJOR MECHANICAL LOSS
  - HYDRAULIC IN NATURE, BUT GENERALLY DO NOT EFFECT η_{hd}

DISC TO WALL CLEARANCE - POWER REQUIRED TO DRIVE A DISC INCREASES AS THE CLEARANCE BETWEEN THE DISC AND WALL IS INCREASED

OVERALL EFFECT OF MECHANICAL LOSSES ON HEAD OUTPUT



OVERALL PUMP EFFICIENCY

η_{OP} = OVERALL EFFICIENCY

$$\eta_{OP} = \frac{LHP_{OUT}}{SHP_{IN}}$$

$$\eta_{OP} = \frac{\Delta h_{IDEAL}}{\Delta h_{ACTUAL}}$$

$$\eta_{OP} \propto \eta_{hd}\eta_{v}\eta_{m}$$

$$\eta_{OP} = \frac{\dot{\omega}\Delta H_{A}/550}{\left[\frac{(\dot{\omega} + \dot{\omega}_{L})(U_{t}^{2}\Psi)}{550g} + fh_{Pd}\right]\left[\frac{1}{1 - K_{BS}}\right]}$$

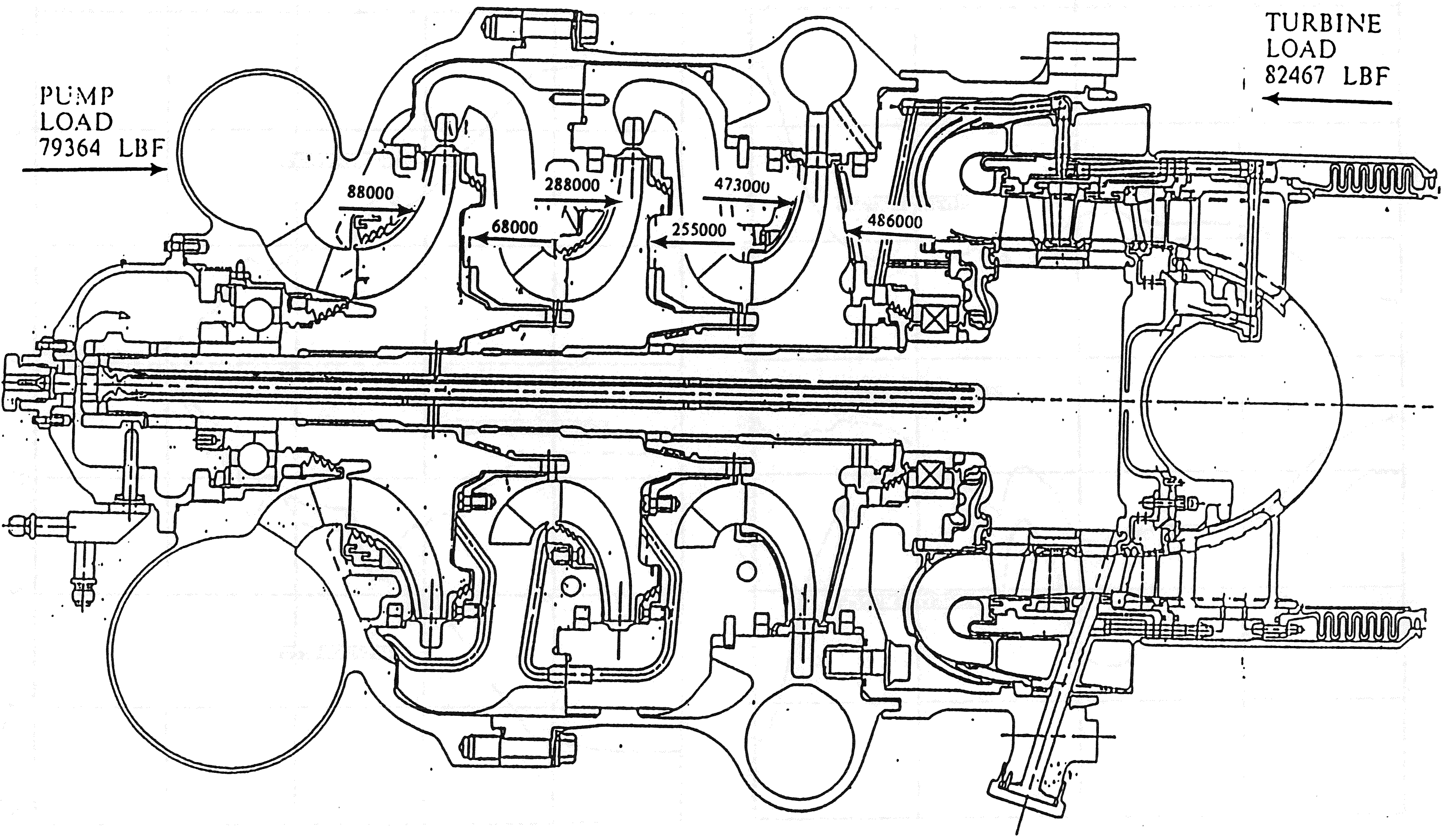
WHERE:

fh_{Pd} = *DISC FRICTION LOSSES, hp*

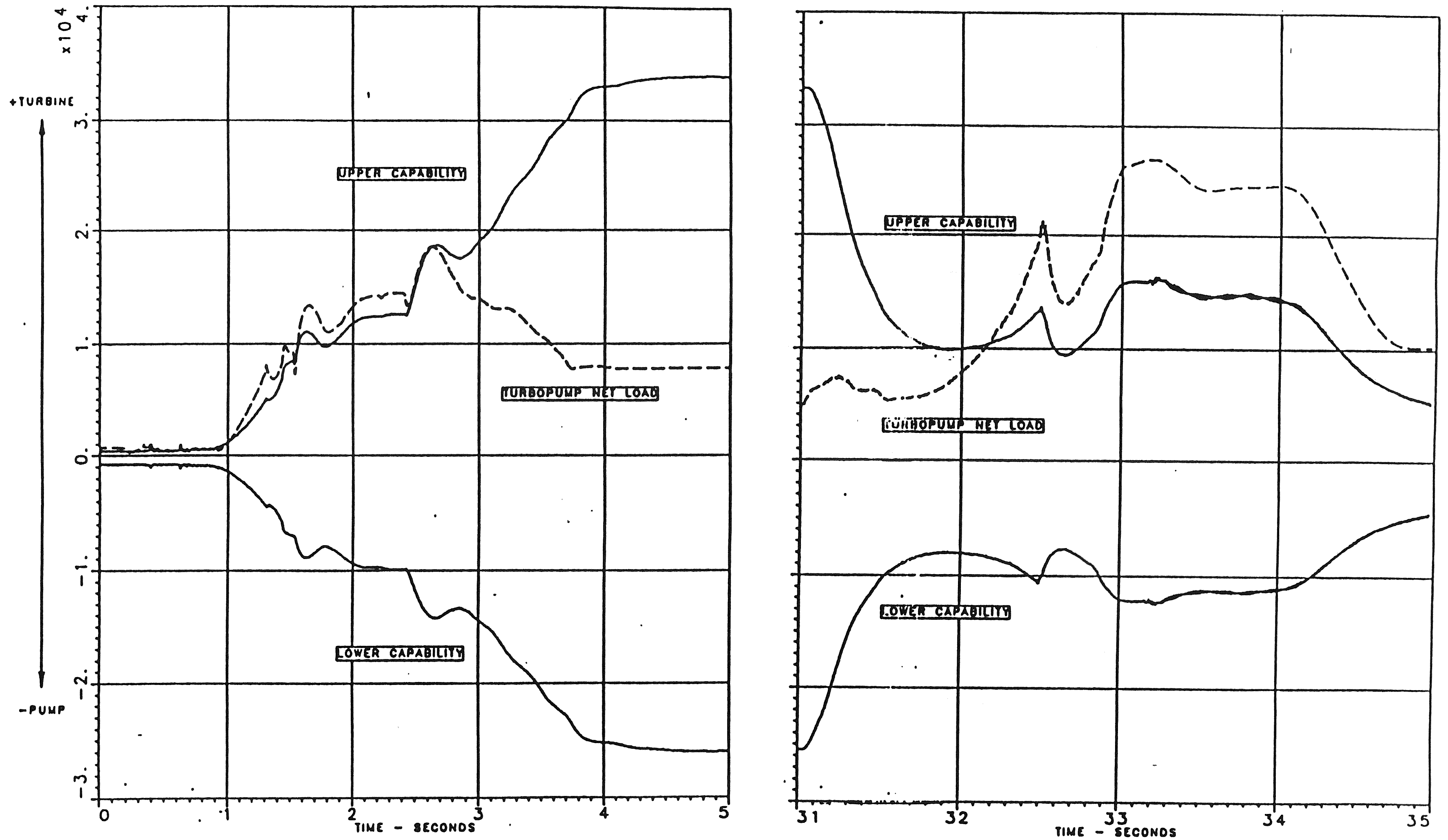
K_{BS} = *BEARING AND SEAL LOSSES, hp*

AXIAL LOADS

HIGH PRESSURES AND LARGE AREAS RESULT IN HIGH AXIAL LOADS ON COMPONENTS



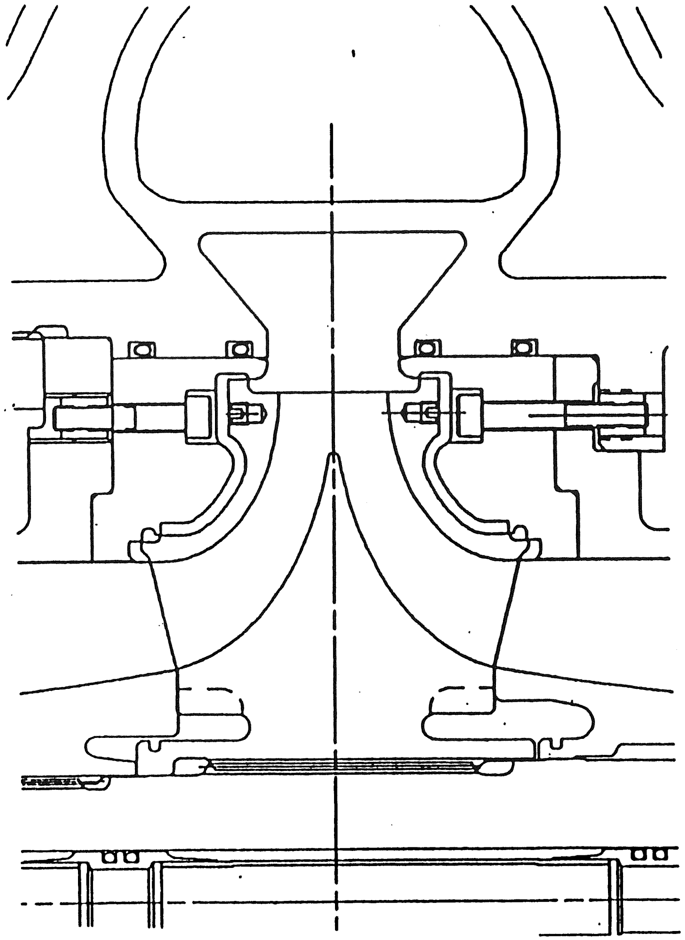
START UP AND SHUTDOWN
USUALLY RESULTS IN IMBALANCED LOADS

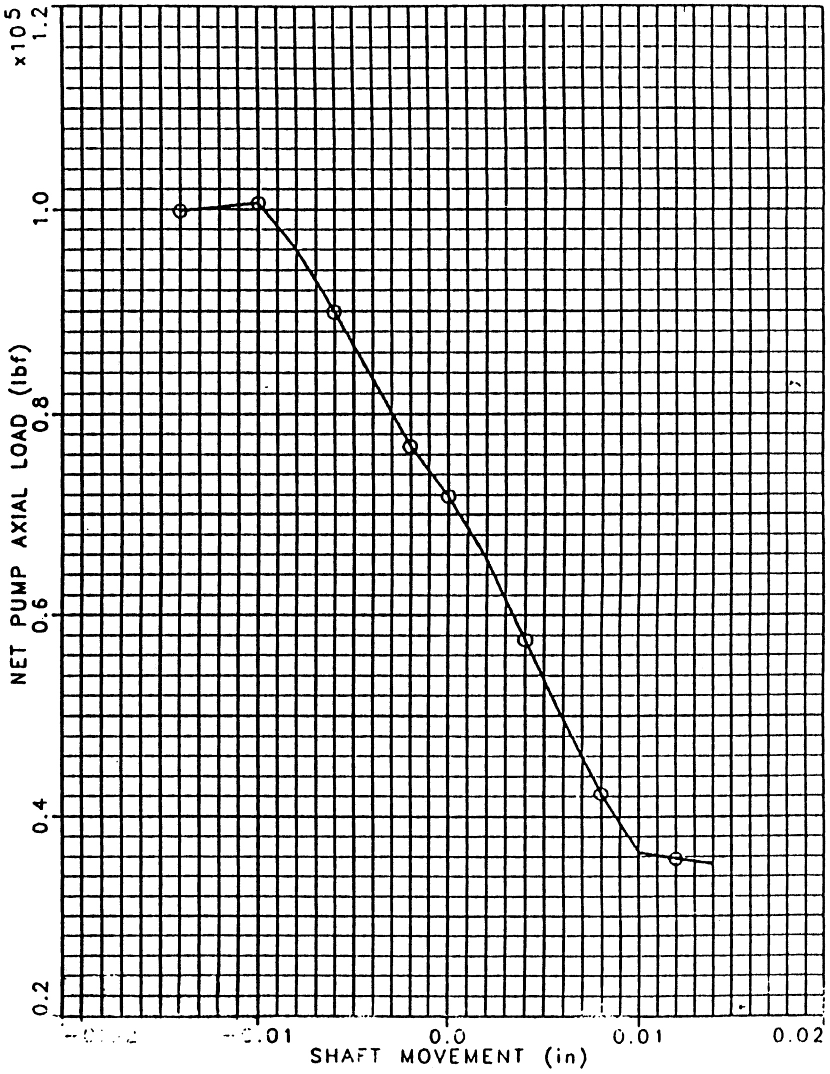


- IMBALANCES RESULT IN BEARING LOADS AND / OR RUB LOADS
- THESE CAN EFFECT BEARING LIFE AND PUMP PERFORMANCE
- TO ELIMINATE ALLEVIATE IMBALANCES DURING STARTUP AND / OR SHUTDOWN
  - USE AN EXTERNAL PRESSURE SOURCE / VENT
  - BEARING PRELOAD MAGNITUDE AND / OR DIRECTION
  - PUMP AND / OR TURBINE DESIGNS
  - STARTUP AND / OR SHUTDOWN PROCEDURES

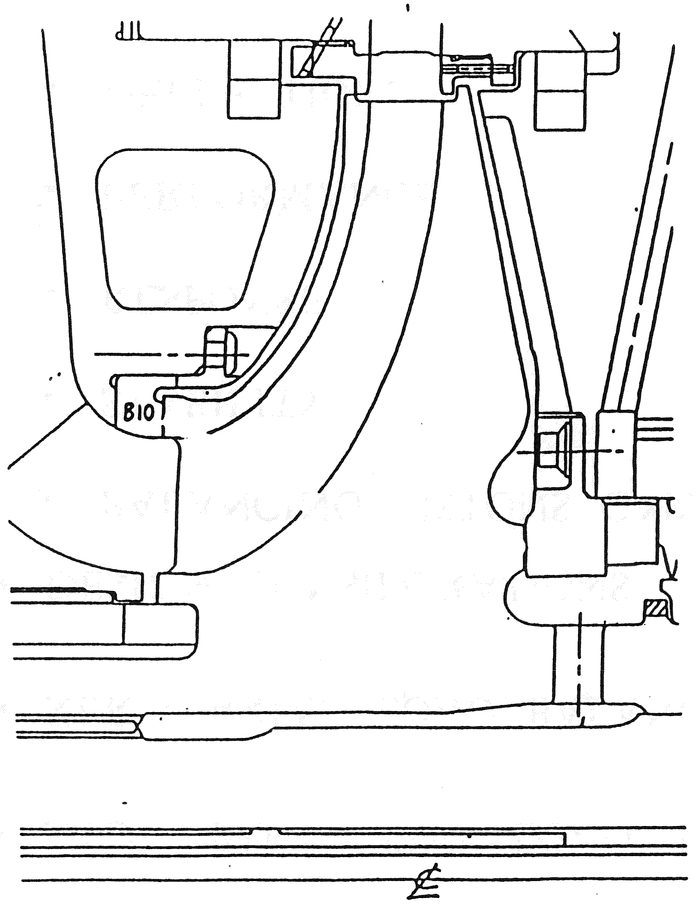
AXIAL LOAD BALANCING PISTONS USED TO ELIMINATE IMBALANCES

HPOTP





HPFTP



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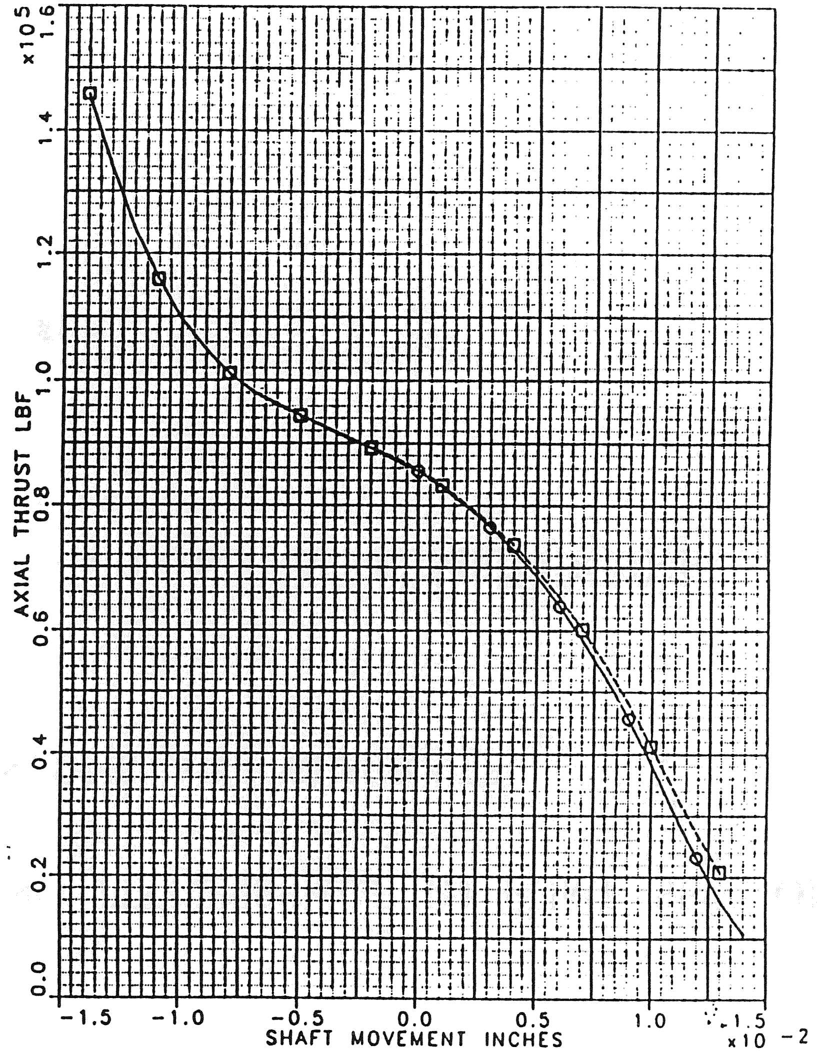
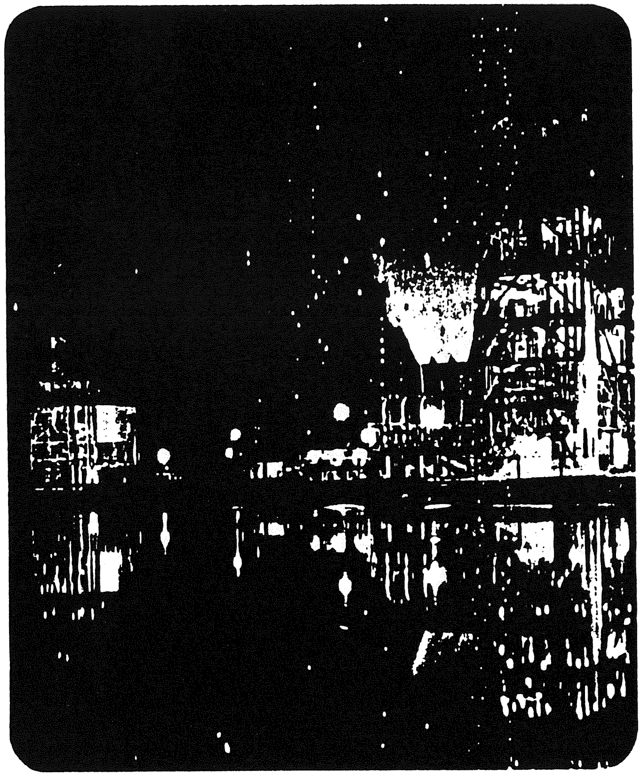


AXIAL THRUST LBF
x10 5
0.0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
SHAFT MOVEMENT INCHES
x10 -2
-1.5
-1.0
-0.5
0.0
0.5
1.0
1.5

- ATD / RD - AXIAL LOAD BALANCING PISTONS ARE IN PUMP SECTION
- XLR - AXIAL LOAD BALANCING PISTONS ARE SEPERATE
- DESIGN CONSIDERATIONS
  - BALANCING PISTONS CAPABILITY +/- LOAD OF PUMP/TURBINE
  - STABILITY
  - RESPONSE
  - PERFORMANCE
  - DEFLECTIONS

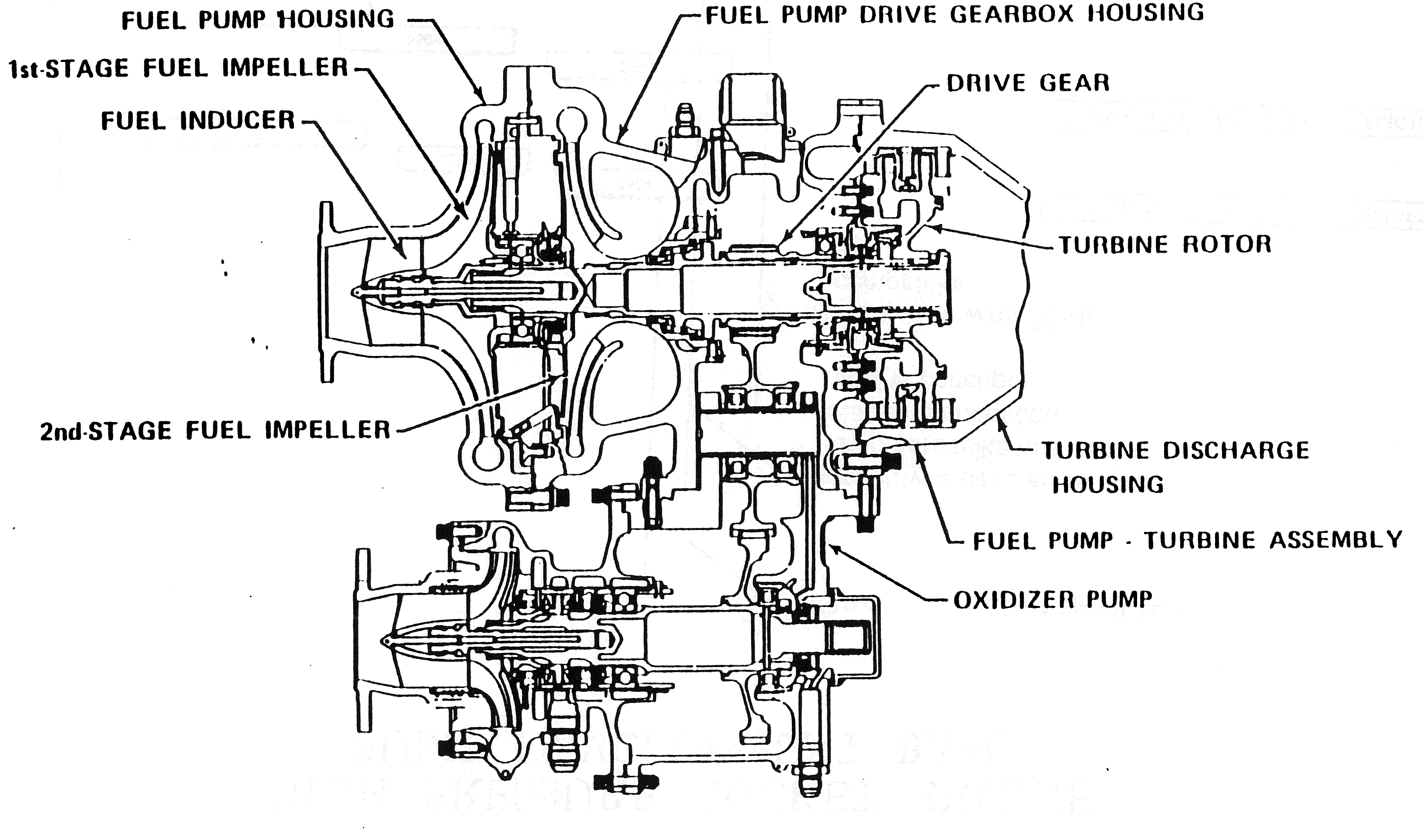
P&W ROCKET ENGINE PUMPS HISTORY

PRATT & WHITNEY ROCKET ENGINE EXPERIENCE

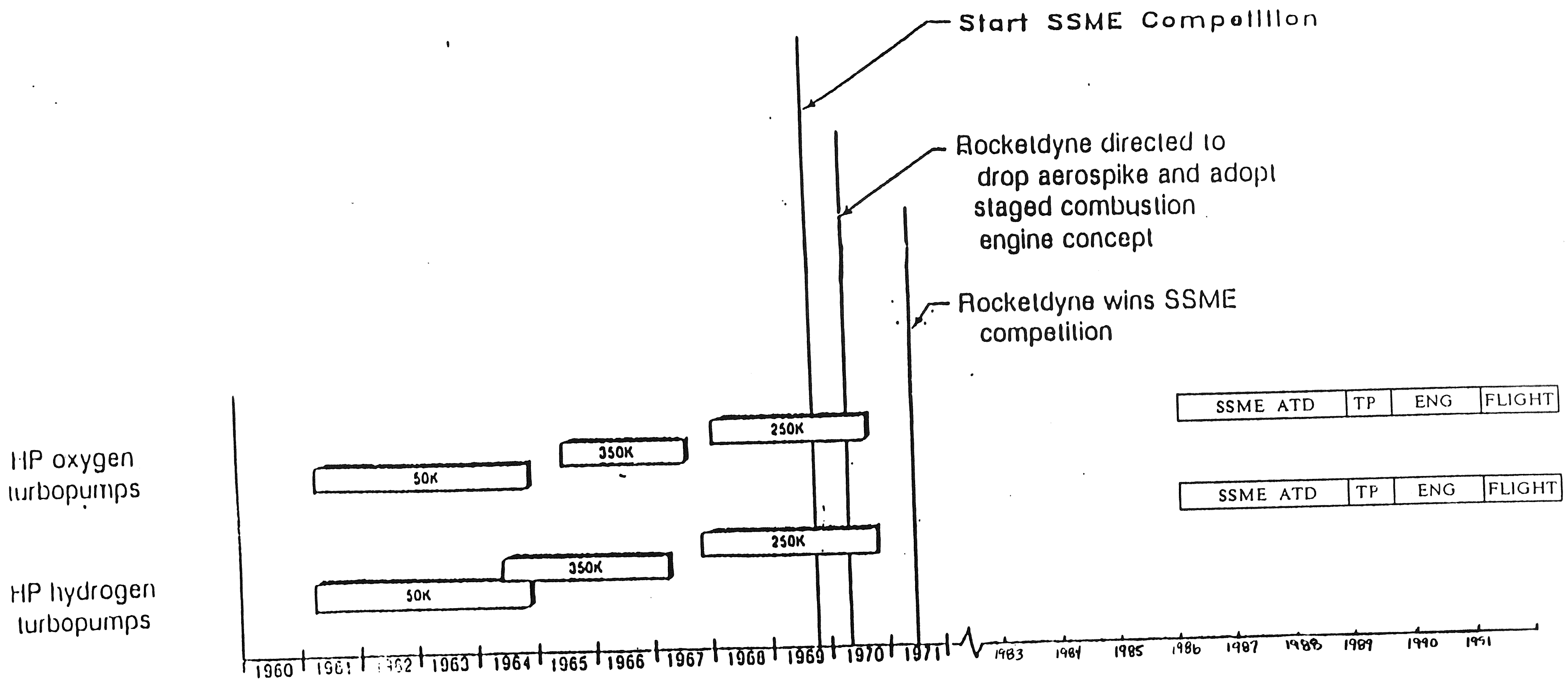


- Ran first liquid hydrogen rocket engine - 1959
- Established feasibility of high pressure staged combustion engine under Air Force sponsorship (AFRPL)
- More flight experience than all other liquid hydrogen rocket engines - 1963 to present
- Perfect flight record - 100% reliable

RL-10 TURBOPUMP ASSEMBLY

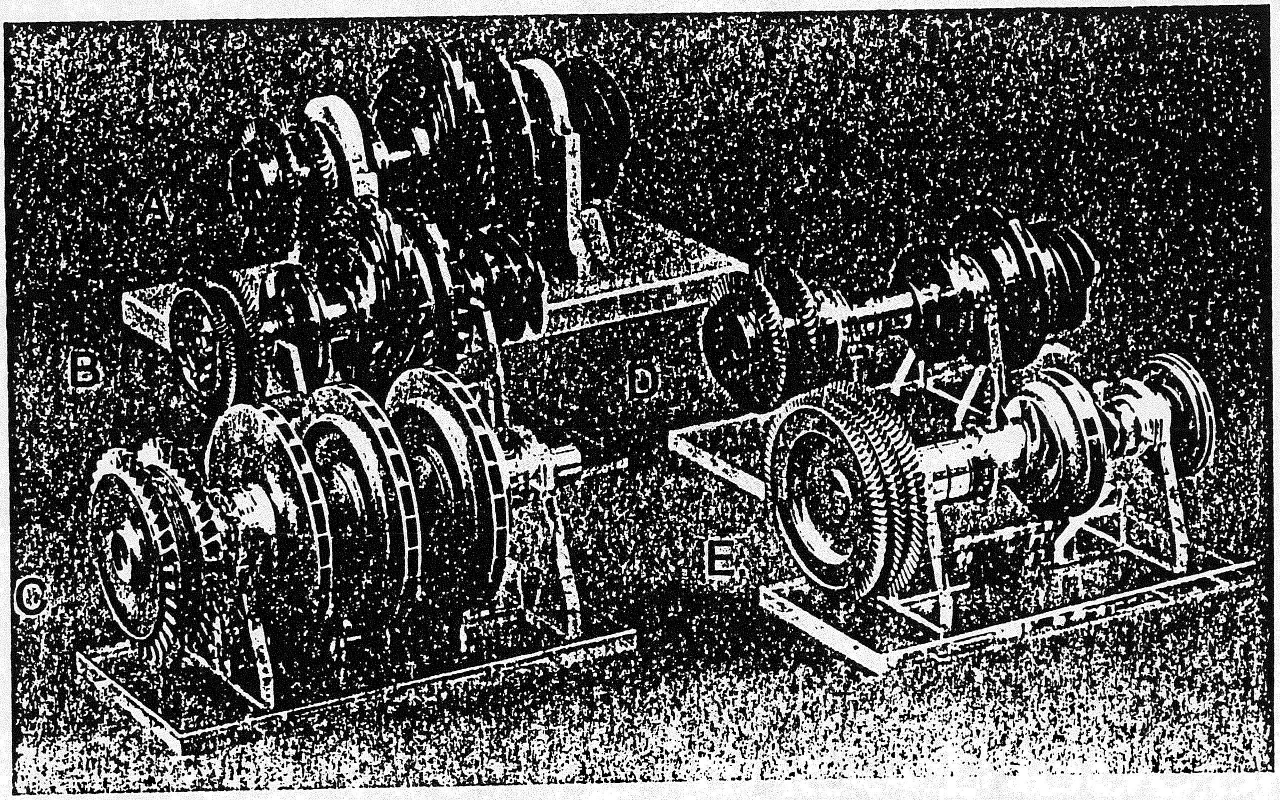


HIGH PRESSURE ROCKET ENGINE PUMP DEVELOPMENT BASE



PRATT & WHITNEY H.P. TURBOPUMP EXPERIENCE

ATD designs will be our 4th generation of H.P. turbopumps carried through test



A. (II) NASA-350K
HPFTP

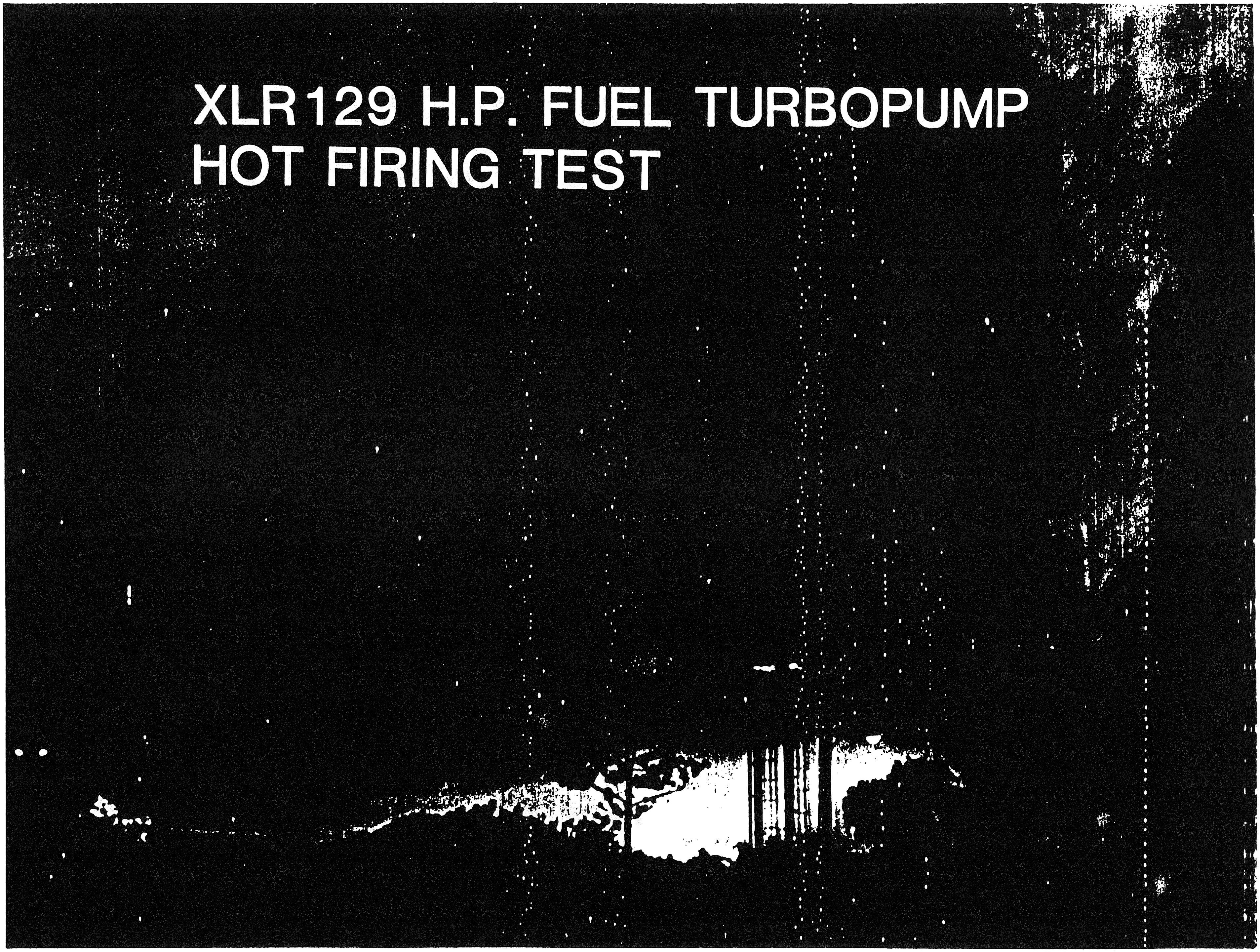
B. (III) XLR129-250K
HPFTP

C. (IV) ATD-HPFTP
Model

D. (II) NASA-350K
HPOTP

E. (IV) ATD-HPOTP
Model

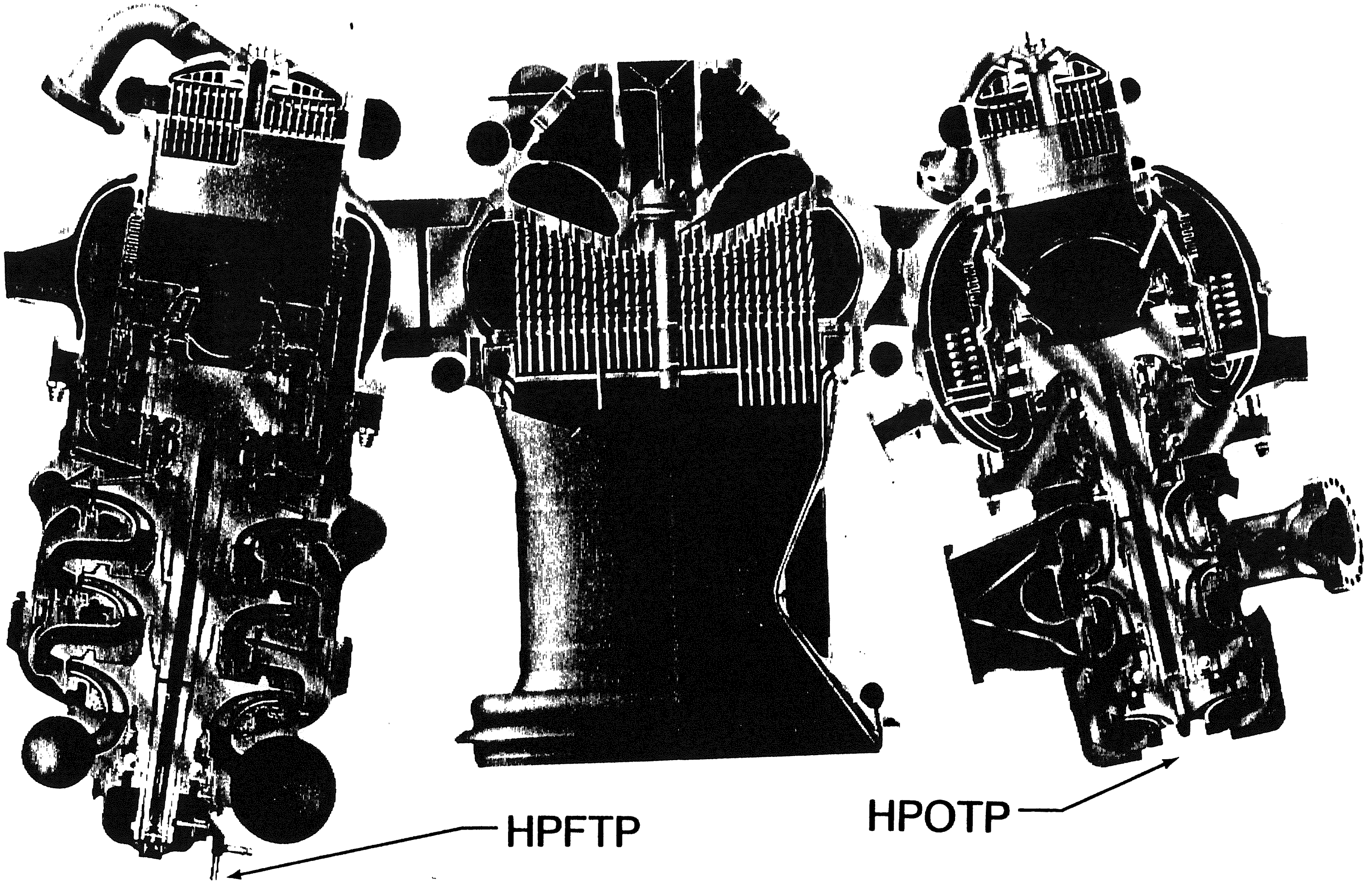
XLR129 H.P. FUEL TURBOPUMP
HOT FIRING TEST



SSME ATD

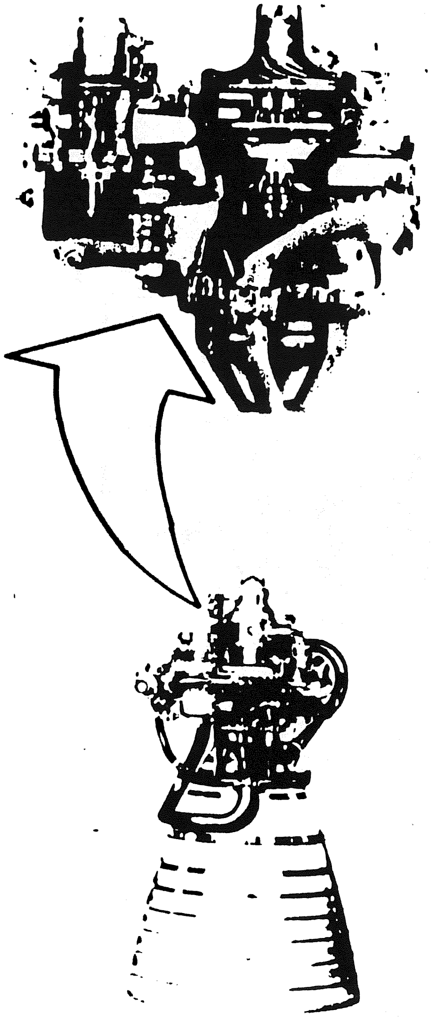
HPOTP AND HPFTP

SSME POWERHEAD WITH P&W (ATD) TURBOPUMPS



RL10 TURBOPUMPS

Differences between RL10 and SSME



	RL10	SSME
Chamber pressure (psia)	465	3,272
Thrust (lbf)	16,500	512,300
Turbine inlet temperature (°R)	380	1605 (Oxidizer) 1890 (Fuel)
Specific impulse (sec)	446.4	452.77
Total propellant flow (lbm/sec)	37.0	1159.26
Fuel pump speed (rpm)	32,800	36,570
Oxidizer pump speed (rpm)	13,100	24,972
Engine weight (lb)	305	7,004

PRELIMINARY PUMP DESIGN

- ESTABLISH LIMITATIONS AND CONSTRAINTS
- PRELIMINARY DESIGN CLOSE TO FINAL DESIGN
- PUMP REQUIREMENTS (DESIGN POINT)
  - STEADY STATE CAPACITY AND PRESSURE REQUIREMENTS (LONGEST ENGINE OPERATING MODE)
  - TRANSIENTS ARE SECONDARY BUT IMPORTANT
  - COMPROMISES

PRELIMINARY DESIGN OF CENTRIFUGAL PUMPS

- PUMP CAPACITY
  - PROPELLANT, ENGINE THRUST LEVEL, NOZZLE AREA RATIO, PC, MIXTURE RATIO, PUMP INLET CONDITIONS
- PUMP HEAD RISE
  - REQUIRED FLOW, SYSTEM RESISTANCE, PC, PUMP INLET CONDITIONS
- DESIGN SPEED AND NUMBER OF STAGES
  - PUMP SIZE AND WEIGHT, EFFICIENCY, STRUCTURAL LIMITATIONS, BEARING DN LIMITS, SEAL RUBBING SPEEDS, SHAFT CRITICAL SPEEDS, SUCTION PERFORMANCE
  - NUMBER OF STAGES $= f(SIZE, WEIGHT, SPEED)$
  - $\Delta H \ PER \ STAGE = \dfrac{\Delta H \ PUMP}{\#STAGES}$
  - TRIAL AND ERROR

- IMPELLER DISCHARGE DIAMETER AND ANGLE
    - SIZED $f(RPM, \Delta H, Q)$
    - CHECK SIZE AND WEIGHT-MAY HAVE TO GO TO MORE STAGES
    - $\beta_2 - \psi - \phi$ CURVE STEEPNESS
    - THROTTLING CAPABILITY STALL CHARACTERISTICS

- STRESS LIMITATIONS IN IMPELLER
    - TIP SPEED
    - MATERIAL
- PUMP INLET
    - INLET TIP DIAMETER AND BLADE ANGLE
        - DESIGN POINT
    - SUCTION CAPABILITY
        - $f(RPM, INLET\ \ TIP\ \ DIAMETER)$
- PUMP INDUCERS (AXIAL FLOW PUMPS)
    - FABRICATION AND GEOMETRY OF INTERNAL IMPELLERS
    - SUCTION PERFORMANCE

- BLADE TIP DIAMETER RATIO $(\frac{D_{T1}}{D_{T2}})$
  - $\frac{D_{T1}}{D_{T2}} = f(\psi, eff(\mathrm{B}\ \&\ \#\ BLADES))$
- NUMBER OF BLADES (Z)
  - $f(SOLIDITY)$
  - $Z = \mathrm{B}_2/3$
- BLADE HEIGHT, HUB TO TIP RATIO, AND THICKNESS
  - HUB TO TIP RATIO = 0.3
  - BLADE HEIGHT = 0.7 D_{T1}
- $N_S = \frac{N\sqrt{Q}}{(\frac{Cm^2}{2gc})^{3/4}}$
  - AREA = .321 $\frac{Q}{Cm^2}$
  - EXIT HEIGHT

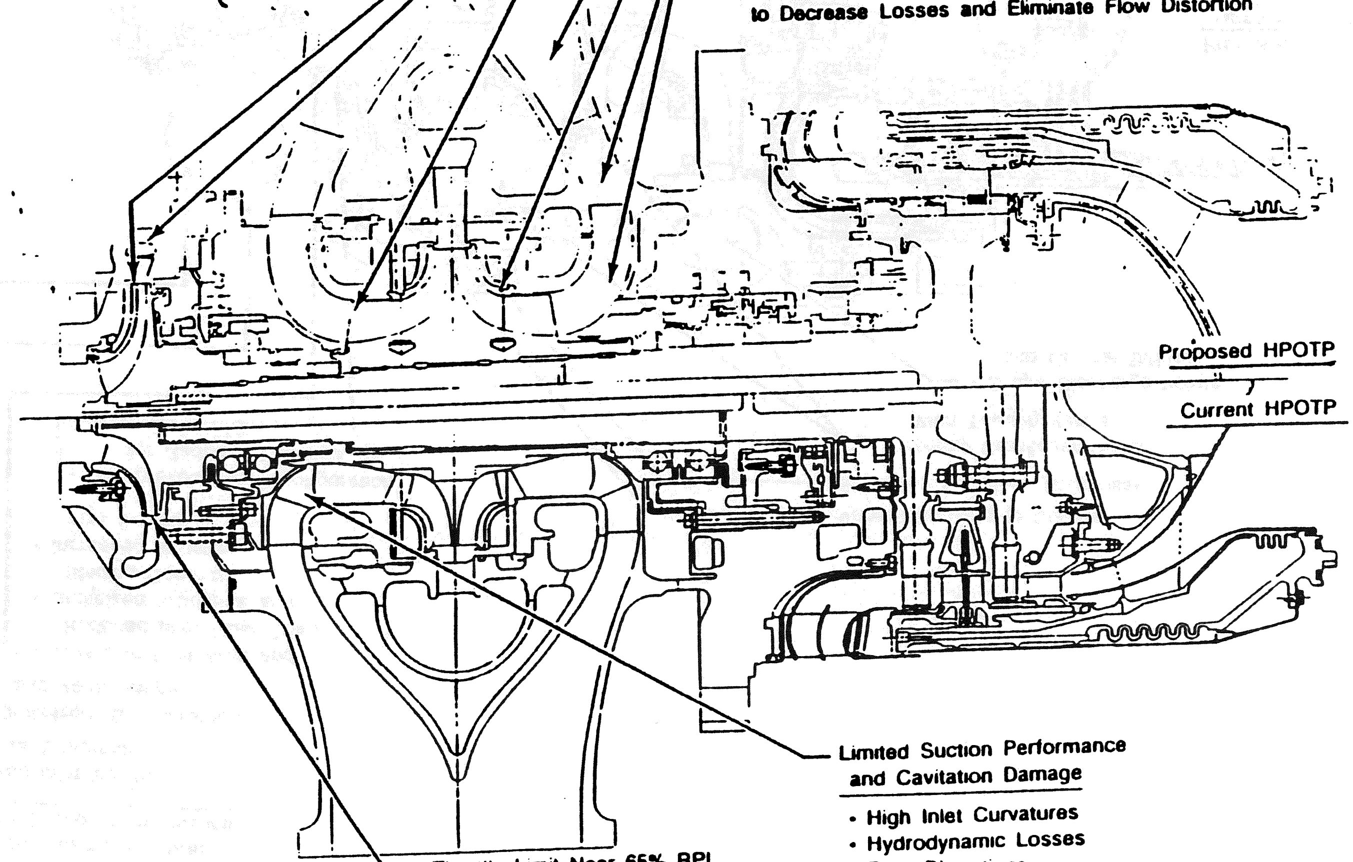
- BLADE THICKNESS
  - $b_T = 0.060 in$ 0.080 *AVERAGE*
  - $b_T = (0.1) b_{HT}$
- BEARING DN LIMITS (MECHANICAL LIMITATIONS)
  - NEED SHAFT DIAMETER-SHEARING STRESSES
- IMPELLER AXIAL LENGTH
  - $l = 0.25 \; D_{T2}$
  - LARGER LENGTH MAY REDUCE TURBULENCE
  - *f TURBOPUMP ENVELOPE*
- SUMMARY
  - PRELIMINARY DESIGN MUST BE FEASIBLE IN EVERY RESPECT BECAUSE CHANGES WILL EFFECT THE TURBINE AS WELL AS OTHER PARTS

NEW TECHNOLOGIES APPLIED IN P&W-ATD PROGRAM

Technologies developed since original SSME turbopump designs

Design systems (3D aero analysis, 3D structural analysis, low loss positive outflow coolant injection, fracture mechanics, etc.)	41
Mechanical features (Solid bore turbine disk, drum rotor, damper seals, blade dampers, proven thermal compliant airfoils, etc.)	26
Instrumentation (Laser holography inspection, laser temperature probing, non-intrusive sensors, etc.)	7
Materials (Blade and vane alloys, GATORIZED® IN100 disks, M-50 NIL bearing alloy)	4
Mfg processes (Single crystal castings, ion implanted coatings, Microcast-X™ castings, etc.)	8
Total	86

CURRENT HPOTP HYDRODYNAMIC LIMITATIONS AND PROPOSED IMPROVEMENTS



Proposed Hydrodynamic Features to Eliminate Throttle Limitation

- Stall Prone Vaned Diffuser Replaced With Volute Collector
- Reduced Design Head-Coefficient for Stable H-Q Characteristic

HPOTP

Proposed Hydrodynamic Features to Eliminate Cavitation

- Reduced Pump Speed, Optimized Inducer Incidence and Blade Leading Edge Contour
- Optimized Inlet for Flow Uniformity Into IGV's
- Improved Seal Configuration for Reduced Leakage
- IGV's Relocated and Optimized for Incidence and Vane Shape to Eliminate Flow Distortion and Wakes
- Recontoured Inlet - Reduced Curvature to Decrease Losses and Eliminate Flow Distortion

Proposed HPOTP

Current HPOTP

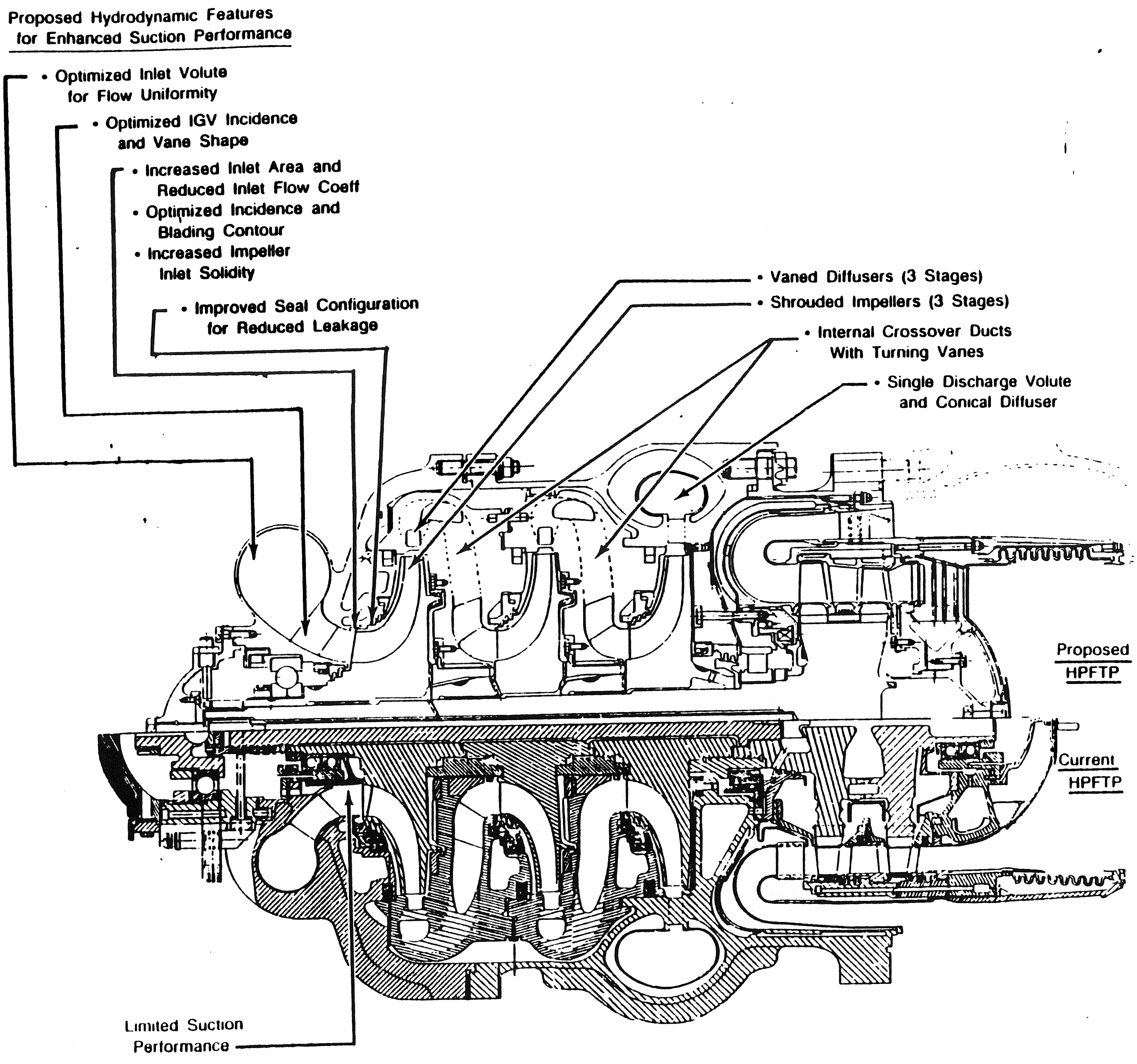
Limited Suction Performance and Cavitation Damage

- High Inlet Curvatures
- Hydrodynamic Losses
- Flow Distortions
- IGV Wakes in Close Proximity to Inducer
- Inducer Incidence and Blading Misalignment

Throttle Limit Near 65% RPL

- Preburner Vane Diffuser Stall
- High Design Head Coefficient and Flat H-Q Characteristic

ALTERNATE ATD HPFTP HYDRODYNAMIC DESIGN FEATURES



REFERENCES

- HYDRODYNAMIC DESIGN OF TURBOPUMPS
  G.L. CLARK 7/16/85
- PUMP DESIGN ANALYSIS SEMINAR
  G.L. CLARK 1965
- FLUID MECHANICS, THERMODYNAMICS OF TURBOMACHINERY
  S.L. DIXON
- CENTRIFUGAL AND AXIAL FLOW PUMPS
  STEPANOFF
- BASIC CONSIDERATION IN PRELIMINARY PUMP DESIGN
  M.Y. YOUNG 1965

ROCKET ENGINE FUNDAMENTALS

SESSION 5

ROCKET ENGINE PERFORMANCE II

Part I: TURBINES

Instructor: Jon Spryer

February 1991

AN INTRODUCTION TO ROCKET TURBINES

OBJECTIVE:

REVIEW THE FUNCTIONAL PARTS AND BASIC PERFORMANCE FEATURES OF ROCKET TURBINES, AND ILLUSTRATE THE BASIC DIFFERENCES BETWEEN ROCKET AND JET ENGINE TURBINES.

MAJOR POINTS:

- DEFINITION, EXAMPLES, & HISTORICAL BACKGROUND
- BASIC PRINCIPLES, FEATURES, & TYPES
- ROCKET- & JET-TURBINE DIFFERENCES
- AIRFOIL NOMENCLATURE & AERODYNAMICS
- IMPULSE & REACTION DESIGNS
- PERFORMANCE PARAMETERS

DEFINITION & EXAMPLES

DEFINITION:

Turbines are rotating machines that extract mechanical power from moving fluids (gases or liquids).

EXAMPLES:

- Water wheels (Note: These are not entirely immersed in the working fluid, & basically don't turn the flow.)
- Windmills
- Steam turbines (widely used for electrical power generation)
- Turbochargers (no turbines in belt-driven superchargers)
- Jet engines } (turbines are 1 of 3 or 4
- Rocket engines } major components)

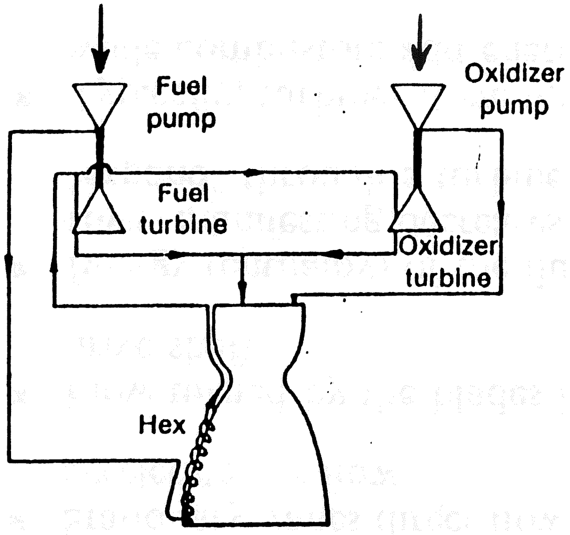
HISTORICAL BACKGROUND

ORIGIN OF MODERN TURBINES:

- First working gas-turbine jet engine was invented by Sir Frank Whittle of England (knighted after World War II):
    - born 1907
    - attended Royal Air Force technical college; 1928 senior thesis "Future Developments in Aircraft Design" supported jet propulsion concept
    - issued patent for gas turbine engine design January 1930
    - formed Power Jets Ltd. in June 1935 to develop his engine
    - fired up Whittle engine on 12th April, 1937
    - development engine provided 860 lbs. thrust for the British Gloster E.28/39 which flew on 15th May 1941, reaching 338 mph.
- In 1936, Hans von Ohain started work on his jet engine under Ernst Heinkel in Germany, unaware of Whittle's work:
    - von Ohain's 838-lb.-thrust engine in the special experimental Heinkel HE178 (first gas-turbine jet aircraft) flew on 27th August 1939 & reached 435 mph.
    - von Ohain was brought to the United States after World War II, & led advanced propulsion group at the Air Force's Aeronautical Research Laboratory at Wright-Patterson AFB (Dayton, Ohio)
    - later affiliated with the U.S.A.F. Aeropropulsion Laboratory at Wright Field

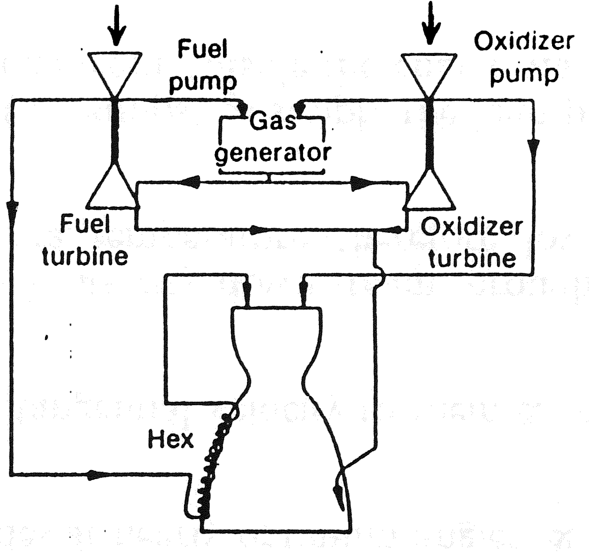
LIQUID ROCKET ENGINE POWER CYCLES

EXPANDER (RL-10)



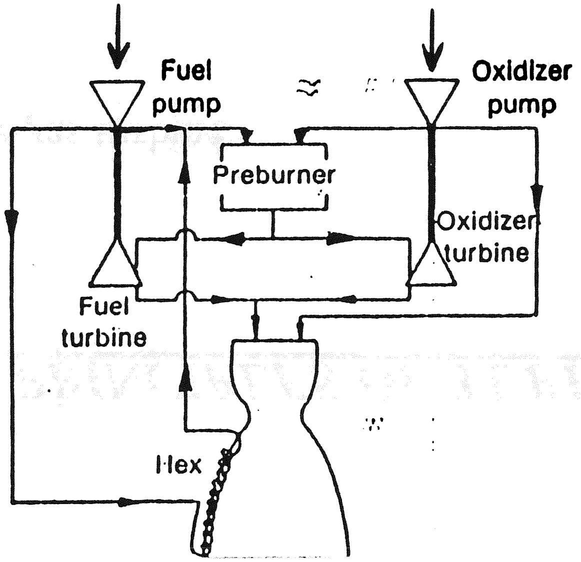
Turbines:
- low P's
- low T's

GAS GENERATOR (ALS)



Turbines:
- medium P's
- high T's

STAGED COMBUSTION (SSME)



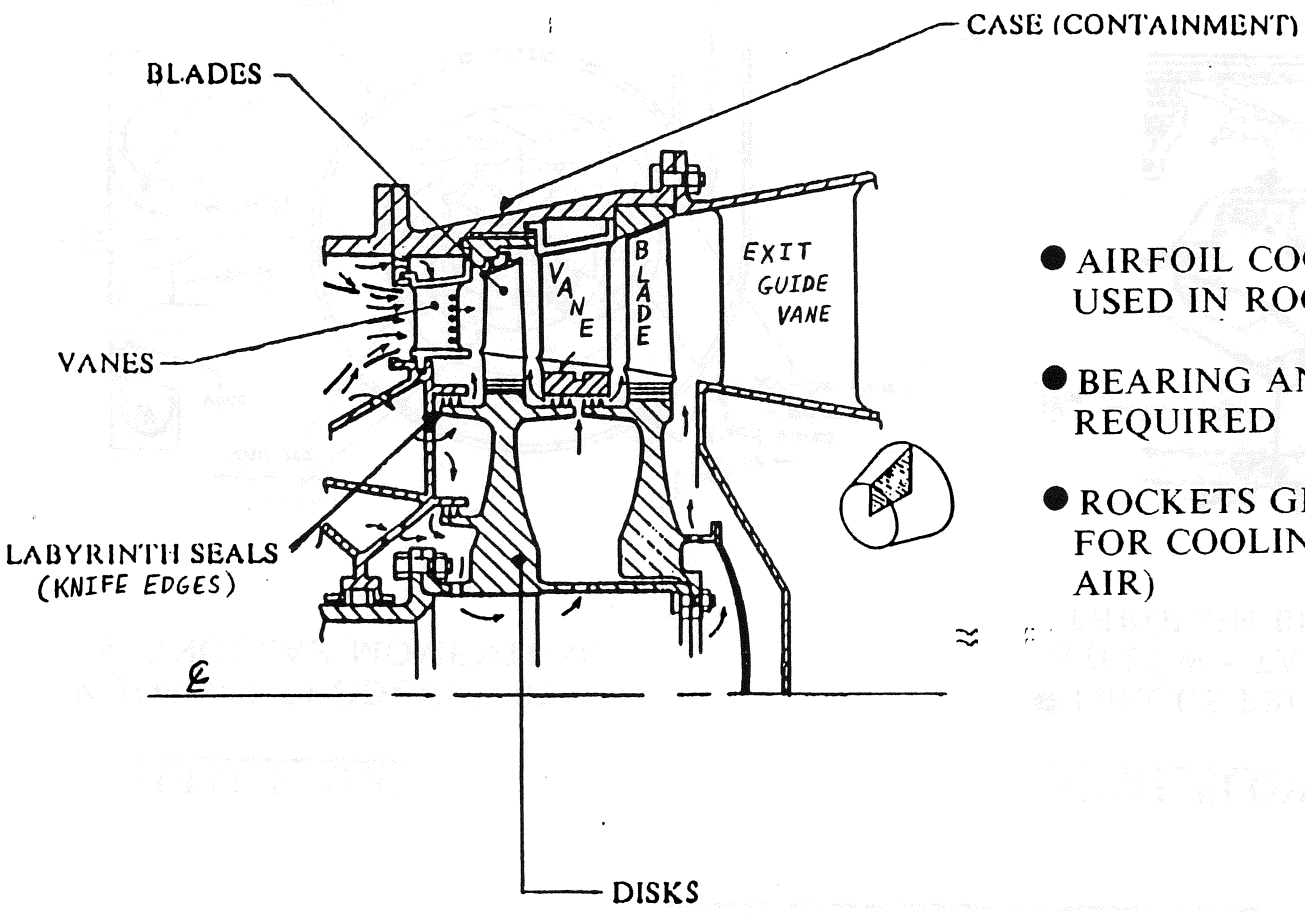
Turbines:
- high P's
- high T's

BASIC PRINCIPLES OF TURBINES

Considering a modern axial-flow gas turbine:

- Flow at elevated pressure & temperature is taken in
- Stationary vanes direct flow toward rotating blades at nearly optimum angles, & usually also accelerate the flow
- Flow turned by the blades imparts its change in tangential velocity to them & thence to a drive shaft
- Energy (enthalpy) of the fluid decreases as it works its way downstream through the blade rows, manifesting decreases in total pressure & total temperature (therefore flow is said to "expand" through a turbine)
- Typically, turbines drive upstream compressors (or pumps) to supply the high pressures, while combustors add energy (higher temperatures) from outside the engine via fuel

GENERAL TURBINE FEATURES

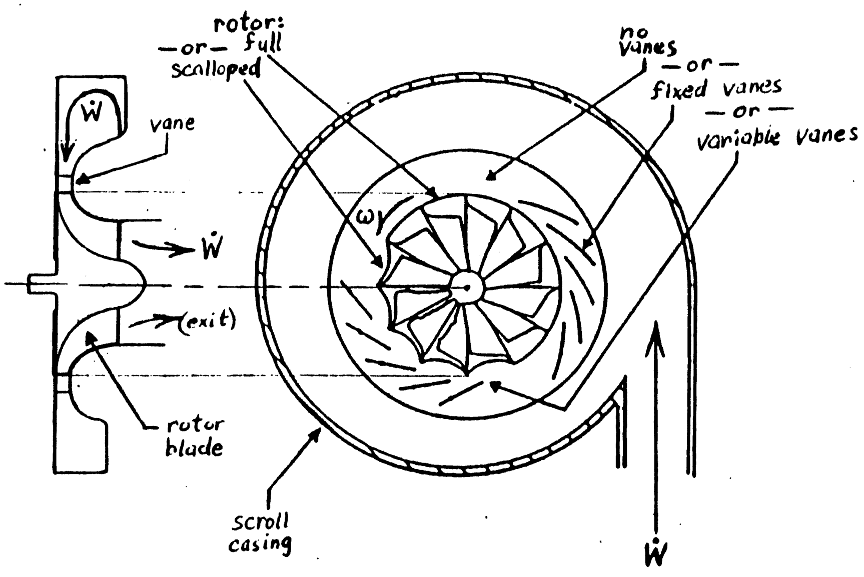


- AIRFOIL COOLING NOT GENERALLY USED IN ROCKET TURBINES
- BEARING AND DISC COOLING REQUIRED
- ROCKETS GENERALLY USE FUEL FOR COOLING (JETS USE COMPRESSOR AIR)

BASIC TURBINE TYPES

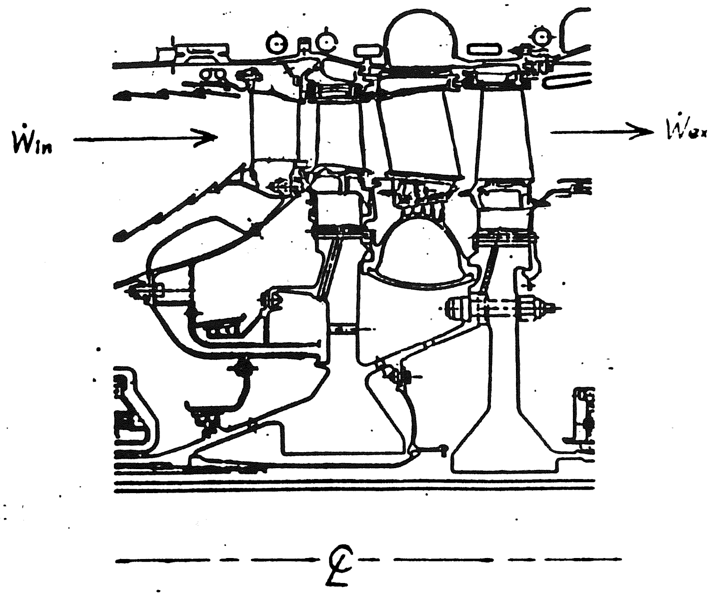
RADIAL FLOW

- TORQUE PRODUCED BY
  Δ (ANGULAR MOMENTUM)



AXIAL FLOW

- TORQUE PRODUCED BY
  Δ (FLOW * TANGENTIAL VELOCITY)
  THROUGH BLADES

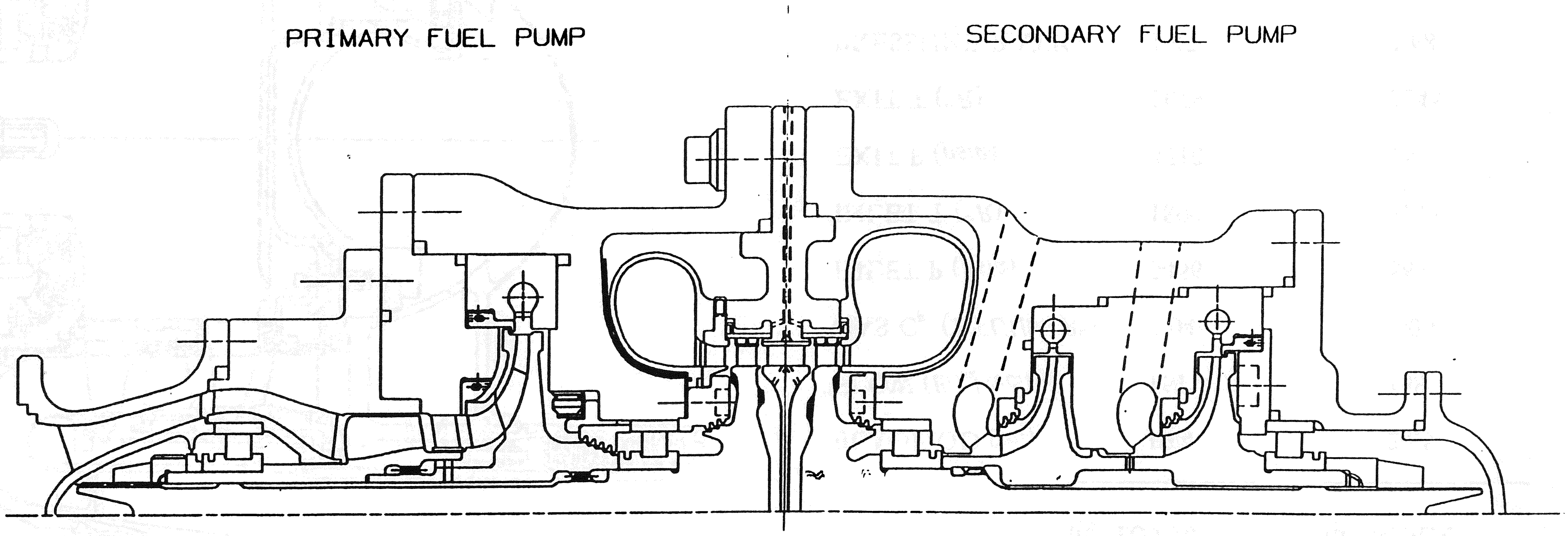


OTHERS:

- LIQUID (HYDRAULIC)
- PARTIAL ADMISSION
- CONTRA-ROTATING

TURBINE TYPES (continued)

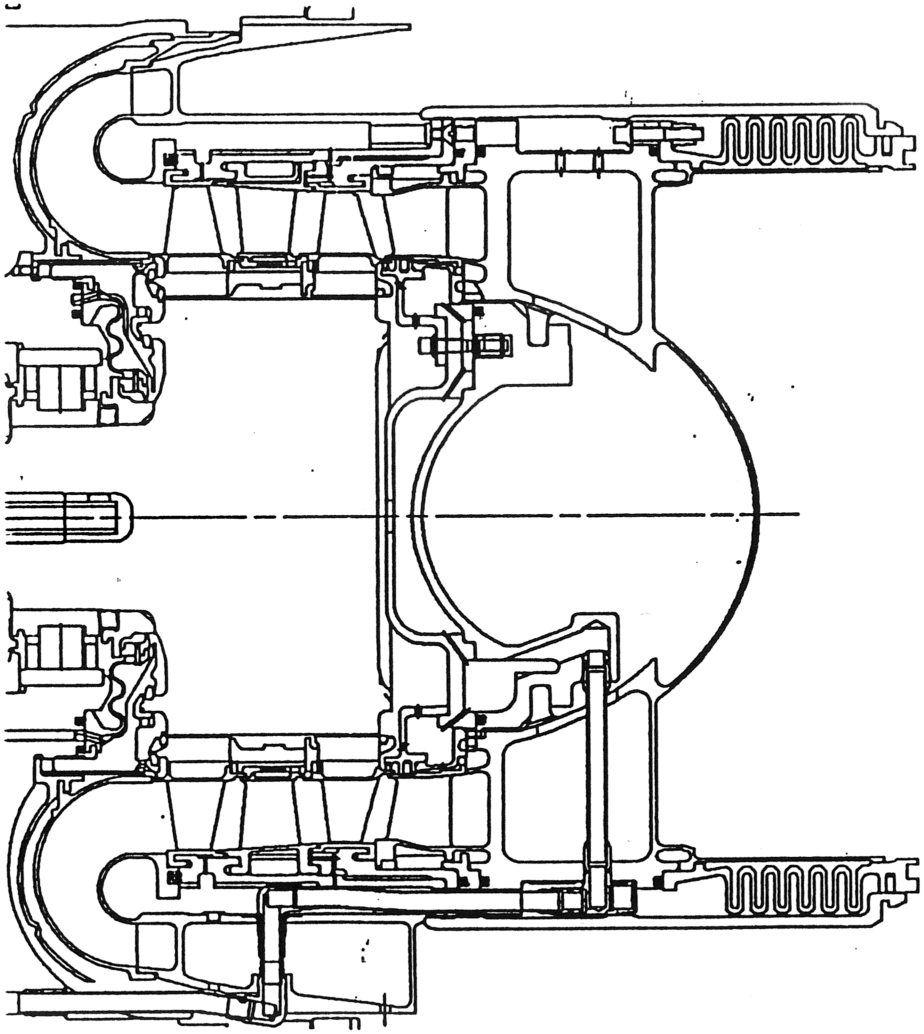
CO-AXIAL, CONTRA-ROTATING



ADVANCED-EXPANDER TEST BED (AETB) FUEL TURBOPUMP

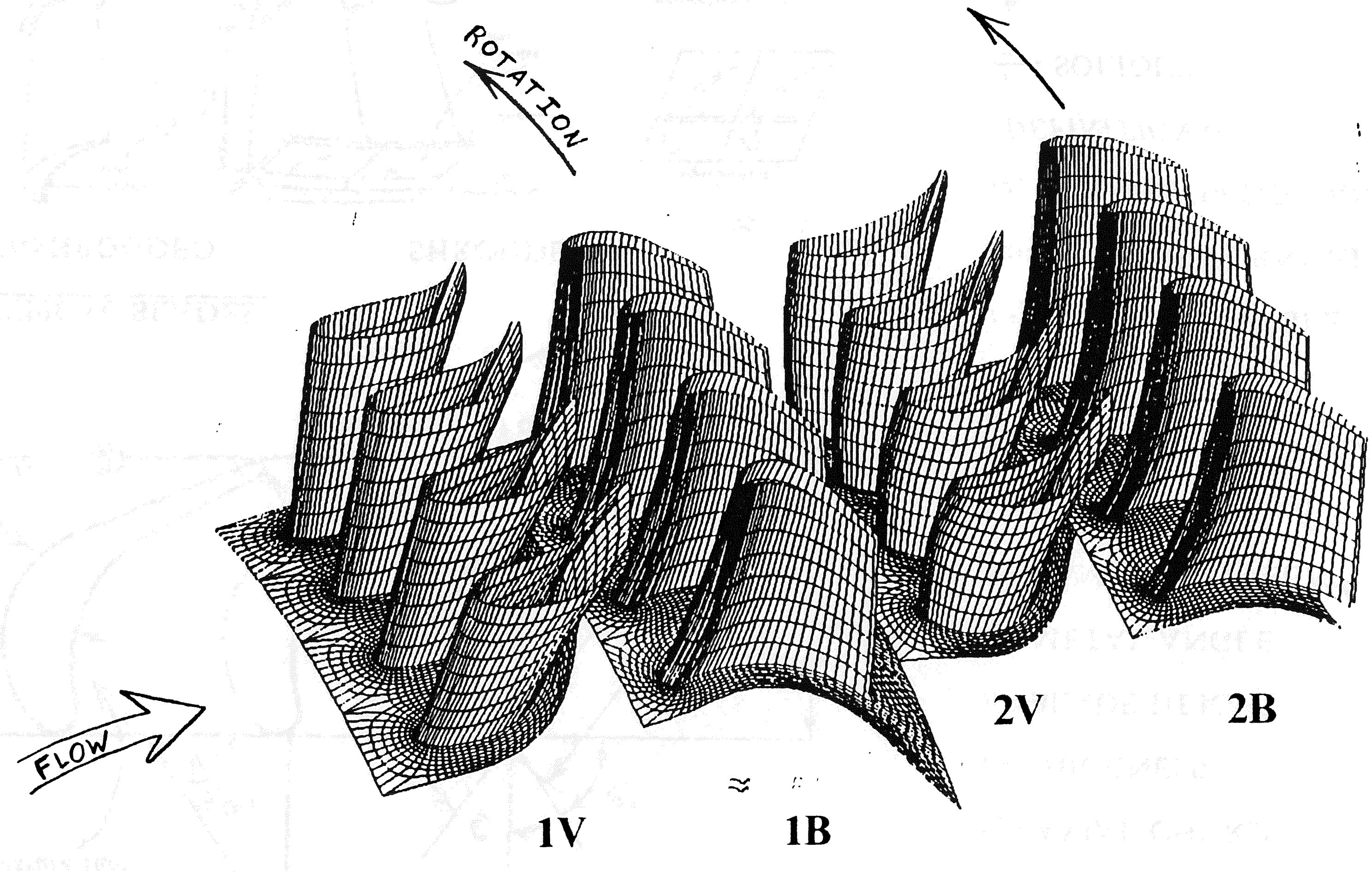
SSME ATD HP-FUEL PUMP TURBINE

OPERATING CONDITIONS COMPARED TO PW1129:



	SSME ATD @ 109%	PW1129 HPT @ MAX
BLADE OD (*in.*)	10.4	24.5
FLOW ($lb._m/sec.$)	161	168
GAS C_p ($BTU/lb._m R°$)	2.04	.305
INLET P (*psia*)	5456	563
INLET T (°*R*)	1890	3211
EXIT P (*psia*)	3710	153
EXIT T (°*R*)	1679	2245
PRESSURE RATIO	1.47	3.68
~ RPM	36,540	13,400
HORSEPOWER	73,928	43,000
EFFICIENCY (%)	87	90

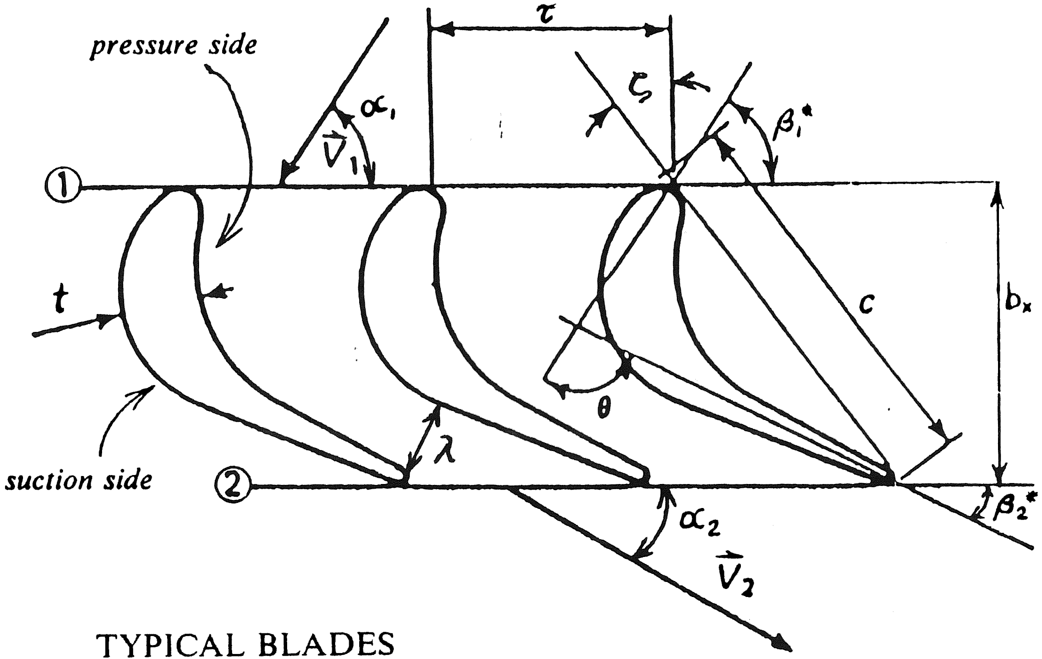
3-DIMENSIONAL AERODYNAMIC ANALYSIS MESH



ATD SSME High-Pressure Fuel Turbine

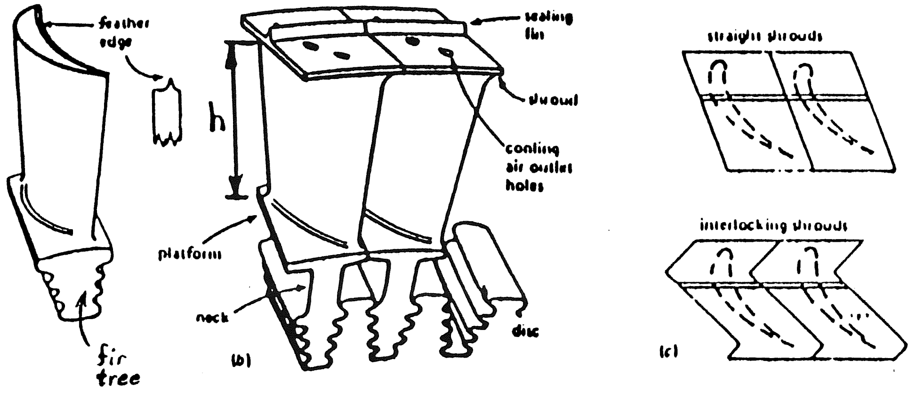
AIRFOIL NOMENCLATURE

TYPICAL VANES' CROSS-SECTION



TYPICAL BLADES

UNSHROUDED SHROUDED



τ - PITCH

c - CHORD

b_x - AXIAL CHORD

t - THICKNESS

h - BLADE HEIGHT

β^*- METAL ANGLE

θ - CAMBER ANGLE $(\beta_1^* - \beta_2^*)$

ζ - STAGGER ANGLE

λ - GAGE (THROAT WIDTH)

α - GAS ANGLE

i - INCIDENCE ANGLE $(\alpha_1 - \beta_1^*)$

$(\alpha_1 - \alpha_2)$ - GAS TURNING

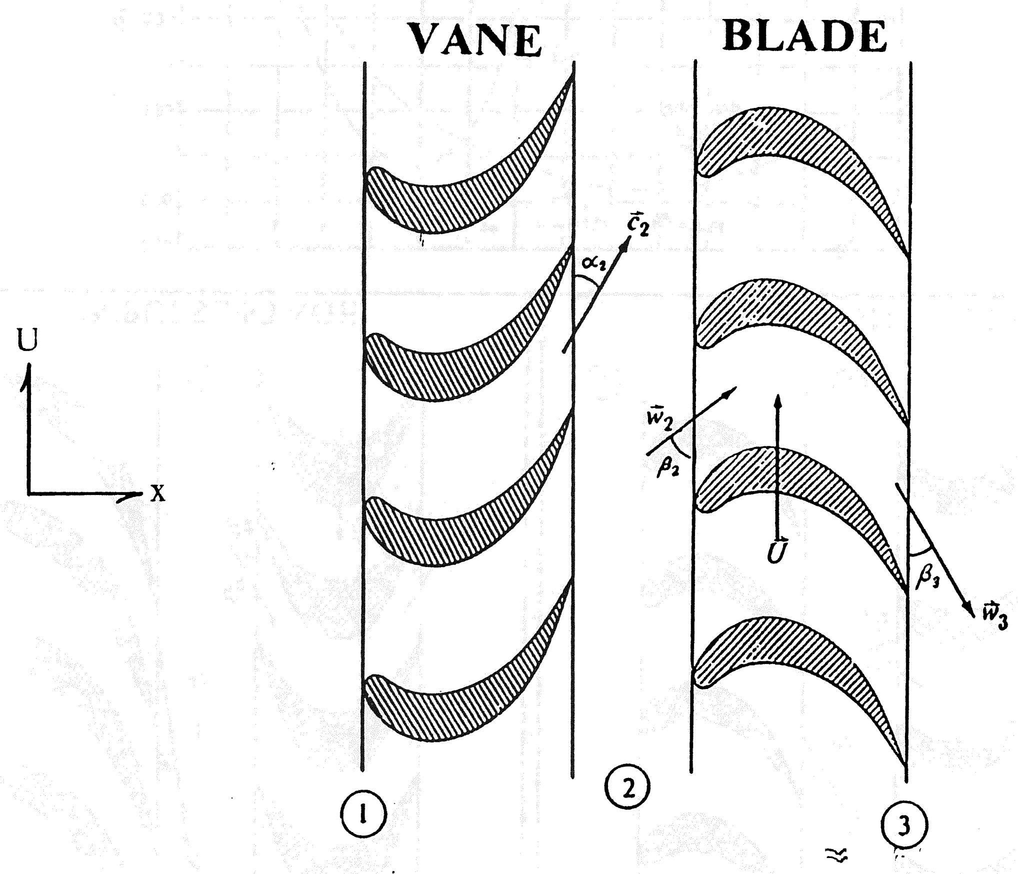
$(\alpha_2 - \beta_2^*)$ - DEVIATION ANGLE

DEFINITIONS:

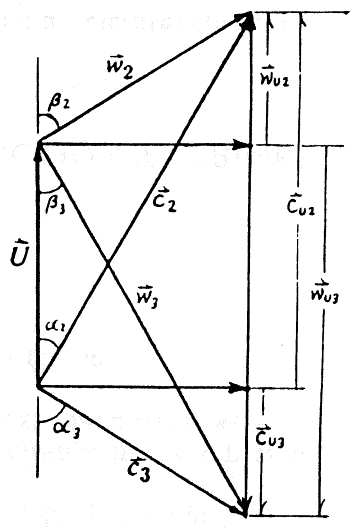
$\frac{c}{\tau}$ - SOLIDITY

$\frac{h}{c}$ - ASPECT RATIO

2-D TURBINE VELOCITY TRIANGLES



ACROSS BLADE:



Special Case:
Constant Axial Velocity

c & α = ABSOLUTE FRAME

w & β = RELATIVE FRAME

$c_x = c \times \sin\alpha$

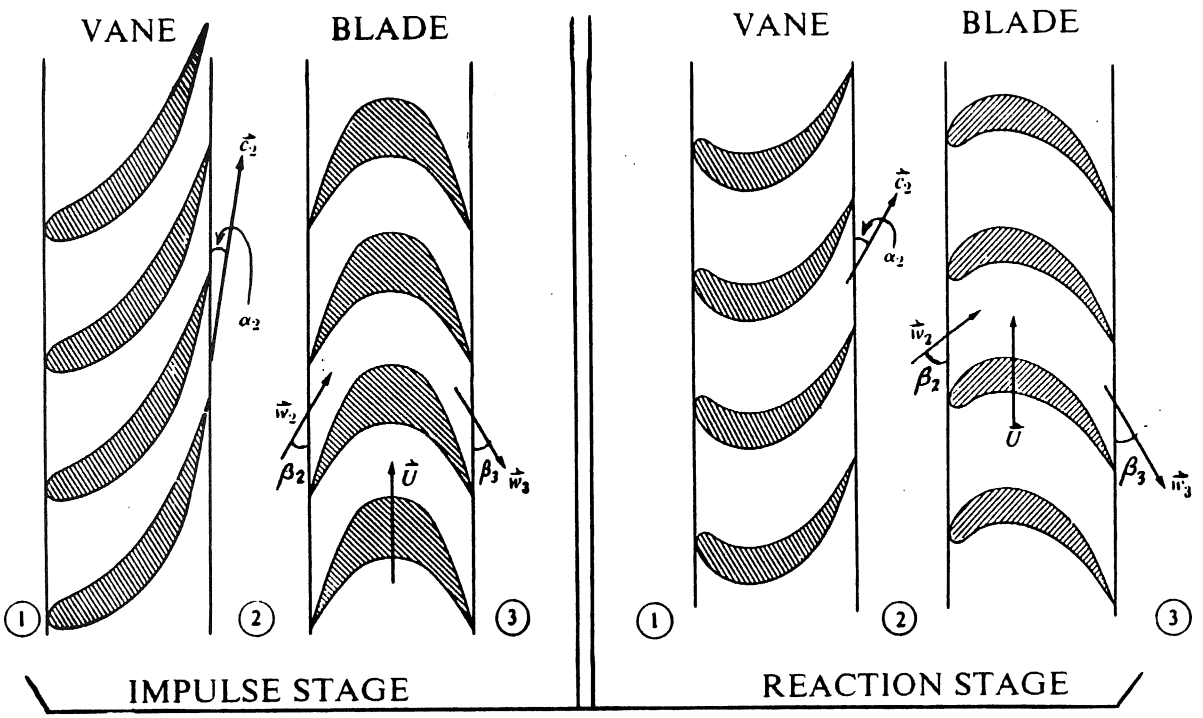
$c_U = c \times \cos\alpha$

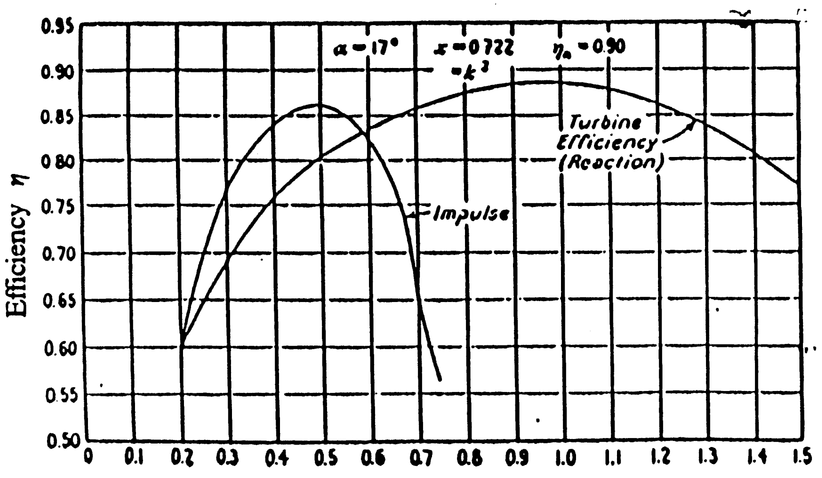
$w_x = w \times \sin\beta$

$w_U = w \times \cos\beta$

$\therefore c_U = w \times \cos\beta + U$

IMPULSE AND REACTION STAGE DESIGN





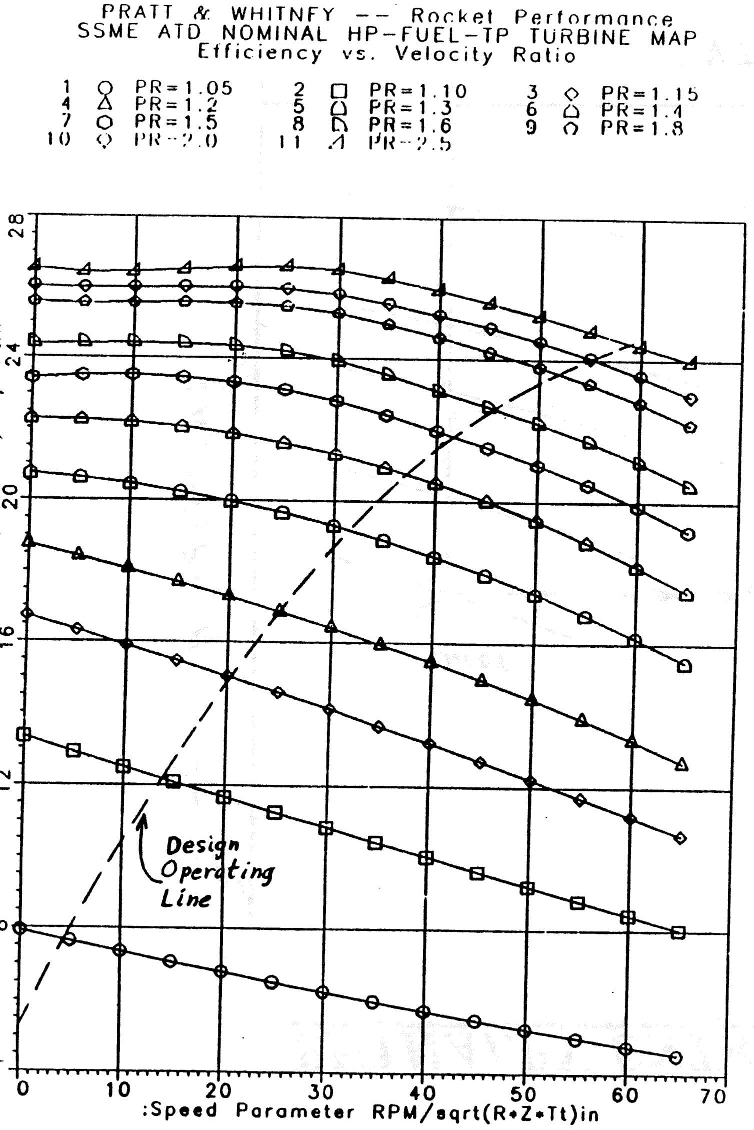
IMPULSE TURBINES:

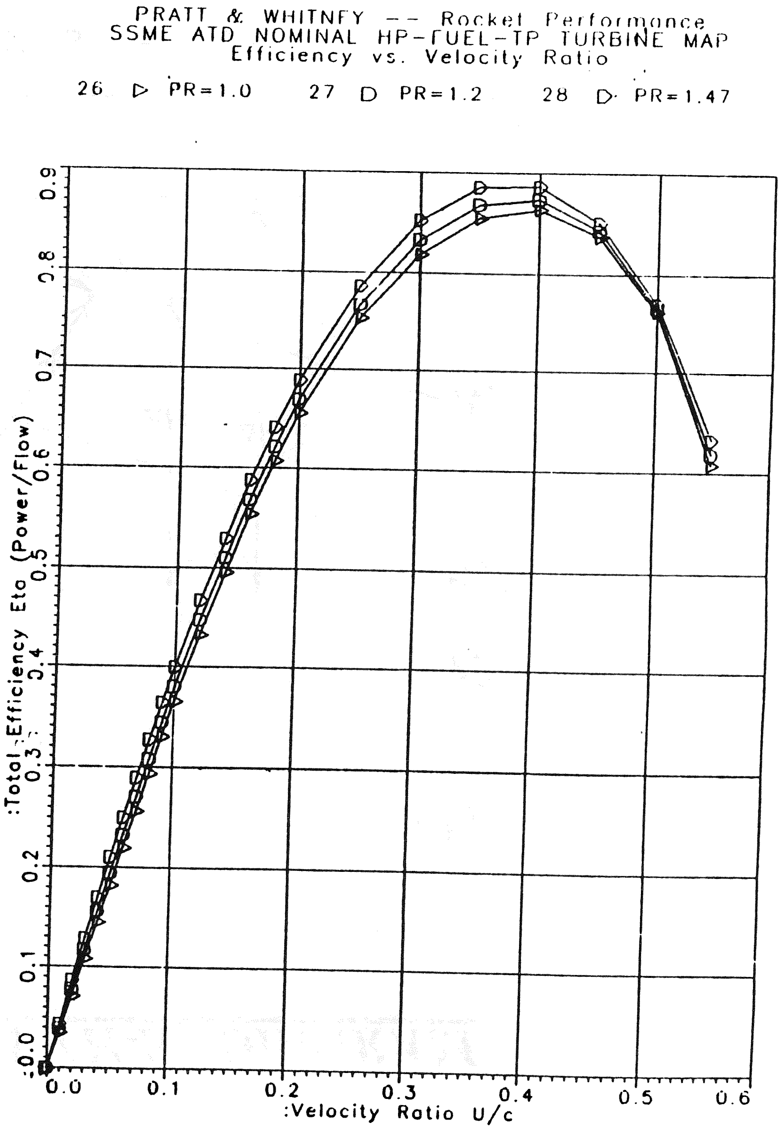
- Older designs
- Small & light weight
- No change in static pressure across blade rows, $w_3 = w_2$
- Less efficient

REACTION TURBINES:

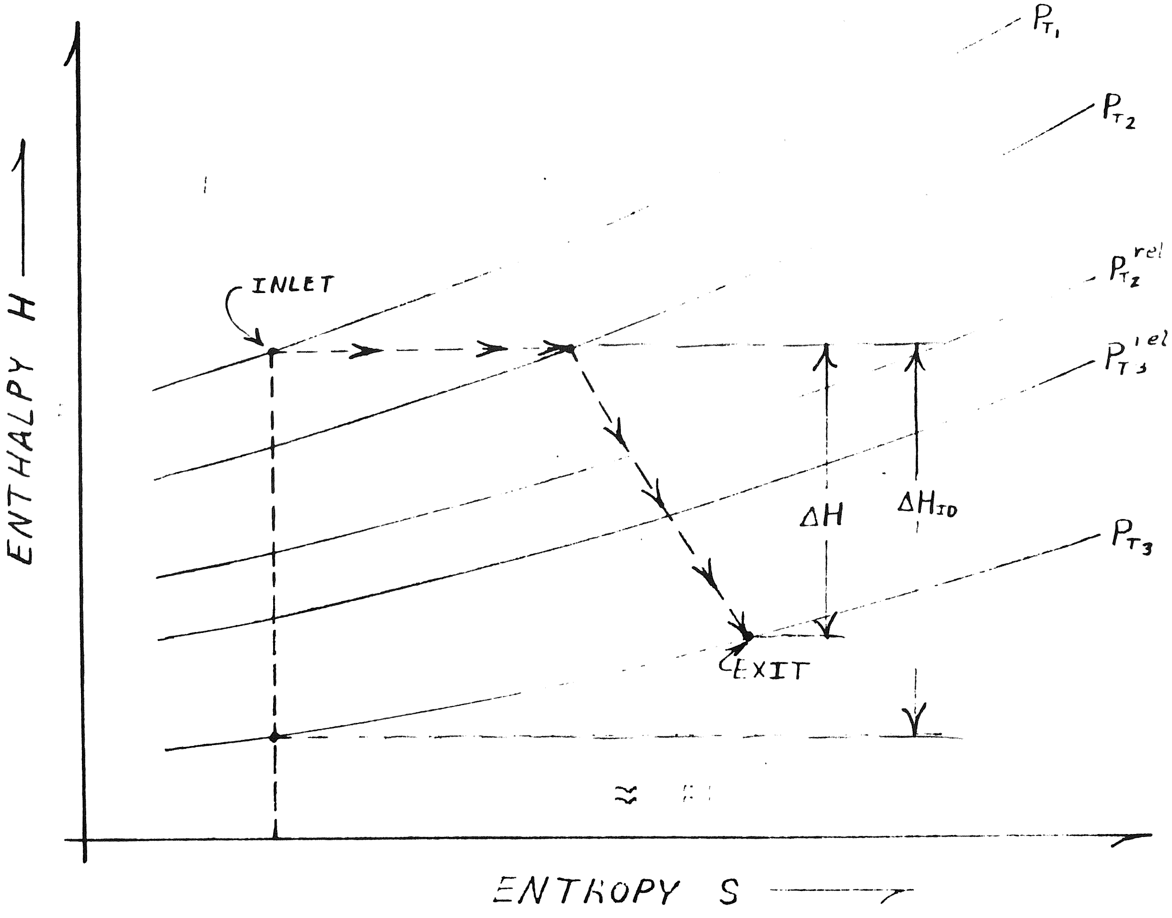
- More stages
- Fluid accelerates through blade rows in relative frame (P_s drop), $w_3 > w_2$
- Higher efficiencies
- Less sensitive to off-design operation

CHARACTERISTIC TURBINE MAPS





ENTHALPY-ENTROPY (H-S) DIAGRAM



Enthalpy & entropy are "insensible" thermodynamic state properties of fluids; for a given gas, both are strong functions of temperature & weaker functions of pressure.

TURBINE PERFORMANCE PARAMETERS

(Note: Definitions may vary with applications.)

1) PRESSURE RATIO: $PR = \frac{P_{Tin}}{P_{Tex}}$

2) FLOW PARAMETER: $FP = \frac{\dot{W}_{in} \times \sqrt{(R \times Z_{in} \times T_{Tin}/\gamma_{in})}}{P_{Tin}}$

where: $\dot{W}_{in}$ = inlet flow rate $[lb._{m}/sec.]$

$R \times Z_{in}$ = inlet gas constant × compressibility $[\frac{ft. - lbs.}{lb._{m} - {}^{\circ}R}]$

γ_{in} = inlet specific heat ratio

3) SPEED PARAMETER: $SP = RPM \div \sqrt{(R \times Z_{in} \times T_{Tin})}$

4) WHEEL SPEED: $U = (2\pi) \times (\frac{RPM}{60}) \times (\frac{D/2}{12})$

where: D = characteristic diameter $[inches]$

(usually blade row mean)

TURBINE PERFORMANCE PARAMETERS (continued)

5) VELOCITY RATIO: $U/c = (\pi \times D_{rms} \times RPM/720) \div \sqrt{2 \times g_c \times J \times (h_{Tin} - h_{Tex}^{ideal})}$

where: $D_{rms} =$ root-mean-square of blade mean diameters $[in.]$

$g_c = 32.174\ [\frac{lb._m - ft.}{lbs. - sec.^2}]$

$J = 778.26\ [ft. - lbs./BTU]$

$h_T =$ total enthalpy $[BTU/lb._m]$

6) REACTION: Enthalpy Reaction $R_H = \frac{\Delta H_{(blades)}}{\Delta H_{(stage)}}$ -OR- Pressure Reaction $R_P = \frac{\Delta P_s(blades)}{\Delta P_s(stage)}$

7) TURBINE POWER: $Horsepower = \dot{W} \times \eta \times (h_{Tin} - h_{Tex}^{ideal}) \times (J\ /\ 550)$

8) ISENTROPIC EFFICIENCY: $\eta = \frac{(h_{Tin} - h_{Tex}^{actual})}{(h_{Tin} - h_{Tex}^{ideal})}$ (sometimes × 100%)

SUMMARY

TYPICAL ROCKET TURBINE CHARACTERISTICS:

- AXIAL FLOW
- IMPULSE OR MODERATE-REACTION TYPE
- SMALL DIAMETER
- NO AIRFOIL COOLING (TO DATE)
- RELATIVELY LOW INLET TEMPERATURES (BUT SOMETIMES VERY HIGH PRESSURES)
- $\eta = f\ (PR,\ T_{Tin},\ RPM,\ D,\ Aerodynamics,\ etc.)$
- $HORSEPOWER = f\ (\dot{W},\ PR,\ T_{Tin},\ \eta,\ etc.)$
- DESIGNS WITH VANELESS STAGES & CONTRA-ROTATING BLADES BEING DEVELOPED

PRATT & WHITNEY
DEPENDABLE ENGINES®

ROCKET ENGINE FUNDAMENTALS

SESSION 5

ROCKET ENGINE PERFORMANCE II

Part II: THRUST CHAMBERS

Instructor: Jon Spryer

October 1990

AN INTRODUCTION TO LIQUID ROCKET THRUST CHAMBERS

OBJECTIVE: REVIEW THE MAJOR COMPONENTS OF LIQUID ROCKET THRUST CHAMBERS AND ILLUSTRATE THE MAJOR PARAMETERS AND DESIGN FEATURES WHICH AFFECT PERFORMANCE.

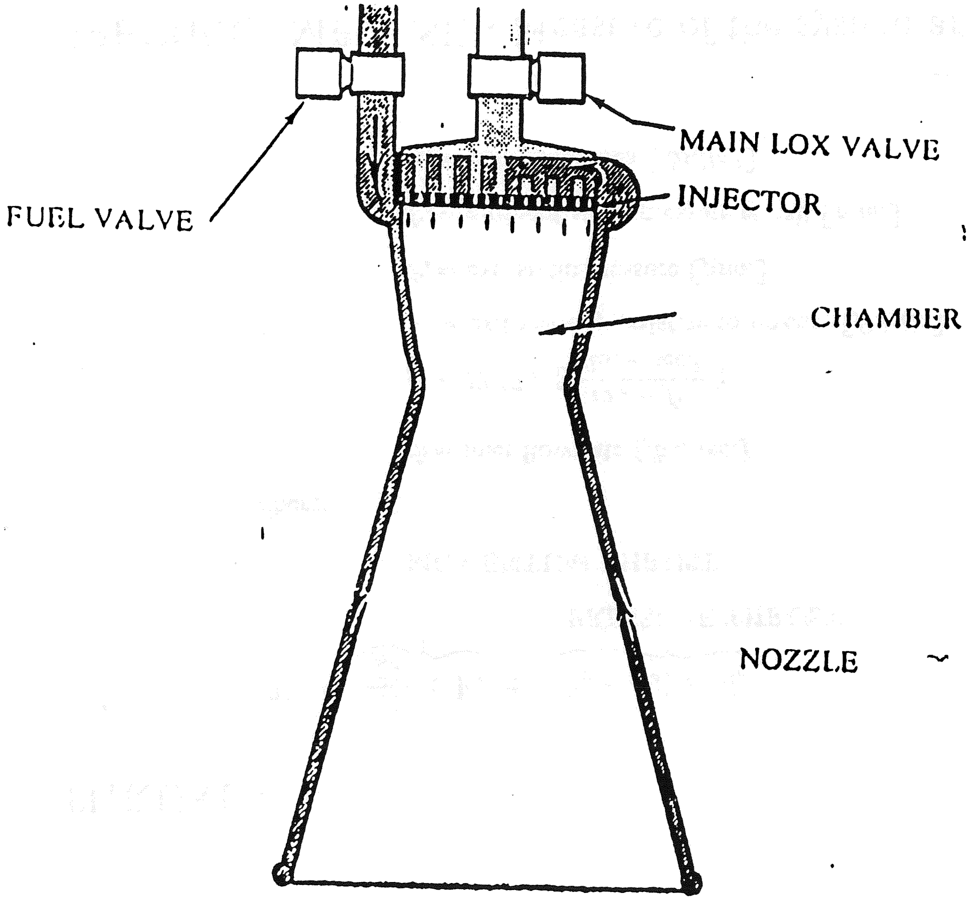
MAJOR TOPICS:

* INJECTORS & COMBUSTION

* IGNITORS

* NOZZLES & HOT GAS EXPANSION

THRUST CHAMBER COMPONENTS



INJECTOR:
ATOMIZES & MIXES PROPELLANTS (FUEL AND OXIDIZER)

CHAMBER:
CONTAINS THE PROPELLANT COMBUSTION PROCESS

NOZZLE:
EXPANSION OF HOT GAS TO OPTIMIZE THRUST

ADDITIONAL COMPONENTS:
- IGNITORS
- PREBURNERS
- GAS GENERATORS

PERFORMANCE PARAMETERS

THRUST -

$$T = \underbrace{\frac{\dot{m}}{g_c} \times V_e}_{\text{MOMENTUM THRUST}} + \underbrace{(P_e - P_a) \times A_e}_{\text{PRESSURE THRUST}}$$

where:

$\dot{m}$ = inlet flow rate $[lb._m/sec.]$

$g_c = 32.174 \ [\frac{lb._m - ft.}{lbs. - sec.^2}]$

V_e = exit velocity (relative to nozzle) $[ft./sec.]$

P_e = exit static pressure $[psia.]$

P_a = ambient static pressure at exit $[psia.]$

A_e = exit plane area $[lbs./in.^2]$

SPECIFIC IMPULSE - Measure of the system and working fluid capability for energy conversion

$$I_s = \frac{T}{\dot{m}} = \sqrt{\frac{2 \times J \times \delta H}{\overline{M}}}$$

PERFORMANCE PARAMETERS (continued)

where:

$J = 778.26 \; [ft. - lbs./BTU]$

$\delta H =$ heat released per mole $[BTU/mole]$

$\overline{M} =$ mean molecular "weight" $[lb._m/mole]$

THRUST COEFFICIENT - Effect of nozzle expansion on increasing thrust above that of P_c acting only over A_T

$$T = C_T \times A_T \times P_c$$

$$C_T = C_{T_{vac}} - \frac{P_a}{P_c} \times \left(\frac{A_e}{A_T} \right)$$

PERFORMANCE PARAMETERS (continued)

where:

$A_T =$ throat area $[in.^2]$

$P_c =$ chamber pressure $[psia.]$

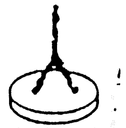
$C_{Tvac} =$ thrust co-efficient *in vacuo*

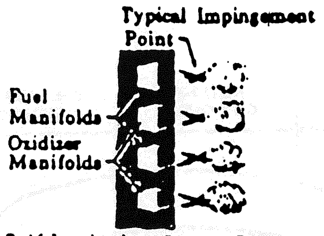
CHARACTERISTIC VELOCITY - Measures combustion performance by indicating how many *lb.$_m$/sec.* propellant must be burned to maintain chamber pressure

$$C^* = \frac{P_c \times A_T \times g_c}{\dot{m}} = \frac{g_c \times I_s}{C_T}$$

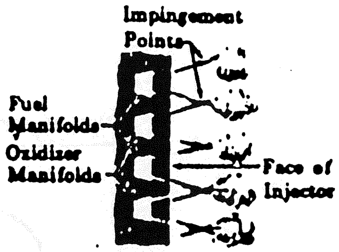
INJECTOR CONFIGURATIONS

GOOD MIXING IS IMPORTANT
TO GOOD COMBUSTION EFFICIENCY

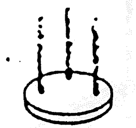

Impinging Head

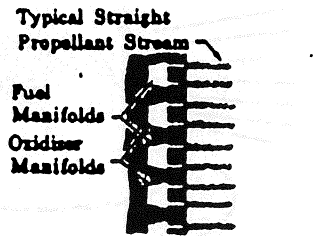


Self-Impinging Stream Pattern

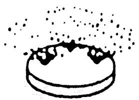


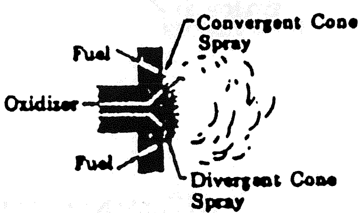
Doublet Impinging Stream Pattern


Shower Head

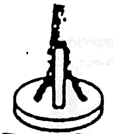


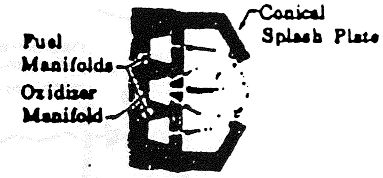
Shower Head Stream Pattern


Spray Head

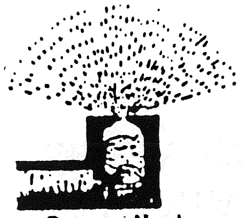


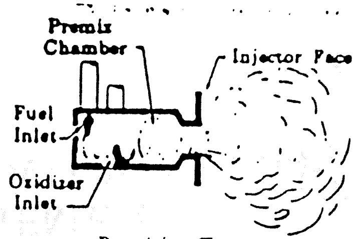
Spray Injection Pattern


Target Plate (Splash)



Splash Plate Pattern

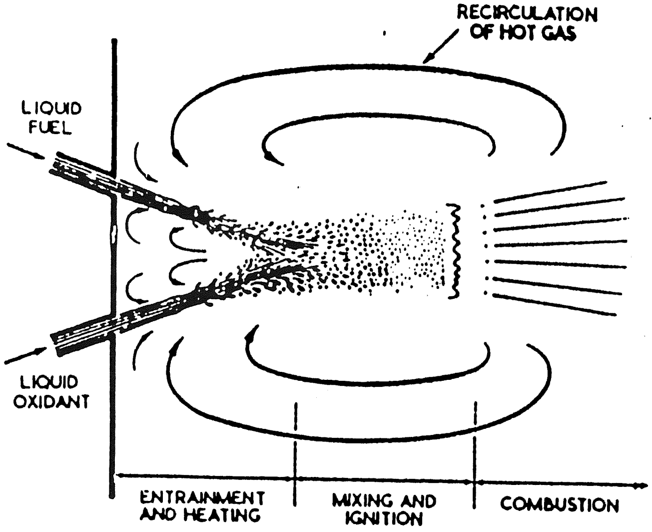

Pressure Nozzle



Premixing Type

INJECTOR FLOW.

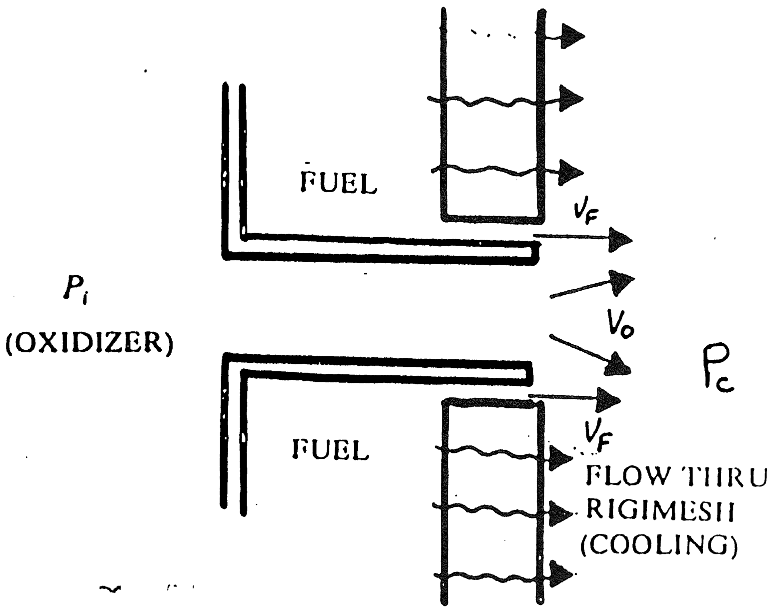
LIQUID ENTRAINMENT AND MIXING



MIXTURE RATIO:

$$\frac{\text{OXIDIZER MASS FLOW}}{\text{FUEL MASS FLOW}} = \text{O/F}$$

CO-AXIAL IMPINGEMENT SPRAY CONFIGURATION (RL-10)



LIQUID FLOW:

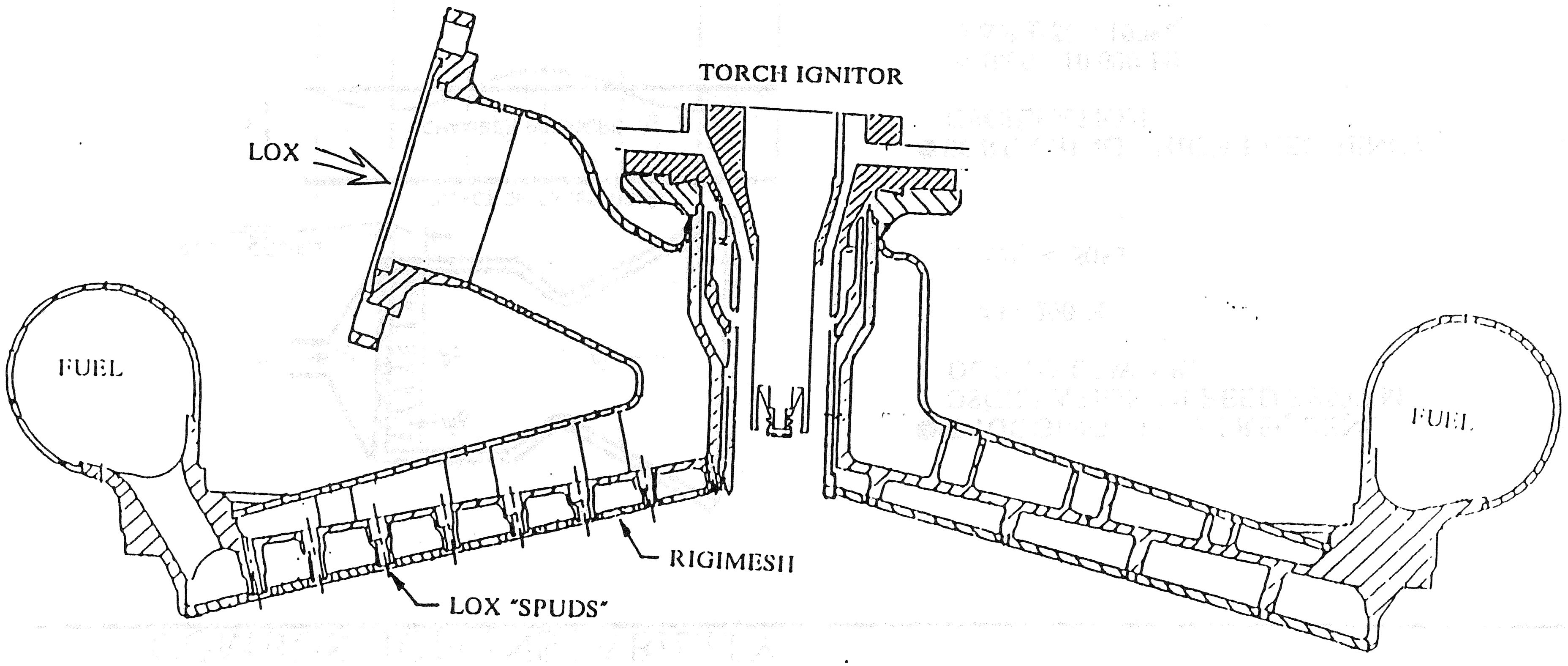
$$\dot{m} = A c_D \sqrt{2 g \rho (P_i - P_c)}$$

A = FLOW AREA

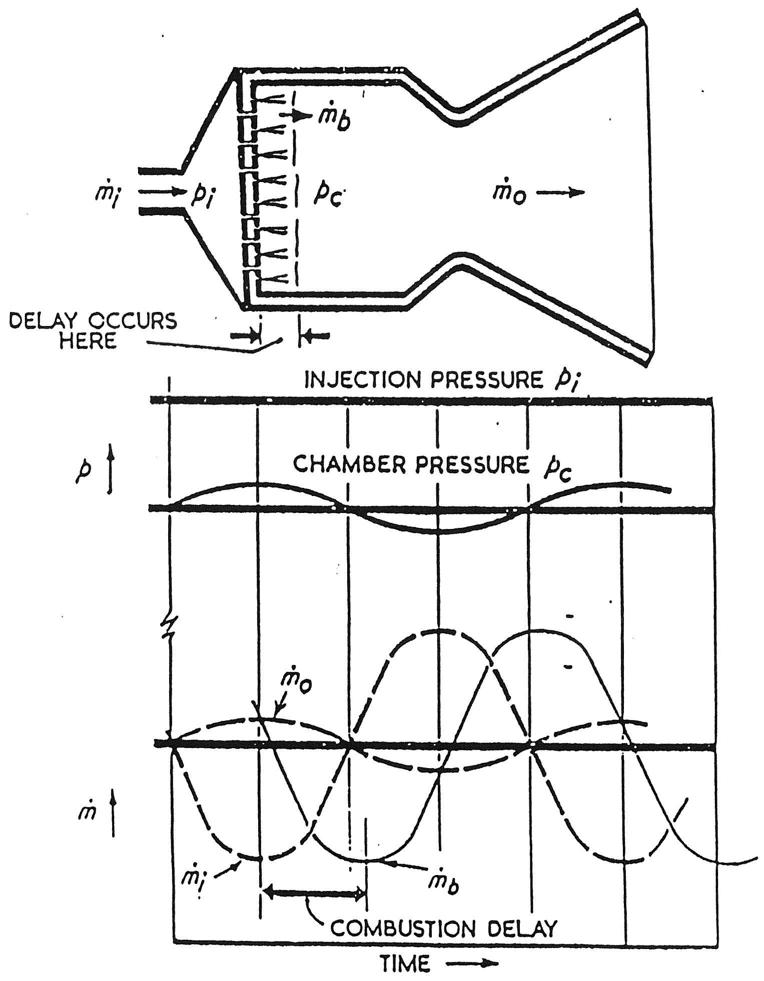
ρ = FLUID DENSITY

RL-10 INJECTOR & IGNITOR ASSEMBLY

HYDROGEN FUEL (GASEOUS) AND LIQUID OXYGEN (LOX)



COMBUSTION INSTABILITY



‘ Chugging ’ combustion oscillations.

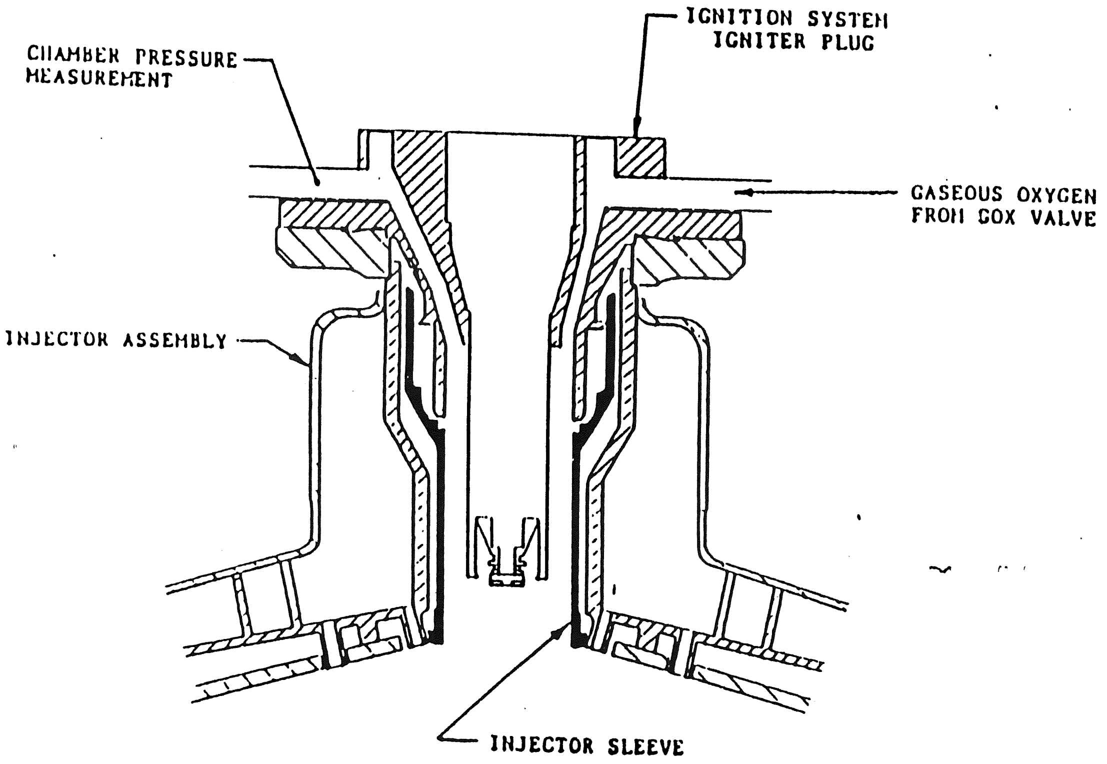
- CHUGGING - LOW FREQUENCY OSCILLATION IN FEED SYSTEM DUE TO LOW ΔP_{inj}
  - o 40 - 200 Hz
  - o $\Delta P_C < 50\%$
- SCREAMING - HIGH FREQUENCY OSCILLATION
  - o 1000 - 10,000 Hz
  - o $\Delta P \pm 50 - 100\%$
- MINIMIZED THROUGH USE OF BAFFLES AND ACOUSTIC LINERS
- DESTRUCTIVE TO VEHICLE AND ENGINE HARDWARE

IGNITOR TYPES

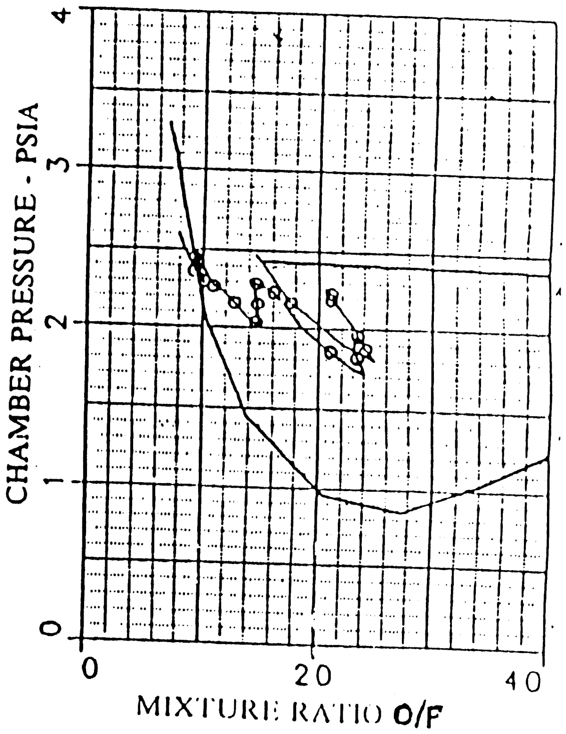
- SPARK PLUG
- GLOW PLUG
- PYROTECHNIC
- CATALYTIC
- HYPERGOLIC

RL-10 SPARK PLUG TYPE TORCH IGNITOR

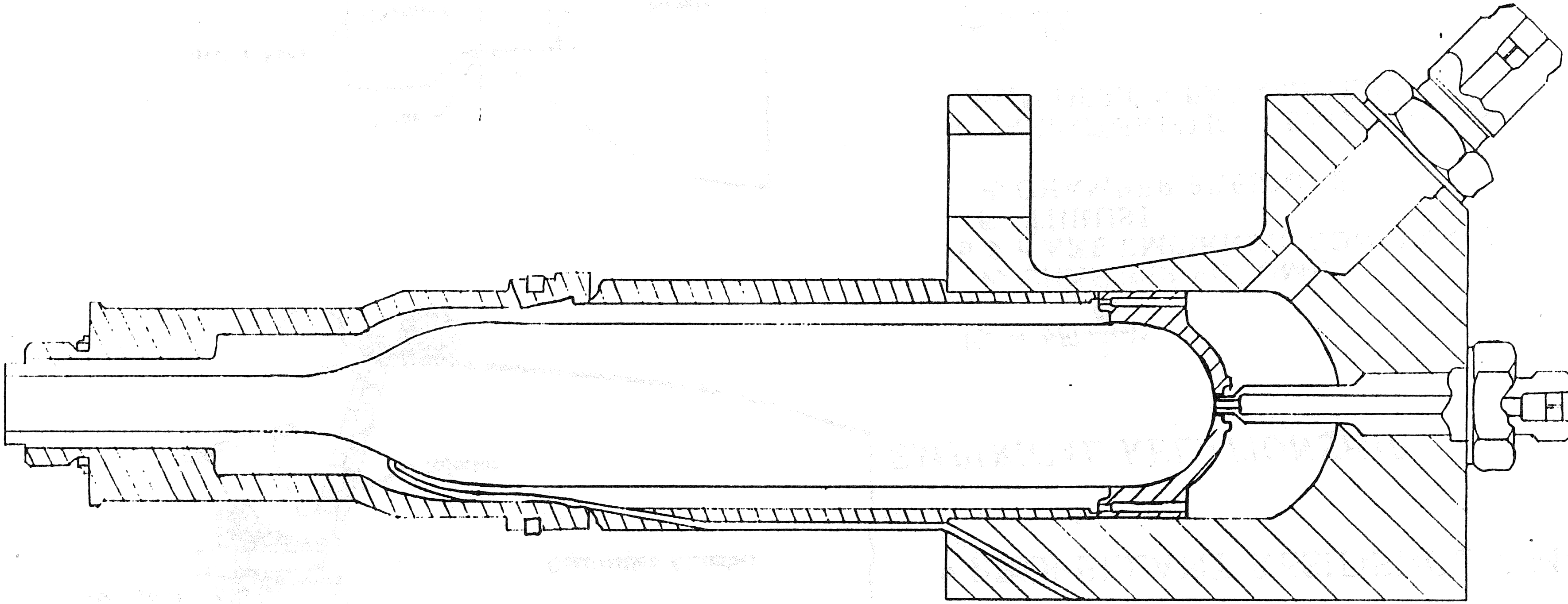
GASEOUS HYDROGEN AND OXYGEN FED:



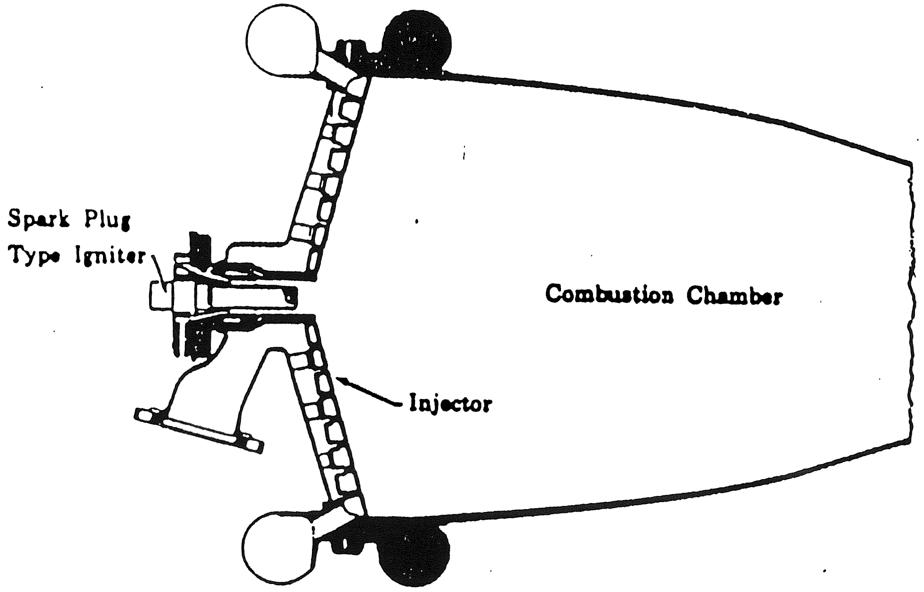
Pc vs O/F DEFINES IGNITION BUCKET:

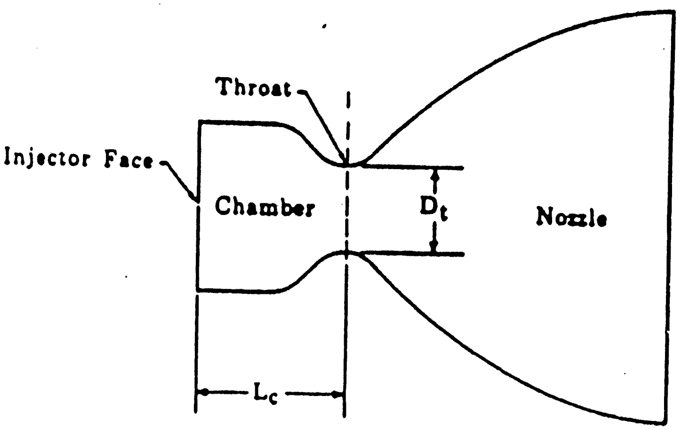


SSME ATD STE IGNITOR



COMBUSTION CHAMBER





CHAMBER DESIGN CONSIDERATIONS:

- MIXTURE RATIO DISTRIBUTION
- HOT SPOTS
- CHAMBER WALL COOLING
- PROPELLANT RESIDENCE TIME

EMPIRICAL RELATIONSHIP:

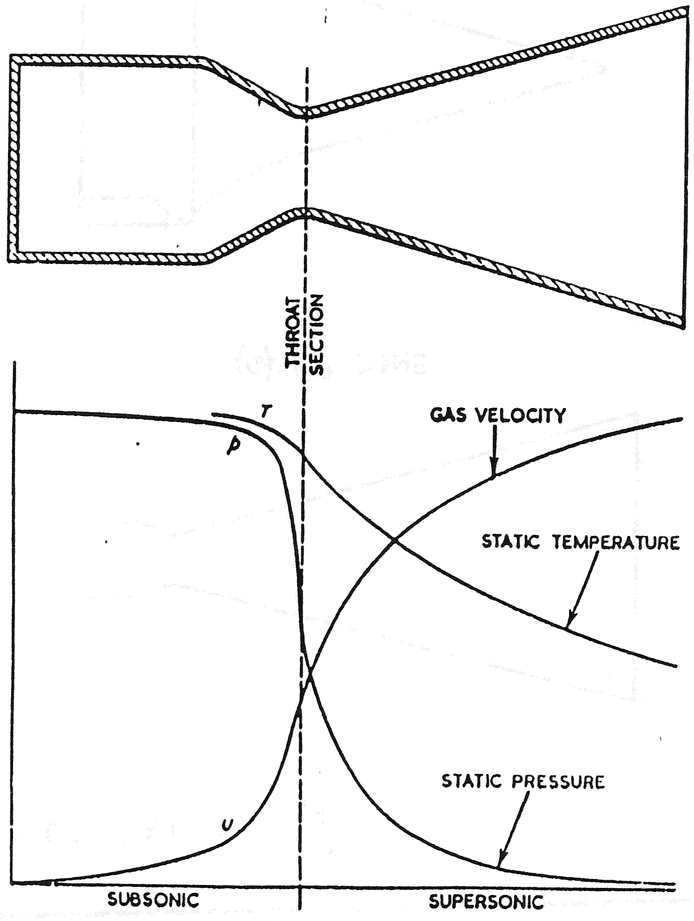
$$V_C = bF\left(\frac{1}{P_C}\right)^n$$

V_C CHAMBER VOLUME
b & n ARE EMPIRICAL CONSTANTS
F - THRUST
P_C CHAMBER PRESSURE

CHARACTERISTIC LENGTH (L*) IS MAIN DESIGN PARAMETER:

$$L^* = \frac{V_C}{A_T}$$

NOZZLE EXPANSION PROCESS



- CONVERGENT / DIVERGENT NOZZLE

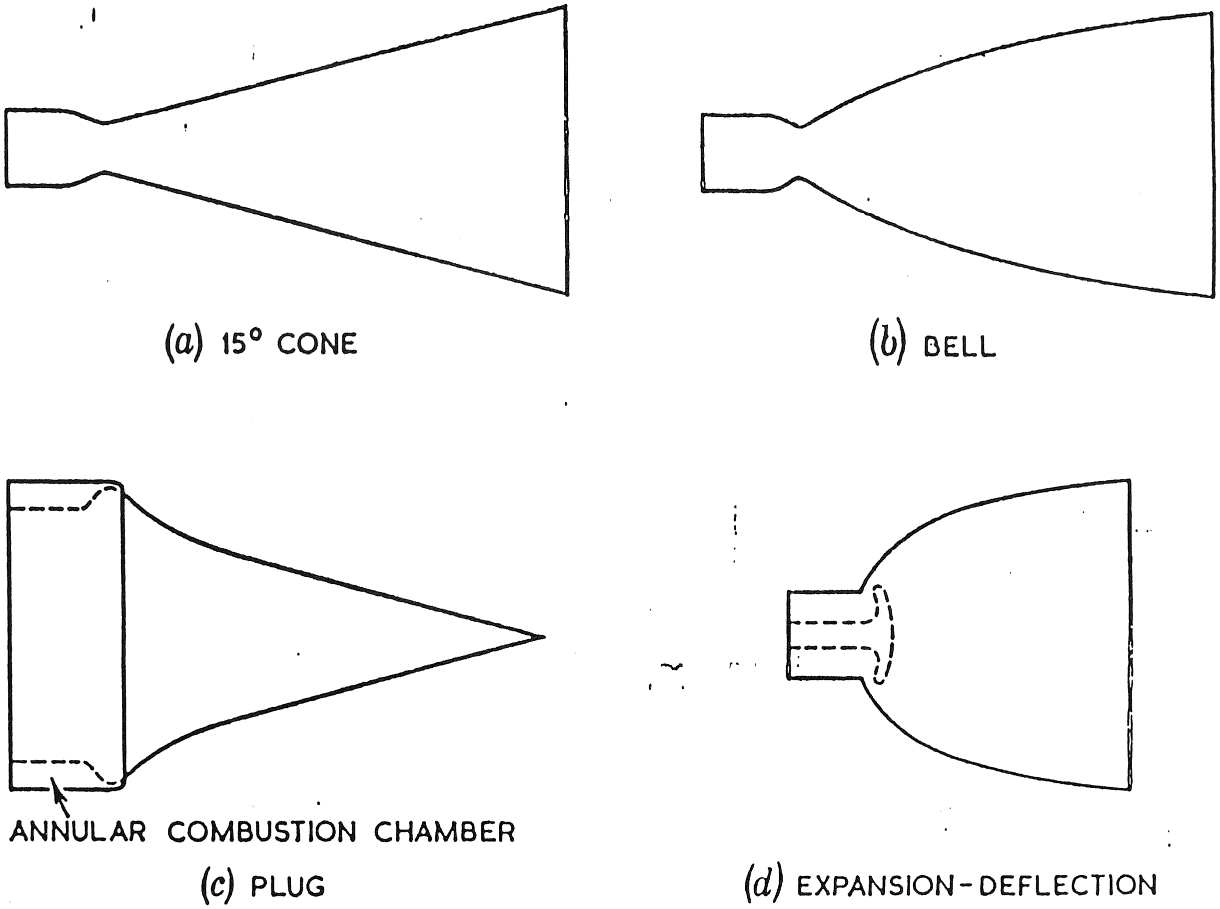
- *CHEMICAL COMPOSITION:*
  FROZEN:
  - CHEMICAL EQUIL. IS "FROZEN" AT CHAMBER CONDITIONS THROUGHOUT EXPANSION PROCESS

  - SHIFTING
    CHEMICAL COMPOSITION IS AT EQUILIBRIUM AT EVERY POINT ALONG NOZZLE

- DIFFERENCE BETWEEN FROZEN & SHIFTING EQUILIBRIUM app. 5%

NOZZLE TYPES

EXPANSION RATIO $= \frac{AE}{AT}$



(a) 15° CONE

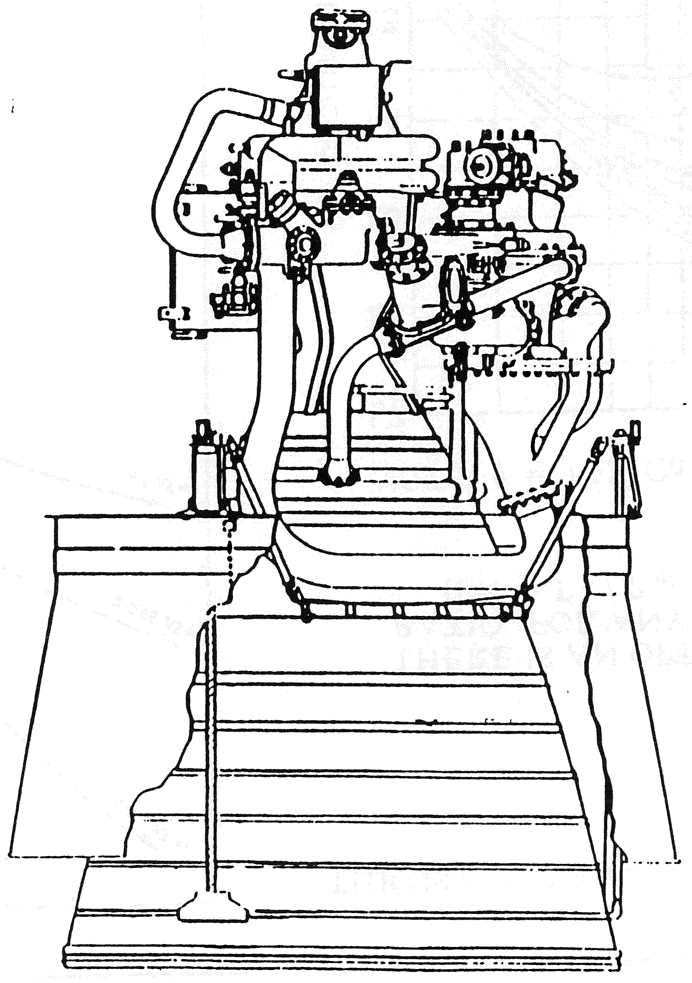
(b) BELL

(c) PLUG

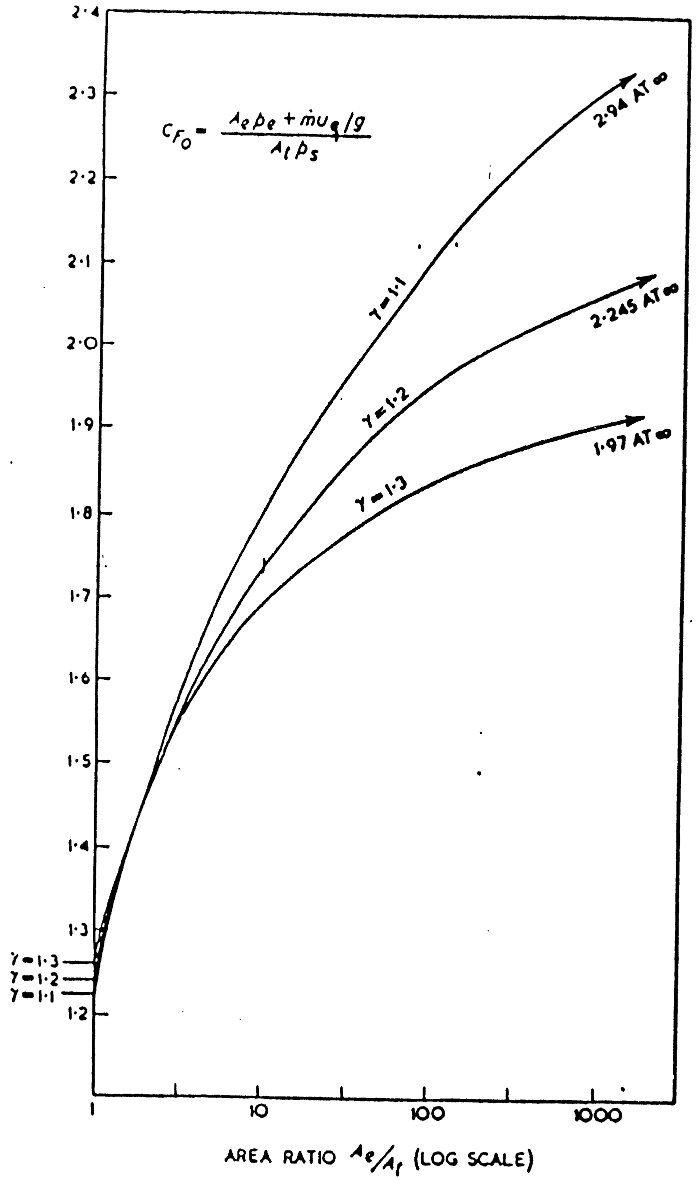
(d) EXPANSION-DEFLECTION

Nozzle contours for 40 : 1 area ratio.

RL-10 TRANSLATING NOZZLE



THRUST COEFFICIENT – C_F

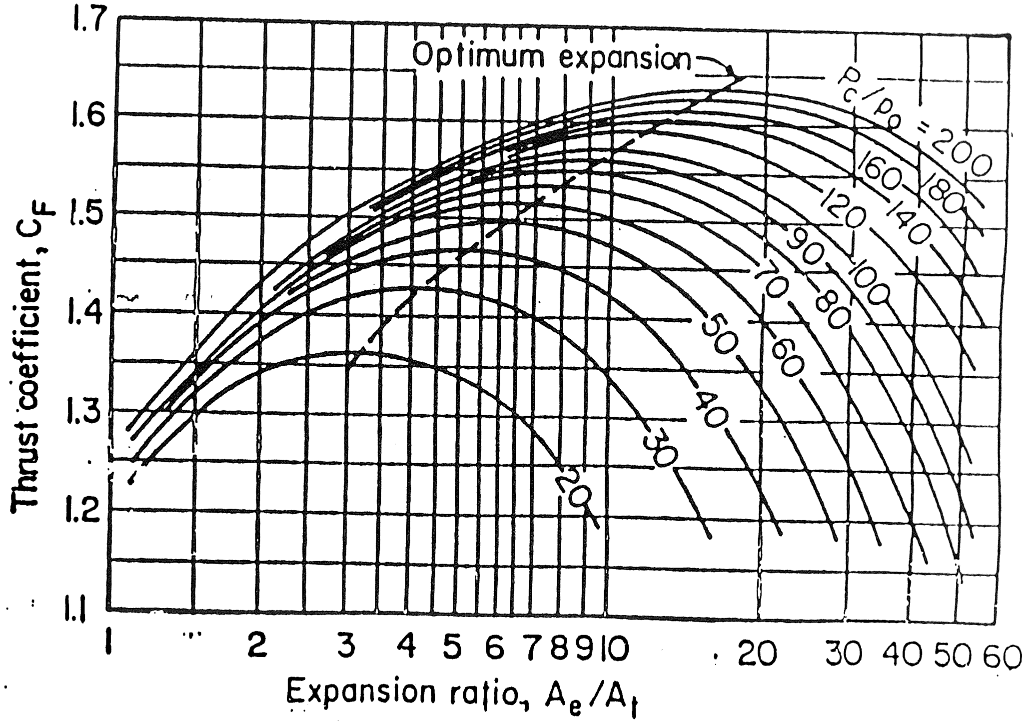


Thrust coefficient *in vacuo*.

THRUST = $C_F A_T P_C$

THERE IS AN OPTIMUM EXPANSION RATIO FOR ANY PRESSURE RATIO: (PR = P_c / P_0)

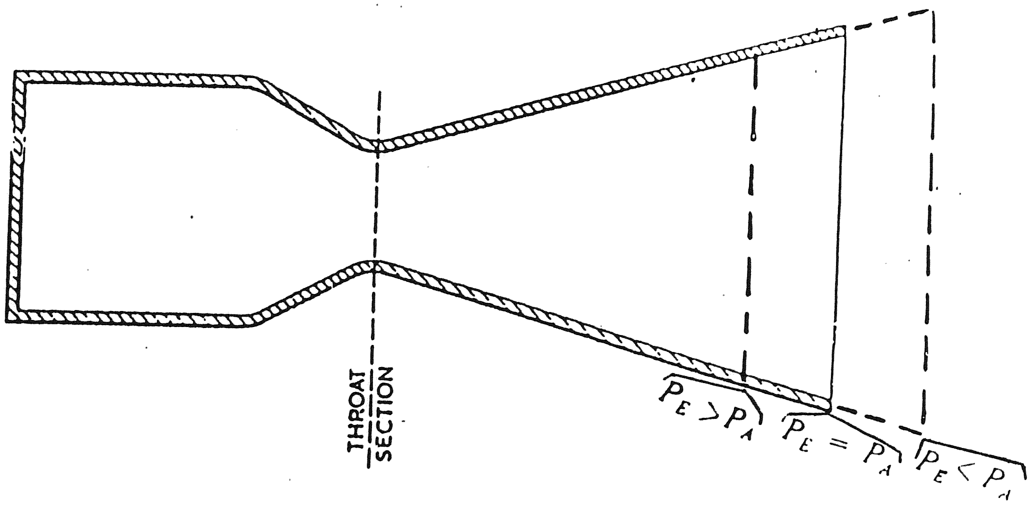
NOTE: γ = 1.28, Cd = 1.0, NO SEPARATION

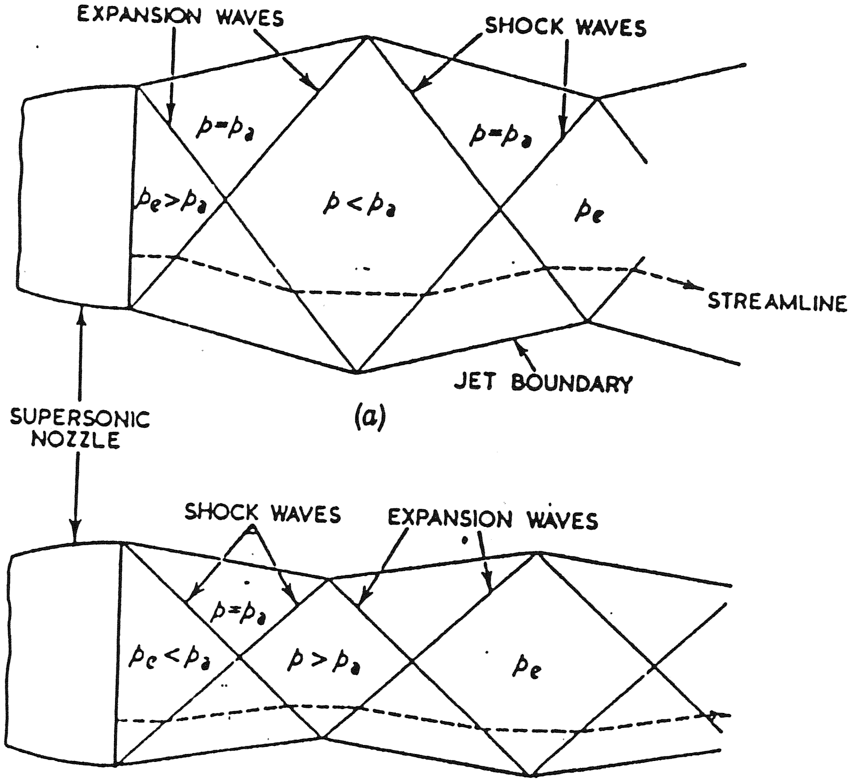


UNDER /OVER EXPANSION

OPTIMUM EXPANSION

RATIO : $P_E = P_A$





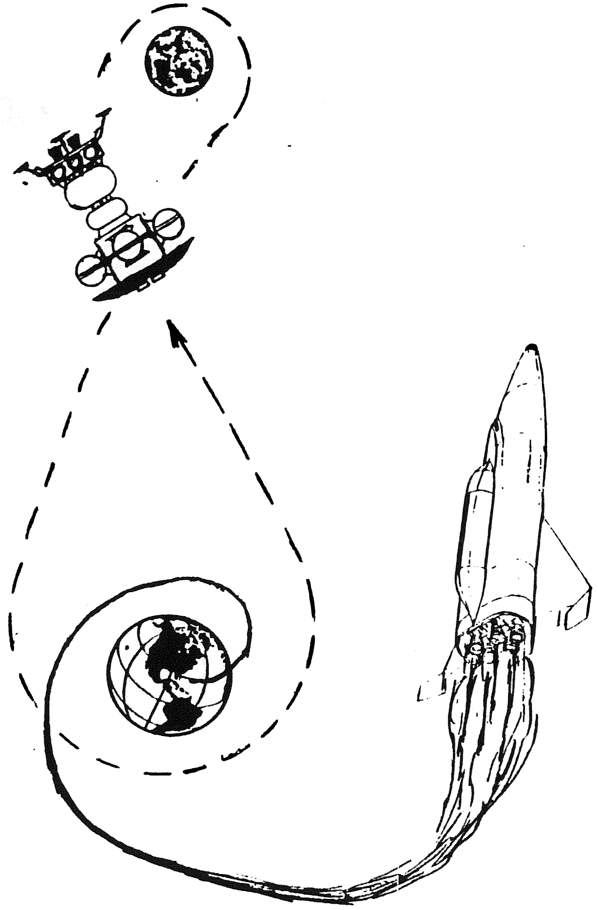
(a)

(b)

⇐ UNDER EXPANDED EXHAUST

⇐ OVER EXPANDED EXHAUST

TRAJECTORY AND VEHICLE ANALYSIS



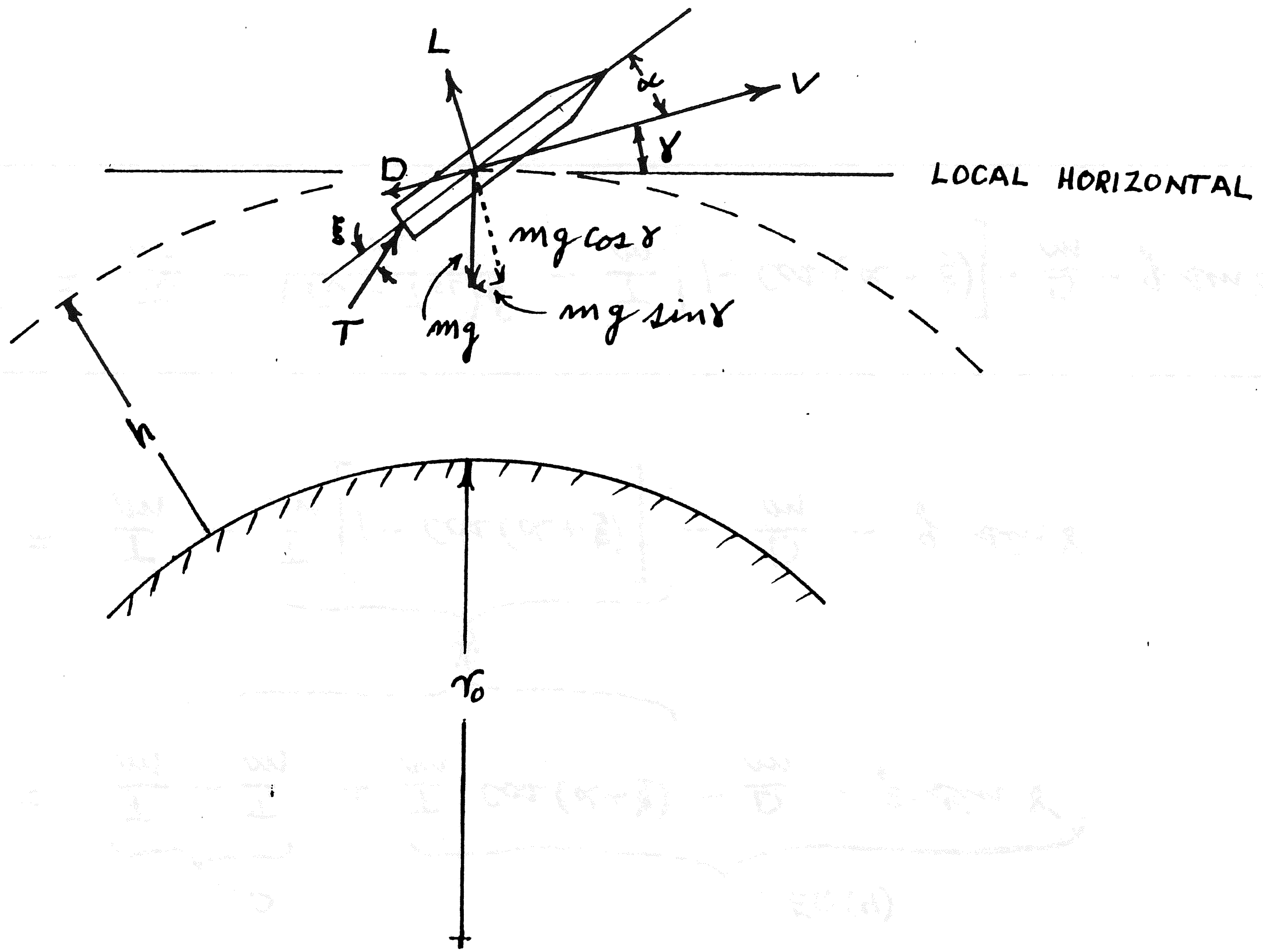
PRATT & WHITNEY
VEHICLE ANALYSIS GROUP

Joseph Sabatella
Russell Joyner

TOPICS

- TRAJECTORY CONSIDERATIONS
- VEHICLE WEIGHT RELATIONSHIPS
- VEHICLE/ENGINE SIZING METHODOLOGY

FREE BODY FORCE DIAGRAM



MANIPULATIONS OF $m\dot{V} = \sum F_T$ (FOR A PURPOSE!?)

$$\dot{V} = \overbrace{\frac{T}{m} - \frac{T}{m}}^{=0} + \overbrace{\frac{T}{m}\cos(\alpha+\xi) - \frac{D}{m} - g\sin\gamma}^{=EQ(4)}$$

$$\Downarrow$$

$$\dot{V} = \frac{T}{m} - \overbrace{\frac{T}{m}\left[1-\cos(\alpha+\xi)\right]} - \frac{D}{m} - g\sin\gamma$$

$$\boxed{\dot{V} = \frac{T_V}{m} - \frac{(T_V - T_{SL})\delta}{m} - \frac{T}{m}\left[1-\cos(\alpha+\xi)\right] - \frac{D}{m} - g\sin\gamma} \quad (6)$$

IDEAL VELOCITY

$\left\{ \begin{array}{l} \text{THE VELOCITY THAT WOULD} \\ \text{BE ACHIEVED WITHOUT LOSSES} \\ \text{(DRAG, GRAVITY, ETC.)} \end{array} \right.$

$$V_i = \int_{t_1}^{t_2} \frac{T_v}{m}\, dt = \int_{t_1}^{t_2} \frac{g_o I_v}{m} \frac{dm}{dt}\, dt = I_v g_o \int_{m_1}^{m_2} \frac{dm}{m}$$

(8) $$\boxed{V_i = I_v g_o \ln \frac{m_1}{m_2}}$$

$\left\{ \begin{array}{l} \text{FAMOUS} \\ \text{ROCKET} \\ \text{EQUATION} \end{array} \right.$

ALTERNATE DEFINITION OF IDEAL VELOCITY

THE CONCEPT OF TRAJECTORY AVERAGE I_{sp}

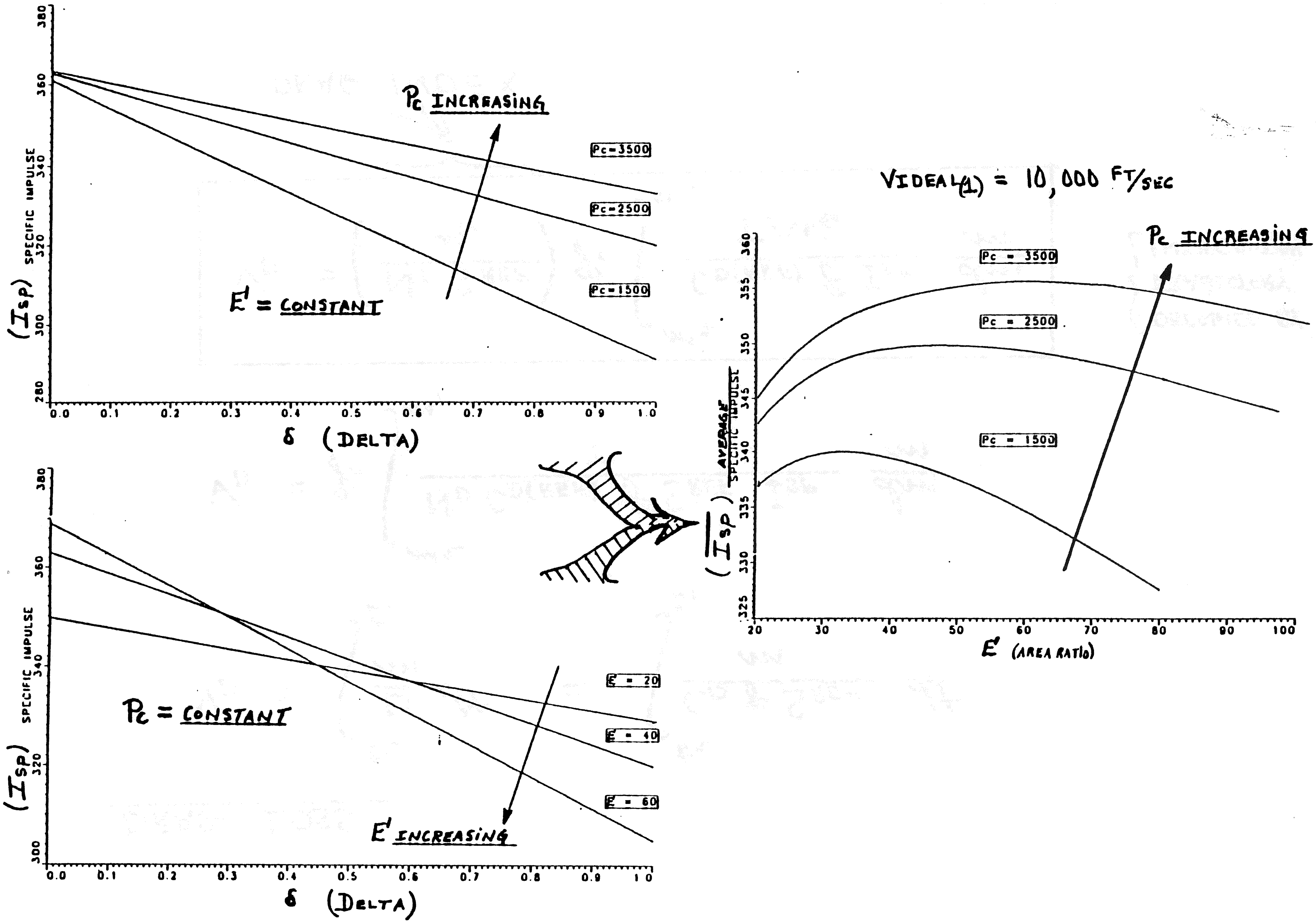
$$V_i' = \int_{t_1}^{t_2} \frac{T}{m}\, dt = \int_{t_1}^{t_2} \frac{g_0 I_{sp}}{m} \frac{dm}{dt}\, dt = \bar{I}_{sp}\, g_0 \int_{m_1}^{m_2} \frac{dm}{m}$$

$$\boxed{V_i' = \bar{I}_{sp}\, g_0 \ln \frac{m_1}{m_2}}$$

NOTE: $V_i' = V_i - V_{ATM}$

$\bar{I}_{SP} = I_V - (I_V - I_{SL})\bar{\delta}$

INFLUENCE OF ϵ' (AREA RATIO) AND Pc (CHAMBER PRESSURE) ON Isp



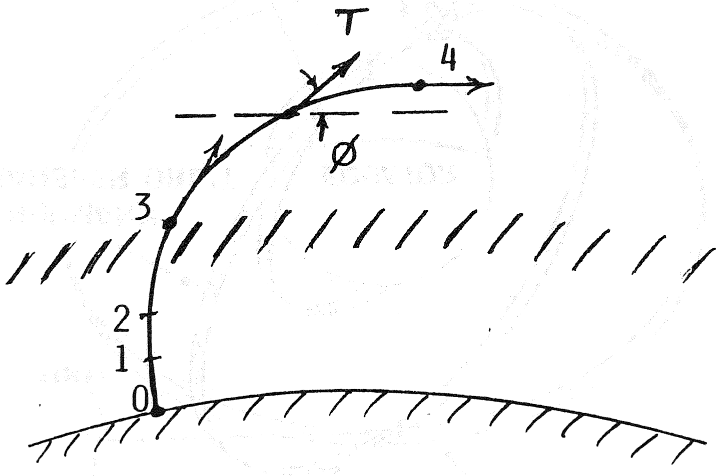
DRAG LOSS

$$V_D = \int_{t_1}^{t_2} \frac{D}{m}\, dt = \int_{t_1}^{t_2} \frac{C_D\, q\, S_{REF}}{m}\, dt$$

$$V_D = g_0 \int_{m_1}^{m_2} \frac{N_D\, C_{D(REF)}\, q\, S_{REF}\, I_{SP}}{T} \frac{dm}{m}$$

$$\boxed{V_D = \underbrace{\left(\frac{N_D\, S_{REF}}{W_G}\right)}_{\text{DRAG INDEX}} g_0 \int_{m_1}^{m_2} \frac{C_{D(REF)}\, q\, I_{SP}}{T/W_G} \frac{dm}{m}} \quad \left\{ \begin{array}{l} \text{OBTAINED BY} \\ \text{TRAJECTORY} \\ \text{INTEGRATION} \end{array} \right.$$

LAUNCH VEHICLE STEERING SCHEDULE



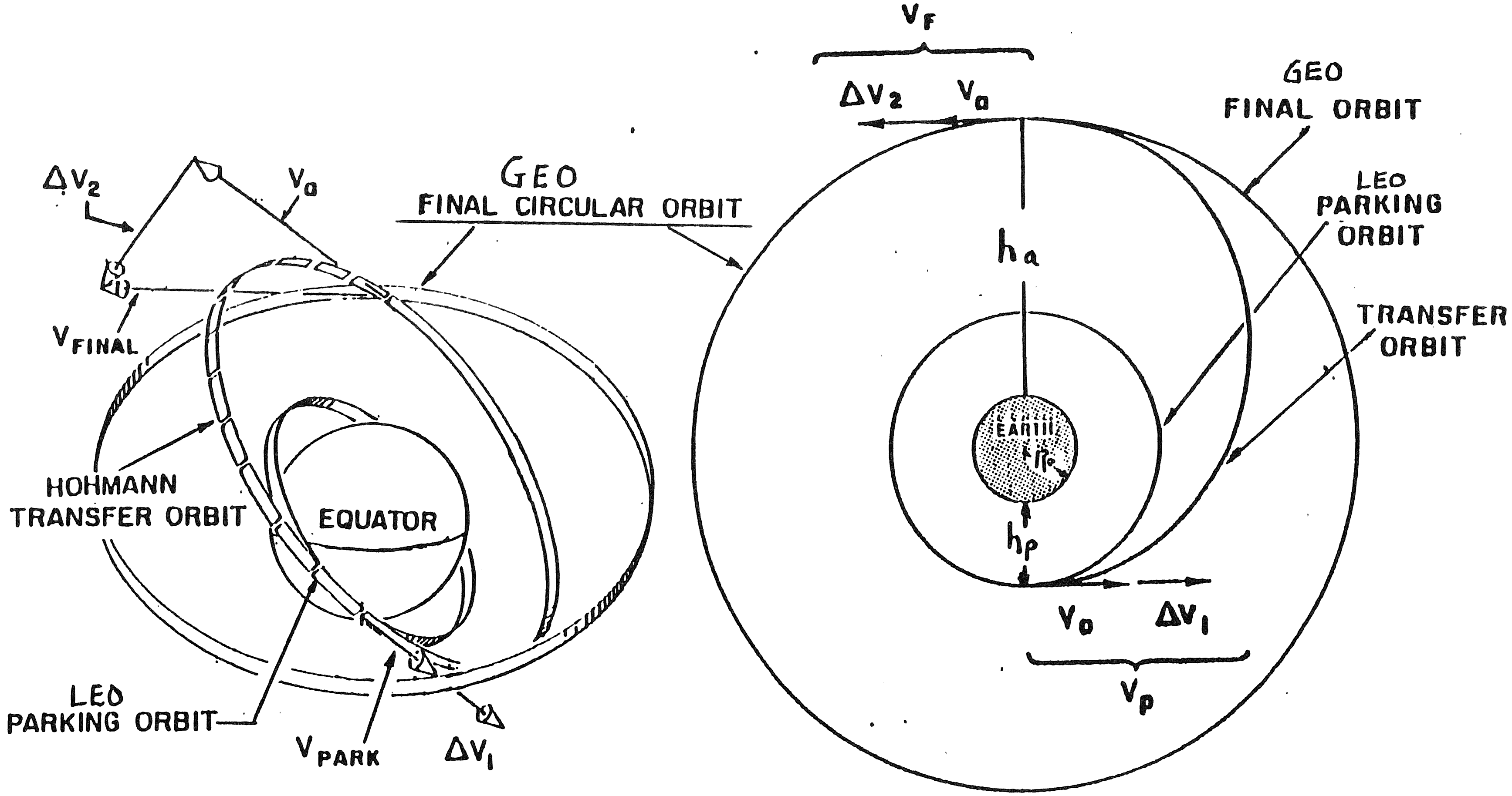
NOTE: τ = BURN TIME
FROM 3 TO 4

0 - 1	VERTICAL RISE
1 - 2	PITCH-OVER MANEUVER
2 - 3	GRAVITY TURN ($\alpha \approx 0$) TO POINT WHERE DRAG ≈ 0
3 - 4	OPTIMUM PITCH SCHEDULE: $\tan\phi = \left(1 - \frac{at}{\tau}\right)\tan\gamma_0$
4	BURNOUT: $\gamma = 0$, $50 \leq h \leq 80$ NM

INTER-ORBIT TRANSFER

GEO- GEO-STATIONARY ORBIT (22,700 NM)

LEO- LOW EARTH ORBIT (100 NM)



Hohmann Transfer Maneuver

TYPICAL IDEAL VELOCITY REQUIREMENT

$$\begin{array}{ccccccccccc} V_i & = & V & + & V_{ATM} & + & V_{TV} & + & V_D & + & V_G \\ \Downarrow & & \Downarrow & & \Downarrow & & \Downarrow & & \Downarrow & & \Downarrow \\ 30{,}030 & = & 24{,}430^{*} & + & 250 & + & 350 & + & 300 & + & 4700 \end{array}$$

* EAST LAUNCH / 28.5° LATTITUDE (75 NM x 150 NM)

$V_{INERTIAL} = 24{,}430 + 1370 = 25{,}800$ FT/SEC

WEIGHT (MASS) PARAMETERS ($w \equiv m g_0$)

WEIGHT (MASS) RATIO, MR

GROSS LIFT-OFF WEIGHT

$$\frac{W_1}{W_2} = \frac{W_G}{W_G - W_P} = \frac{1}{1 - \frac{W_P}{W_G}} \leftarrow \mu$$

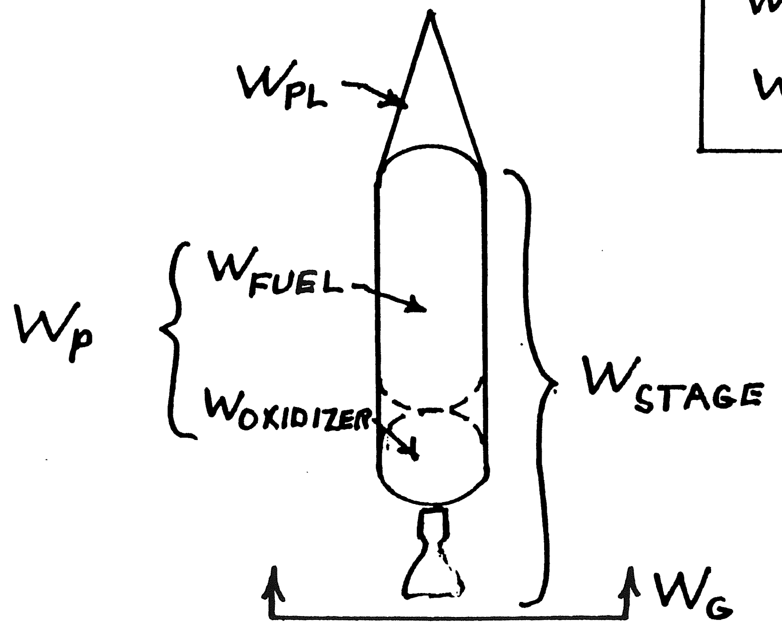
$\frac{\text{WEIGHT START}}{\text{WEIGHT END}}$

PROPELLANT WEIGHT

PROPELLANT MASS FRACTIONS

$W_P/W_G = \mu$ = PROPELLANT FRACTION

$W_P/W_{STG} = \lambda$ = STAGE PROP. FRACTION



$$W_G = W_{STG} + W_{PL}$$

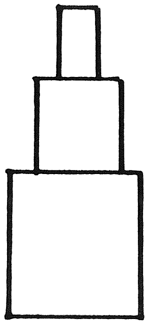
MAY BE WEIGHT OF UPPER STAGE

BASIC VEHICLE SIZING RELATIONSHIPS

- $V_i = I_v g_o \ln MR \Longrightarrow \boxed{MR = e^{\frac{V_i}{I_v g_o}}}$
- $MR = \frac{W_G}{W_G - W_P} \Longrightarrow \boxed{\mu = \frac{MR - 1}{MR}}$
- $W_{DRY} = W_{STG} - W_P \Longrightarrow \boxed{W_{DRY} = (1-\lambda) W_{STG}}$
- $W_G = W_{STG} + W_{PL} \Longrightarrow \boxed{\frac{W_{PL}}{W_G} = 1 - \frac{\mu}{\lambda}}$

SIMPLIFIED SAMPLE PROBLEM (TWO STAGE VEHICLE)

PAYLOAD =
10,000 LB



$$\left.\begin{aligned} I_{sp} &= 450 \text{ SEC} \\ \lambda &= 0.9 \\ V_i &= 15{,}000 \text{ FT/SEC} \end{aligned}\right\} \text{EACH STAGE}$$

2ND STAGE

$$MR_2 = e^{\frac{15000}{450(32.2)}} = 2.816$$

$$\mu_2 = \frac{2.816 - 1}{2.816} = 0.645$$

$$\frac{W_{PL}}{W_{G2}} = 1 - \frac{0.645}{0.9} = 0.2835$$

$$W_{G2} = \frac{10{,}000}{0.2835} = 35{,}270 \text{ LB}$$

1ST STAGE

$$MR_1 = e^{\frac{15000}{450(32.2)}} = 2.816$$

$$\mu_1 = \frac{2.816 - 1}{2.816} = 0.645$$

$$\frac{W_{G2}}{W_{G1}} = 1 - \frac{0.645}{0.9} = 0.2835$$

$$W_{G1} = \frac{35{,}270}{0.2835} = 124{,}400 \text{ LB}$$

SAMPLE PROBLEM FOR OPTIMUM STAGING VELOCITY

ASSUME: $V_{i\,(\text{TOTAL})} = 30{,}000$ FT/SEC

1ST STAGE

$I_{SL_1} = 330$ SEC

$I_{V_1} = 360$ SEC

$\lambda_1 = 0.90$

2ND STAGE

$I_{V_2} \approx \bar{I}_{SP_2} = 450$ SEC

$\lambda_2 = 0.88$

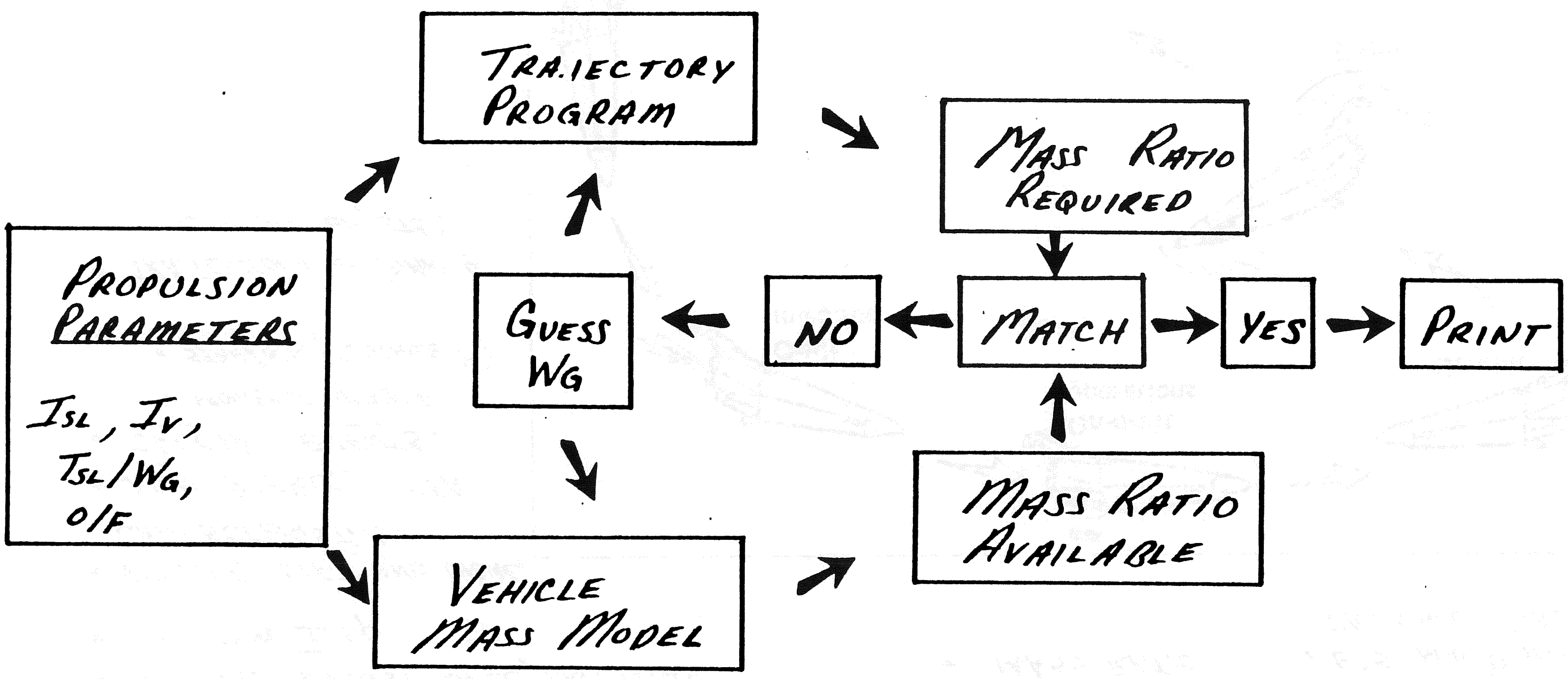
	V_{i_1}	$\bar{\delta}_1$	$\bar{I}_{SP_1}$	MR_1	$\frac{\bar{I}_{SP_1}}{\bar{I}_{SP_2}}$	MR_2	V_{i_2}	V_{i_T}
	6000	0.50	345	1.7162	0.767	3.0386	16090	22090
	8000	0.38	348.6	2.0395	0.775	3.1922	16800	24800
	10000	0.31	350.7	2.4243	0.779	3.4154	17780	27780
	12000	0.26	352.2	2.8810	0.783	3.6882	18890	30890
*	11440	0.265	352	2.7437	0.782	3.6046	18560	30000

VEHICLE SIZING METHODOLOGY: PART I

IN THE FIRST PART OF THIS COURSE, VEHICLE SIZING WAS SIMPLIFIED TO ILLUSTRATE CONCEPTS

- V_{IDEAL} = CONSTANT
- λ = CONSTANT

SIMPLIFIED VEHICLE SIZING FLOWPATH



"POST" PROGRAM MODEL

EARTH - TO - ORBIT TRAJECTORY

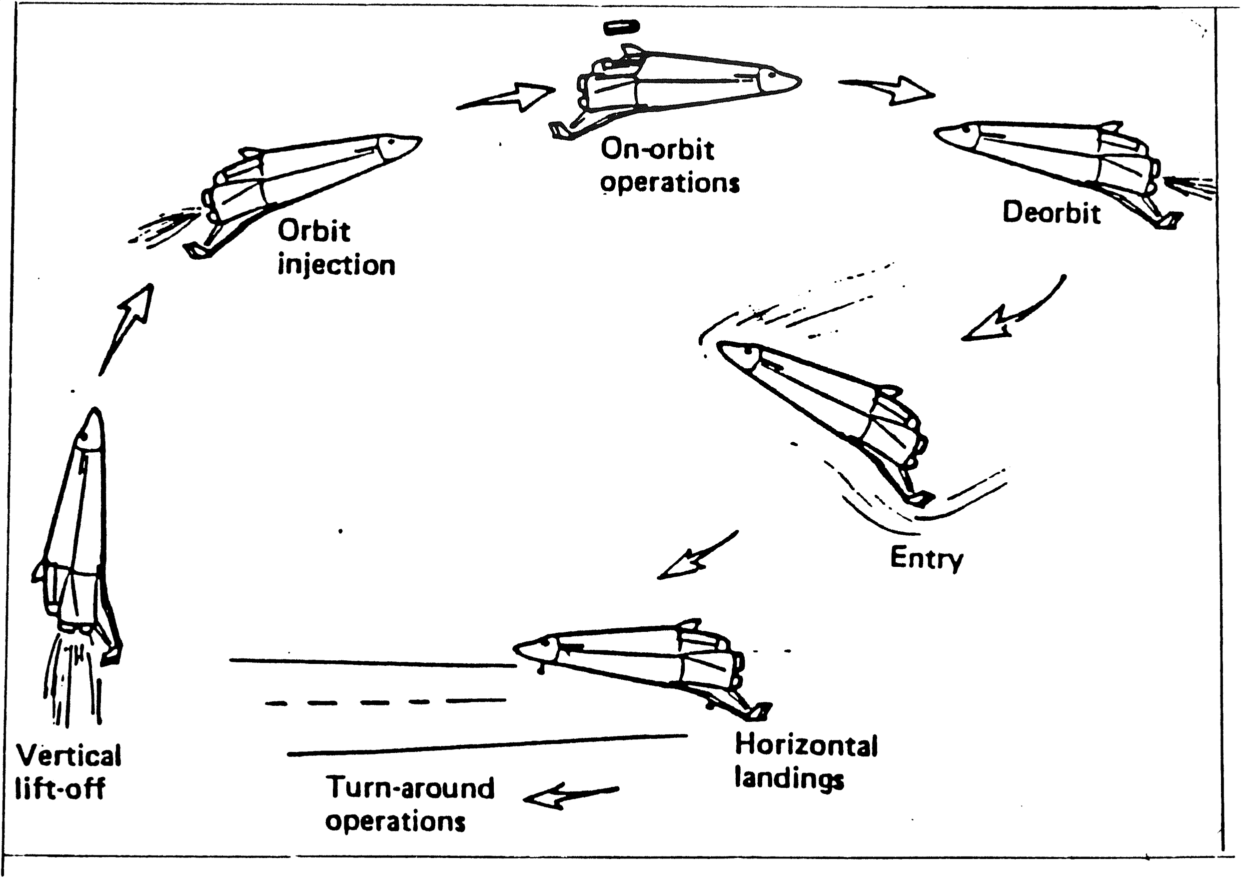
(SINGLE - STAGE - TO - ORBIT)

"POST" INPUTS:

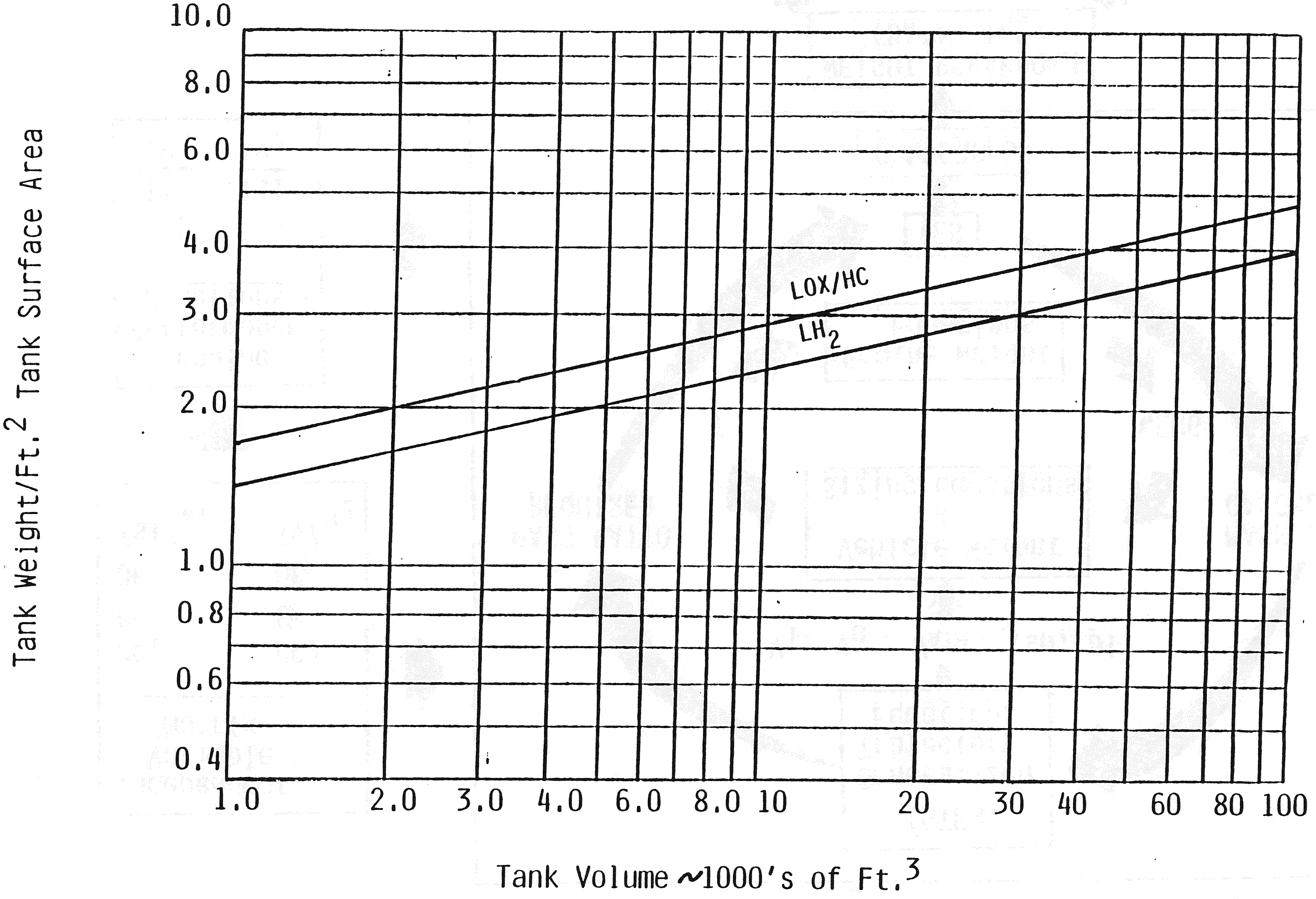
- GROSS LIFT-OFF WEIGHT
- ENGINE THRUST SIZE, EXIT AREA
- VACUUM I_{sp}
- VEHICLE LIFT AND DRAG CHARACTERISTICS
- ATMOSPHERIC DATA
- SPECIAL EVENTS:
  - MOVABLE NOZZLE
  - STAGING (2 STAGE +)
- TRAJECTORY GUIDANCE (CLOSED-LOOP, ETC)

"POST" OUTPUT

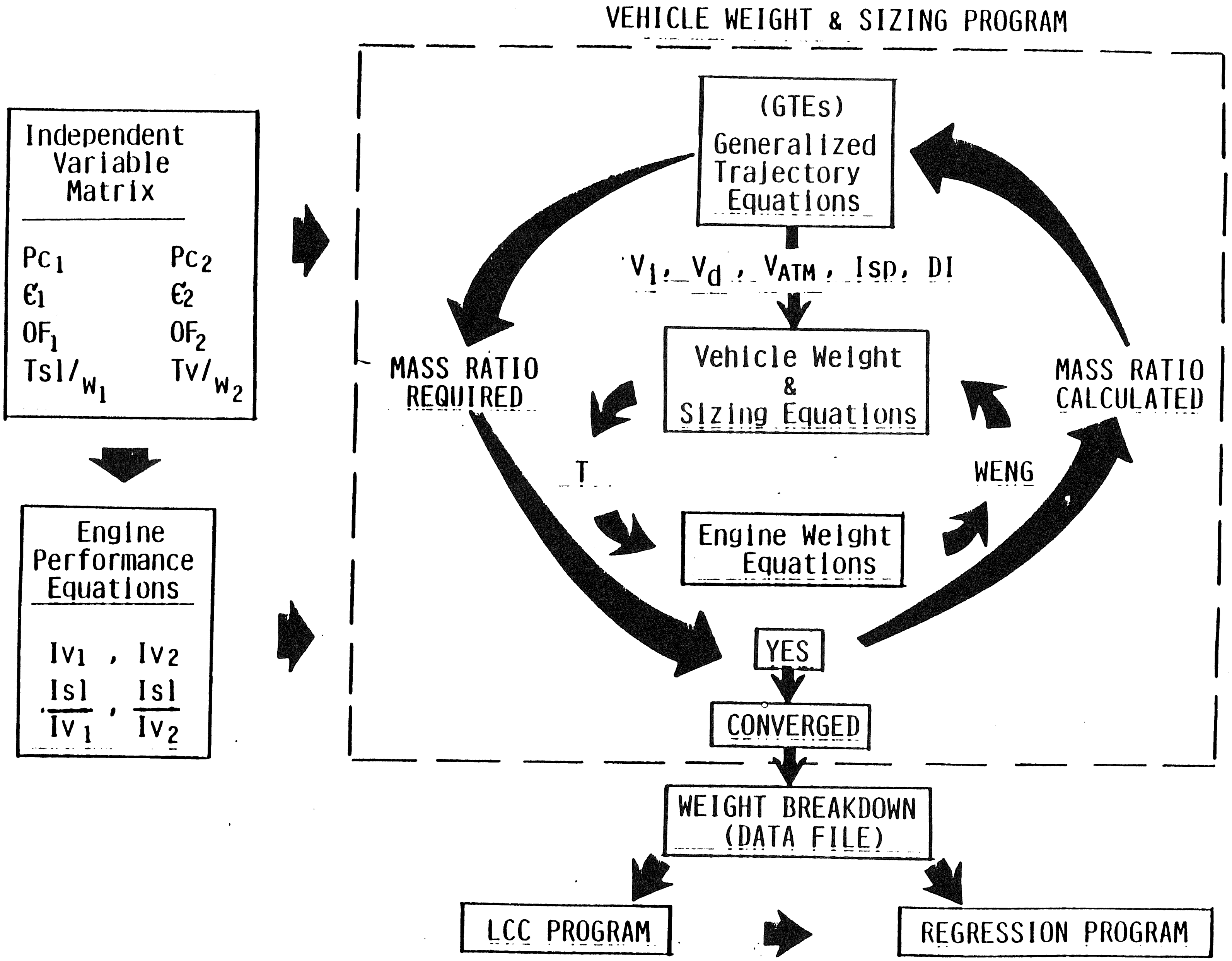
- ΔV_{IDEAL}
- MASS RATIO
- Propellant BURNED
- G'S AND DYNAMIC PRESSURE TRACE



SPECIFIC TANK WEIGHT VS. PROPELLANT TANK VOLUME



VEHICLE WEIGHT & SIZING PROCEDURE



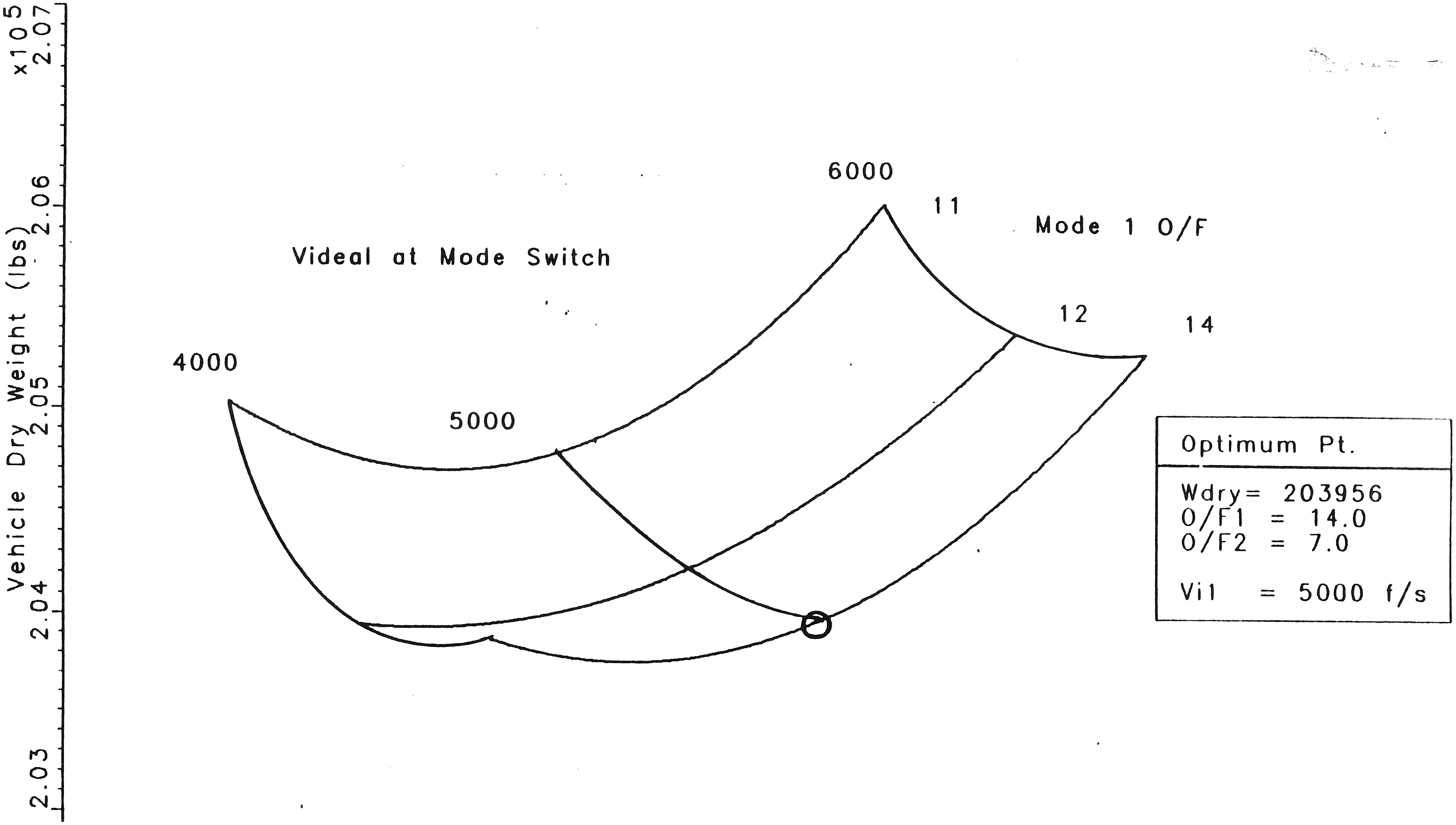
SINGLE STAGE TO ORBIT
VEHICLE WEIGHT BREAKDOWN

VEHICLE WEIGHT GROUP
WEIGHT - LBS

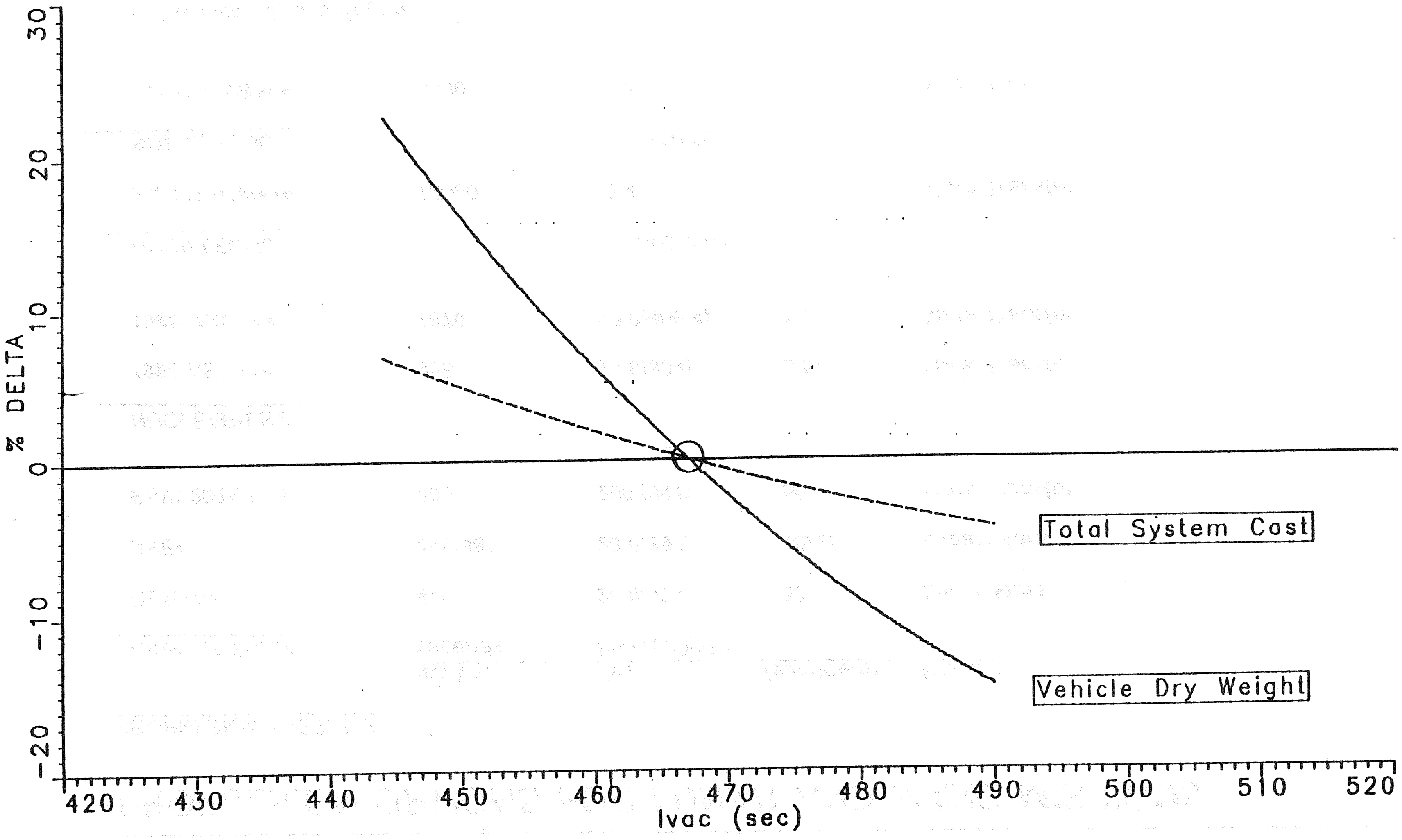
COMPONENT	WEIGHT	
WING	26371.	
TAIL	945.	
BODY (BASIC STRUCTURE)	32774.	
BODY FLAP	774.	
HC FUEL TANK	0.	
OXIDIZER TANK	21295.	
HYDROGEN TANK	17977.	
THRUST STRUCTURE	5250.	
LANDING GEAR	6026.	
TOTAL STRUCTURE WEIGHT	111413.	
HC FUEL TANK INSUL	0.	
OXIDIZER TANK INSUL	1420.	
HYDROGEN TANK INSUL	3115.	
INT. THERMAL PROTECT.	1419.	
EXT. THERMAL PROTECT.	23645.	
TOTAL T.P.S. WEIGHT	29598.	
CREW PROVISIONS	1000.	
SURFACE CONTROLS	4226.	
AVIONICS	4600.	
ENVIRONMENTAL CNTL SYS	6500.	
ELECTRICAL POWER UNIT	2400.	
ELECTRICAL SYSTEMS	11273.	
PAYLOAD PROVISIONS	420.	
TOTAL VEHICLE SYSTEMS	30419.	
NUMBER OF ENGINES	6.	
TVAC/ENGINE (MODE1/MODE2)	599818.	430950.
MAIN ENGINES	21184.	
TVC,HEAT SHIELD,PLUMB	3599.	
FUEL/OXIDIZER SYSTEMS	8050.	
FLYBACK PROPULSION	0.	
FLYBACK FUEL SYSTEMS	0.	
RCS INERTS	2598.	
OMS INERTS	3065.	
TOTAL PROPULSION SYSTEMS	38496.	
VEHICLE MARGIN ALLOWANCE	16374.	
TOTAL STAGE DRY WEIGHT *	226300.	

COMPONENT	WEIGHT	
STAGE DRY WEIGHT **	226300.	
RESIDUAL PROPELLANTS	14889.	
ASCENT RESERVE PROP	6574.	
CREW & GEAR	2100.	
PAYLOAD BAY/SHROUD	9911.	
LANDING/WET WEIGHT	259773.	
RCS PROPELLANT	4326.	
OMS PROPELLANT	26414.	
BURN-OUT WT (W/O PAYLOAD)	290513.	
PAYLOAD	35000.	
BURN-OUT WT (W/PAYLOAD)	325513.	
	MODE 1	MODE 2
HYDROCARBON BOIL-OFF	0.	0.
HYDROGEN BOIL-OFF	239.	850.
OXIDIZER BOIL-OFF	2512.	4459.
BOIL-OFF PROPELLANT	2751.	5309.
HYDROCARBON (ASCENT)	0.	0.
HYDROGEN (ASCENT)	59590.	211595.
OXIDIZER (ASCENT)	715083.	1269568.
ASCENT PROPELLANT	774673.	1481163.
ASCENT + B/O PROPELLANT	777414.	1486452.
GROSS WT (W/PAYLOAD)	2589379.	1811921.
PROP RATIO(WPROP/WGROSS)	0.2991732	0.8174545
MASS RATIO(WGROSS/WBO)	1.4290791	5.5663519
TOTAL BOIL-OFF PROPELLANT	8060.	
TOTAL ASCENT PROPELLANT	2255771.	
TOTAL PROPELLANT	2263866.	
TOTAL STAGE WEIGHT	2554379.	
TOTAL GROSS L/O WEIGHT	2589379.	
TOTAL PROP FRACTION	0.8831113	
TOTAL PROP RATIO	0.8711743	
TOTAL MASS RATIO	7.9547586	

Advanced O2/H2 Engine Study
Vehicle Dry Weight Sensitivity to Mode Switching
Advanced Stg. Combustion Engine – Dual Mixture Ratio
Advanced SSTO Vehicle/50x150nm Mission/35k Payload



Advanced O2/H2 Engine Study
SSTO Vehicle Sensitivity to Propulsion System Isp
Advanced Stg. Combustion Engine – Dual Mixture Ratio
ADVANCED SSTO VEHICLE/50X150NM MISSION/35K PAYLOAD



UTC/SEI SYSTEM REQUIREMENTS

PROPULSION OPTIONS FOR LUNAR AND MARS MISSIONS

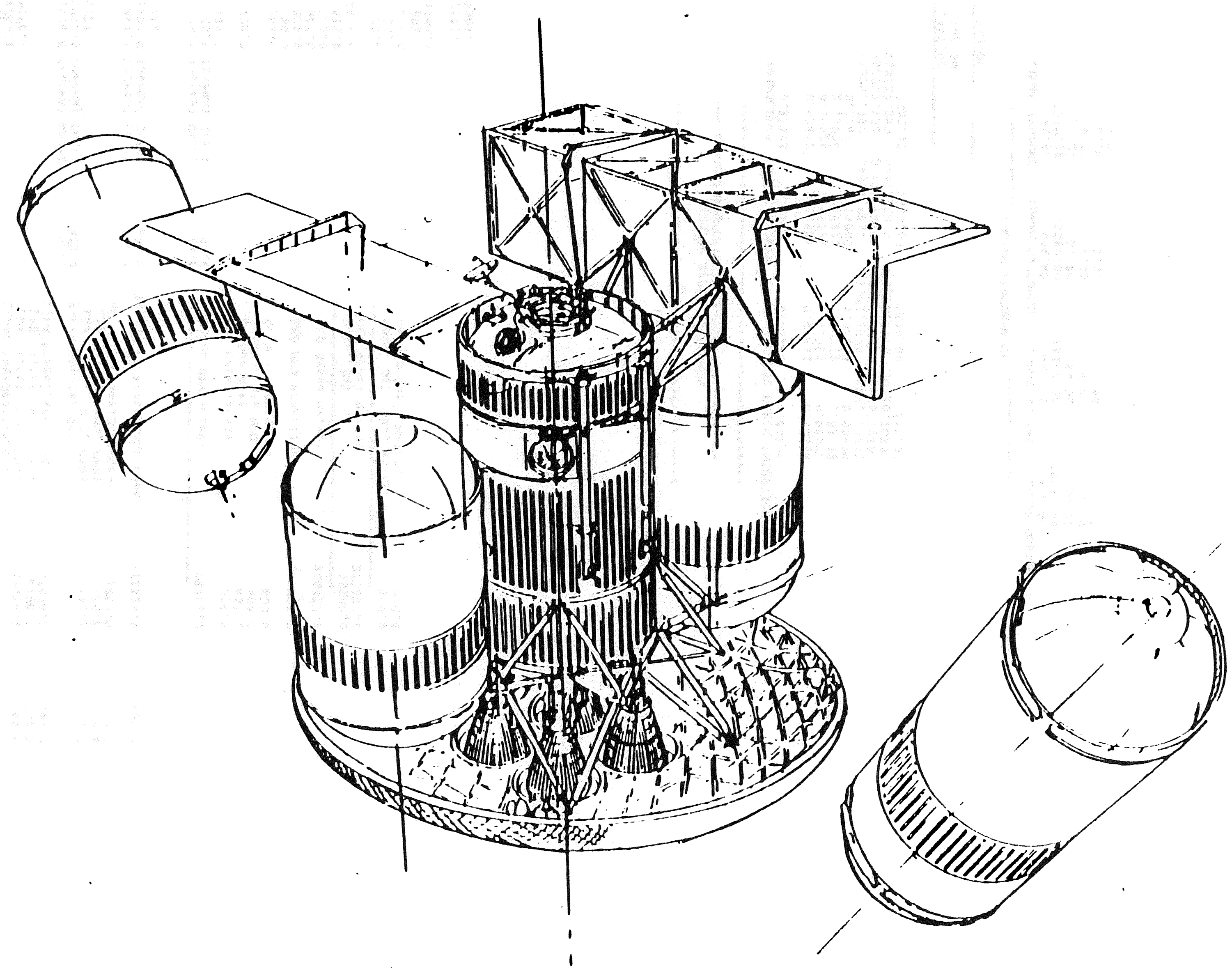
PROPULSION SYSTEMS

Chem LO2/LH2	*Isp Vac seconds*	*Tvac lbsx1000(kN)*	*Tvac/Weight*	*Mission*
RL10-A4	*449*	*20.8(92.6)*	*57*	*Lunar/Mars*
ASE•	*465/481*	*20.0(89.0)*	*48/26*	*Lunar/Mars*
P&W 200K Exp	*460*	*200.(891)*	*50*	*Mars Transfer*
NUCLEAR/LH2				
1990 NSCR••	*925*	*75.0(334)*	*3.5*	*Mars Transfer*
1990 NGCR••	*1870*	*92.0(409.4)*	*1.3*	*Mars Transfer*
NUC/ELEC/Ar		*(KG/KW)*		
Pw 2/20MW•••	*10000*	*5.4*		*Mars Transfer*
SOL/ELEC/Ar		*(KG/KW)*		
Pw 11.4MW•••	*5000*	*8.5*		*Mars Transfer*

• Advanced Space Engine
•• NSCR = Nuclear Solid Core Rocket
•• NGCR = Nuclear Gas Core Rocket
•••Primary Propulsion Power

ILLUSTRATION OF LUNAR TRANSFER VEHICLE (LTV)

(Boeing)



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                          ********************************************
                                Mass Summary of Vehicle Systems
                          ********************************************
       COMPONENT                  MASS (Kg)

 Vehicle "DRY" Weights

   Crew Module (SM=5%)            1801.1
   Micro/Solar Shield              115.3
   Support Str.(CP=5%)            1176.2
   Separation/Docking Sys          502.5
   Lunar Landing Legs             3014.7
   AeroBrake TPS & Struc.         3281.1
Total Structure Weight            9890.8

 TransLunar :
  LO2 Tanks. Struc.. OFS          1578.5 Thermal Cntrl         192.3
  LH2 Tanks. Struc.. OFS          2992.6 Thermal Cntrl         973.0
Total Of TLI  Tanks               5737.4

 Lunar Ins/Desc/Ascent :
  LO2 Tanks. Struc.. OFS           847.3 Thermal Cntrl         130.9
  LH2 Tanks. Struc.. OFS          1609.8 Thermal Cntrl         576.8
Total Of LIDA Tanks               3165.3

 Earth Orbital Insert  :
  LO2 Tanks. Struc.. OFS .          24.1 Thermal Cntrl          16.8
  LH2 Tanks. Struc.. OFS .          45.5 Thermal Cntrl          52.6
Total Of EOO  Tanks                139.2

Total RCS Tankage Weight           278.2

   Prime Elec. Power               181.4
   Sec. Elec. Power(Batt)           50.0
   Electrical Dist. Sys.           273.0
   Environ. Control                907.0
   G.N. & Control                  500.0
   COM & Data Systems              515.0
Total Sub-Systems Wt.             2426.4

   LOX/LH2 Feed  System             28.6
   Engine Gimbal System             33.0
   RCS  Propulsion Sys.            650.0
   Main Propulsion Sys.            693.0
Total Propulsion Sys. Wt.         1404.6

   Contingency/Margin             2301.9
Total Stage Dry Weight           25321.0
```

```
      COMPONENT                         MASS (Kg)       VOLUME (m**3)

Vehicle "WET" Weights

TransLunar :
 LOX  (Incl. B/O. FPR)                   96452.3            84.6
 LH2  (Incl. B/O. FPR)                   16310.7           230.4
TLI  Propellant Load                    112762.9           314.9

Lunar Ins/Desc/Ascent :
 LOX  (Incl. B/O. FPR)                   49486.3            63.4
 LH2  (Incl. B/O. FPR)                    8388.0           118.5
LIDA Propellant Load                     57874.3           161.9

Earth Orbital Ops. :
 LOX  (Incl. B/O. FPR)                    1102.3             1.0
 LH2  (Incl. B/O. FPR)                     185.5             2.6
EO  Ops Propellant Load                   1287.8             3.6

Total LEO Propellant Load               171924.9           480.6

______________________________________________
   Stage Dry Weight                      25321.0

   RCS Propellants                         295.7
   Consumables                             431.6
   Solar Flare H2O                        1800.0
   Crew + Provisions                       800.0

   LEO MPS Propellant                   171924.9

   LEO Stage Weight                     200357.2
______________________________________________
   PAYL. Del. LLO                        13000.0
   LEO Gross Weight                     213357.2
______________________________________________
   PAYL. Del. Back to LEO                  500.0
   Return Entry Weight                   16810.8
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                                ********************************************

                                          MISSION SIZING ELEMENTS
                                --- LUNAR TRANSPORTATION SYSTEM BREAKOUT ---

                                ********************************************

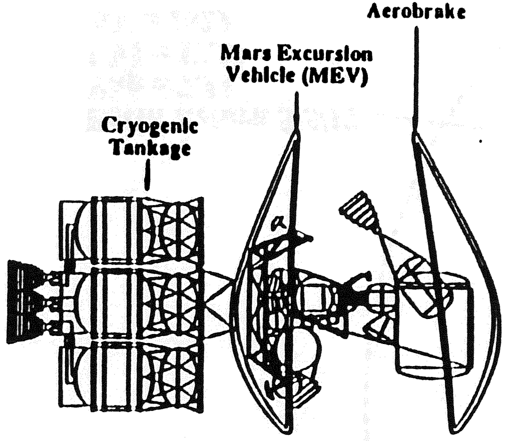
                                TRANSLUNAR  LUNAR INS/DES  LUNAR ASC/ESC EARTH INSERTION
Wprop/Wg(step)Req.(Mu):            0.53103      0.43783        0.43783       0.06866
____ Calculated _____:
Wprp/Wgr (step)   (Mu):            0.52612      0.42926        0.41662       0.06818
Wprp/Wstg(step)(Lamda):            0.56026      0.49776        0.42053       0.07172
Mass Ratio(step) W0/Wf:            2.11024      1.75212        1.70770       1.07317
Wprp(step)/WIMLEO     :            0.52612      0.19008        0.08018       0.00598
Stage Propellant  (Kg):         112251.750   40554.9531     17107.2578    1276.79712
Stage Weights     (Kg):         200357.250   81675.6875     40780.1953   18226.7070
Mission Start Wt. (Kg):         213357.250   94675.6875     41280.1953   18726.7070
Delta Velocities (m/s):            3300.00      2510.00        2510.00        310.00

____________________________________________
Delta-V (AEROBRAKE)   :      2997.00
Delta-V (RCS)         :       105.00
____________________________________________
TOTAL Delta-V s       :     11732.00

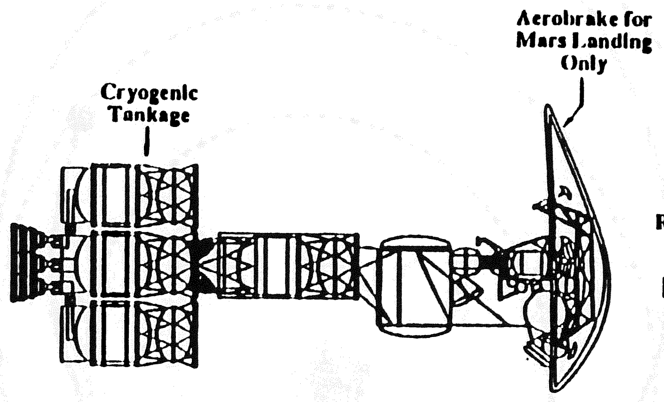
                            ___________ ENGINE PERFORMANCE ___________

                           LUNAR INBOUND   LUNAR OUTBOUND   LUNAR ASCENT   LUNAR DESCENT
VAC SPECIFIC IMPULSE (SEC):     444.40          444.40          444.40         444.40
VAC THRUST PER ENGINE     :   73392.00        73392.00        73392.00       73392.00
VAC THRUST/ENGINE MASS    :      54.00           54.00           54.00          54.00
CHAMBER MIXTURE RATIO     :       6.00            6.00            6.00           6.00
THRUST VAC/VEHICLE MASS   :       0.18            2.01            0.96           0.60
NUMBER OF ENGINES         :       5.00
```

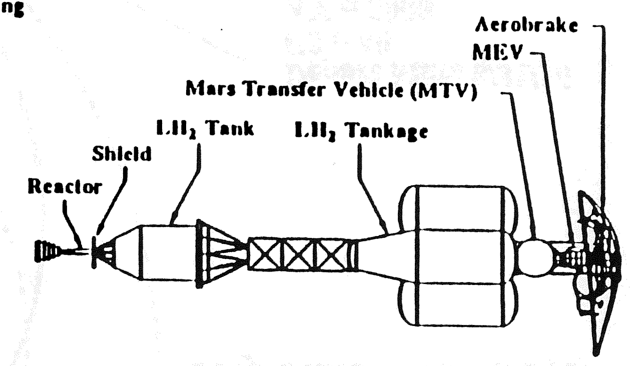
Mars Transportation Concepts



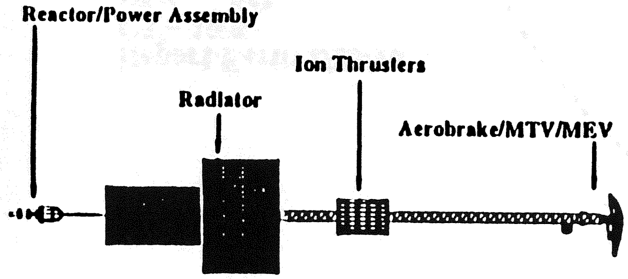
Cryogenic/Aerobrake Reference Configuration



Cryogenic/All Propulsive Configuration

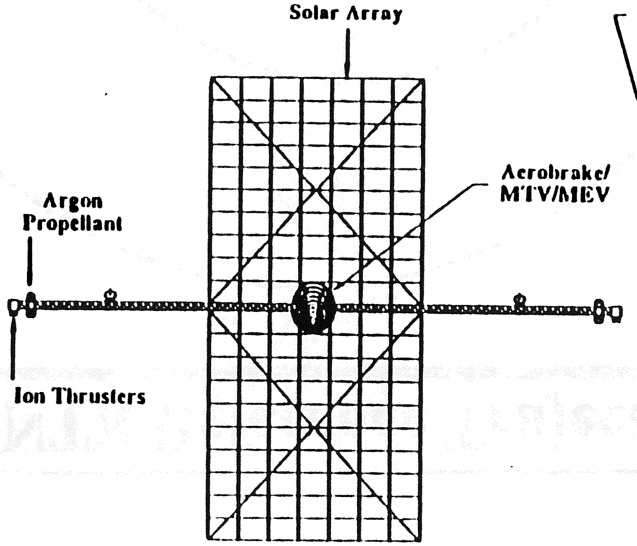


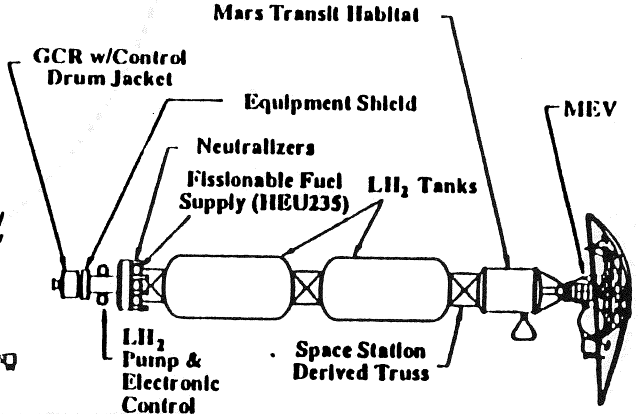
Nuclear Thermal Propulsion Configuration



Nuclear Electric Propulsion Configuration

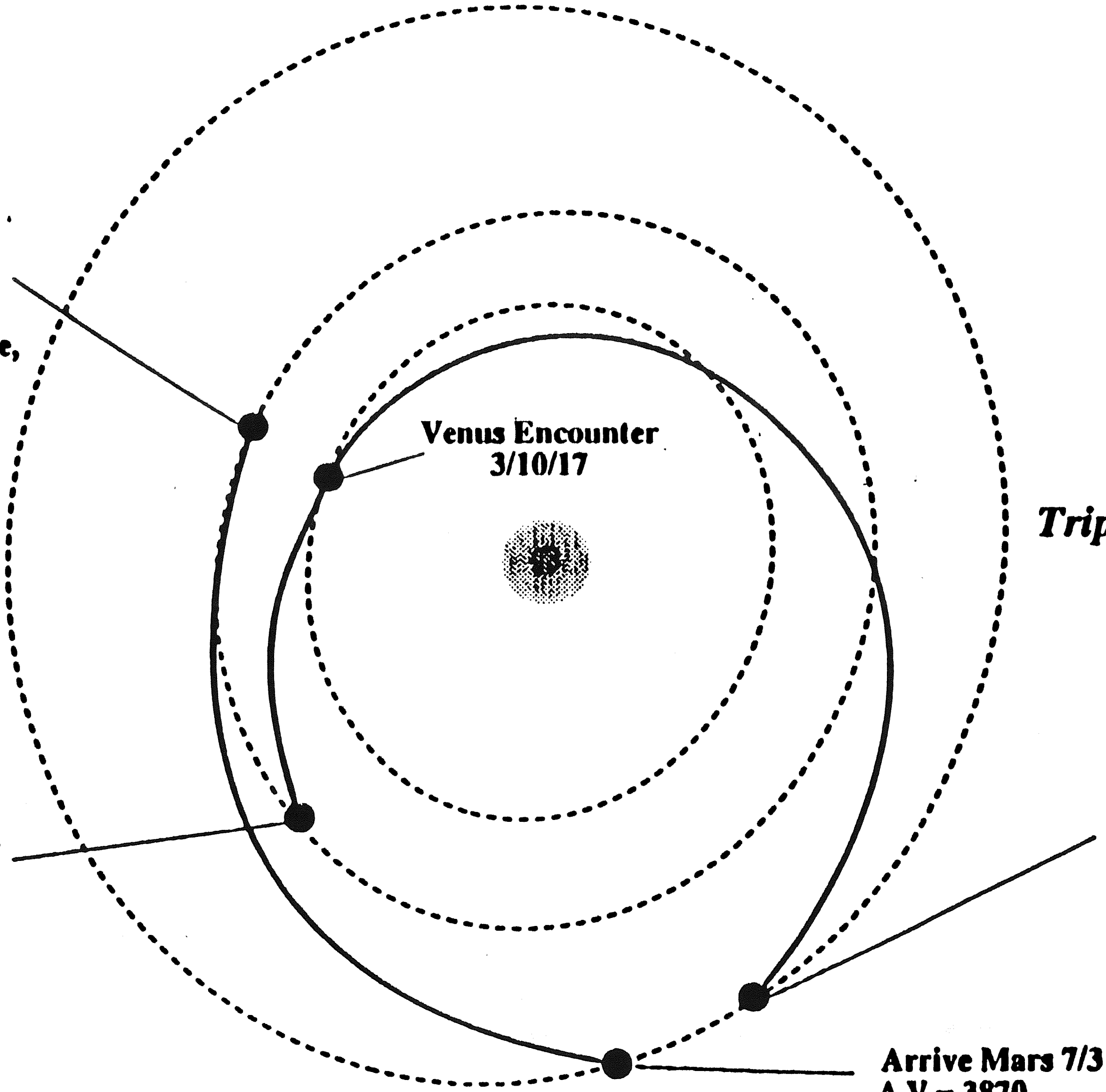
Solar Electric Propulsion Configuration





Gas Core Nuclear Reactor Configuration

2016 NTR Reference Trajectory



Depart Earth 2/25/16
C3 = 10.3
DLA = -35°
ΔV = 4182
including plane change,
window, and g loss

Venus Encounter
3/10/17

Trip Time = 434 Days

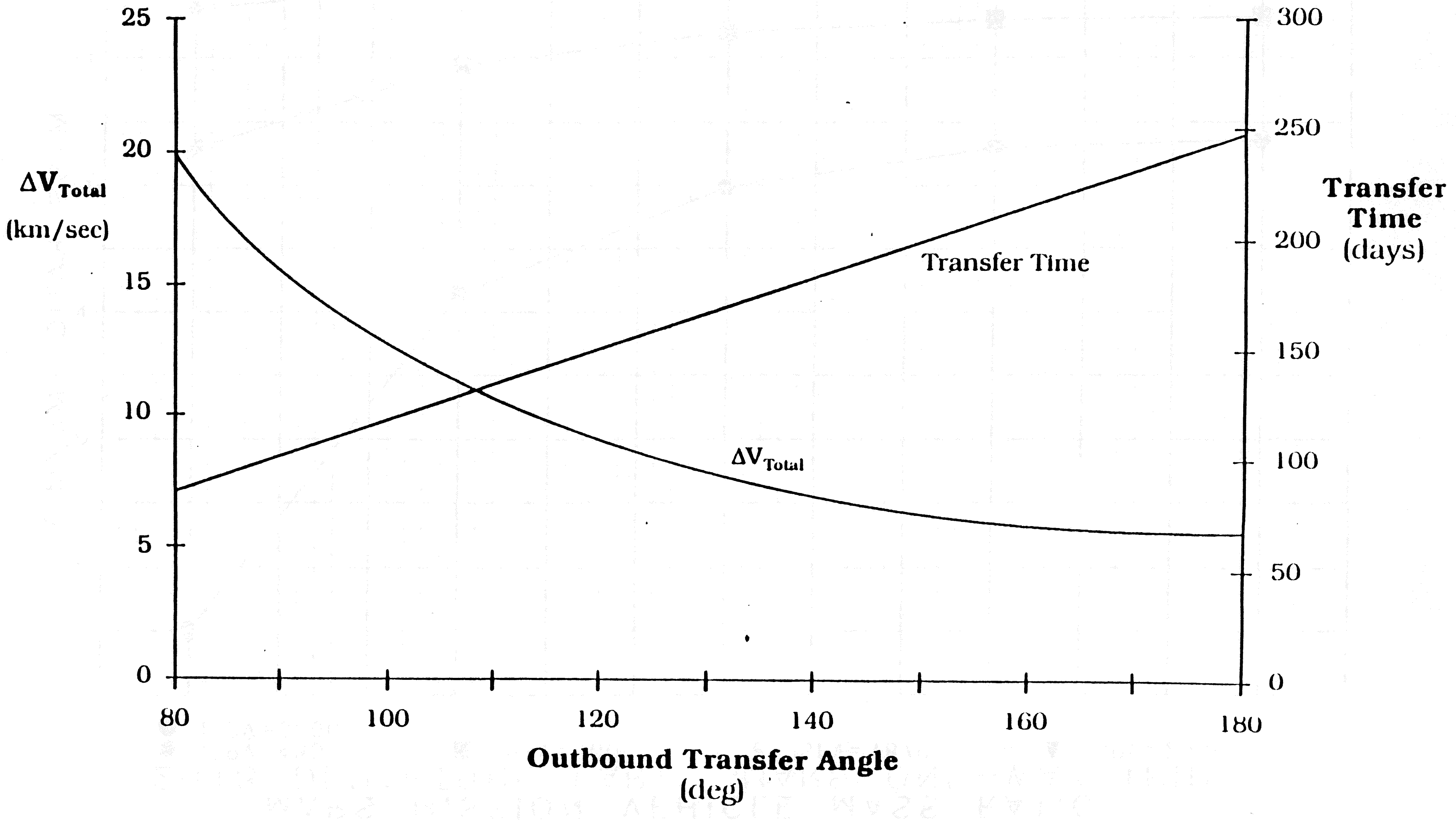
Earth Return 5/5/17
Vhp = 7.14
LVI = 17.2°
ΔV = 2629

Depart Mars 8/31/16
C3 = 40
ΔV = 3900
including plane change
apsidal misalignment
penalties

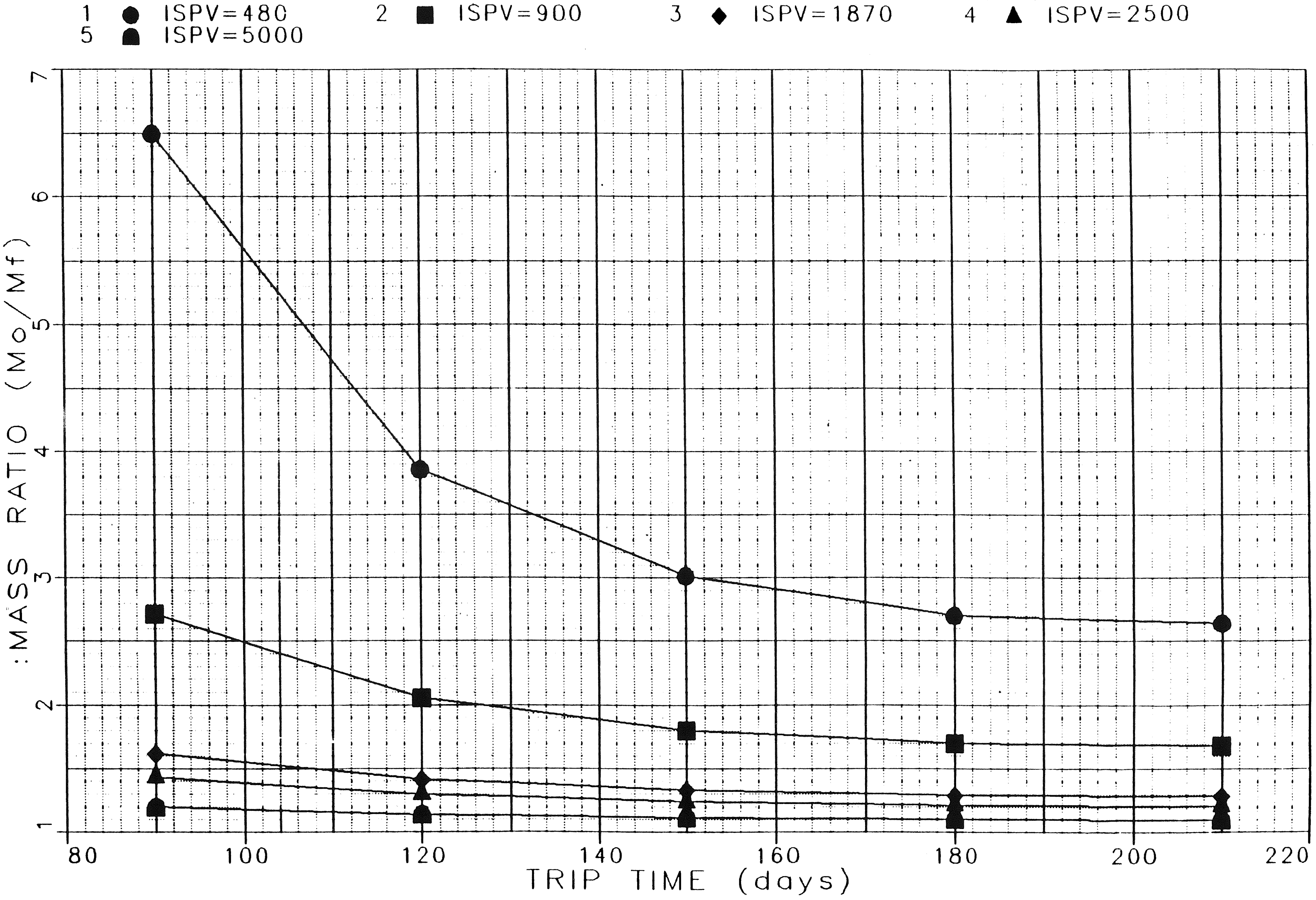
Arrive Mars 7/31/16
Δ V = 3870

BOEING

Outbound Transfer

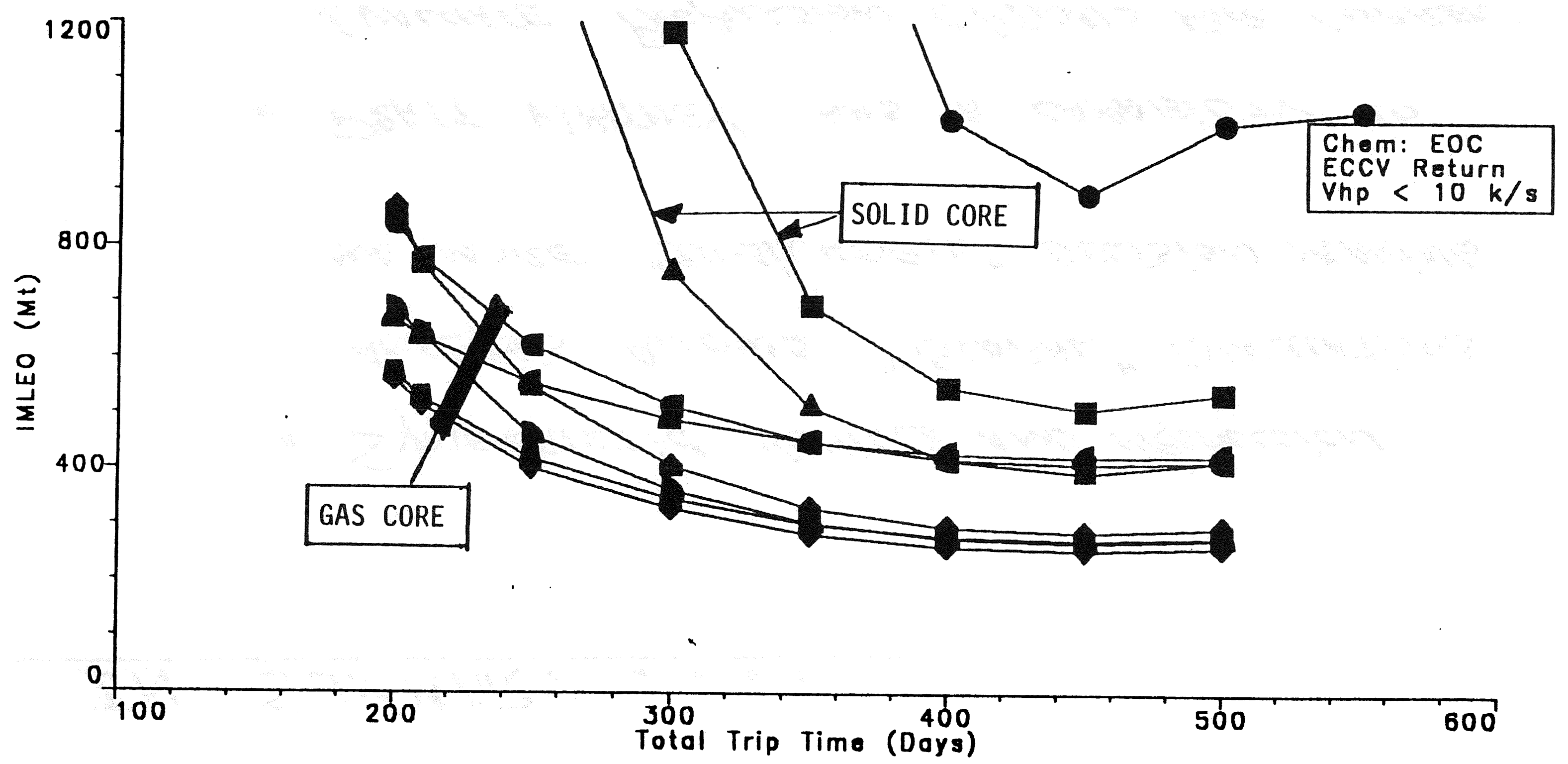


MARS MISSION VEHICLE MASS RATIO
2018 DEPARTURE EARTH-MARS ONE-WAY TRIP



02/06/91
J. E. EMPERT

PRATT WHITNEY VEHICLE ANALYSIS
MARS Vehicle/Propulsion Analysis
IMLEO TRENDS : 200–500 Day Round Trip (30 Stay)
14% (Wtk/Wprop), Drop Tanks: TMI, MOC, TEI



IN SUMMARY . . . .

- SYNERGISTIC VEHICLE AND PROPULSION ANALYSIS ALLOWS "GLOBAL" EVALUATIONS AND MORE "IN-SIGHTFULL" DECISION MAKING
- PRATT WHITNEY HAS A CAPABILITY TO EVALUATE PROPULSION OPTIONS FOR CURRENT AS WELL AS "ANTICIPATED" BUSINESS

— Ad ASTRA

MATERIALS AND FABRICATION TECHNOLOGY

DAN BALES
MAY 1987

MATERIALS AND FABRICATION TECHNOLOGY

PARTICIPANTS WILL ....

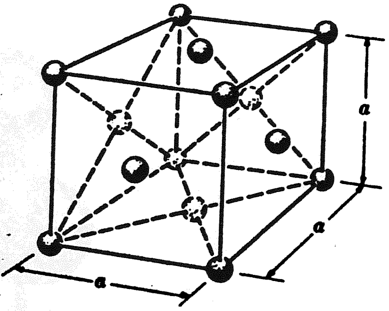
- UNDERSTAND THE EFFECTS OF HIGH PRESSURE ROCKET PROPELLANTS ON COMMONLY USED MATERIALS
- BECOME ACQUAINTED WITH ALLOYS, MATERIAL PROCESSES AND FABRICATION TECHNIQUES USED IN LIQUID PROPELLANT ROCKET ENGINE ENVIRONMENTS

HYDROGEN EMBRITTLEMENT

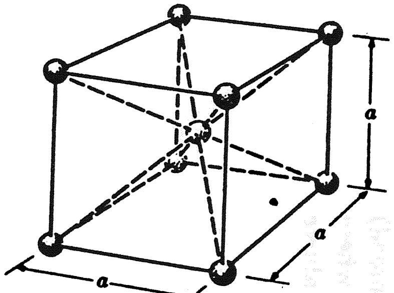
A CONDITION OF LOW DUCTILITY IN METALS RESULTING FROM THE ABSORPTION OF HYDROGEN

3 FORMS OF HYDROGEN DAMAGE

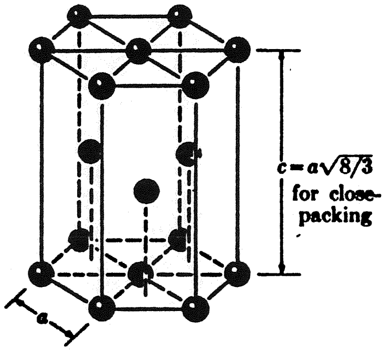
- INTERNAL VOIDS, CRACKS OR OTHER DEFECTS ARISE FROM THE DIFFUSION OF HYDROGEN THROUGH THE METAL LATTICE TO CAUSE THE DEFECTS
- REDUCED MECHANICAL PROPERTIES AND TOUGHNESS RESULT FROM HYDRIDES WHICH ARE FORMED WHEN HYDROGEN ASSUMES SPECIFIC LATTICE POSITIONS
- CRACK FORMATION AND GROWTH IS AIDED BY THE PRESENCE OF CONCENTRATED HYDROGEN DISSOLVED IN THE LATTICE JUST AHEAD OF THE CRACK TIP



Face-centered cubic
(f.c.c.)

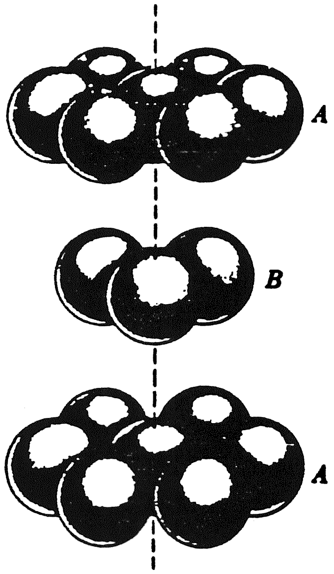


Body-centered cubic
(b.c.c.)

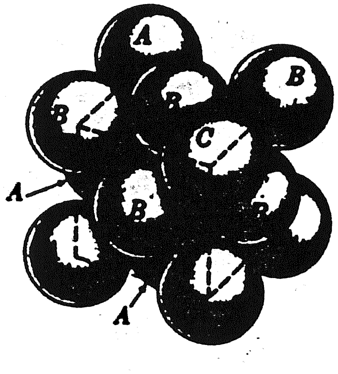


Hexagonal close-packed
(h.c.p.)

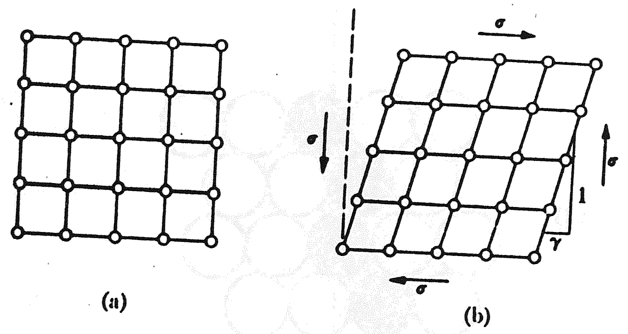
SOME OF THE MOST COMMON UNIT CELLS OF CRYSTALS



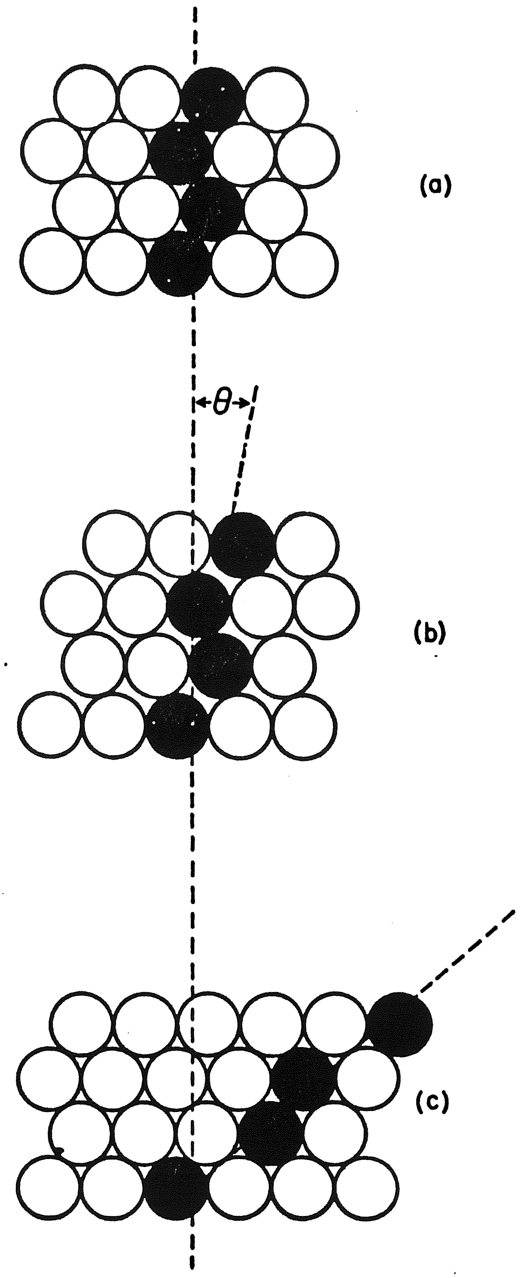
Vertical stacking (expanded) of close-packed planes of spheres, where the layers marked *A* are directly above each other.



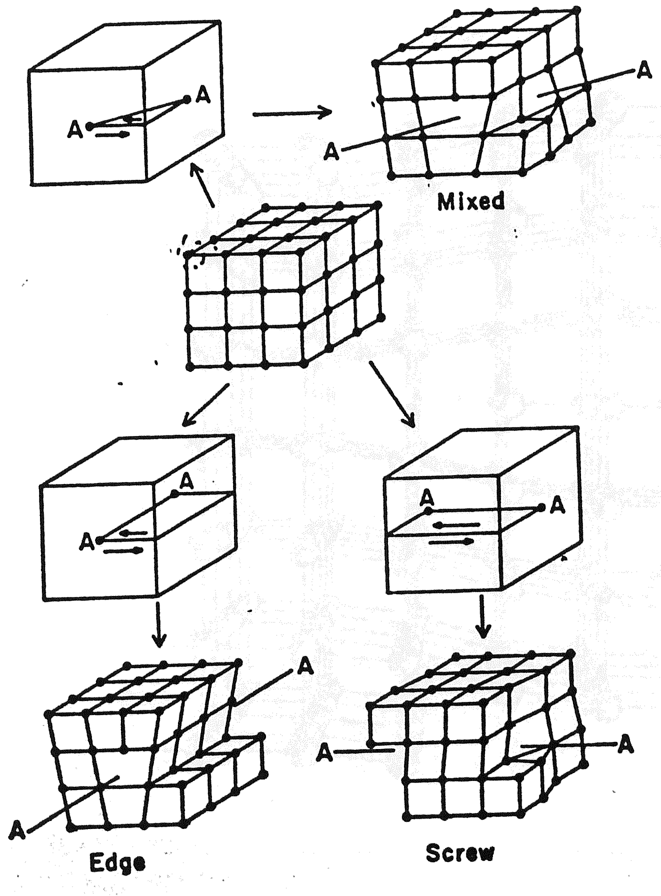
Face-centered cubic structure, showing the *ABCABCA* sequence of close-packed (111) planes.



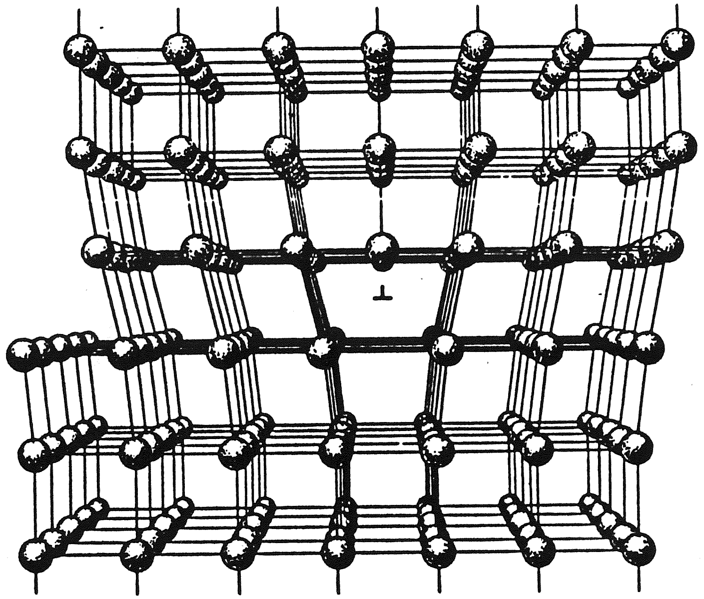
(a) An unstrained simple cubic lattice. (b) A simple cubic lattice strained elastically.



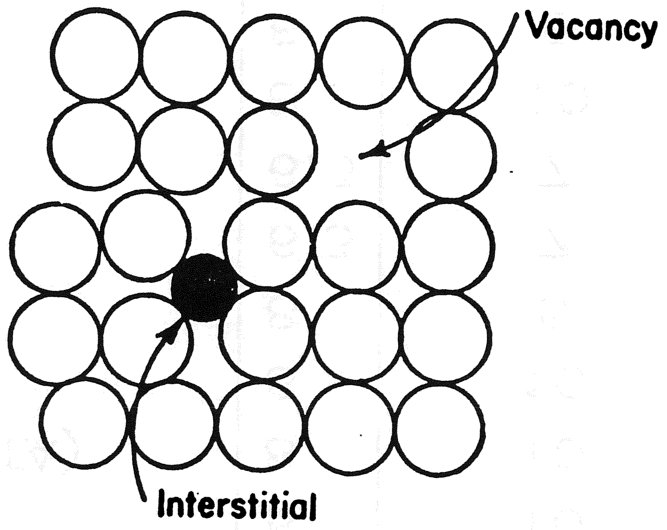
SHEAR DISPLACEMENT IN A PERFECT LATTICE



The creation of an edge, a screw, and a mixed dislocation.



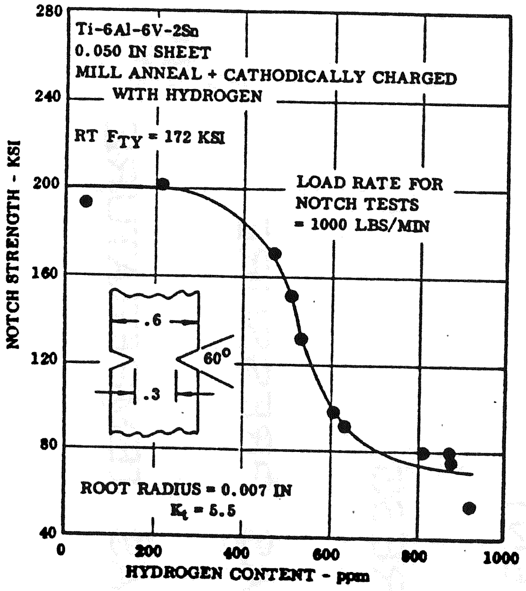
AN EDGE DISLOCATION



VACANCY AND INTERSTITIAL ATOM

ALLOY	K_t	PRESSURE KSI	RATIO H_2/He
17–7 PH (TH 1050)	8	10	.23
RENE 41	8	10	.27
INCONEL 718	8	10	.46
Ti–6Al–4V (STA)	8	10	.58
NICKEL 270	8	10	.70
INCONEL 625	8	5	.76
WASPALOY	6.3	7	.78
INCONEL 706	6.3	7	.80
304	8	10	.87
HASTELLOY X	8	5	.87
ASTROLOY	8	5	.90
HAYNES 188	6.3	7	.92
A–286	8	10	.97
NITRONIC 40	6.3	7	.97
316	8	10	1.00
INCOLOY 903	8	5	1.00

Rocketdyne data presented under contract NAS8-25579



NOTCH TENSILE STRENGTH OF ANNEALED SHEET AT ROOM TEMPERATURE AS INFLUENCED BY HYDROGEN CONTENT.

FACTORS WHICH AFFECT THE DEGREE OF HYDROGEN EMBRITTLEMENT

- MATERIAL COMPOSITION
- HEAT TREATMENT
- RESIDUAL STRESS
- HYDROGEN GAS PRESSURE
- HYDROGEN GAS TEMPERATURE
- GRAIN SIZE
- SURFACE CONDITION (MILLED, EDM, ECM....)
- MATERIAL FORM (PLATE, FORGING, CASTING....)
- SURFACE ROUGHNESS

ROOM TEMPERATURE TENSILE PROPERTIES OF INCONEL 718 SPECIMENS FABRICATED FROM 1-1/2 IN. FORGING; SUPPLIED BY CARLTON FORGE, MILL SUPPLIER: SPECIAL METALS

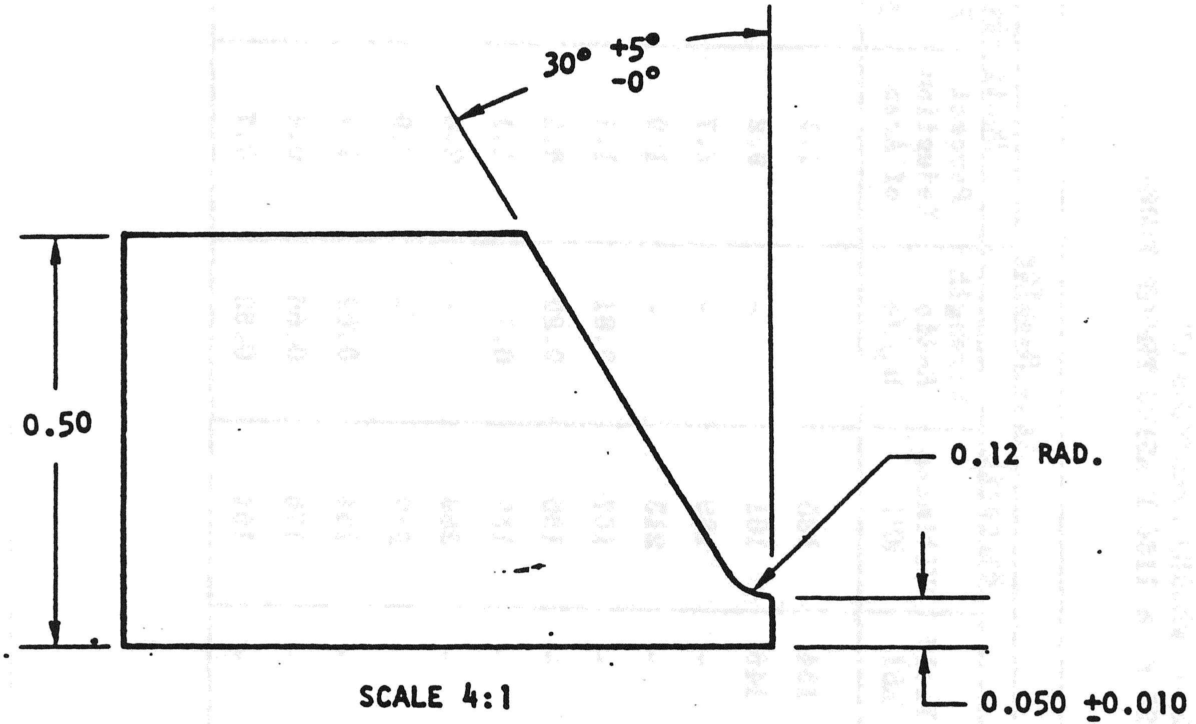
Heat Treatment			Specimen			Environment		Test Results				
								Strength			Ductility	
Solution Temp. F	Aging Temps		No.	Type	Stress Conc. Factor	Type	Pressure psig	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area	Percent Elongation
	First	Second										
1725	1325	1150	IG-1	UN	-	Air	0	154	199	-	29	22
			IG-2	UN	-	Air	0	164	197	-	32	21
			IG-6	N	8.7	Helium	5000	-	293	-	2.4	-
			IG-7	N	8.5	Helium	5000	-	286	-	3.5	-
			IG-3	N	8.3	Hydrogen	5000	-	180	0.62	1.3	-
			IG-4	N	8.9	Hydrogen	5000	-	159	0.55	0.9	-
			IG-5	N	8.3	Hydrogen	5000	-	172	0.59	1.1	-
1925	1400	1200	IH-1	UN	-	Air	0	170	200	-	42	26
			IH-2	UN	-	Air	0	168	196	-	39	26
			IH-6	N	8.0	Helium	5000	-	340	-	3.0	-
			IH-7	N	8.3	Helium	5000	-	338	-	6.2	-
			IH-3	N	8.9	Hydrogen	5000	-	250	0.74	1.3	-
			IH-4	N	8.7	Hydrogen	5000	-	259	0.76	2.2	-
			IH-5	N	8.3	Hydrogen	5000	-	265	0.78	2.0	-

Rocketdyne data presented under contract NAS8-25579

ROOM TEMPERATURE TENSILE PROPERTIES OF INCONEL 718 SPECIMENS FABRICATED FROM 1-1/4 IN. x 2-3/4 IN. ROLLED BAR SUPPLIED BY ALLVAK

Heat Treatment			Specimen			Environment		Test Results				
								Strength			Ductility	
Solution Temp. F	Aging Temps											
	First	Second	No.	Type	Stress Conc. Factor	Type	Pressure psig	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area	Percent Elongation
1725	1325	1150	IA-1	UN	-	Air	0	161	201	-	34	23
			IA-2	UN	-	Air	0	165	203	-	36	22
			IA-6	N	8.7	Helium	5000	-	284	-	2.8	-
			IA-7	N	8.7	Helium	5000	-	281	-	2.9	-
			IA-3	N	8.9	Hydrogen	5000	-	150	0.53	0.8	-
			IA-4	N	8.9	Hydrogen	5000	-	113	0.40	0.5	-
			IA-5	N	8.5	Hydrogen	5000	-	192	0.68	1.5	-
1925	1400	1200	IB-1	UN	-	Air	0	160	195	-	36	25
			IB-2	UN	-	Air	0	161	195	-	37	26
			IB-6	N	8.5	Helium	5000	-	322	-	4.6	-
			IB-7	N	8.9	Helium	5000	-	321	-	5.3	-
			IB-3	N	8.2	Hydrogen	5000	-	228	0.71	1.9	-
			IB-4	N	8.7	Hydrogen	5000	-	228	0.71	2.1	-
			IB-5	N	8.7	Hydrogen	5000	-	234	0.73	1.1	-

Rocketdyne data presented under contract NAS8-25579



Weld Design for Gas Tungsten Arc Welding of Inconel 718
With Inconel 718 Filler Metal

ROOM TEMPERATURE TENSILE PROPERTIES OF WELDED SPECIMENS OF INCONEL 718 1/2 IN. PLATE, 1725 F SOLUTION, 1325 F & 1150 F AGING TEMPERATURES

Specimen			Environment		Stress Conc. Factor	Test Results				
						Strength			Ductility	
No.	Type		Type	Pressure psig		Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Percent Reduction of Area	Percent Elon-gation
IJW-11	UN	Weld	Air	0	-	154	185	-	1.5	6.3
IJW-12	UN	Weld	Air	0	-	146	161	-	9.2	3.4
IJW-1	N	Weld	Helium	5000	8.7	-	198	-	1.7	-
IJW-2	N	Weld	Helium	5000	8.7	-	213	-	1.0	-
IJW-3	N	Weld	Hydrogen	5000	8.7	-	167	0.81	1.1	-
IJW-4	N	Weld	Hydrogen	5000	8.9	-	165	0.80	0.8	-
IJW-5	N	Weld	Hydrogen	5000	8.9	-	159	0.76	1.1	-
IJW-6	N	HAZ	Helium	5000	8.3	-	289	-	2.6	-
IJW-10	N	HAZ	Helium	5000	8.1	-	242	-	0.9	-
IJW-7	N	HAZ	Hydrogen	5000	8.5	-	164	0.62	1.1	-
IJW-8	N	HAZ	Hydrogen	5000	8.5	-	176	0.66	0.4	-
IJW-9	N	HAZ	Hydrogen	5000	8.3	-	164	0.62	0.7	-

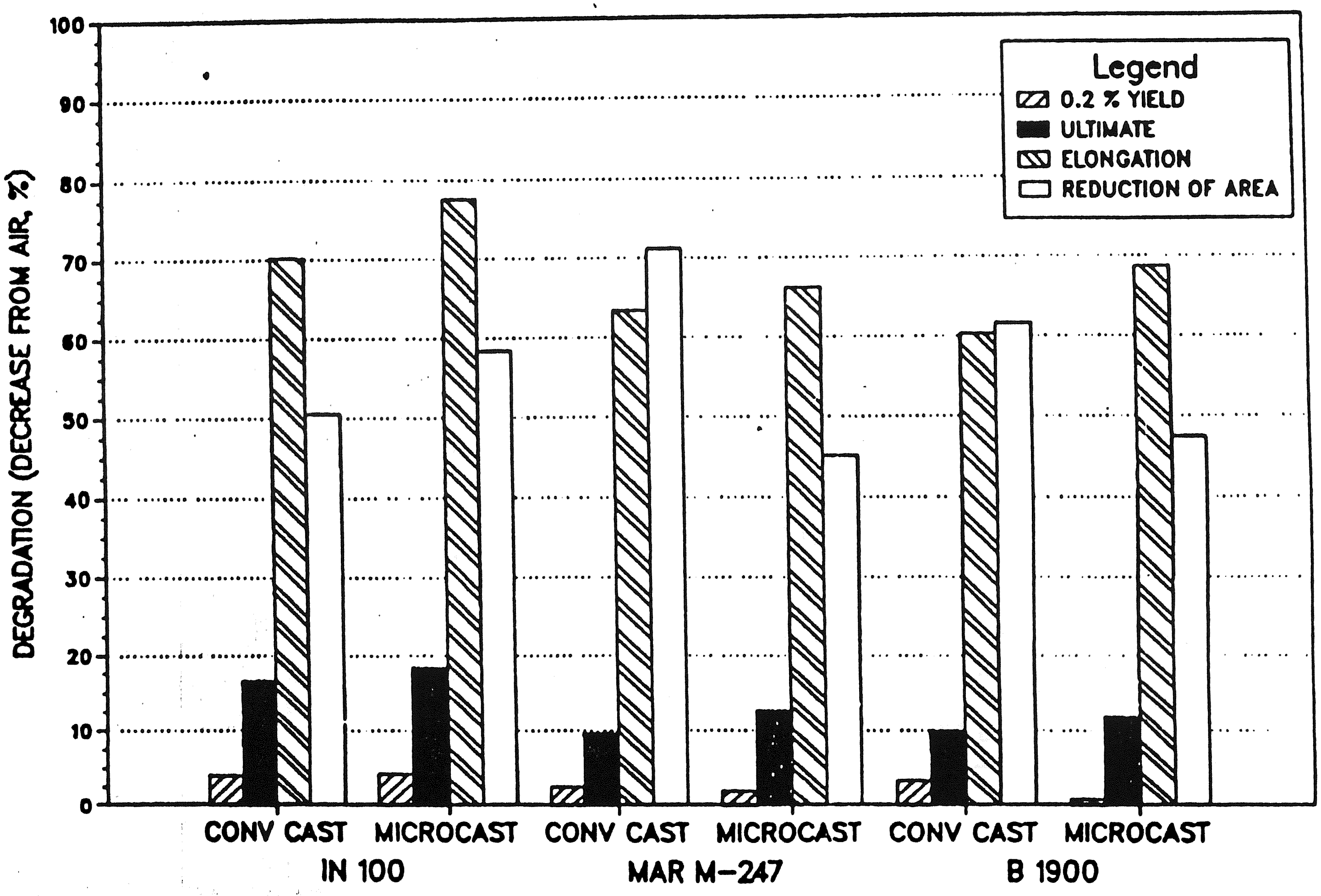
Rocketdyne data presented under contract NAS8-25579

ROOM TEMPERATURE TENSILE PROPERTIES OF WELDED SPECIMENS OF INCONEL 718 1/2 IN. PLATE, 1925 F SOLUTION, 1400 F & 1200 F AGING TEMPERATURES

Specimen			Environment		Stress Conc. Factor	Test Results				
						Strength			Ductility	
No.	Type		Type	Pressure psig		Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Percent Reduction of Area	Percent Elon-gation
IKW-11	UN	Weld	Air	0	-	166	199	-	19	11
IKW-12	UN	Weld	Air	0	-	164	198	-	26	15
IKW-1	N	Weld	Helium	5000	8.7	-	262	-	2.6	-
IKW-4	N	Weld	Helium	5000	8.7	-	274	-	2.5	-
IKW-2	N	Weld	Hydrogen	5000	8.7	-	148	0.55	0.8	-
IKW-3	N	Weld	Hydrogen	5000	8.7	-	175	0.65	0.9	-
IKW-5	N	Weld	Hydrogen	5000	8.9	-	131	0.49	0.7	-
IKW-6	N	HAZ	Helium	5000	8.1	-	310	-	5.0	-
IKW-7	N	HAZ	Helium	5000	8.9	-	292	-	2.6	-
IKW-8	N	HAZ	Hydrogen	5000	8.7	-	237	0.79	0.4	-
IKW-9	N	HAZ	Hydrogen	5000	8.7	-	181	0.60	0.8	-
IKW-10	N	HAZ	Hydrogen	5000	8.7	-	232	0.77	2.1	-

Rocketdyne data presented under contract NAS8-25579

DEGRADATION IN SMOOTH TENSILE PROPERTIES OF VANE ALLOY
CANDIDATES DUE TO 5000-psig HYDROGEN AT AMBIENT TEMPERATURE



TESTS USED TO DETERMINE HYDROGEN EMBRITTLEMENT

- TENSILE TEST
- ERICKSON CUP TEST
- ALTERNATING STRESS TEST
- BEND TEST
- IMPACT TEST
- LOW CYCLE FATIGUE TEST
- FRACTURE TOUGHNESS TEST

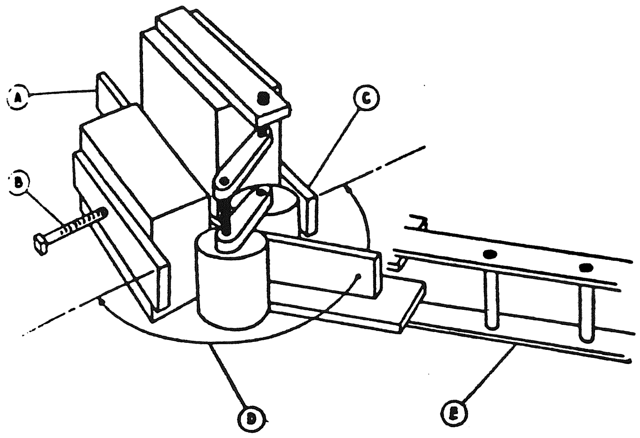


Figure 1. Pictorial diagram of bend tester

A. Specimen held in tester
B. Set screw for holding specimen in place
C. Metal stop positioned for 90° sweep of specimen
D. Arc showing total 180° sweep of specimen
E. Arm used to bend tester and specimen

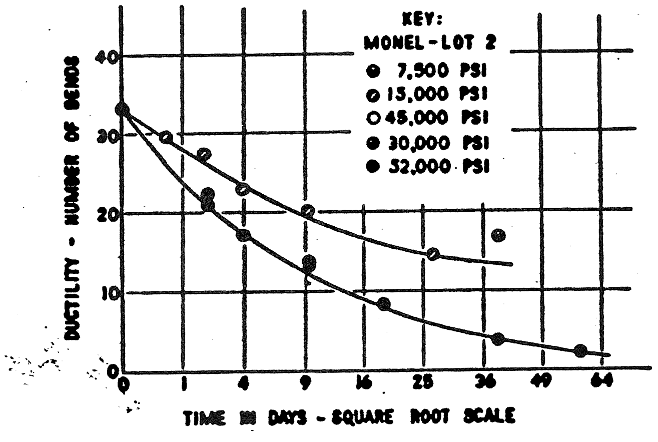
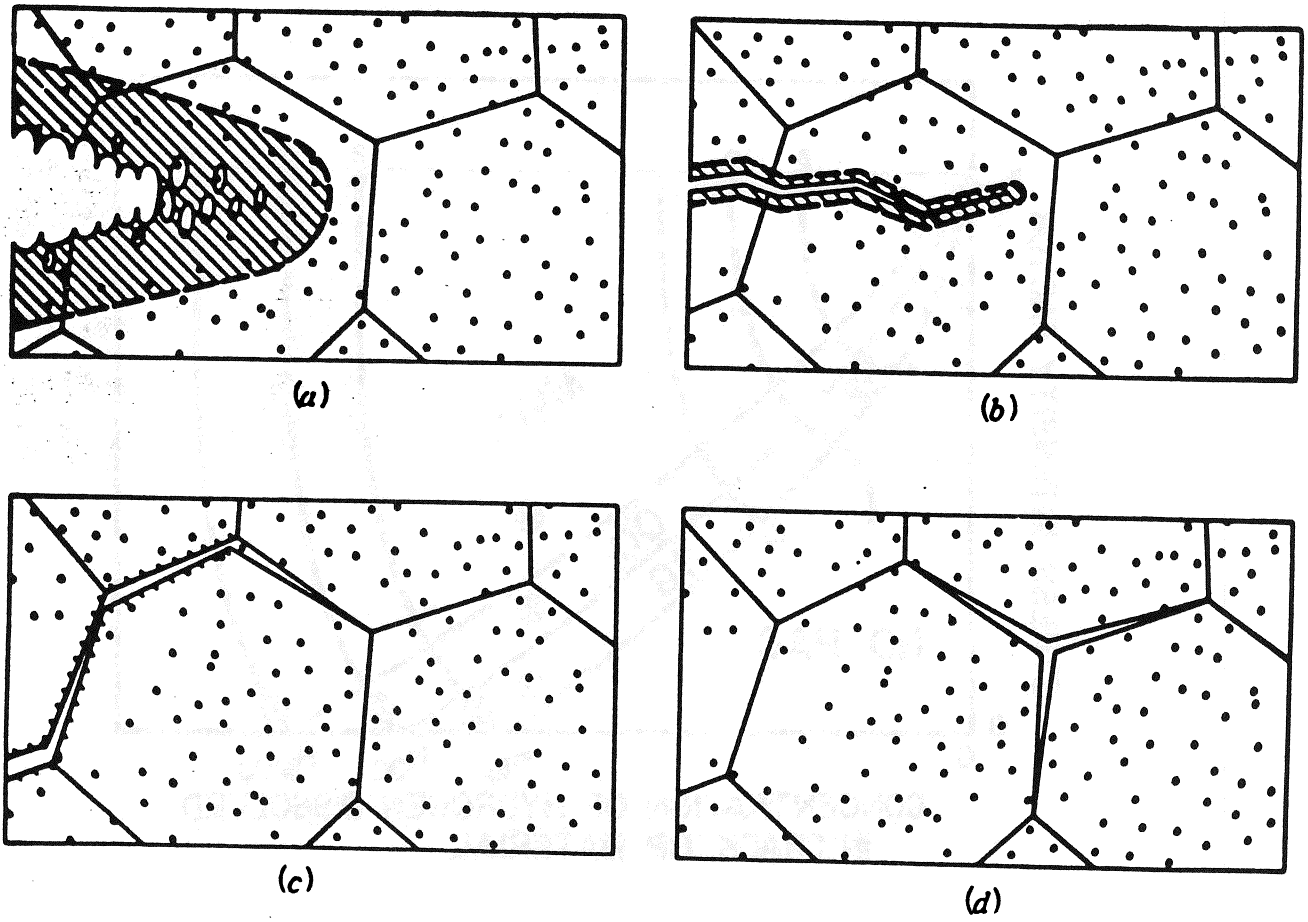


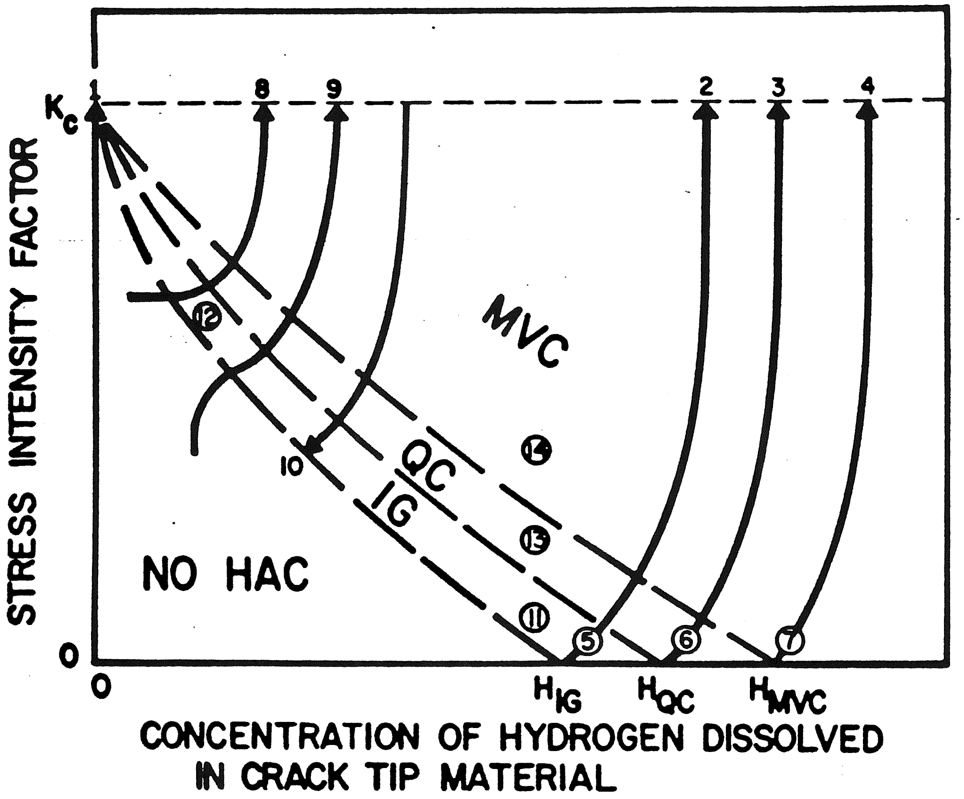
Figure 2. Ductility of Monel exposed to hydrogen gas at room temperature

Perlmutter & Dodge, Industrial & Engineering Chemistry
May 1956, p.885.



Sketches of microscopic fracture modes observed in these experiments as a function of decreasing stress intensity factor and concomitant decreasing cracking rate. (*a*) High *K*, MVC; (*b*) intermediate *K*, QC; (*c*) low *K*, IG; (*d*) IG cracking with an assist from hydrogen pressure.

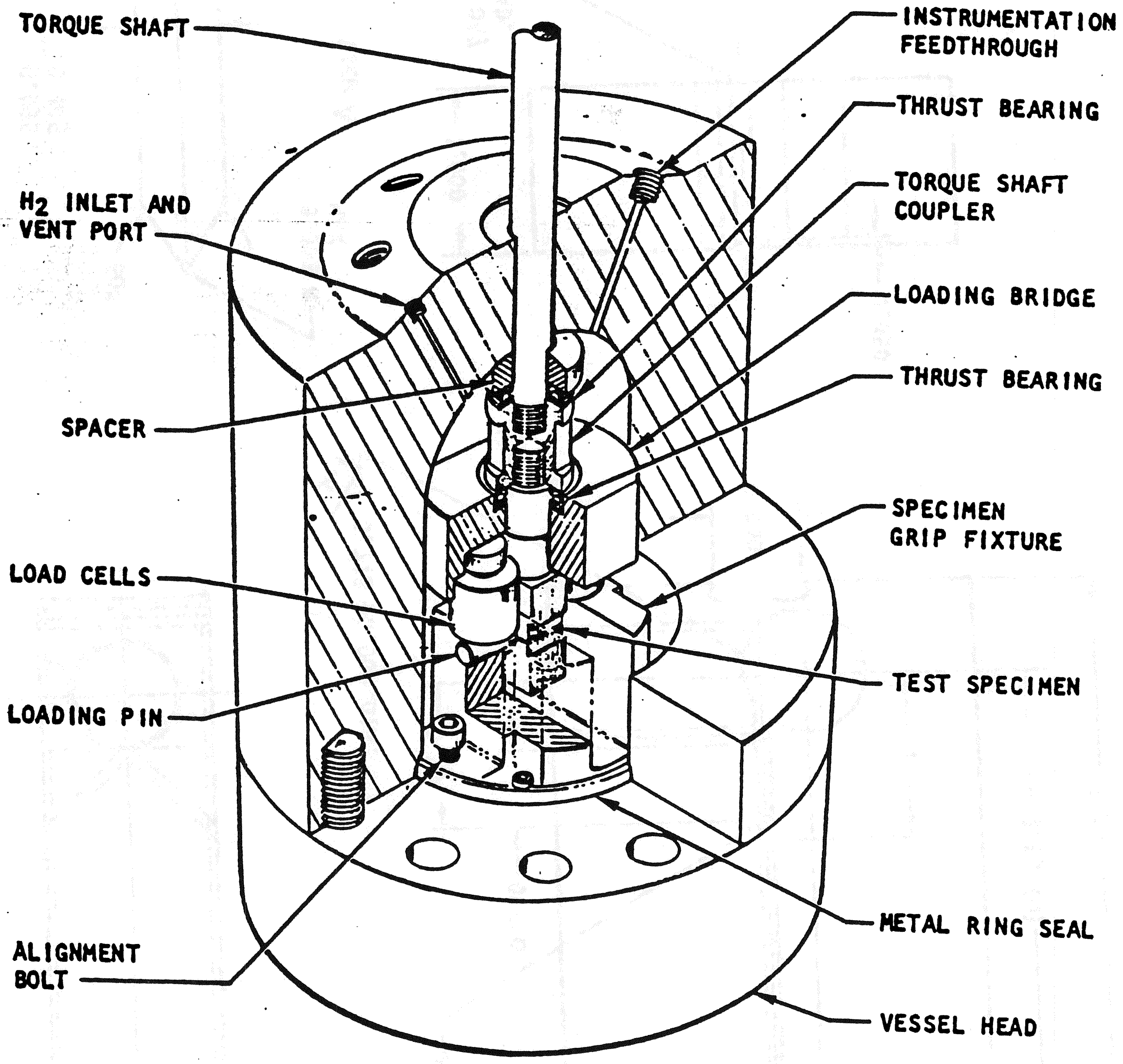
Metallurgical Transactions, Vol. 3, Feb. 1972, The Metallurgical Society of AIME ASM



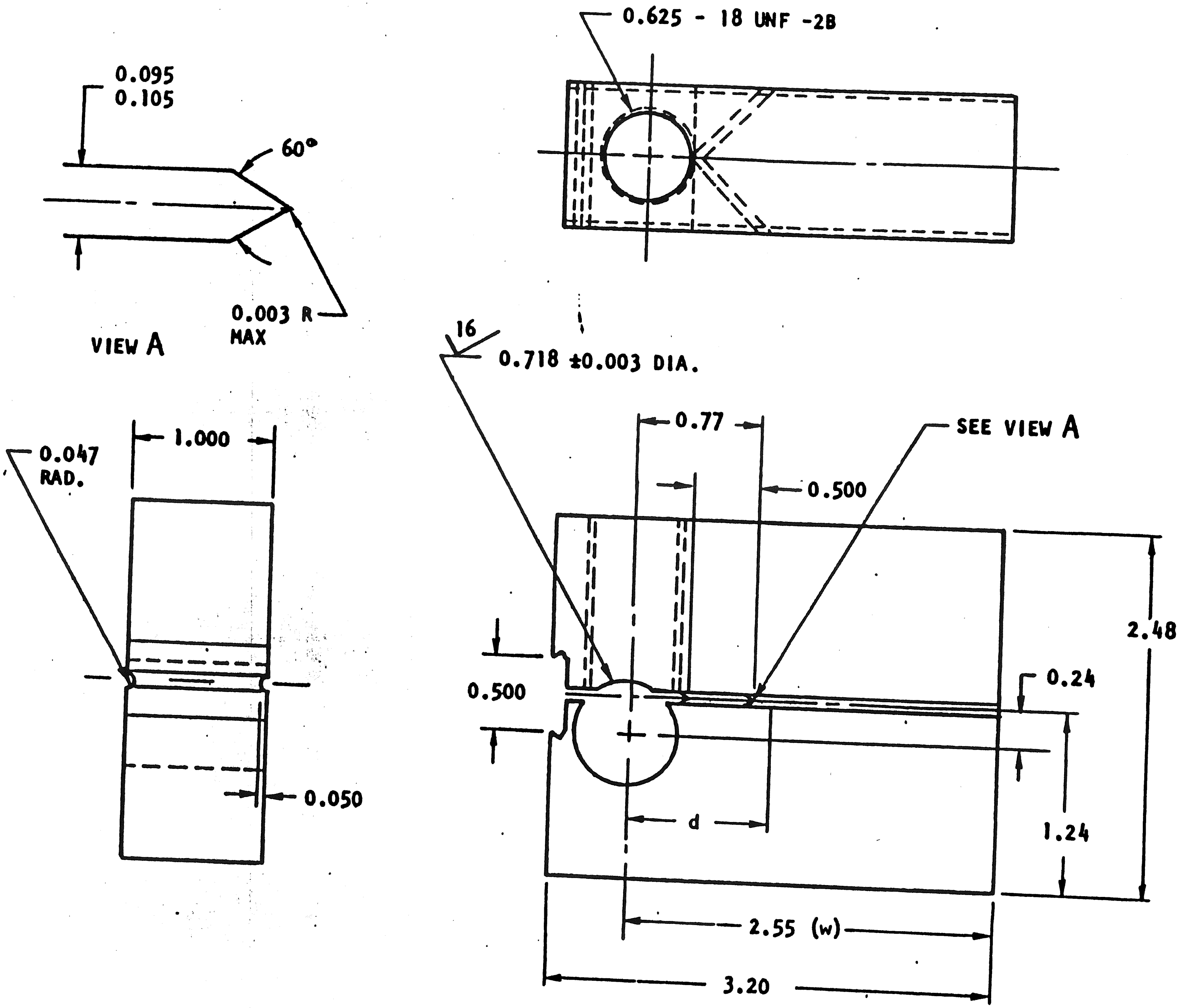
Suggested interrelationship between stress intensity factor, dissolved hydrogen content, and HAC deformation mode in microscopically small volumes of crack-tip material.

Metallurgical Transactions, Vol. 3, Feb. 1972. The Metallurgical Society of AIME ASM

Rocketdyne data presented under contract NAS8-25



Apparatus for Performing Threshold Stress Intensity Measurements


0.095
0.105
60°
0.003 R
MAX
VIEW A
0.625 - 18 UNF -2B
16
0.718 ±0.003 DIA.
0.047
RAD.
1.000
0.050
0.77
0.500
SEE VIEW A
0.500
0.24
2.48
d
1.24
2.55 (w)
3.20

AVERAGE K_{IC} AND K_{TH} VALUES FOR VARIOUS METALS IN 34.5 MN/m^2 (5000 PSI) HYDROGEN AND HELIUM ENVIRONMENTS

Material	Environment	295 K (70 F)				144 K (-200 F)				K_{TH}/K_{IC}	
		K_{IC}		K_{TH}		K_{IC}		K_{TH}		295 K (70 F)	144 K (-200 F)
		$MN/m^2 \sqrt{m}$	KSI $\sqrt{in.}$	$MN/m^2 \sqrt{m}$	KSI $\sqrt{in.}$	$MN/m^2 \sqrt{m}$	KSI $\sqrt{in.}$	$MN/m^2 \sqrt{m}$	KSI $\sqrt{in.}$		
Inconel 718											
1214, 991 to 894 K	He	78	71	58	53	98	89	81	74	0.73	0.83
(1725, 1325 to 1150 F)	H_2	--	--	~14	~13	--	--	63	57	0.18	0.65
Inconel 718											
1325, 1033 to 922 K	He	119*	108*	112	102	122	111	139	126	0.94	1.13
(1925, 1400 to 1200 F)	H_2	110	100	38	35	121	110	123	112	0.33	1.01
Inconel 625	He	84*	76*	--	--	110*	100*	--	--	--	--
	H_2	67*	61*	--	--	108*	98*	--	--	--	--
AISI 321 Stainless Steel	He	36*	33*	--	--	34	31	38**	35**	--	1.13
	H_2	34*	31*	--	--	32	29	36	33	--	1.07
A-286 Stainless Steel	He	145*	132*	--	--	138	126*	--	--	--	--
	H_2	100*	91*	<113**	<103**	152	138*	--	--	<0.78	--
Ti-5Al-2.5 Sn ELI	He	79	72	69	63	87	79	60	55	0.88	0.70
	H_2	--	--	34	31	--	--	58	53	0.43	0.67
2219 T87 Aluminum	He	33	30	32	29	41	37	40	36	0.97	0.97
	H_2	29	26	33	30	37	34	40	36	1.00	0.97
OFHC Copper	He	20*	18*	--	--	18*	16*	--	--	--	--
	H_2	17*	16*	--	--	19*	17*	--	--	--	--

*Not valid K_{IC} according to ASTM 399

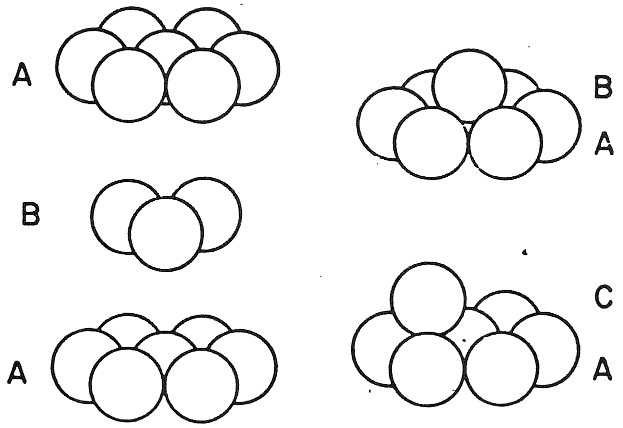
**Stress at $K_{TH} > \sigma_y$

Rocketdyne data presented under contract NAS8-25579

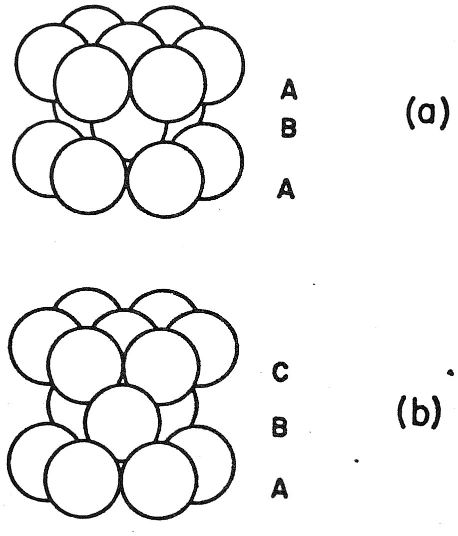
AREAS OF CONCERN ON ROCKETDYNE SSME

- THRUST CHAMBER
- TURBINE VANES
- TURBINE BLADES
- SEALS
- BEARINGS
- PLATINGS/COATINGS
- WELD INSPECTABILITY

APPENDIX

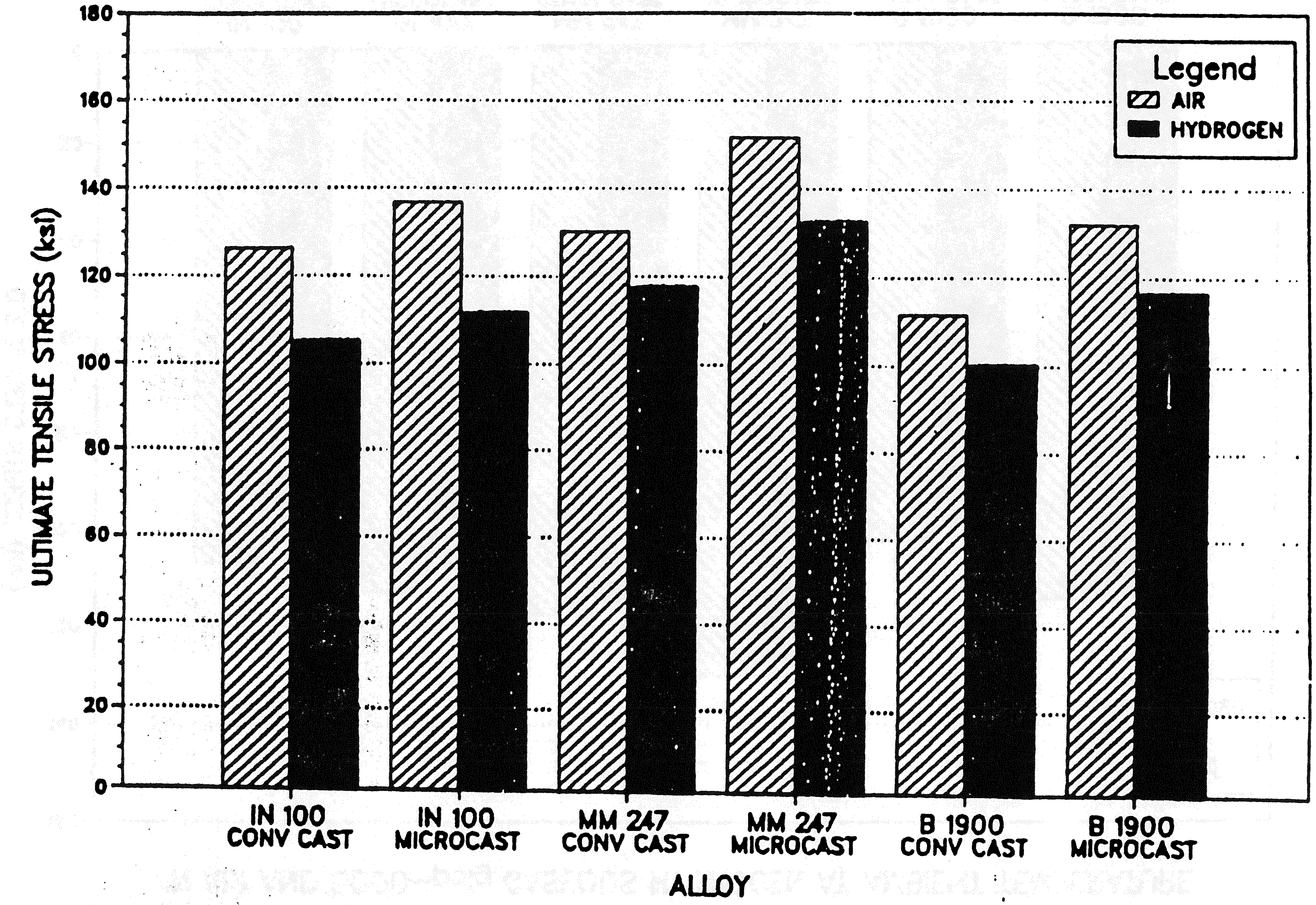


Stacking of close-packed atomic layers.

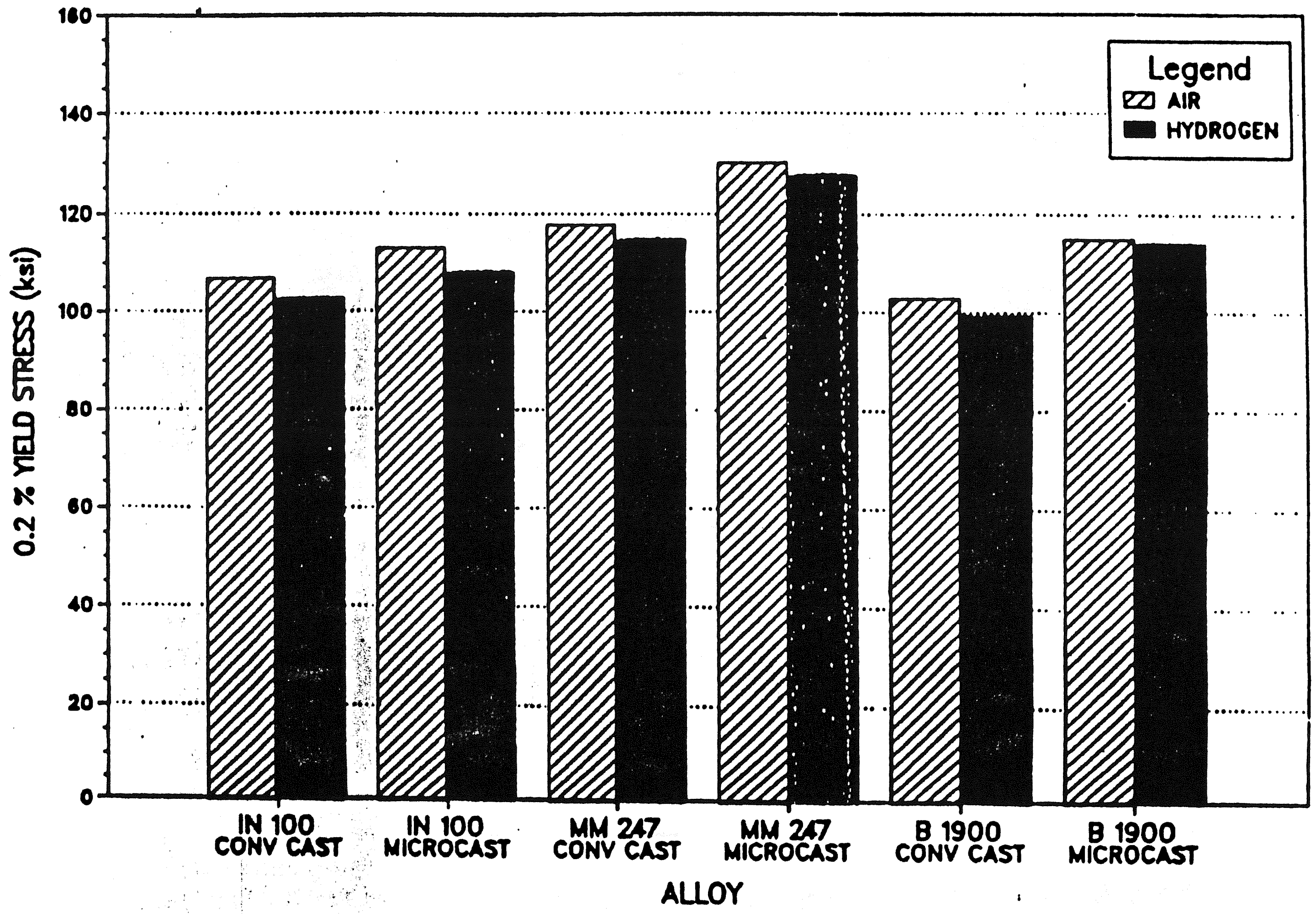


(a) Hexagonal close-packed; (b) face-centered cubic.

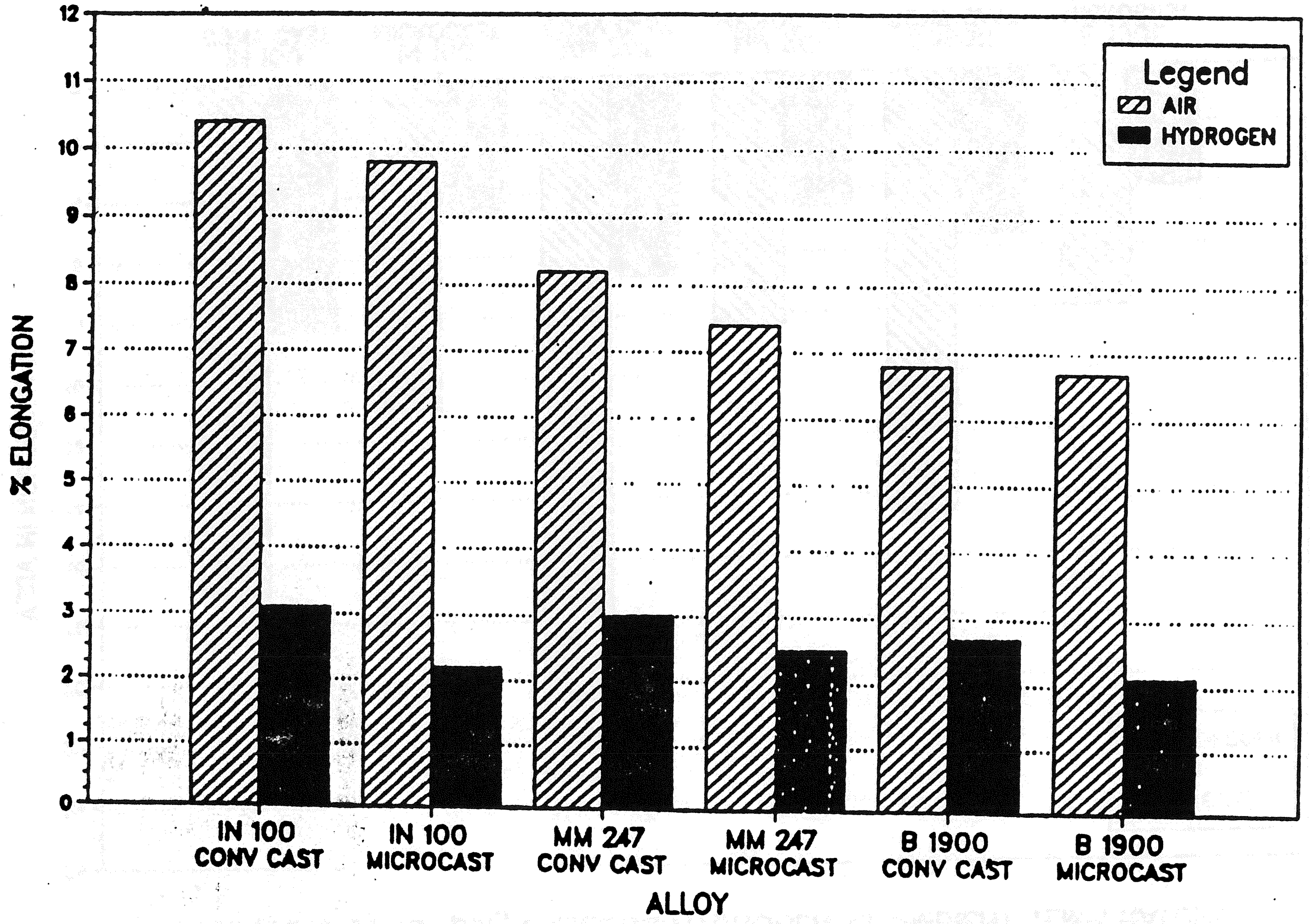
AVERAGE ULTIMATE TENSILE STRESS OF VARIOUS SSME VANE ALLOY CANDIDATES
IN AIR AND 5000–psig GASEOUS HYDROGEN AT AMBIENT TEMPERATURE



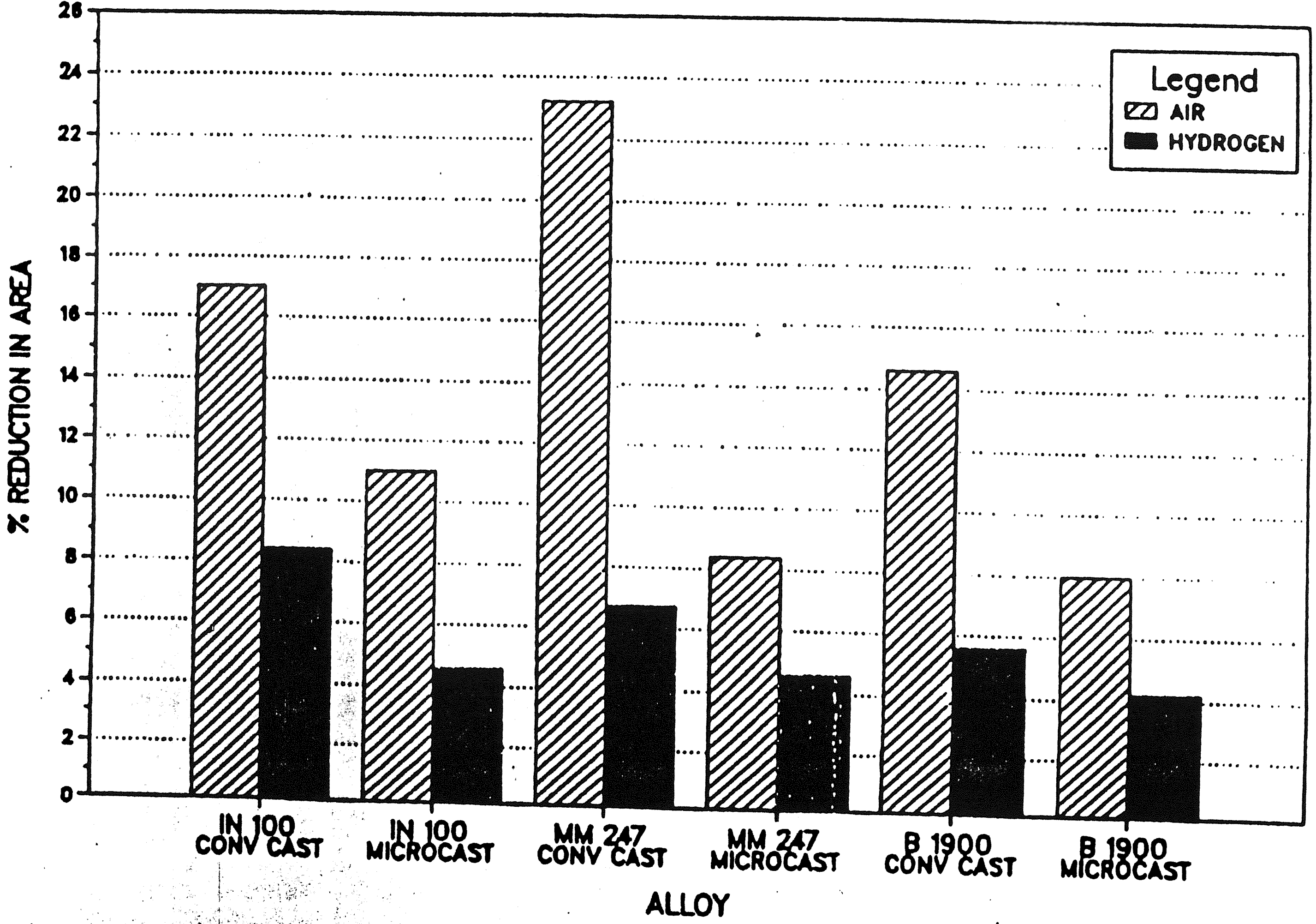
AVERAGE 0.2 % YIELD STRESS OF VARIOUS SSME VANE ALLOY CANDIDATES
IN AIR AND 5000–psig GASEOUS HYDROGEN AT AMBIENT TEMPERATURE



AVERAGE % ELONGATION OF VARIOUS SSME VANE ALLOY CANDIDATES
IN AIR AND 5000-psig GASEOUS HYDROGEN AT AMBIENT TEMPERATURE



AVERAGE % REDUCTION IN AREA OF VARIOUS SSME VANE ALLOY CANDIDATES IN AIR AND 5000–psig GASEOUS HYDROGEN AT AMBIENT TEMPERATURE



TENSILE PROPERTIES OF Ti-5Al-2.5Sn ELI IN VARIOUS ENVIRONMENTS

Specimen			Environment			Test Results				
						Strength			Ductility	
No.	Type	Stress Conc. Factor	Type	Pressure psig	Temp. F	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area %	Elongation %
T-1	UN	-	Air	0	Rm	114	119	-	30	17
T-2	UN	-	Air	0	Rm	114	119	-	32	19
T-6	UN	-	Helium	5000	-200	-	152	-	26	11
T-7	UN	-	Helium	5000	-200	-	149	-	26	16
T-3	UN	-	Hydrogen	5000	-200	95	151	-	33	10
T-4	UN	-	Hydrogen	5000	-200	116	152	-	29	10
T-5	UN	-	Hydrogen	5000	-200	111	144	-	28	8
T-8	N	8.9	Helium	5000	-200	-	230	-	1.6	-
T-9	N	8.7	Helium	5000	-200	-	225	-	1.7	-
T-10	N	8.9	Hydrogen	5000	-200	-	228	-	1.4	-
T-11	N	8.7	Hydrogen	5000	-200	-	228	-	1.5	-
T-12	N	8.7	Hydrogen	5000	-200	-	226	-	0.9	-

Rocketdyne data presented under contract NAS8-25579

TENSILE PROPERTIES OF OFHC COPPER IN VARIOUS ENVIRONMENTS

Specimen			Environment			Test Results				
						Strength			Ductility	
No.	Type	Stress Conc. Factor	Type	Pressure psig	Temp. F	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area %	Elongation %
C-1	UN	-	Air	0	Rm	15	28	-	84	61
C-2	UN	-	Air	0	Rm	18	28	-	83	53
C-11	UN	-	Helium	5000	Rm	11	28	-	81	62
C-12	UN	-	Helium	5000	Rm	12	27	-	88	63
C-3	UN	-	Hydrogen	5000	Rm	11	27	-	86	62
C-4	UN	-	Hydrogen	5000	Rm	11	27	-	83	65
C-5	UN	-	Hydrogen	5000	Rm	10	26	-	84	68
C-21	N	8.7	Helium	5000	Rm	-	44	-	25	-
C-22	N	8.8	Helium	5000	Rm	-	41	-	21	-
C-13	N	8.7	Hydrogen	5000	Rm	-	40	-	24	-
C-14	N	8.8	Hydrogen	5000	Rm	-	43	-	26	-
C-15	N	8.7	Hydrogen	5000	Rm	-	44	-	25	-
C-19	N	8.7	Helium	5000	-200	-	40	-	30	-
C-20	N	8.8	Helium	5000	-200	-	41	-	28	-
C-16	N	8.7	Hydrogen	5000	-200	-	41	-	30	-
C-17	N	8.6	Hydrogen	5000	-200	-	46	-	23	
C-18	N	8.7	Hydrogen	5000	-200	-	45	-	20	-

Rocketdyne data presented under contract NAS8-25579

TENSILE PROPERTIES OF AISI TYPE 321 STAINLESS STEEL IN VARIOUS ENVIRONMENTS

Specimen			Environment			Test Results				
						Strength			Ductility	
No.	Type	Stress Conc. Factor	Type	Pressure psig	Temp. F	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area %	Elongation %
S-1	UN	-	Air	0	Rm	32	87	-	71	77
S-2	UN	-	Air	0	Rm	31	87	-	70	77
S-11	UN	-	Helium	5000	Rm	28	85	-	65	62
S-12	UN	-	Helium	5000	Rm	30	83	-	67	64
S-4	UN	-	Hydrogen	5000	Rm	36	86	-	59	63
S-6	UN	-	Hydrogen	5000	Rm	37	85	-	61	64
S-21	N	8.4	Helium	5000	Rm	-	111	-	6.3	-
S-22	N	8.7	Helium	5000	Rm	-	115	-	6.5	-
S-13*	N	8.7	Hydrogen	5000	Rm	-	101	0.89	1.2	-
S-14**	N	8.7	Hydrogen	5000	Rm	-	98	0.87	2.0	-
S-15	N	8.9	Hydrogen	5000	Rm	-	97	0.86	3.6	-
S-25	UN	-	Helium	5000	-200	-	120	-	67	45
S-29	UN	-	Helium	5000	-200	-	128	-	66	51
S-26	UN	-	Hydrogen	5000	-200	-	113	-	64	38
S-27***	UN	-	Hydrogen	5000	-200	-	118	-	53	39
S-28	UN	-	Hydrogen	5000	-200	-	134	-	51	51
S-16	N	8.5	Helium	5000	-200	-	142	-	14	-
S-17	N	8.7	Helium	5000	-200	-	143	-	10	-
S-18	N	8.7	Hydrogen	5000	-200	-	145	-	9.6	-
S-20	N	8.7	Hydrogen	5000	-200	-	141	-	14	-

Rocketdyne data presented under contract NAS8-25579

*Specimen bent and straightened prior to testing.

**5000 psi hydrogen was established during test, but prior to plastic deformation.

***Leakage occurred near end of test and the final test temperature was -50 F.

TENSILE PROPERTIES OF INCONEL 625 IN VARIOUS ENVIRONMENTS

Specimen			Environment			Test Results				
						Strength			Ductility	
No.	Type	Stress Conc. Factor	Type	Pressure psig	Temp. F	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Reduction of Area %	Elongation %
I-1	UN	-	Air	0	Rm	96	146	-	56	45
I-2	UN	-	Air	0	Rm	91	142	-	51	54
I-11	UN	-	Helium	5000	Rm	90	142	-	51	56
I-12	UN	-	Helium	5000	Rm	94	146	-	49	54
I-3	UN	-	Hydrogen	5000	Rm	86	126	0.88	18	21
I-4	UN	-	Hydrogen	5000	Rm	93	133	0.92	21	19
I-5	UN	-	Hydrogen	5000	Rm	82	128	0.89	16	19
I-21	N	8.7	Helium	5000	Rm	-	197	-	8.9	-
I-22	N	8.7	Helium	5000	Rm	-	219	-	9.8	-
I-13	N	8.9	Hydrogen	5000	Rm	-	161	0.77	3.9	-
I-14	N	8.9	Hydrogen	5000	Rm	-	154	0.74	5.0	-
I-15	N	8.7	Hydrogen	5000	Rm	-	160	0.77	5.0	-
I-6	UN	-	Helium	5000	-200	105	166	-	52	45
I-7	UN	-	Helium	5000	-200	101	162	-	51	44
I-8	UN	-	Hydrogen	5000	-200	102	162	-	48	40
I-9	UN	-	Hydrogen	5000	-200	95	152	-	47	42
I-10	UN	-	Hydrogen	5000	-200	106	172	-	49	48
I-16	N	8.7	Helium	5000	-200	-	217	-	8.0	-
I-17	N	8.9	Helium	5000	-200	-	206	-	8.2	-
I-18	N	8.9	Hydrogen	5000	-200	-	228	-	7.4	-
I-19	N	8.9	Hydrogen	5000	-200	-	215	-	6.7	-
I-20	N	8.9	Hydrogen	5000	-200	-	219	-	6.5	-

Rocketdyne data presented under contract NAS8-25579

ROOM TEMPERATURE TENSILE PROPERTIES OF INCONEL 718 SPECIMENS FABRICATED FROM 1/2 IN. THICK PLATE SUPPLIED BY STELLITE DIV., CABOT CORP.

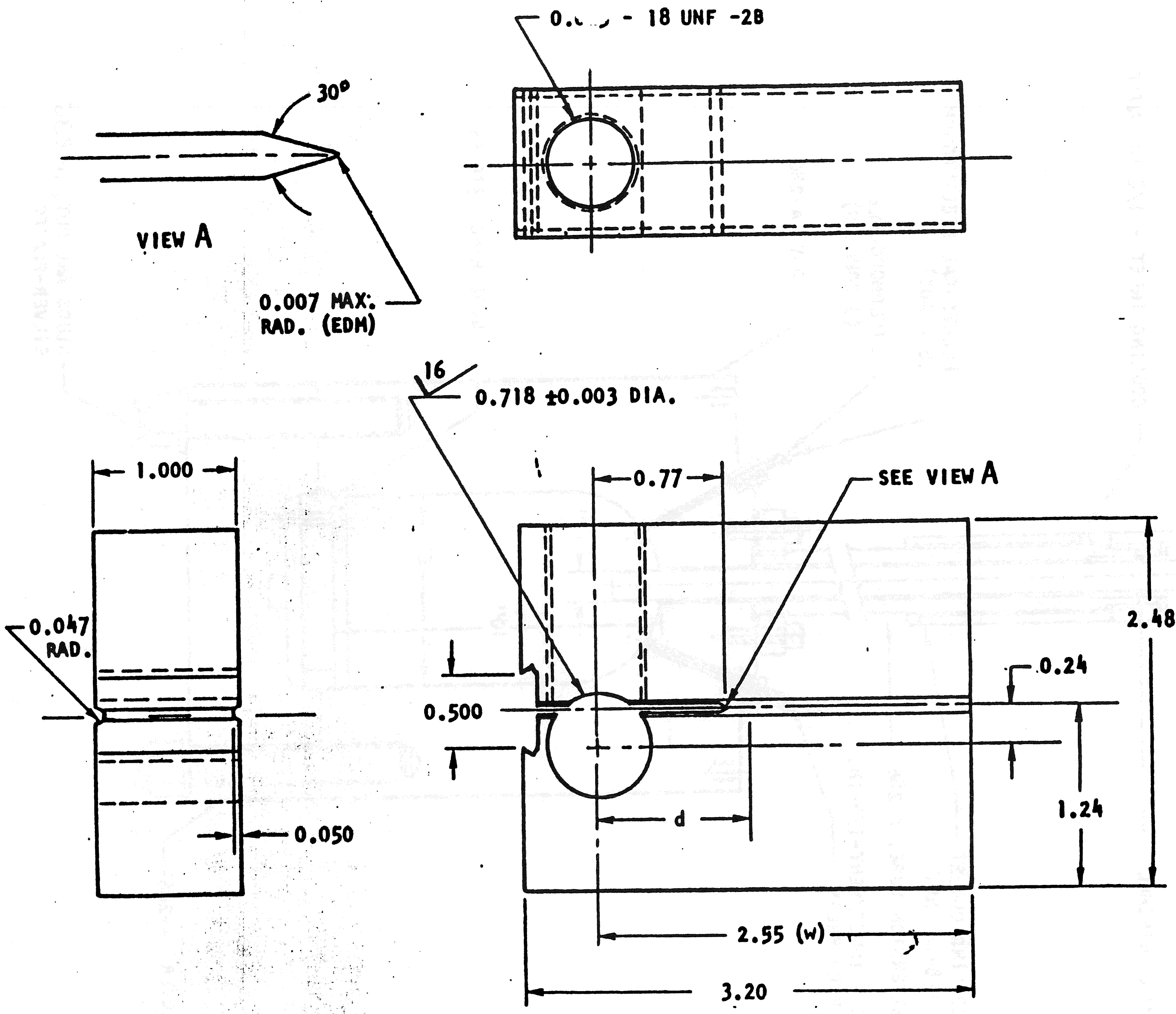
Heat Treatment			Specimen			Environment		Test Results				
								Strength			Ductility	
Solution Temp. F	Aging Temps		No.	Type	Stress Conc. Factor	Type	Pressure psig	Yield KSI	Ultimate KSI	Strength Ratio H_2/He	Percent Reduction of Area	Percent Elongation
	First	Second										
1725	1325	1150	ID-1	UN	-	Air	0	160	206	-	36	22
			ID-2	UN	-	Air	0	158	204	-	36	23
			ID-6	N	8.0	Helium	5000	-	277	-	2.0	-
			ID-7	N	8.9	Helium	5000	-	296	-	4.0	-
			ID-3	N	8.7	Hydrogen	5000	-	257	0.89	2.4	-
			ID-4	N	8.4	Hydrogen	5000	-	233	0.81	2.0	-
			ID-5	N	8.3	Hydrogen	5000	-	249	0.87	1.7	-
1925	1400	1200	IE-1	UN	-	Air	0	163	204	-	37	25
			IE-2	UN	-	Air	0	170	203	-	39	24
			IE-6	N	8.7	Helium	5000	-	319	-	3.6	-
			IE-7	N	8.7	Helium	5000	-	321	-	3.8	-
			IE-5	N	8.7	Hydrogen	5000	-	248	0.78	2.9	-
			IE-3	N	8.3	Hydrogen	5000	-	246	0.77	2.1	-
			IE-4	N	8.5	Hydrogen	5000	-	248	0.78	2.0	-

Rocketdyne data presented under contract NAS8-25579

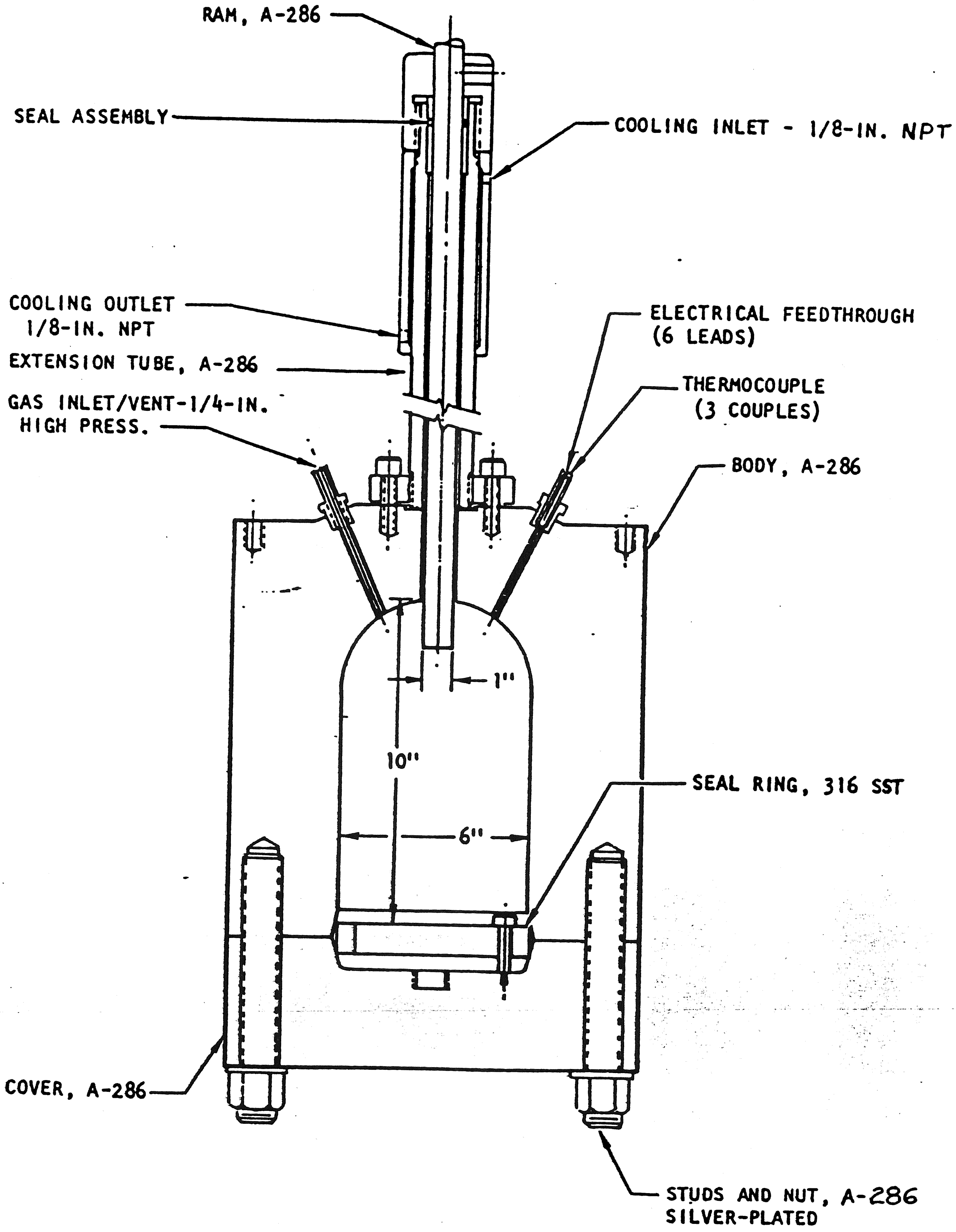
YIELD STRENGTHS AND MAXIMUM STRESS INTENSITIES AT WHICH PLANE STRAIN EXISTS FOR THE 0.0254 m (1.0 IN.) THICK WOL SPECIMEN USED FOR THESE TESTS

Material	Yield Strength				$K_{I\ (max)}$ For Plane Strain			
	295 K (70 F)		144 K (-200 F)		295 K (70 F)		144 K (-200 F)	
	MN/m^2	ksi	MN/m^2	ksi	$MN/m^2\ \sqrt{m}$	$KSI\ \sqrt{in.}$	$MN/m^2\ \sqrt{m}$	$KSI\ \sqrt{in.}$
Inconel 718	1120	162	1240	180	110	100	125	114
Inconel 625	650	94	760	110	64	58	77	70
AISI 321 Stainless Steel	220	32	260	38	22	20	25	23
A-286 Stainless Steel	780	113	879	126	77	70	88	80
T-5Al-2.5 Sn ELI	820	119	1040	151	80	73	105	96
2219-T87 Al Alloy	390	57	430	63	38	35	44	40
OFHC Copper	120	17	160	23	11	10	17	15

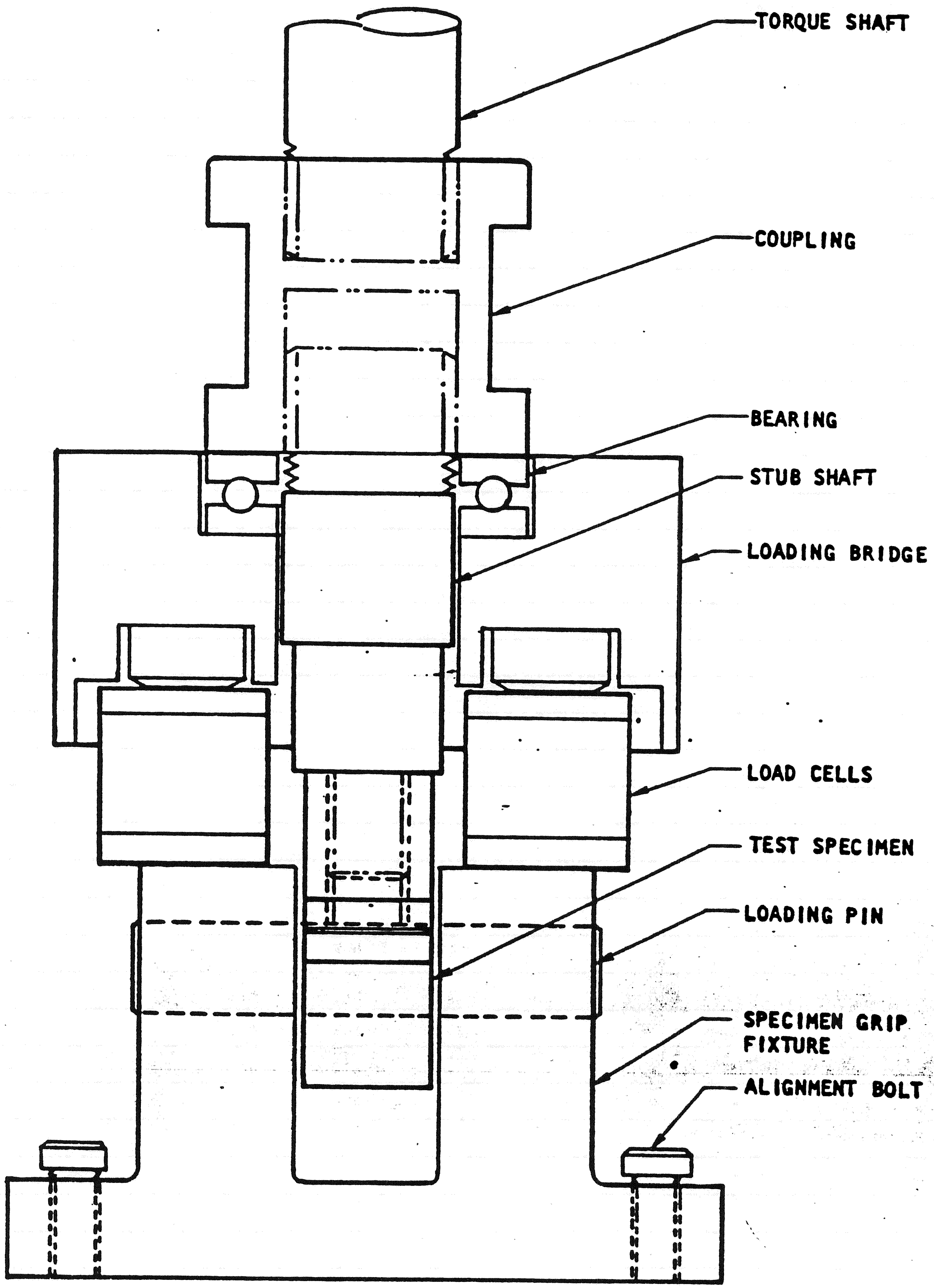
Rocketdyne data presented under contract NAS8-25579



Modified WOL Specimen With Straight EDM Notch



Pressure Vessel Used to Perform Tests on Modified WOL Specimen in High-Pressure Hydrogen



Apparatus for Holding a Constant Deflection and Measuring the Load on the WOL Specimen

NOTES

NOTES

NOTES

Melting Points of the Elements

(Based on the 1968 International Practical Temperature Scale)

Element (and symbol)	Melting point	
	C	F
Actinium (Ac)	1051	1924
Aluminum (Al)	660.37(a)	1220.67(a)
Americium (Am)	996	1825
Antimony (Sb)	630.74(a)	1167.33(a)
Argon (Ar)	−189.33	−308.79
Arsenic (As)	603(b)	1117(b)
Astatine (At)	302(est) (c)	576 (est) (c)
Barium (Ba)	729	1344
Berkelium (Bk)	987(d)	1809(d)
Beryllium (Be)	1289	2352
Bismuth (Bi)	271.442(a)	520.596(a)
Boron (B)	2103 (approx) (e)	3817 (approx) (e)
Bromine (Br)	−7.25	+18.95
Cadmium (Cd)	321.108(a)	609.994(a)
Calcium (Ca)	840	1544
Californium (Cf)	....	....
Carbon (C) (graphite)	3836(b)	6937(b)
Cerium (Ce)	799	1470
Cesium (Cs)	28.39	83.10
Chlorine (Cl)	−100.97	−149.75
Chromium (Cr)	1863(e)	3385(e)
Cobalt (Co)	1494(a)	2721(a)
Columbium (Cb)	2471	4480
Copper (Cu)	1084.5(a)	1984.1(a)
Curium (Cm)	1342(f)	2448(f)
Dysprosium (Dy)	1411	2572
Einsteinium (Es)	....	....
Erbium (Er)	1524	2775
Europium (Eu)	818	1504
Fermium (Fm)	....	....
Fluorine (F)	−219.67	−363.41
Francium (Fr)	27 (est) (c)	81 (est) (c)
Gadolinium (Gd)	1314	2397
Gallium (Ga)	29.75	85.55
Germanium (Ge)	938.3	1720.9
Gold (Au)	1064.43(g)	1947.97(g)
Hafnium (Hf)	2231	4048
Helium (He)	−271.39(h)	−456.50(h)
Holmium (Ho)	1472	2682
Hydrogen (H)	−259.347	−434.825
Indium (In)	156.634(a)	313.941(a)
Iodine (I)	113.6	236.5
Iridium (Ir)	2447(a)	4437(a)
Iron (Fe)	1538	2800
Krypton (Kr)	−157.38	−251.28
Lanthanum (La)	921	1690
Lawrencium (Lr)	....	....
Lead (Pb)	327.502(a)	621.504(a)
Lithium (Li)	180.5	356.9
Lutetium (Lu)	1665	3029
Magnesium (Mg)	649	1200
Manganese (Mn)	1246	2275
Mendelevium (Md)	....	....
Mercury (Hg)	−38.862(a)	−37.952(a)
Molybdenum (Mo)	2623(e)	4753(e)
Neodymium (Nd)	1017	1863
Neon (Ne)	−248.597	−415.475
Neptunium (Np)	637	1179
Nickel (Ni)	1455(a)	2651(a)
Nitrogen (N)	−210.01	−346.02
Nobelium (No)	....	....
Osmium (Os)	3033	5491
Oxygen (O)	−218.80	−361.84
Palladium (Pd)	1554(a)	2829(a)
Phosphorus (P) (white)	44.15	111.47
Platinum (Pt)	1772(a)	3222(a)
Plutonium (Pu)	640	1184
Polonium (Po)	254(c)	489(c)
Potassium (K)	63.2	145.8
Praseodymium (Pr)	932	1710
Promethium (Pm)	1027 (est) (c)	1881 (est) (c)
Protactinium (Pa)	1230 (est) (c)	2246 (est) (c)
Radium (Ra)	700(c)	1292(c)
Radon (Rn)	−71	−96
Rhenium (Re)	3186	5767
Rhodium (Rh)	1963(a)	3565(a)
Rubidium (Rb)	39.48	103.06
Ruthenium (Ru)	2254	4089
Samarium (Sm)	1074	1965
Scandium (Sc)	1541	2806
Selenium (Se)	221	430
Silicon (Si)	1414	2577
Silver (Ag)	961.93(g)	1763.47(g)
Sodium (Na)	97.8	208.0
Strontium (Sr)	769	1416
Sulfur (S) (yellow)	115.21	239.38
Tantalum (Ta)	3020	5468
Technetium (Tc)	2204	3999
Tellurium (Te)	449.57	841.23
Terbium (Tb)	1359	2478
Thallium (Tl)	304	579
Thorium (Th)	1758	3196
Thulium (Tm)	1547	2817
Tin (Sn)	231.968(g)	449.542(g)
Titanium (Ti)	1672	3042
Tungsten (W)	3387(a)	6129(a)
Uranium (U)	1133	2071
Vanadium (V)	1929(e)	3504(e)
Xenon (Xe)	−111.78	−169.20
Ytterbium (Yb)	825	1517
Yttrium (Y)	1528	2782
Zinc (Zn)	419.58(g)	787.24(g)
Zirconium (Zr)	1865(e)	3389(e)

SOURCE: Data except those footnoted to indicate otherwise are from R Hultgren, P. D. Desai, D. T. Hawkins, M. Gleiser, K. K. Kelley and D. D. Wagman, "Selected Values of the Thermodynamic Properties of the Elements", American Society for Metals, 1973. Values corrected to the 1968 International Practical Temperature Scale (IPTS-68).

(a) Secondary reference point on 1968 International Practical Temperature Scale (IPTS-68). (b) Sublimation point. (c) From HANDBOOK, 8th Ed., Vol 1 (American Society for Metals, 1961). (d) From J. R. Peterson, R D. Baybarz and J. A. Fahey, Preparat Some Properties of Berkelium Metal, p 20-34 in Vol 17 of Nuclear Metallurgical Series (W. N. Miner, Ed.), TMS-AIME, 1970. (e) E. Rudy, "Compendium of Phase Diagram Data", AFML-TR-65-2, Part V, 1969. (f) From B. B. Cunningham and J. C. Wallman, *J Inorg Nuc Chem*, Vol 26, 1964, p 271-275. (g) Primary reference point on 1968 International Practical Temperature Scale (IPTS-68). (h) Helium does not solidify at 1 atm; value is at 0.95 atm.

ROTORDYNAMICS OF TURBOMACHINERY

- ☞ INTRODUCTION
- ☞ COMPUTER CODES
- ☞ MATHEMATICAL FOUNDATION
- ☞ INSTABILITY
- ☞ PRACTICAL EXAMPLE – SSME

ROTORDYNAMICS

What do Rotordynamist do?

- Predict rotor system deflection
- Predict rotor critical speeds
- Predict bearing dynamic loads
- Monitor turbopump / engine health
  - General deterioration
  - Rotor system rubs
  - Bearing health

These are accomplished through Critical Speeds, Forced Response, and Stability Analyses.

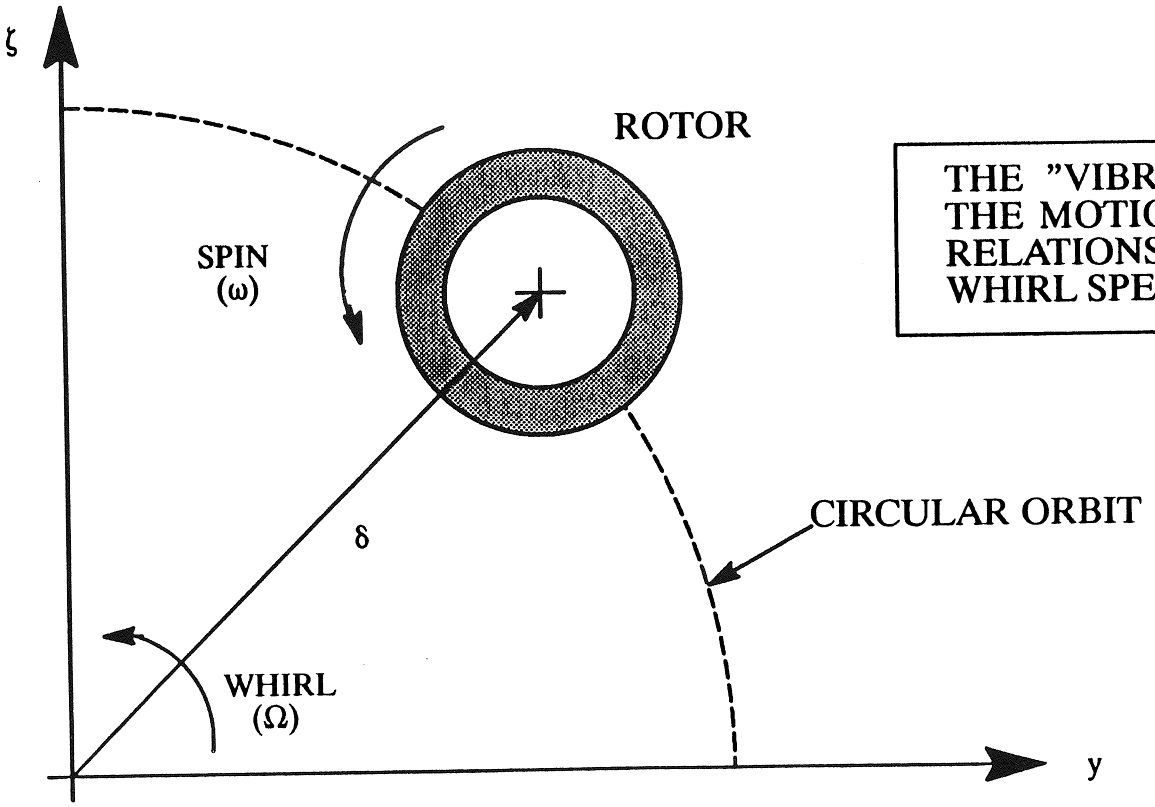
INTRODUCTION TO VIBRATIONS

- SYNCHRONOUS MOTION
- NON–SYNCHRONOUS MOTION

DYNAMIC MOTION

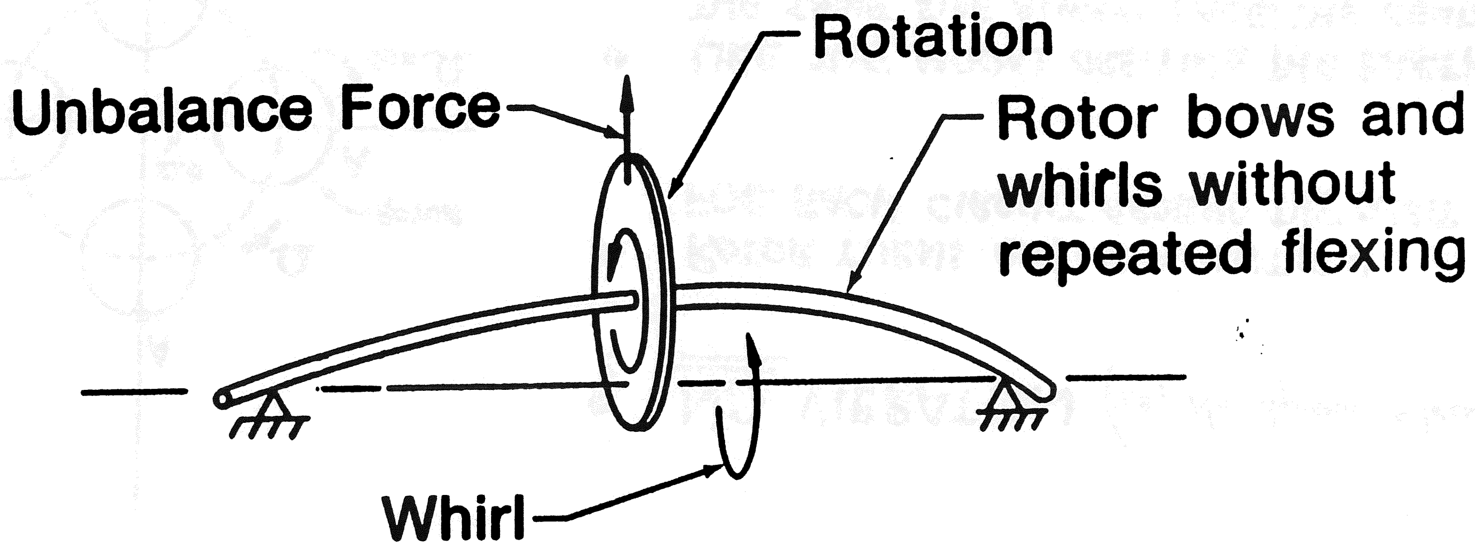
All state variables are rotating vectors

e.g. DEFLECTION



THE "VIBRATORY" NATURE OF THE MOTION DEPENDS ON THE RELATIONSHIP OF SPIN AND WHIRL SPEEDS

ROTOR WHIRL



Unbalance Force

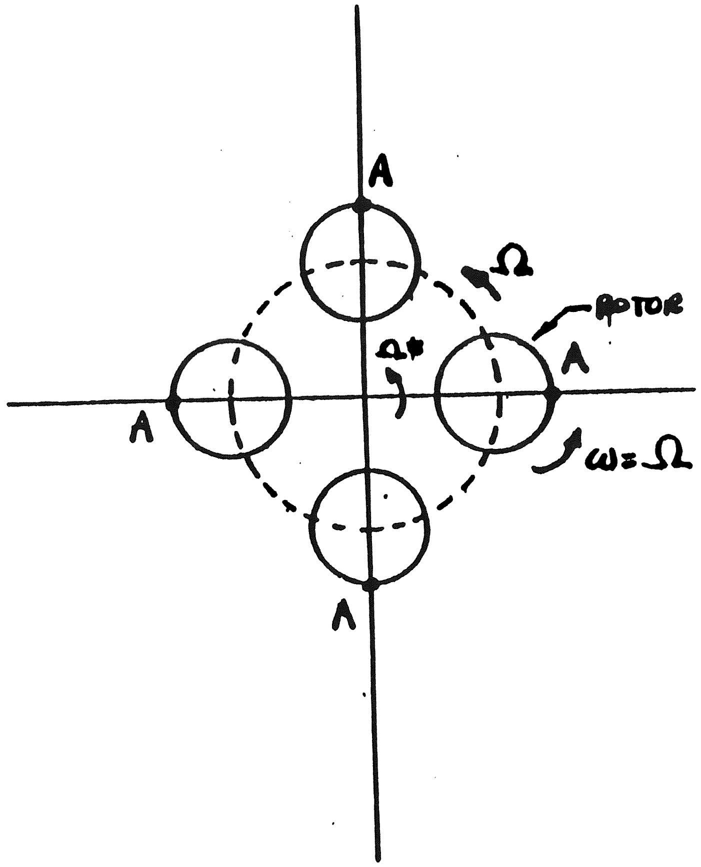
Rotation

Rotor bows and
whirls without
repeated flexing

Whirl

SYNCHRONOUS MOTION

WHIRL AND SPIN SPEED VECTORS ARE IDENTICAL

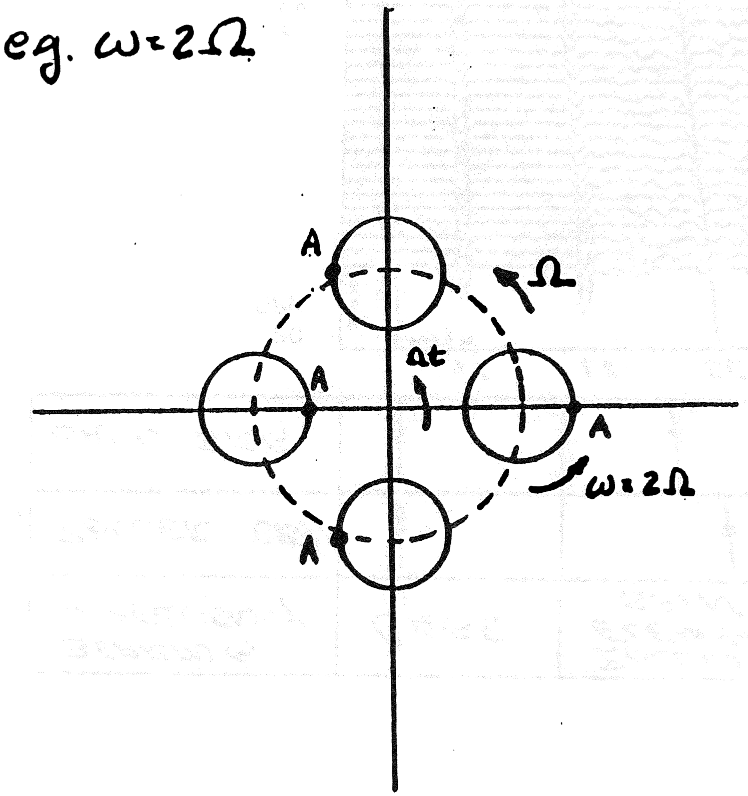


- **NO** VIBRATION (i.e, No dynamic stress)
- ROTOR TURNS ONE REVOLUTION FOR EACH CIRCUIT AROUND THE ORBIT
- LIKE THE MOON ORBITING THE EARTH – THE SAME SIDE ALWAYS FACES THE CENTER ($\omega = \Omega$)

NONSYNCHRONOUS MOTION

WHIRL AND SPIN SPEED VECTORS UNEQUAL

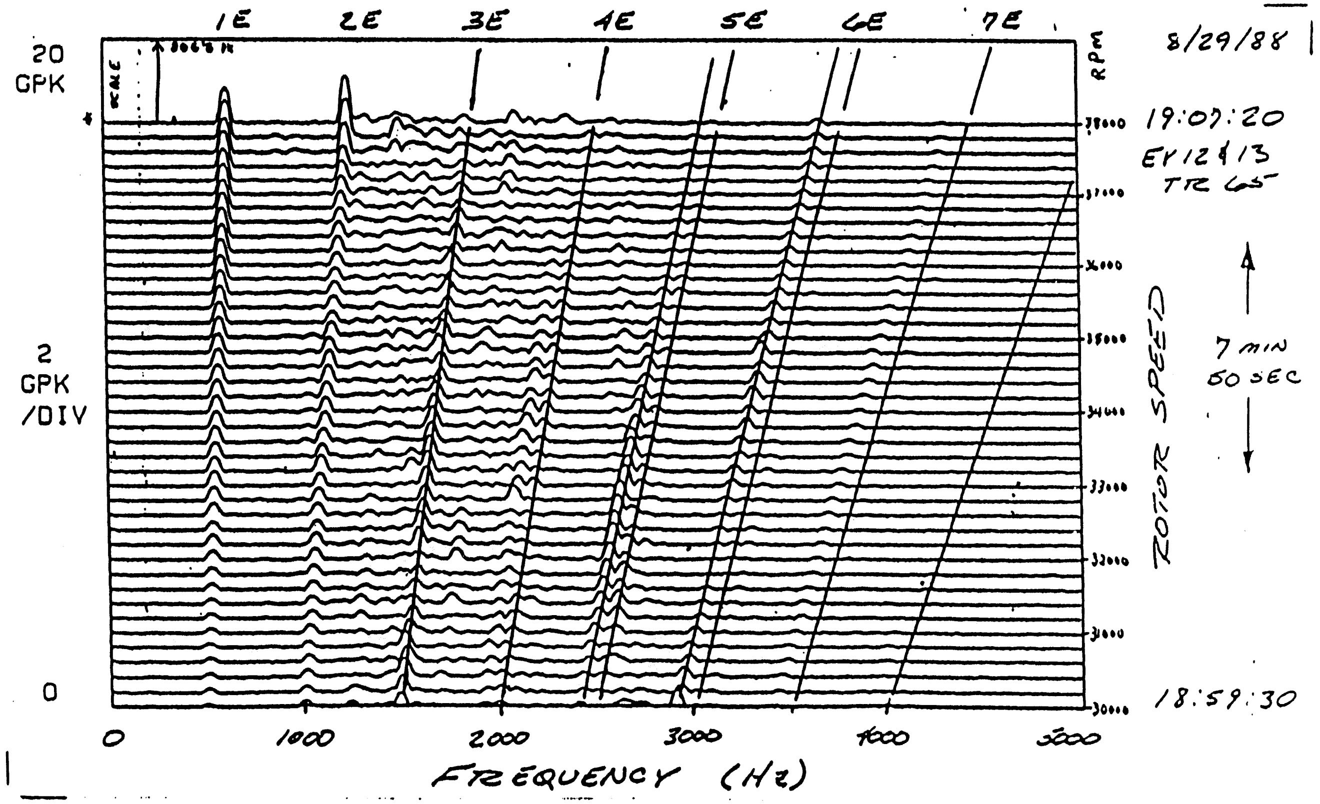
e.g. $\omega = 2\Omega$



- VIBRATION AND STEADY MOTION
- ROTOR TURNS TWICE AROUND ITS OWN AXIS AS ONE ORBIT IS TRAVERSED
- FREQUENCY OF VIBRATORY COMPONENT OF MOTION = $|\omega - \Omega|$ {RELATIVE TO THE ROTOR}
- LIKE THE EARTH ORBITING AROUND THE SUN ($\omega = 365\Omega$)
- SPECIAL CASE $\omega = 0$. MOTION IS PURE VIBRATORY (CIRCULAR) e.g. NONSPINNING STRUCTURES

ATD ROLLER BEARING RIG
SPECTRAL MAP

BEARING FREQUENCY	CAGE	ROLLING ELEMENT SPIN	ROLLING ELEMENT PASSING	INNER RACE DEFECT
ROLLER BRG	↓	↓	↓	↓
BALL BRG	↓	↓	↓	↓



COMPUTER CODES

ROTORDYNAMICS COMPUTER PROGRAMS

Eight programs in use

1. ROTOR DYNAMICS GRAPHICS (SID: U495)

   INTERACTIVE GRAPHICS LAYOUT DIGITIZER AND MODEL GENERATOR
   DOCUMENTATION: FTDM 1906

2. ENGINE CRITICAL SPEEDS (SID: M636)

   UNDAMPED CRITICAL SPEEDS AND MODE SHAPES
   DOCUMENTATION: METHDOC

3. ENGINE FORCED RESPONSE (SID: A346)

   DAMPED UNBALANCE RESPONSE – LINEAR AND NONLINEAR
   DOCUMENTATION: METHDOC

4. ENGINE STATIC DEFLECTION (SID: P871)

   ENGINE DEFLECTION DUE TO MANEUVERS AND STATIC LOADS
   DOCUMENTATION: METHDOC

5. ENGINE TRANSIENT RESPONSE (SID: W600)

   TIME DEPENDENT RESPONSE DUE TO TIME DEPENDENT LOADS
   DOCUMENTATION: IN WORK – ROUGH MANUAL AVAILABLE

6. DESIGN OF ENGINE ROTORS BY VIBRATION ANALYSIS – DERVA (SID: T169)

   INTERACTIVE ANALYSIS SELECTOR AND RESULTS PROCESSOR
   DOCUMENTATION: IN WORK

7. ANALYSIS OF ROTORDYNAMIC SYSTEMS – ARDS (SID: E040)

   DAMPED ROTOR SYSTEM INSTABILITY ANALYSIS
   DOCUMENTATION: METHDOC

8. TEXAS A&M SEAL ANALYSIS CODE

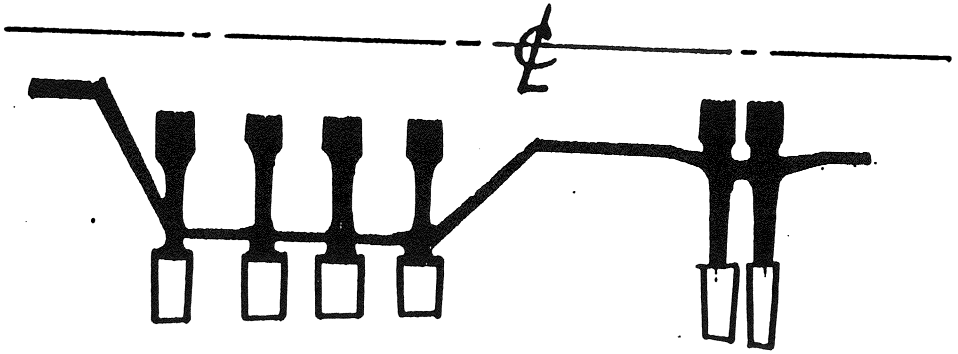
   DAMPER SEAL ANALYSIS SYSTEM TO DETERMINE ROTORDYNAMIC COEFFICIENTS USED IN FORCED RESPONSE AND STABILITY ANALYSES. DOCUMENTATION: ROCKET ENGINE STRUCTURES GROUP

MATHEMATICAL FOUNDATION

DISCRETIZED SYSTEM

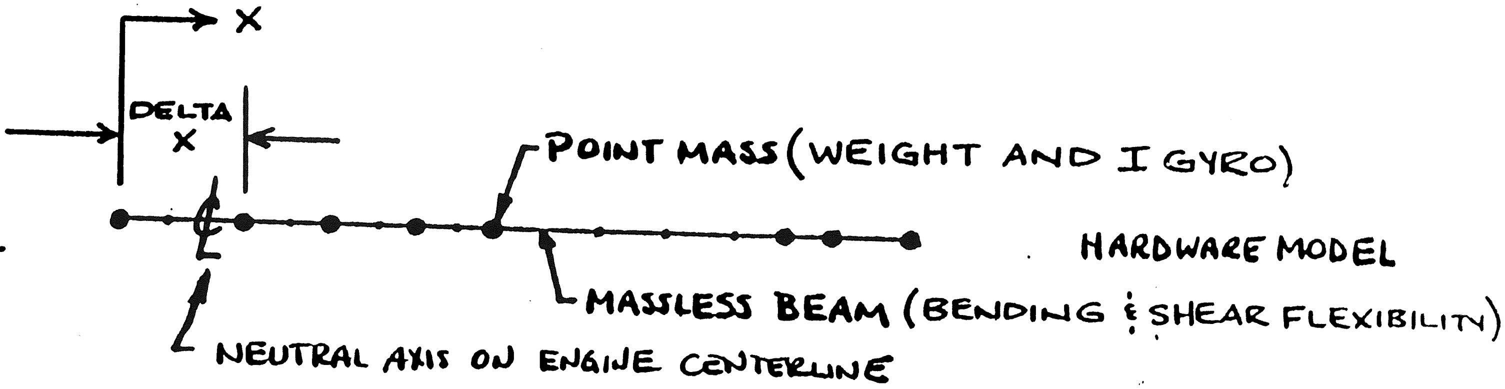
CONTINUOUS MASS AND STIFFNESS MODELLED BY POINT MASSES AND MASSLESS BEAMS

THIS:



ACTUAL HARDWARE

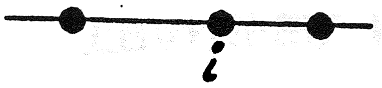
BECOMES THIS:



HARDWARE MODEL

STATE VECTOR

DISPLACEMENT AND LOADING AT A SPECIFIC POINT AND SPECIFIC FREQUENCY



DISCRETIZED MODEL

$$\{Z\}_i = \left\{\begin{array}{c} \bar{\delta} \\ \bar{\theta} \\ \hdashline \bar{M} \\ \bar{V} \\ \hdashline 1 \end{array}\right\}_i \quad \begin{array}{l} \text{DEFLECTION} \\ \text{SLOPE} \\ \text{MOMENT} \\ \text{SHEAR} \\ \text{(unity)} \end{array}$$

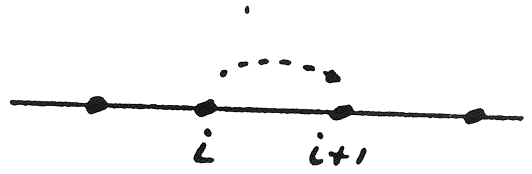
ALTERNATE DEFINITION:

"THE ANSWER"

TRANSFER MATRIX - CONCEPT

IT RELATES ADJACENT STATE VECTORS

MODEL



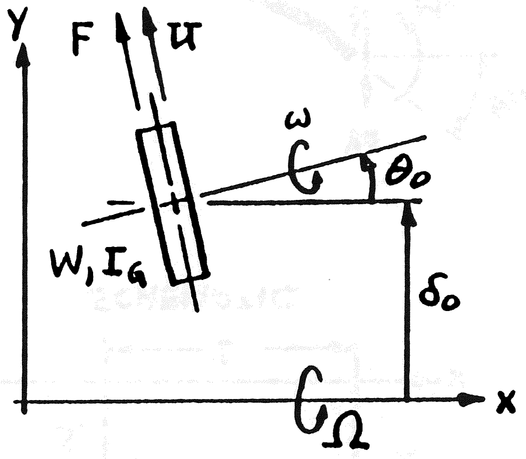
TRANSFER MATRIX EQUATION

$$\{Z\}_{i+1} = [T]_i \{Z\}_i$$

IN WORDS: IF THE STATE VECTOR AT ONE POINT IS KNOWN THE STATE VECTOR AT THE NEXT POINT CAN BE COMPUTED BY MULTIPLYING THE FIRST STATE VECTOR BY THE <u>TRANSFER</u> MATRIX.

MASS POINT TRANSFER MATRIX

CONTAINS THE DYNAMIC MASS PROPERTIES AND APPLIED LOADS OF A STATION POINT



SCHEMATIC

W - WEIGHT

I_G - GYRO INERTIA

F - CONSTANT FORCE

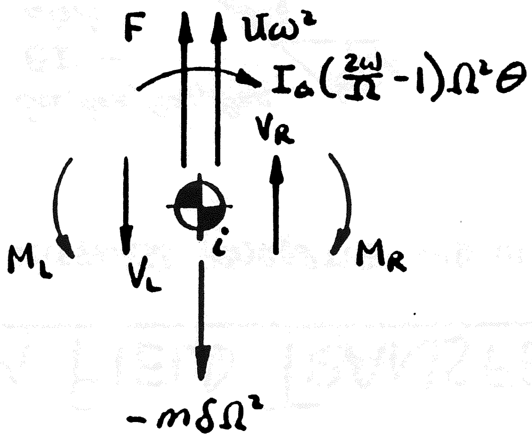
U - UNBALANCE

δ_0 - INITIAL DEFLECTION

θ_0 - INITIAL SLOPE

ω - SPIN SPEED

Ω - WHIRL SPEED



FREE BODY DIAGRAM

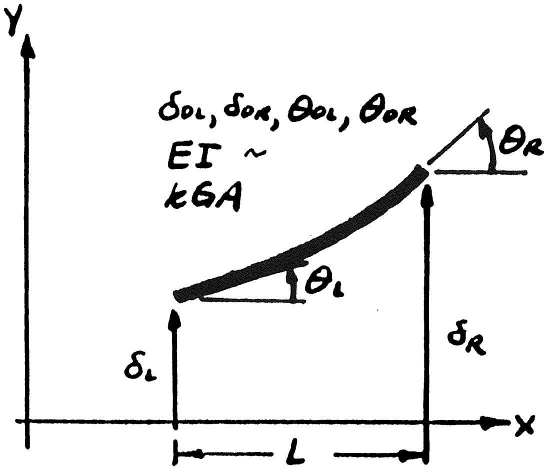
DISPLACEMENT COMPATIBILITY AND FORCE BALANCE LEADS TO TRANSFER MATRIX EQUATION :

$$\begin{Bmatrix} \delta \\ \theta \\ M \\ V \\ 1 \end{Bmatrix}_R = \underbrace{\left[\begin{array}{cc|cc|c} 1 & 0 & 0 & 0 & \\ 0 & 1 & 0 & 0 & \\ \hline 0 & I_G(\frac{2\omega}{\Omega}-1)\Omega^2 & 1 & 0 & f(\theta_0) \\ -m\Omega^2 & 0 & 0 & 1 & f(\delta_0, F, U) \\ \hline 0 & 0 & 0 & 0 & 1 \end{array}\right]_M}_{\text{MASS POINT TRANSFER MATRIX}} \begin{Bmatrix} \delta \\ \theta \\ M \\ V \\ 1 \end{Bmatrix}_L$$

NOTE DEPENDENCY ON WHIRL (Ω) AND SPIN SPEED (ω)

BEAM FIELD TRANSFER MATRIX

CONTAINS STIFFNESS PROPERTIES AND INITIAL DISPLACEMENT LOADS BETWEEN MASS POINTS



SCHEMATIC

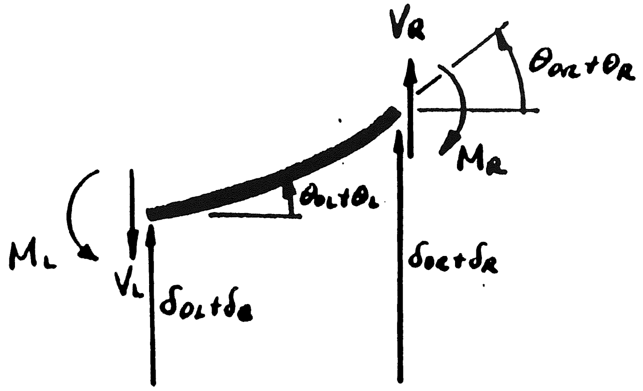
δ_{0L}, δ_{0R} INITIAL RIGHT AND LEFT DEFLECTIONS

θ_{0L}, θ_{0R} INITIAL RIGHT AND LEFT SLOPES

$\beta_B = \frac{L}{EI}$ BENDING FLEXIBILITY

$\beta_S = \frac{L}{kAG}$ SHEAR FLEXIBILITY

ELASTIC BEAM EQUATIONS AND FORCE BALANCE LEADS TO THE TRANSFER MATRIX EQUATION:



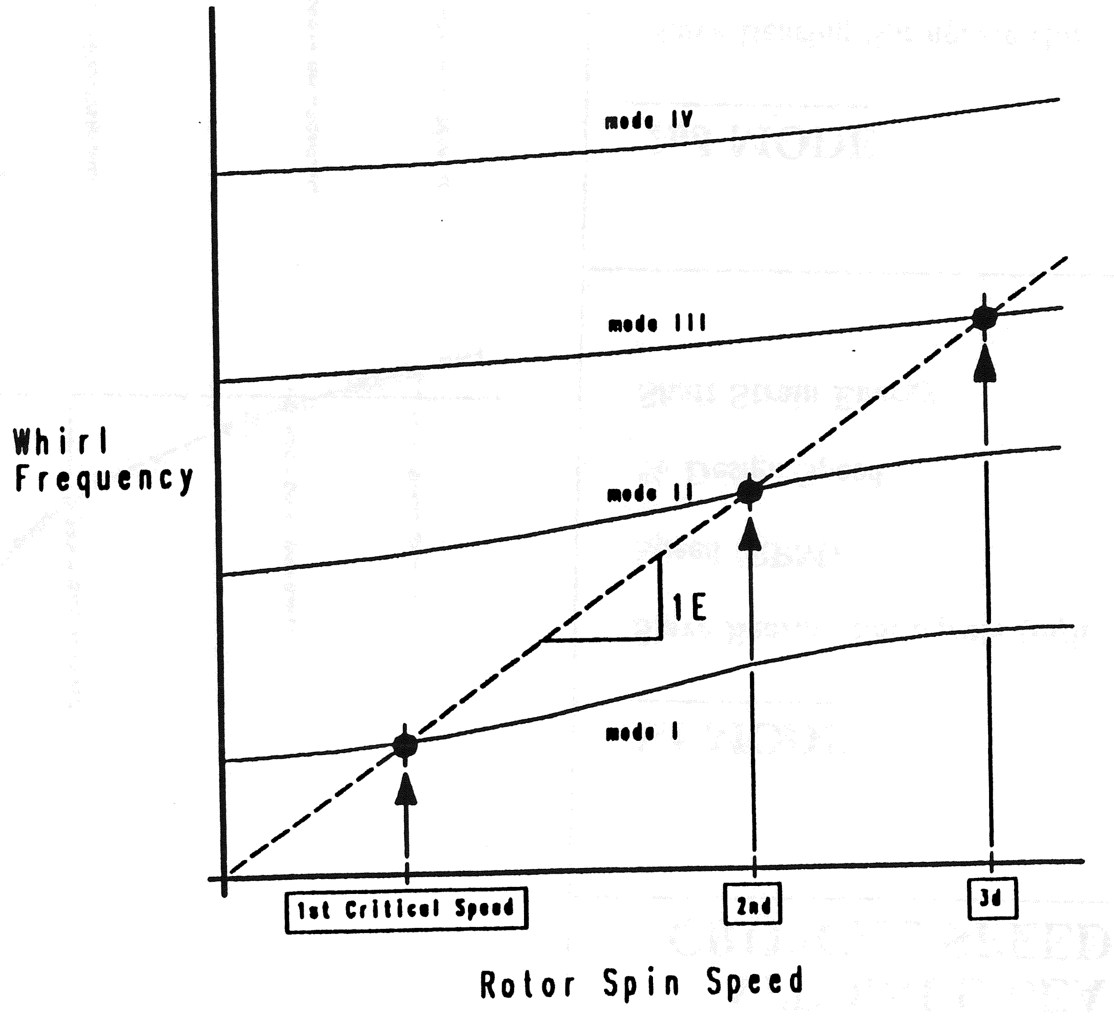
FREE BODY DIAGRAM

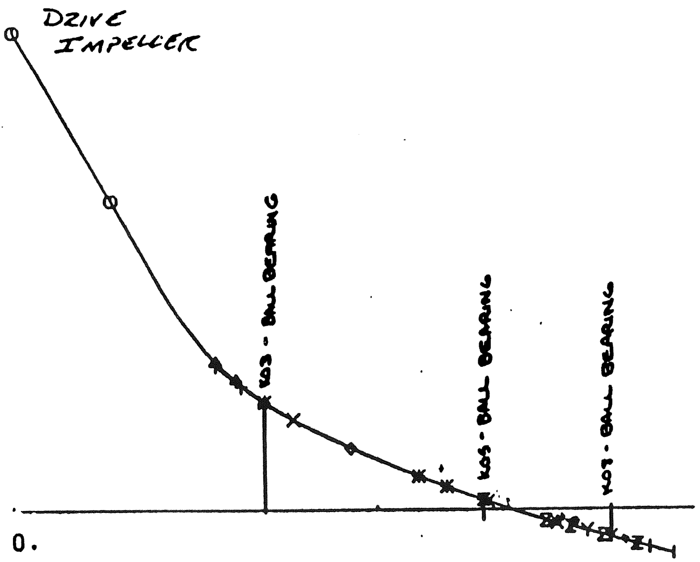
$$\begin{Bmatrix} \delta \\ \theta \\ M \\ V \\ 1 \end{Bmatrix}_R = \underbrace{\left[\begin{array}{cc:cc:c} 1 & L & \beta_B \frac{L}{2} & \left(\beta_B \frac{L^2}{6} - \beta_S\right) & f(\delta_{0L}, \delta_{0R}, \theta_{0L}) \\ 0 & 1 & \beta_B & \beta_B \frac{L}{2} & f(\theta_{0L}, \theta_{0R}) \\ \hdashline 0 & 0 & 1 & L & f(\theta_{0L}) \\ 0 & 0 & 0 & 1 & 0 \\ \hdashline 0 & 0 & 0 & 0 & 1 \end{array}\right]_B}_{\text{BEAM FIELD TRANSFER MATRIX}} \begin{Bmatrix} \delta \\ \theta \\ M \\ V \\ 1 \end{Bmatrix}_L$$

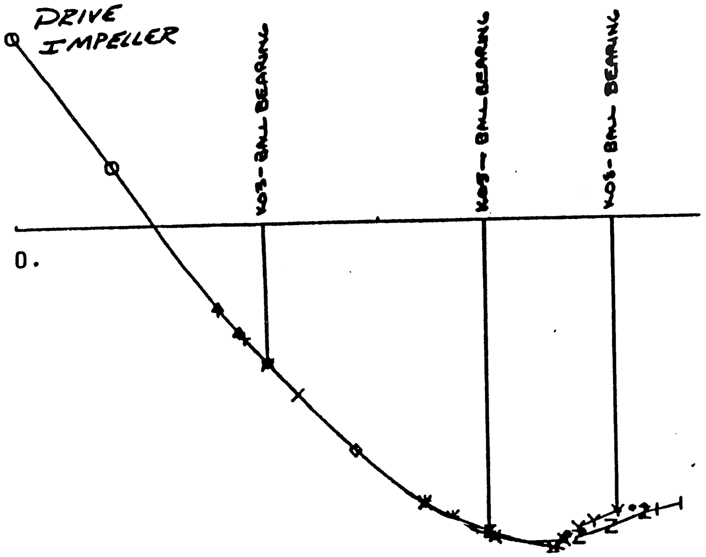
NOTE INDEPENDENCE ON WHIRL OR SPIN SPEED

CRITICAL SPEEDS

Determined by intersections of 1E line and mode lines on whirl map







ATD BALL BEARING RIG
CRITICAL SPEED SUMMARY

1st MODE

Slave Bearing Springrate (lb/in)	0.5×10^6	1.0×10^6
Speed (RPM)	25,500	32,000
% Design Speed	66%	83%
Shaft Strain Energy	23%	38%

2nd MODE

Slave Bearing Springrate (lb/in)	0.5×10^6	1.0×10^6
Speed (RPM)	45,750	60,000
% Design Speed	122%	160%
Shaft Strain Energy	9%	16%

ROTORDYNAMIC INSTABILITY IN TURBOMACHINERY

☐ EXCERPTS FROM V. S. Fehr '86 REVIEW

ROTORDYNAMIC INSTABILITY

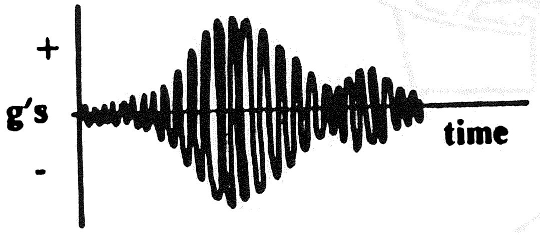
- CHARACTERISTIC VIBRATION
- POTENTIAL DRIVERS
- FLUID – ROTOR COUPLING
  - INTERACTION
  - WORK CONCEPT
  - EXAMPLES
- STABILITY ANALYSIS
- ANALYSIS ASSESSMENT
- PWA EXPERIENCE
- SUMMARY
- RECOMMENDATIONS

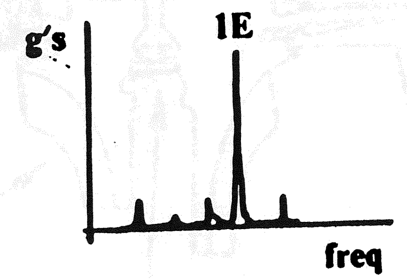
TYPICAL VIBRATION CHARACTERISTICS

Instability easily observed by spectral content of vibration

Overall Vibration

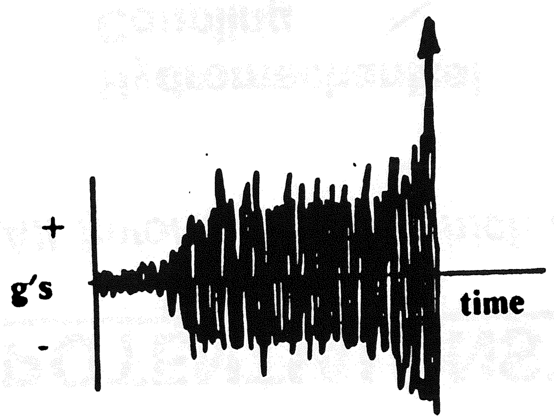
Spectral Content

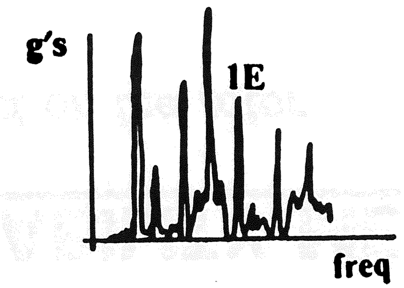




STABLE

- Acceptable levels
- Mostly *SYNCHRONOUS* (1E)
- Peaks at critical speeds



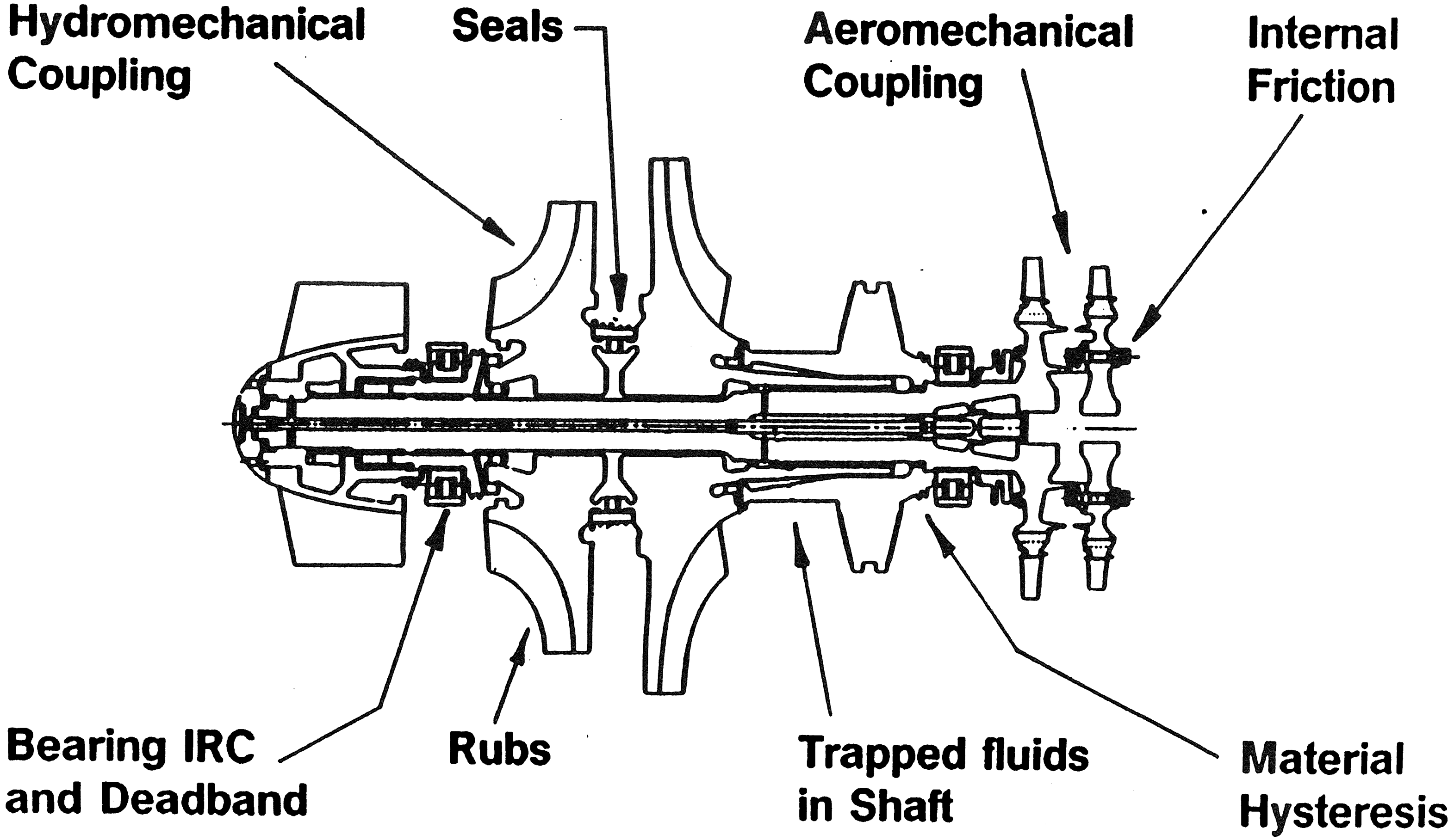


UNSTABLE

- Excessive levels
- Mostly *NONSYNCHRONOUS*
- Peaks at whirl frequencies
- Can be *catastrophic*

POTENTIAL INSTABILITY MECHANISMS

All involve forces which act on the rotor



Hydromechanical Coupling

Seals

Aeromechanical Coupling

Internal Friction

Bearing IRC and Deadband

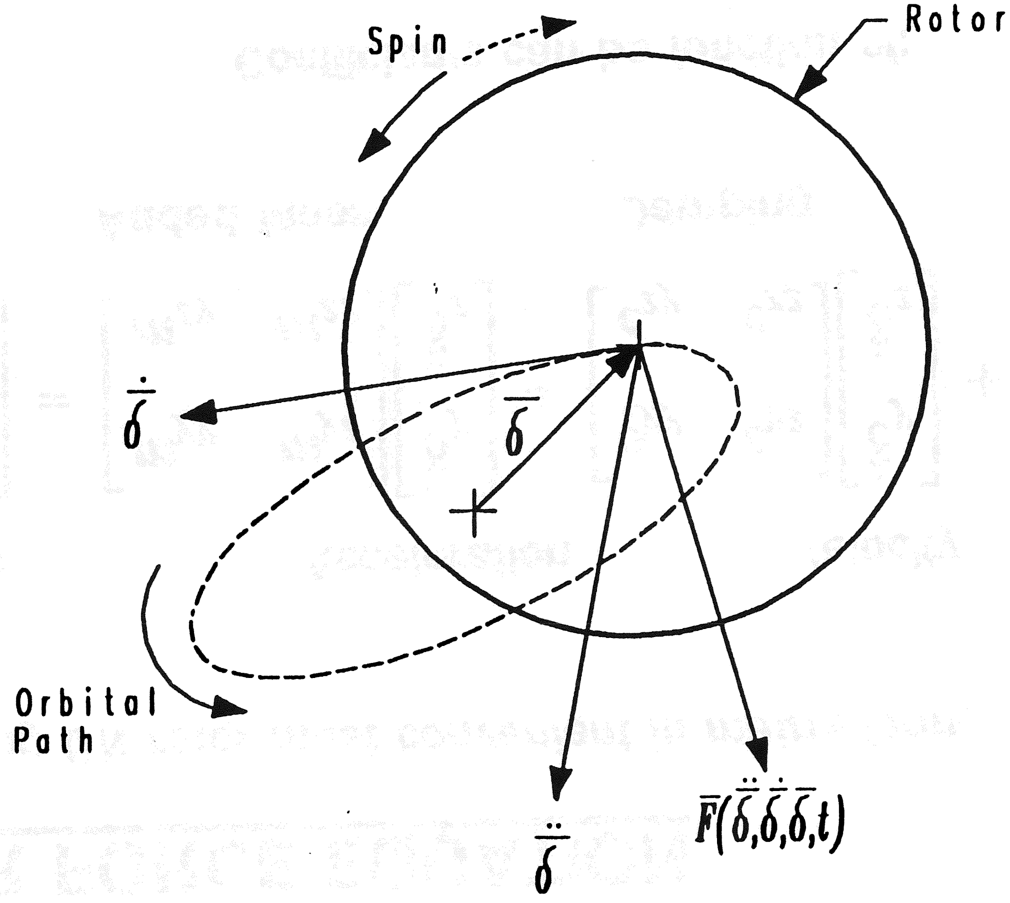
Rubs

Trapped fluids in Shaft

Material Hysteresis

FORCE ON ROTOR

Force acting on rotor depends on displacement and its time derivatives



ROTOR FORCE EQUATION

Force acting ON rotor most convenient in matrix form

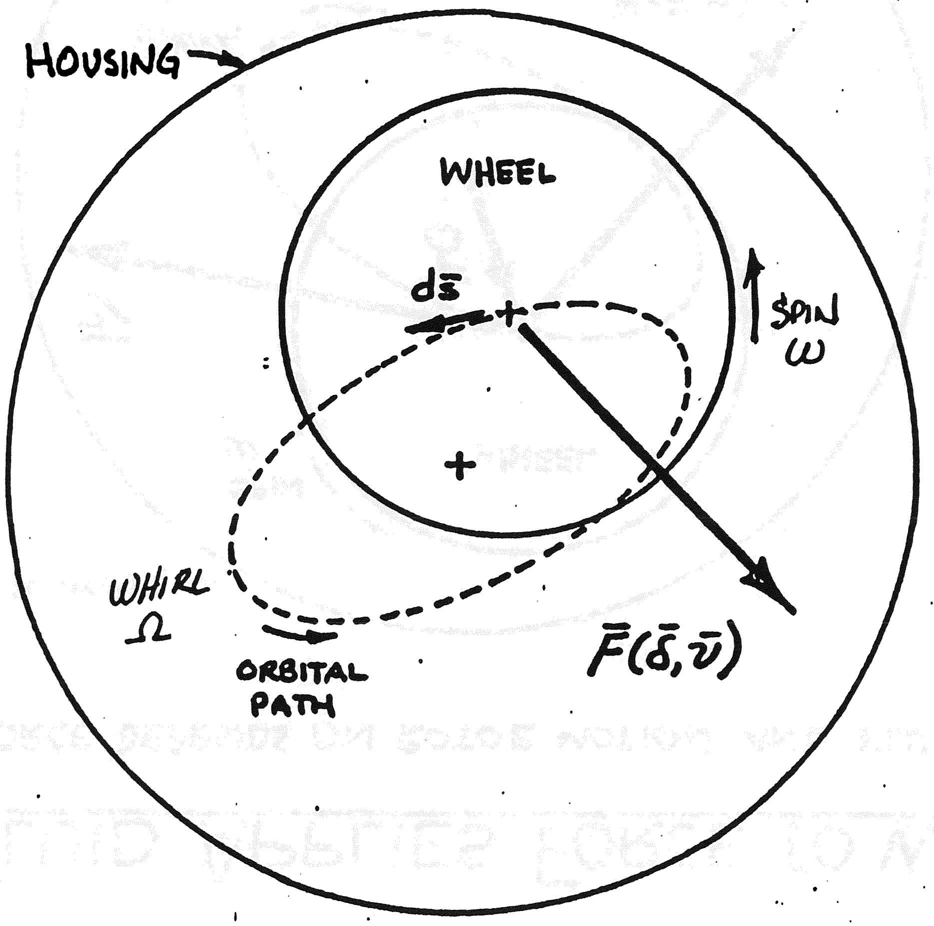
$$-\overset{\text{Force}}{\begin{bmatrix} F_y \\ F_z \end{bmatrix}} = \underset{\text{Added Mass}}{\begin{bmatrix} m_{yy} & m_{yz} \\ m_{zy} & m_{zz} \end{bmatrix}} \overset{\text{Acceleration}}{\begin{bmatrix} \ddot{\delta}_y \\ \ddot{\delta}_z \end{bmatrix}} + \underset{\text{Damping}}{\begin{bmatrix} c_{yy} & c_{yz} \\ c_{zy} & c_{zz} \end{bmatrix}} \overset{\text{Velocity}}{\begin{bmatrix} \dot{\delta}_y \\ \dot{\delta}_z \end{bmatrix}} + \underset{\text{Stiffness}}{\begin{bmatrix} k_{yy} & k_{yz} \\ k_{zy} & k_{zz} \end{bmatrix}} \overset{\text{Displacement}}{\begin{bmatrix} \delta_y \\ \delta_z \end{bmatrix}}$$

Coefficients can be function of:

- Spin speed
- Geometry
- Fluid properties
- Material properties
- Displacement (nonlinear)

FLUID FORCE DOES WORK ON WHEEL

ALGEBRAIC SIGN OF WORK DETERMINES STABILIZING NATURE OF FORCE



WORK DONE BY FLUID FORCE

$$W = \int_{\text{ORBIT}} \bar{F}(\bar{\delta},\bar{v}) \bullet d\bar{s}$$

$W < 0$: STABILIZING FORCE

ENERGY IS **DISSIPATED** FROM SYSTEM

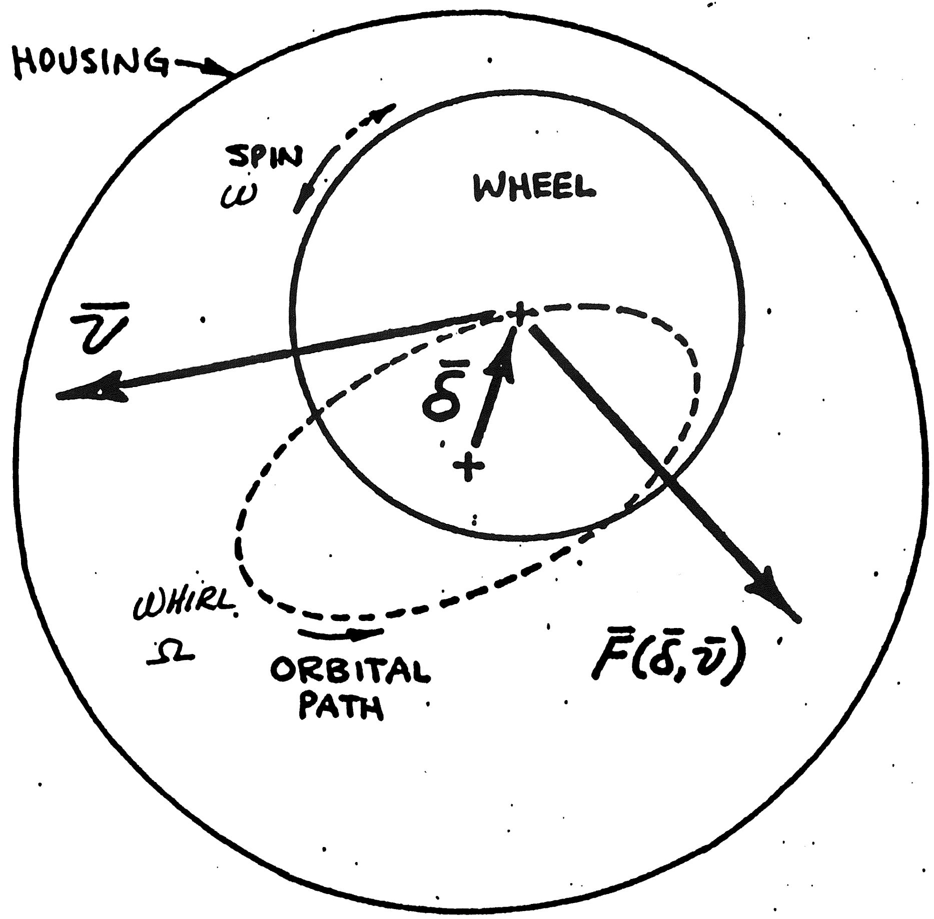
$W > 0$: DESTABILIZING FORCE

ENERGY IS **ADDED** TO SYSTEM

AT INSTANT SHOWN, FLUID FORCE IS STABILIZING SINCE COMPONENT OF $\bar{F}$ ALONG PATH OPPOSES $d\bar{s}$.

FLUID APPLIES FORCE TO WHEEL

FORCE DEPENDS ON ROTOR MOTION AND FLUID BEHAVIOR



$\bar{\delta}$ DISPLACEMENT VECTOR

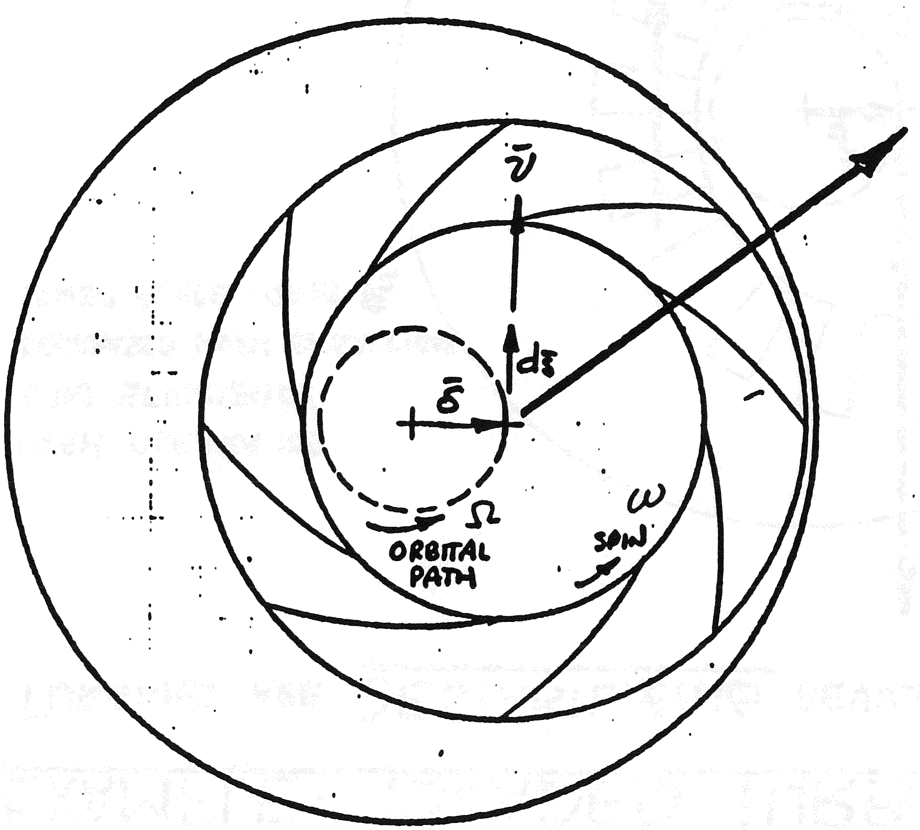
$\bar{v}$ VELOCITY VECTOR

$\bar{F}(\bar{\delta},\bar{v})$ FLUID FORCE ON WHEEL

- ALL VARY WITH TIME AS WHEEL TRAVERSES ORBITAL PATH
- $\bar{F}(\bar{\delta},\bar{v})$ IS OFTEN A VERY COMPLEX MATHEMATICAL QUANTITY, DIFFICULT TO QUANTIFY

EXAMPLE - CENTRIFUGAL IMPELLER

IMPELLERS ARE DESTABILIZING DEVICES - THEY ADD ENERGY



$\bar{F}(\bar{\delta},\bar{v})$: EMPIRICALLY DETERMINED BY TEST ON SHROUDED IMPELLER AT CALIFORNIA INSTITUTE OF TECHNOLOGY, IN DIRECTION SHOWN

SINCE $\bar{F}(\bar{\delta},\bar{v})$ HAS A COMPONENT IN THE SAME DIRECTION AS $d\bar{s}$

$$W > 0$$

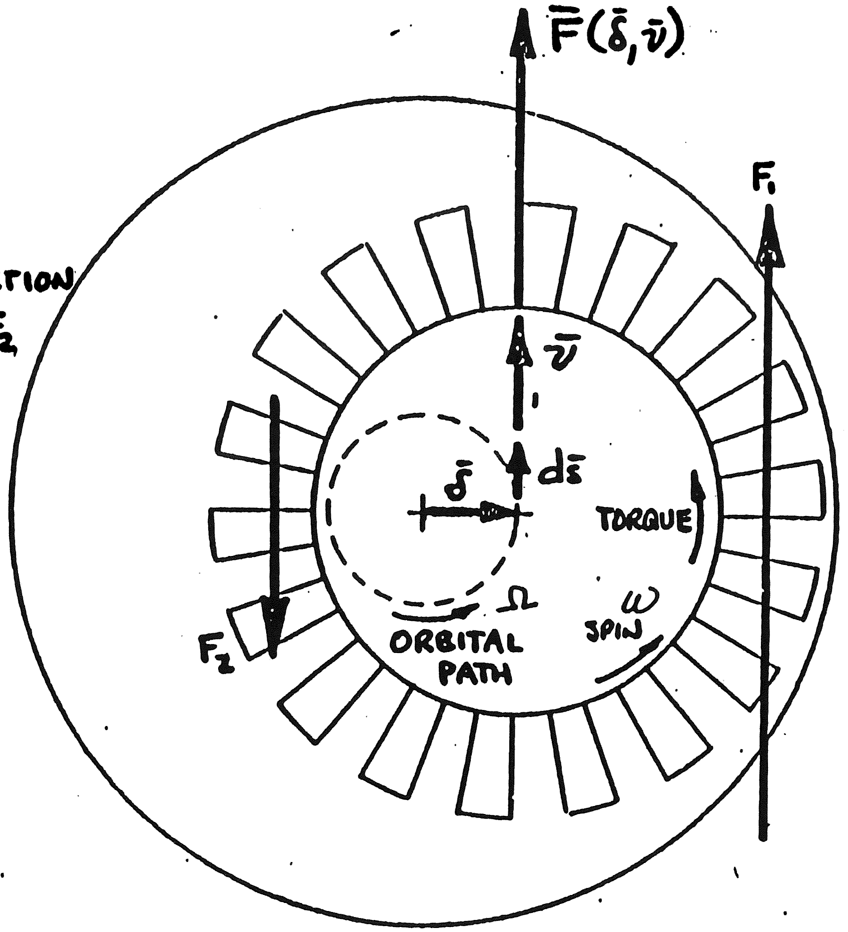
$\therefore$ DESTABILIZING

NOTE THAT RADIAL COMPONENT OF $\bar{F}(\bar{\delta},\bar{v})$ IS IN SAME DIRECTION AS $\bar{\delta}$ WHICH IS EQUIVALENT TO A NEGATIVE SPRING, SOFTENING THE SYSTEM AND LOWERING CRITICAL SPEEDS.

EXAMPLE - BLADED TURBINE DISK

TURBINES ARE **DESTABILIZING** DEVICES - THEY **ADD** ENERGY

HIGH CLEARANCE
LOW EFFICIENCY
DECREASED WORK EXTRACTION
LOWER BLADE LOADS F_2



LOW CLEARANCE
HIGH EFFICIENCY
INCREASED WORK EXTRACTION
HIGHER BLADE LOADS F_1

$\therefore \bar{F}(\bar{\delta},\bar{v}) = F_1 - F_2$

SINCE $\bar{F}(\bar{\delta},\bar{v})$ IS IN THE SAME DIRECTION AS $d\bar{s}$

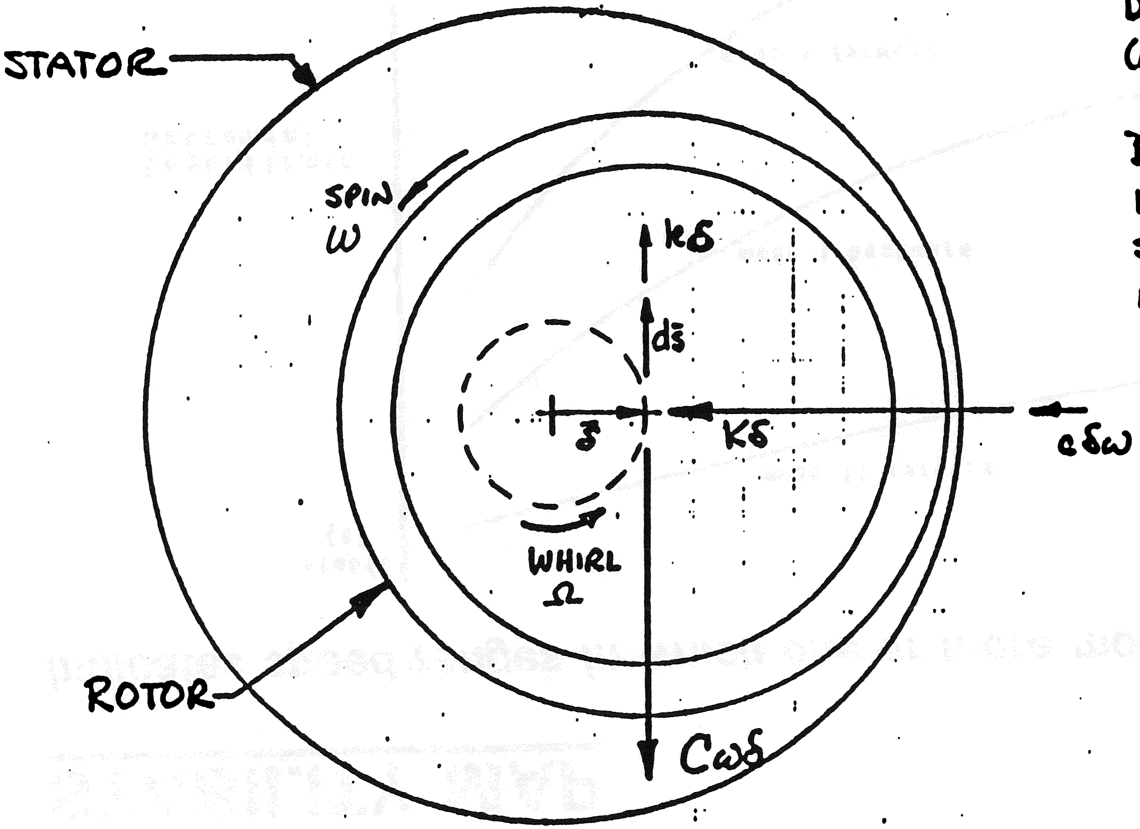
$W > 0$

DESTABILIZING

IN THIS APPLICATION $\bar{F}(\bar{\delta},\bar{v})$ IS ALSO KNOWN AS "ALFORD'S FORCE"

EXAMPLE - DAMPER SEALS

DAMPER SEALS ARE **STABILIZING** DEVICES - THEY **DISSIPATE** ENERGY



DUE TO HIGH AXIAL ΔP, FLOW (AXIAL) IS TURBULENT.

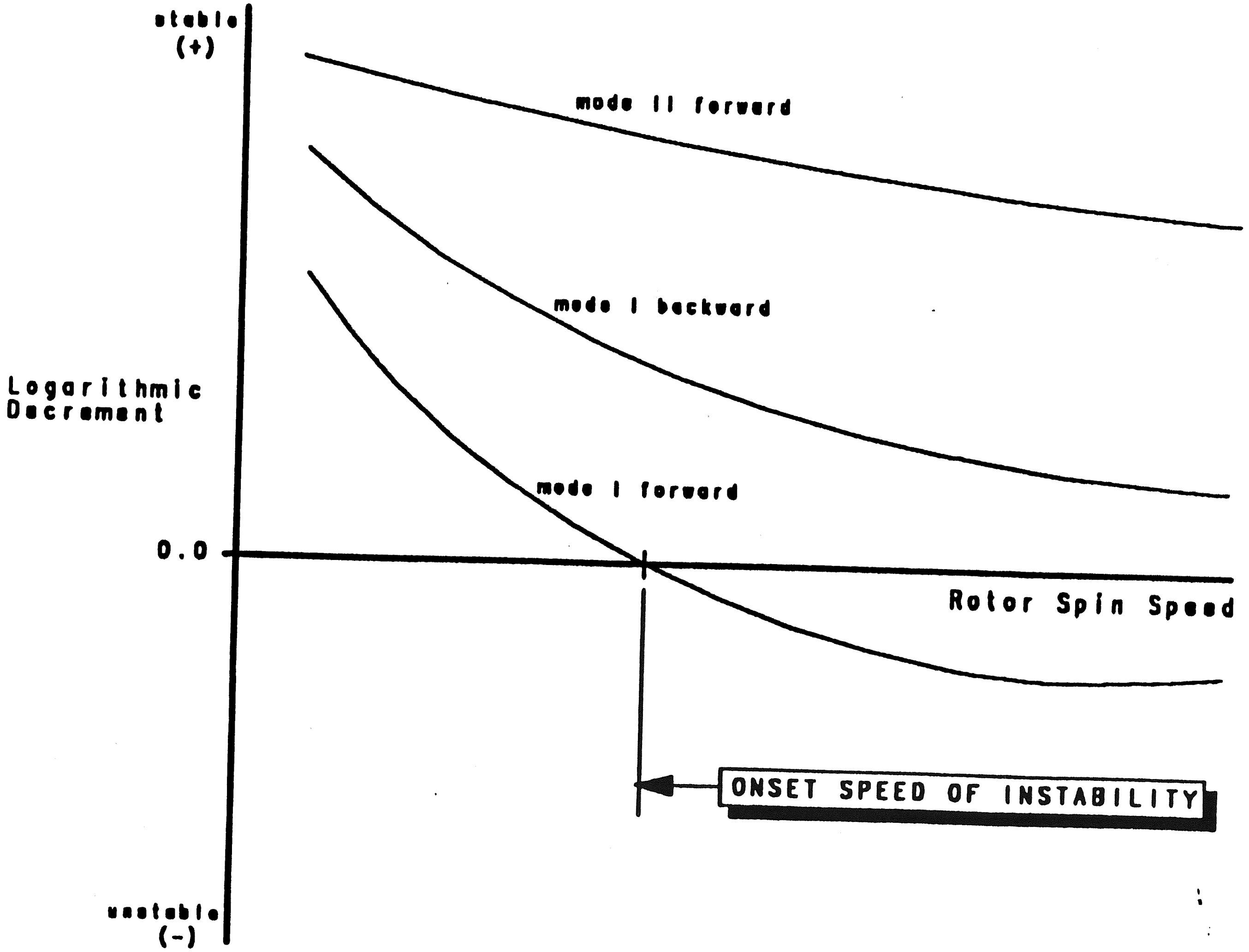
DIRECT DAMPING FORCE $C\omega\delta$ IS MUCH GREATER THAN CROSS-COUPLED STIFFNESS ~~FORCE~~ $k\delta$ THUS NET $\bar{F}(\bar{\delta},\bar{v})$ HAS COMPONENT **OPPOSITE** THE DIRECTION OF $d\bar{s}$.

$$W < 0$$

$\therefore$ STABILIZING

STABILITY MAP

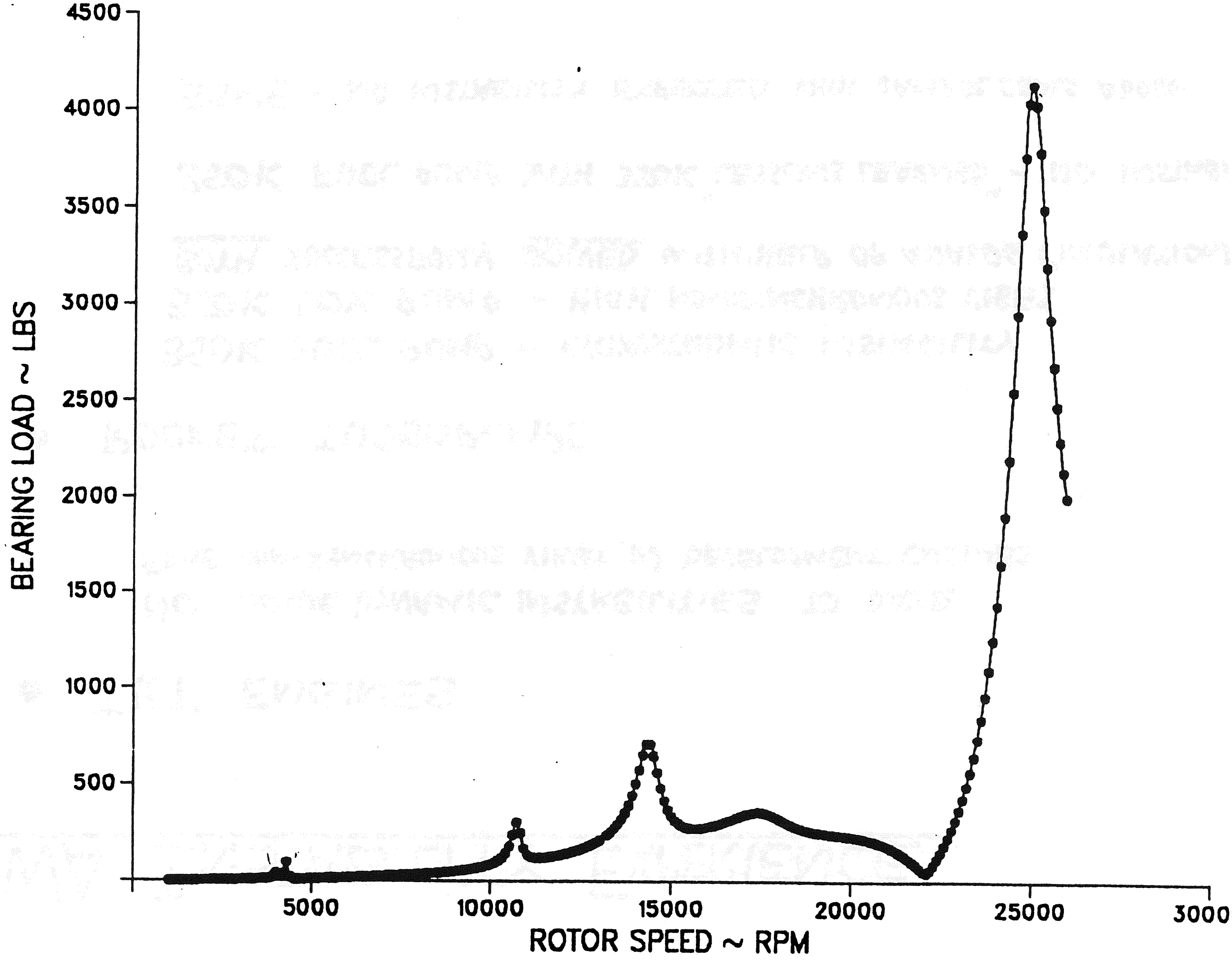
Indicates speed ranges in which one or more modes are unstable



HPFTP ROLLER BEARING LOAD

PWA ANALYSIS - INCLUDES SWIRL BRAKE

ROTOR + HOUSING - ARDS



KB=0.4756E6, KR=1.75E6
6/16/88 SKM

PWA INSTABILITY EXPERIENCE

- JET ENGINES

  NO ROTOR DYNAMIC INSTABILITIES TO DATE
  SOME NON-SYNCHRONOUS VIBES IN DEVELOPMENT ENGINES

- ROCKET TURBOPUMPS

  350K FUEL PUMP - CATASTROPHIC INSTABILITY
  350K LOX PUMP - HIGH NONSYNCHRONOUS VIBES
  BOTH SUCCESSFULLY SOLVED WITH HELP OF ANALOG SIMULATION

  250K FUEL PUMP WITH 350K "LESSONS LEARNED" - NO INSTABILITY

  SSME - NO INSTABILITY EXPECTED WITH DAMPER SEALS ADDED

- TECHNOLOGY PROGRAMS

  - U.S. ARMY CONTRACT - SPLINE FRICTION INSTABILITY
    SOLVED BY PROPER SPLINE LUBRICATION
  - PW3005 ROTOR DYNAMICS RIG

SUMMARY

- INSTABILITY MECHANISM UNDERSTOOD
- MECHANISM DIFFICULT TO QUANTIFY
- DESIGN TO AVOID INSTABILITY TRENDS
- MINIMIZE INSTABILITIES IN DEVELOPMENT

PRACTICAL EXAMPLE – SSME

SSME–ATD ROTORDYNAMICS

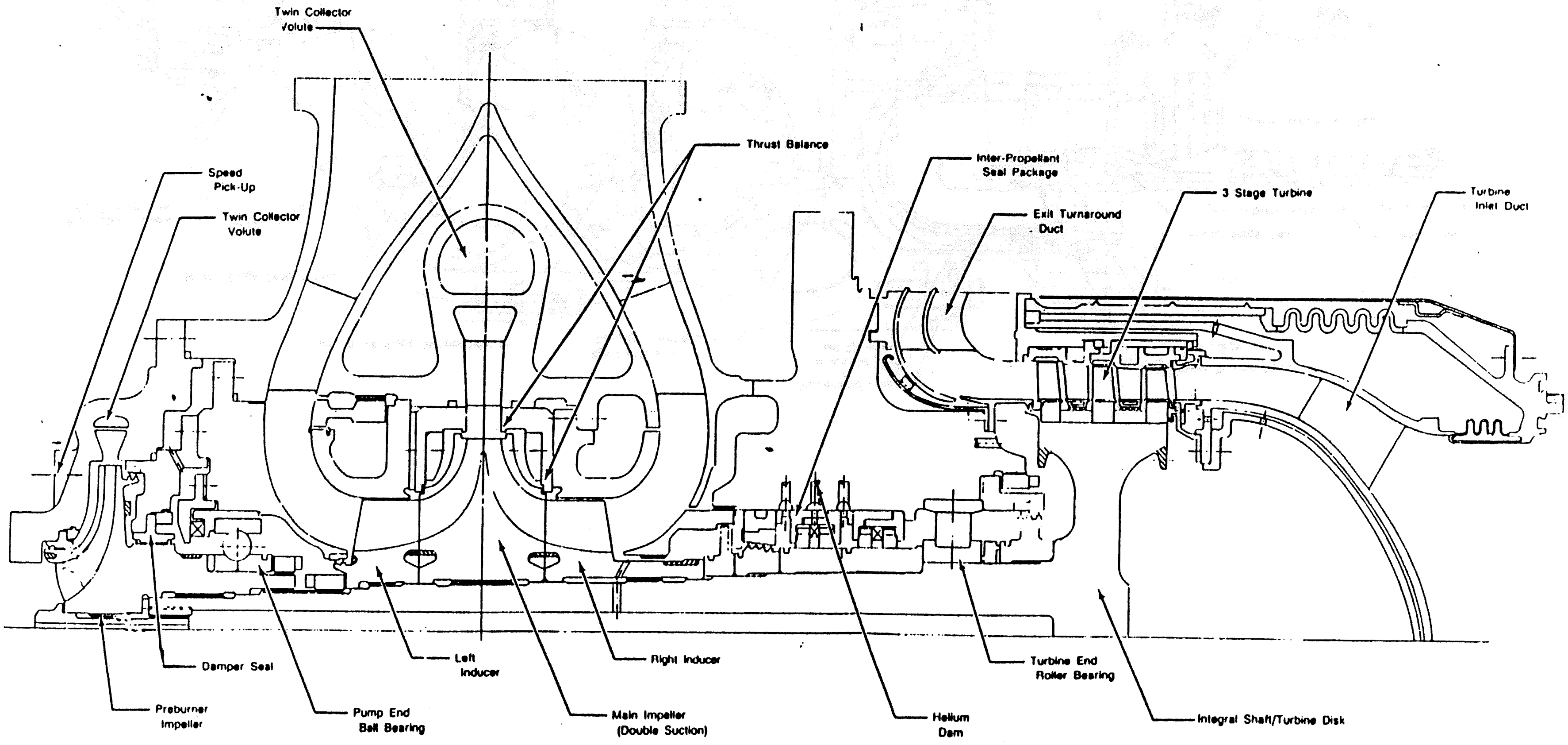
☐ High Pressure Oxidizer Turbopump (HPOTP)

☐ High Pressure Fuel Turbopump (HPFTP)

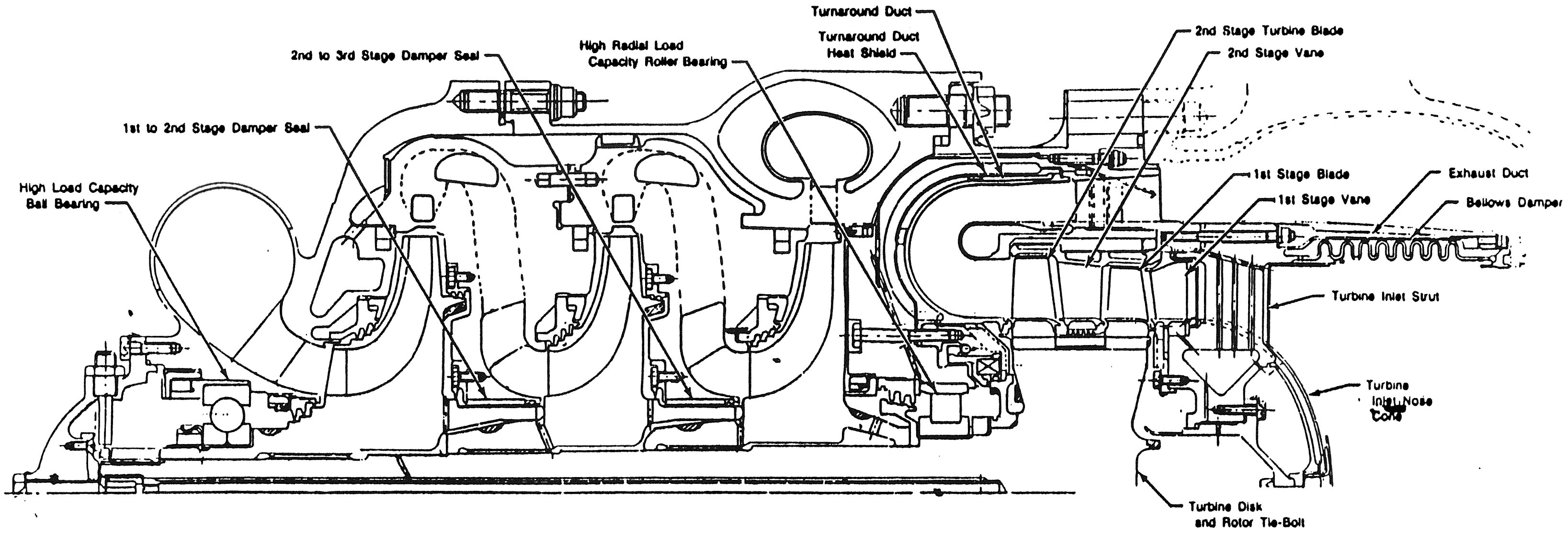
SSME–ATD ROTORDYNAMICS

- ☐ Stiff Rotor, Bearings, & Supports
- ☐ Major Components Double Piloted to Thru Tiebolt
- ☐ Tiebolt Shaft Integral with Turbine Rotor
- ☐ Roughened Stator Damper Seals
- ☐ Two Plane Balance of Assembly & Details
- ☐ Fundamental Rotor Bending Modes > 50% Above Design Speed

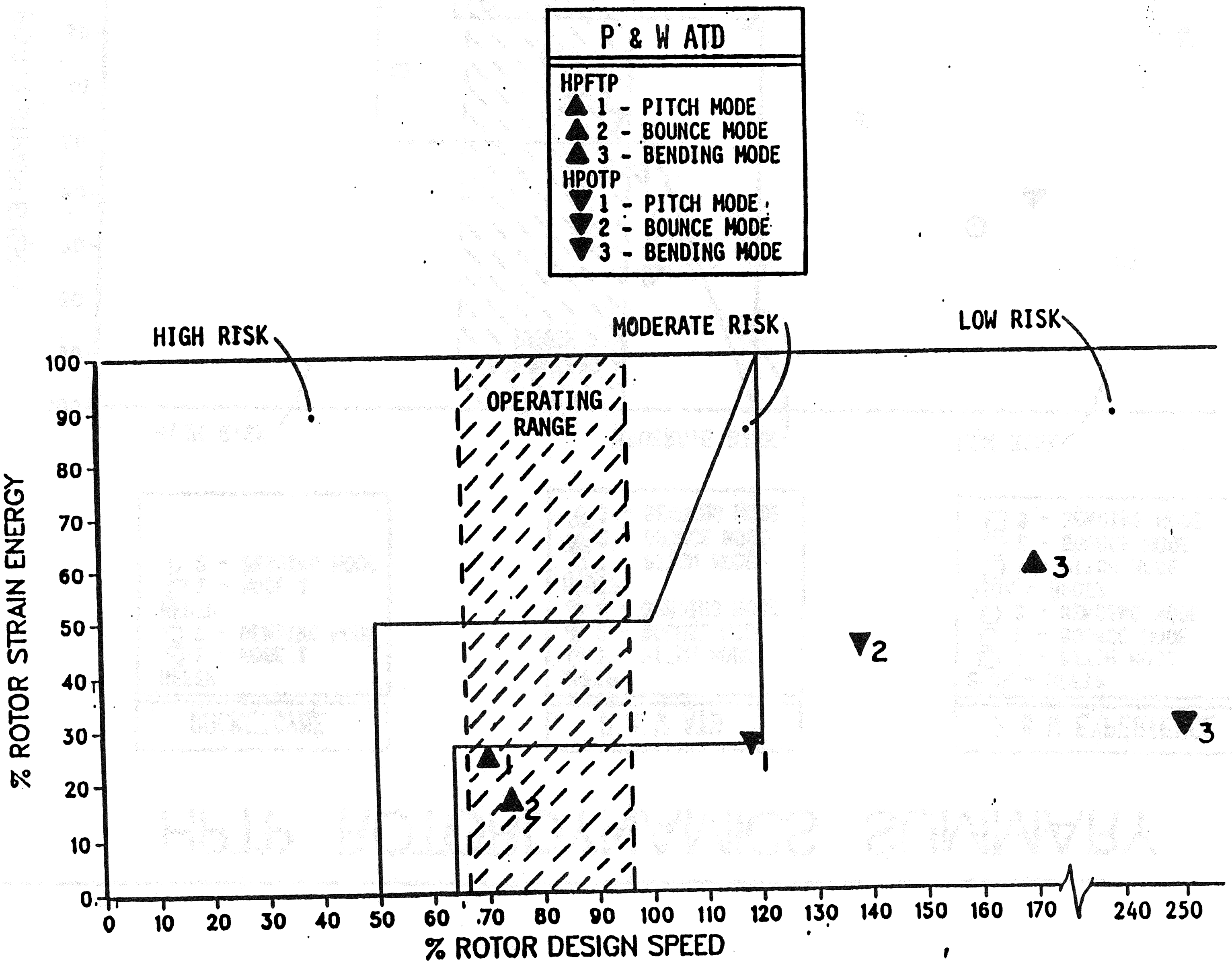
SSME—ATD HPOTP



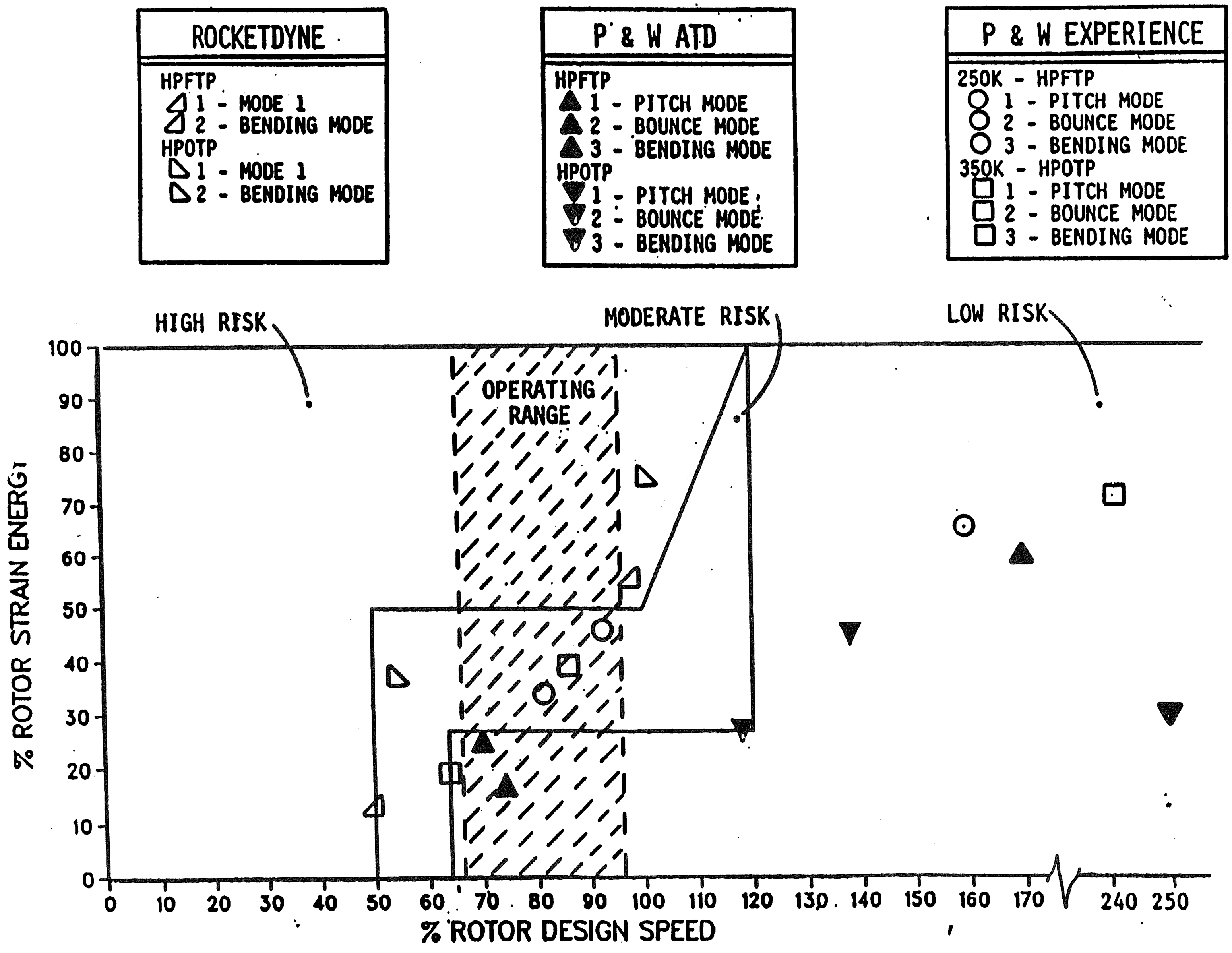
SSME—ATD HPFTP



HPTP ROTORDYNAMICS SUMMARY

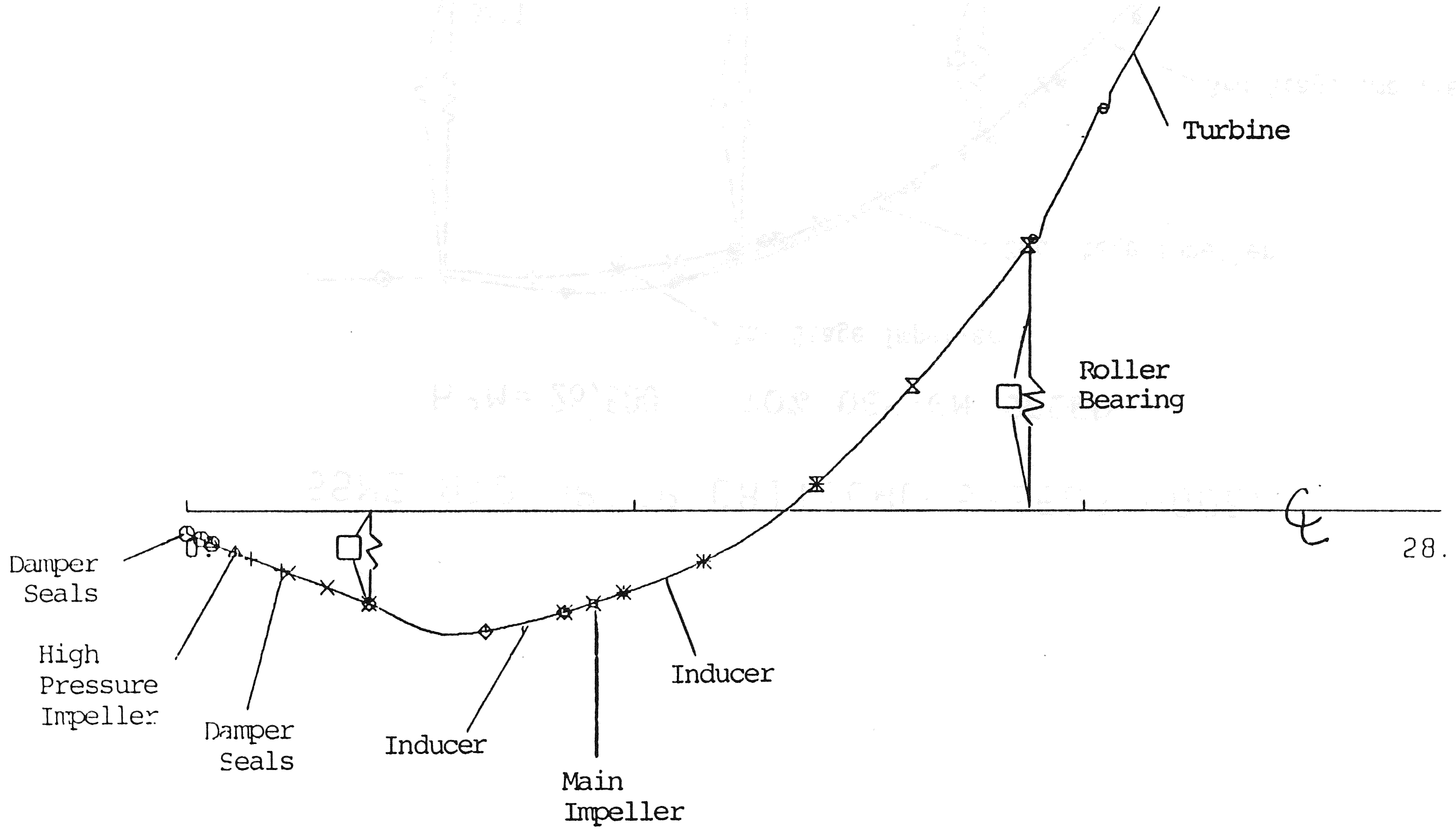


HPTP ROTORDYNAMICS SUMMARY



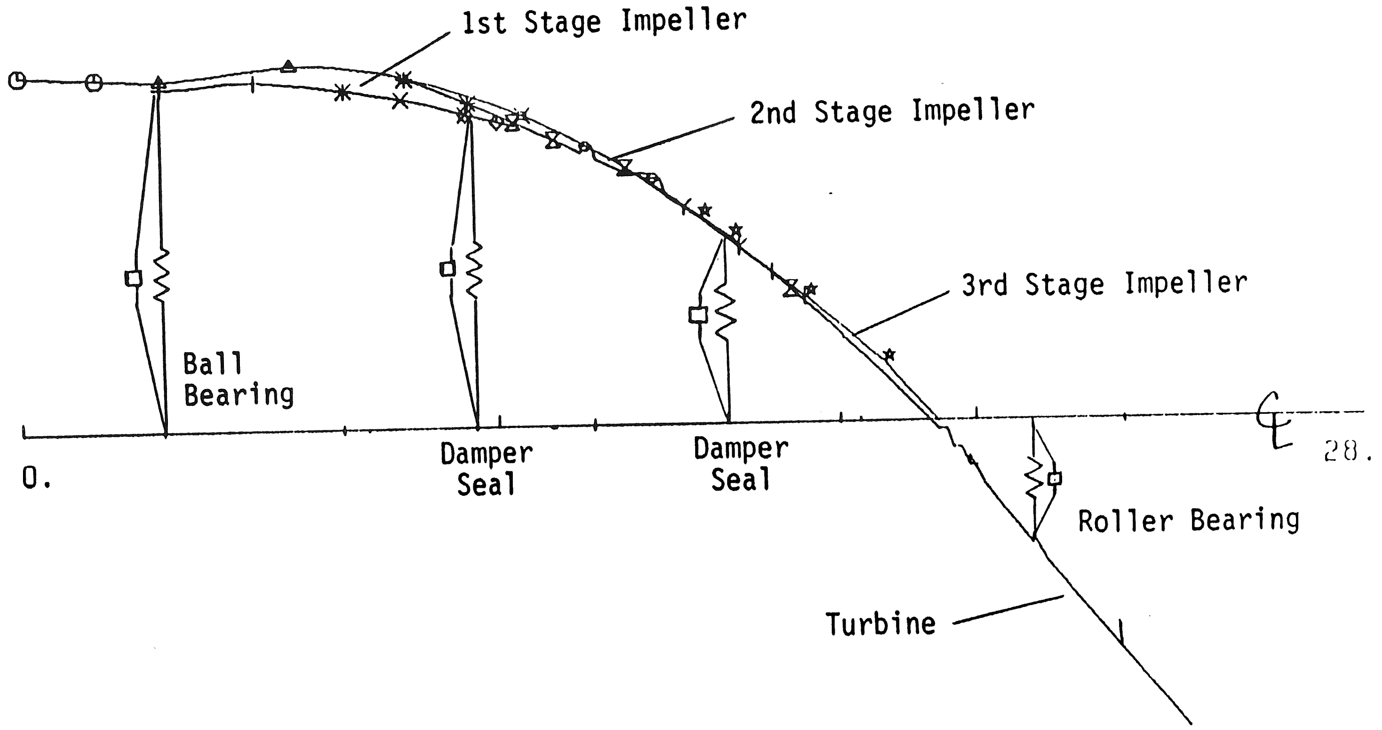
SSME-ATD HPOTP CRITICAL SPEEDS MODEL

RPM= 30,700 118 % DESISIGN SPEED



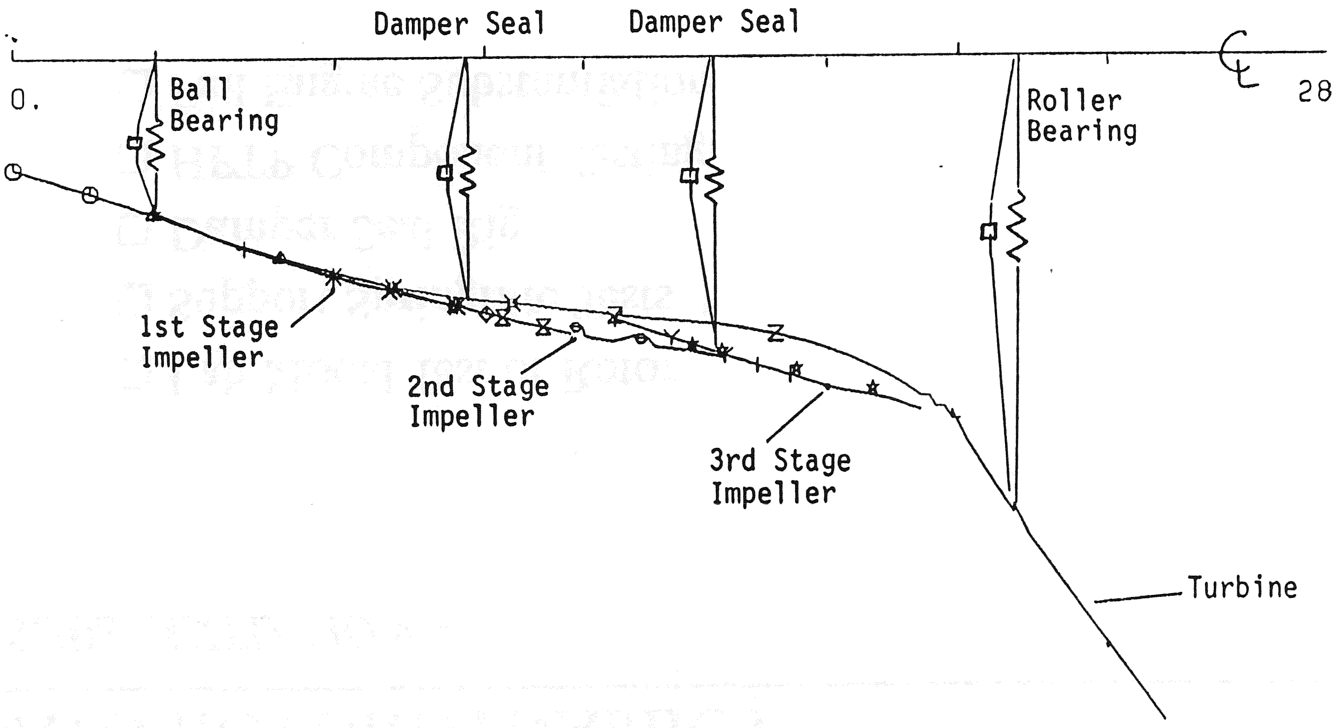
SSME-ATD HPFTP CRITICAL SPEEDS MODEL

RPM= 26,500 **70% DESIGN SPEED**



SSME-ATD HPFTP CRITICAL SPEEDS MODEL

RPM=31500 90% DESIGN SPEED



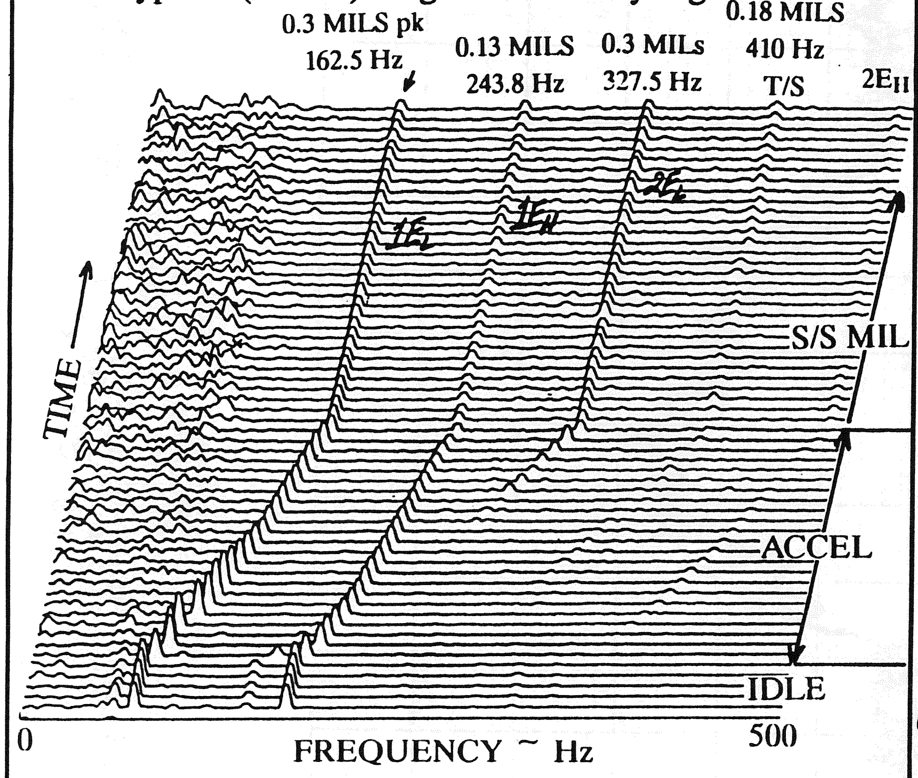
SSME–ATD ROTORDYNAMICS

DESIGN SUBSTANTIATION –

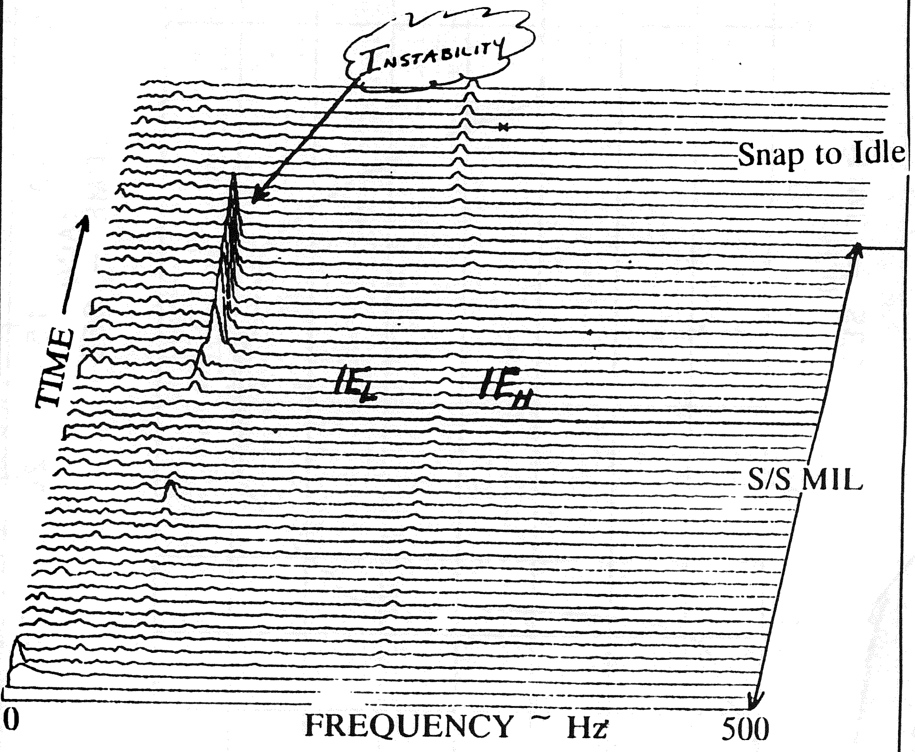
- ☐ Lab Modal Test of Rotor
- ☐ Support Springrate Tests
- ☐ Damper Seal Rig
- ☐ HPTP Component Testing
- ☐ Full Engine Substantiation

ROTORDYNAMIC INSTABILITY

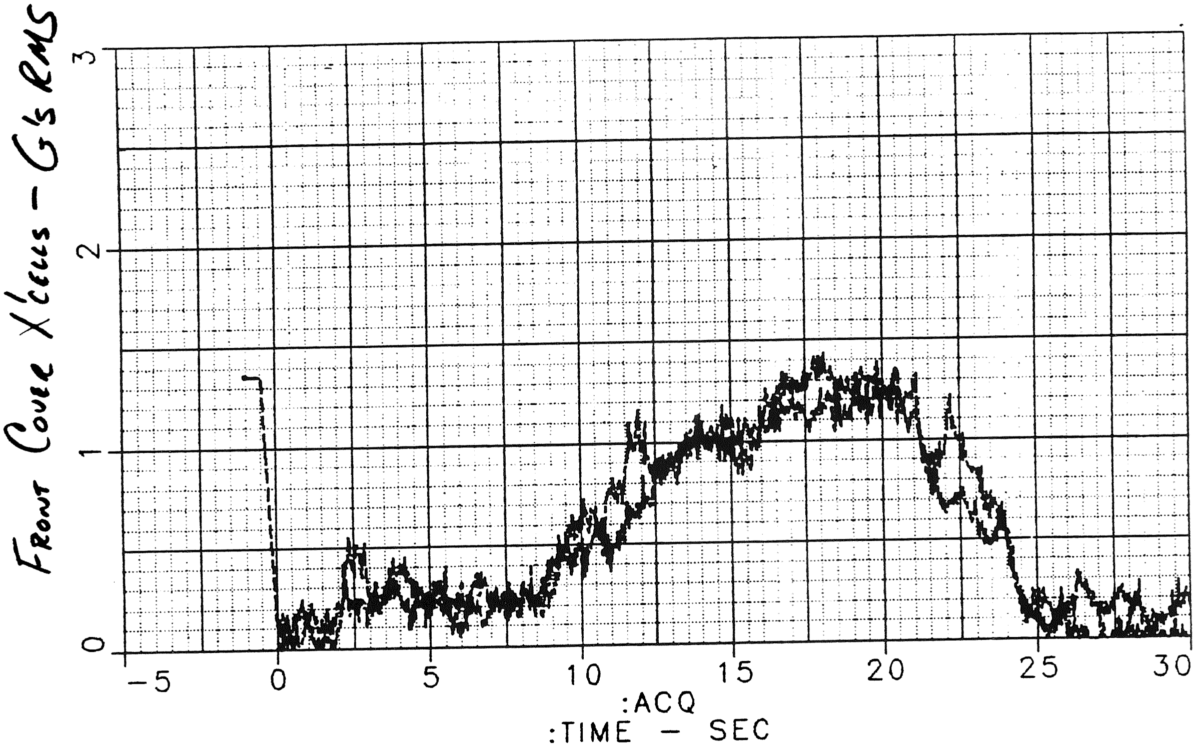
Typical (Stable) Engine Vibratory Signature

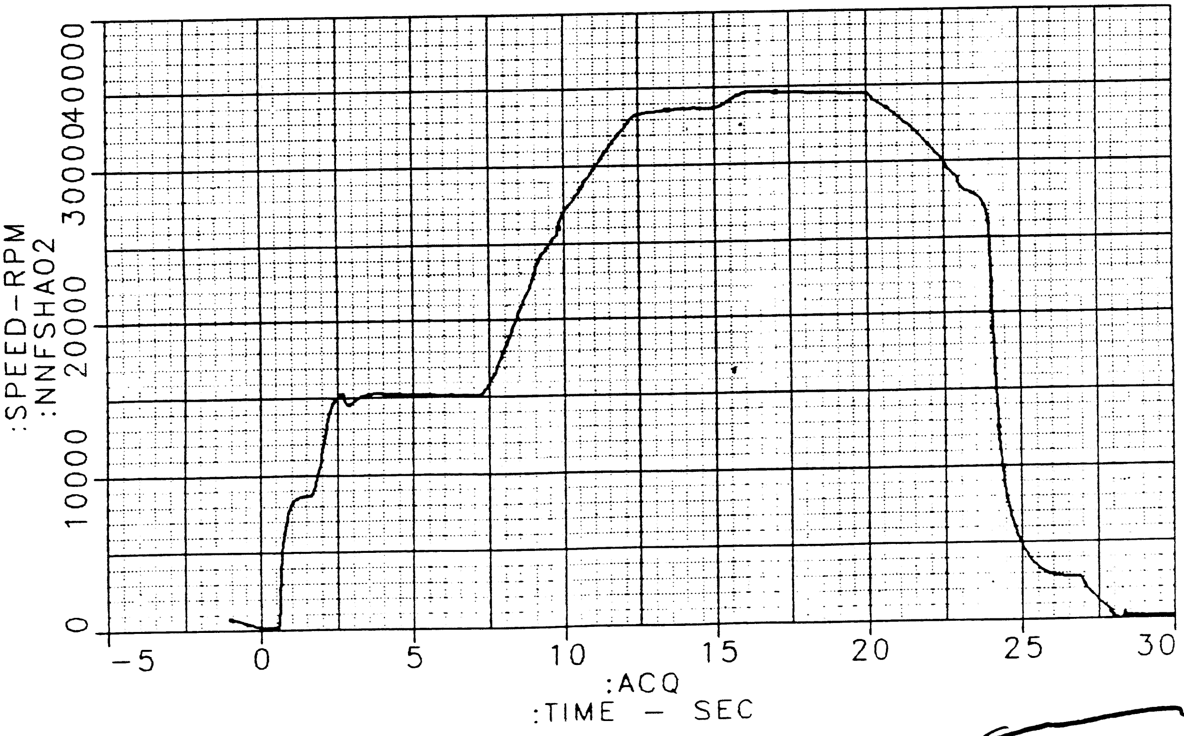


Unstable Engine Vibratory Signature



HPFTP UNIT 4-1 (XFE805-04)
TEST DATE 03/14/91 RUN 111 READING #695 AND 1723
2 XFE805 4 111 695 (0 E08 GPD HIGH FIT ST
______:NNFSHA02 :SPEED-RPM
------:VBFFCR35 :GS - RMS, BANDPASSED
-·-·-·:VBFFCR45 :GS - RMS, BANDPASSED

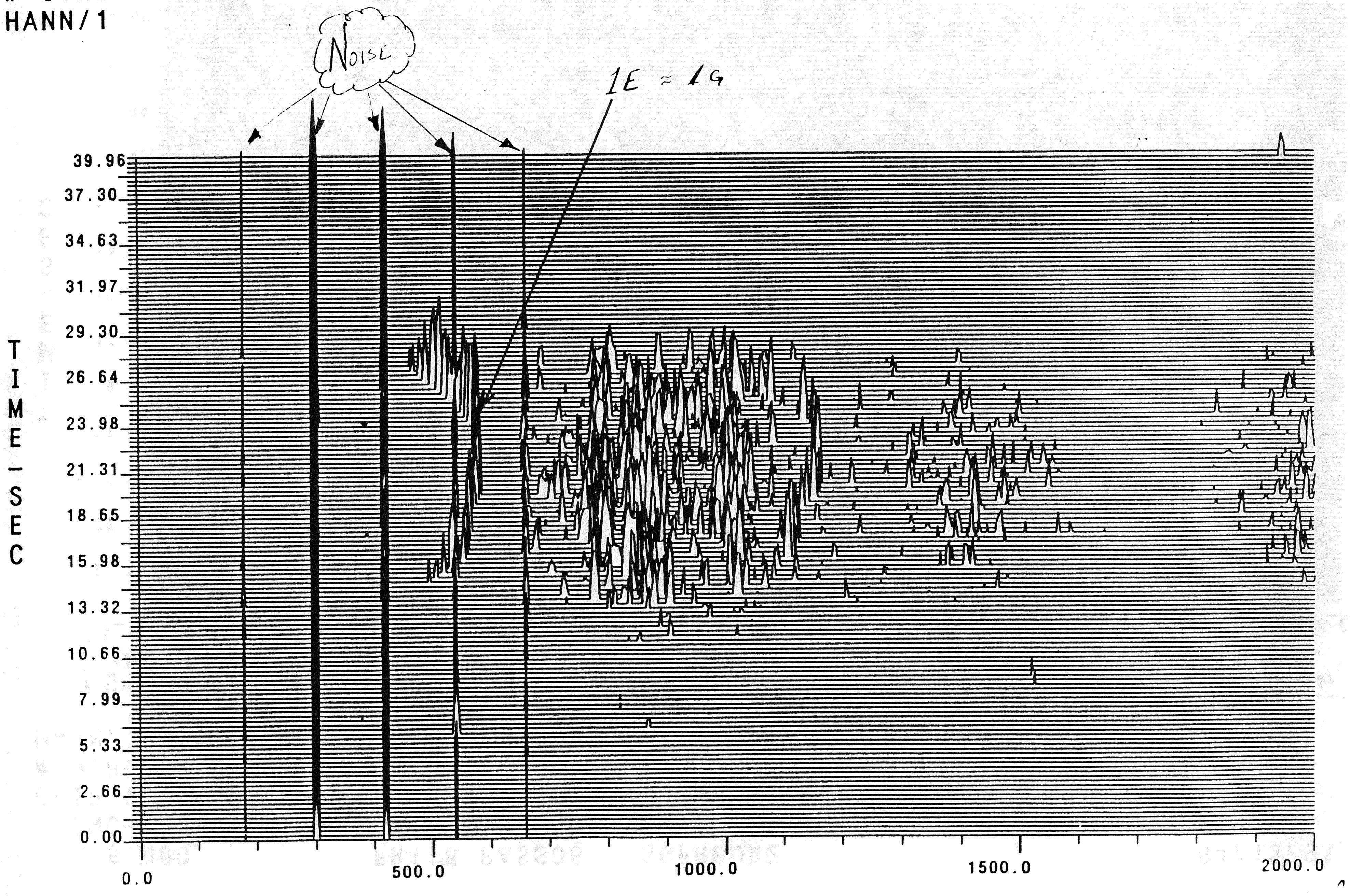




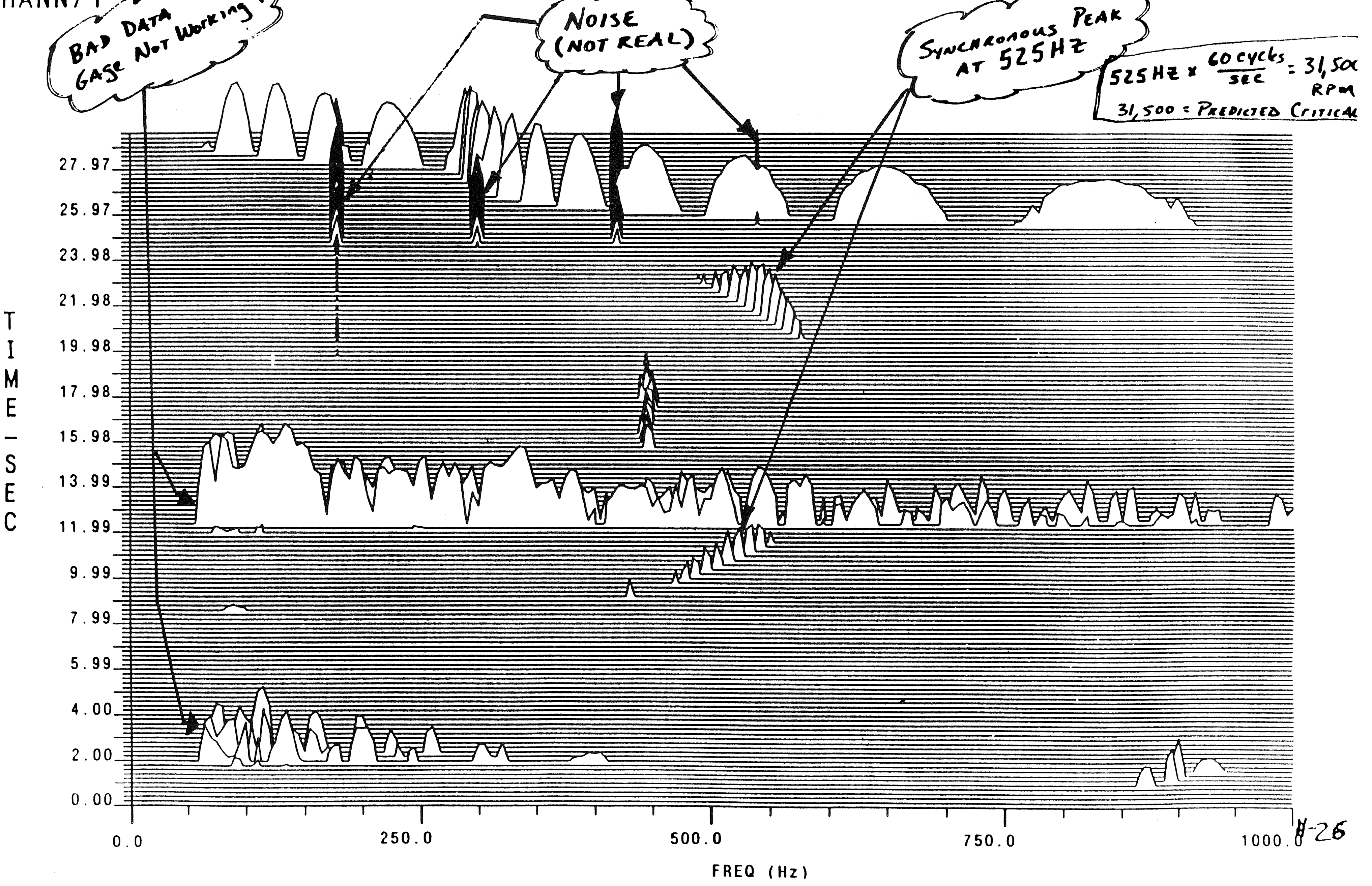
03/14/91 17:23:33
MONTGOMERY, S.

150-800 Hz Bandpass
Limit 12 G's RMS

BW= 5.000 HR111 PASS01 VBFFCR41 03/14/91
Y-INC=.266E+00 sec 4 PSDs SKIPPED <E8MC>
CLIP LEVEL = .289E+00 G-SQ/Hz LOG/ 70.% Freq. Range = 0.0 - 2000.0
OVRLPS= 3
HANN/1



5.000 FR118 PASS06 SGFRBOR2 04/13/91
Y-INC=.200E+00 sec 3 PSDs SKIPPED <E8MC>
CLIP LEVEL = .111E+01 K-SQ/Hz LOG/ 40.% Freq. Range = 0.0 - 1000.0
OVRLPS= 3
HANN/1



BAD DATA GAGE NOT WORKING?

NOISE (NOT REAL)

SYNCHRONOUS PEAK AT 525HZ

525 HZ x $\frac{60\ \text{cycles}}{\text{sec}}$ = 31,500 RPM
31,500 = PREDICTED CRITICAL

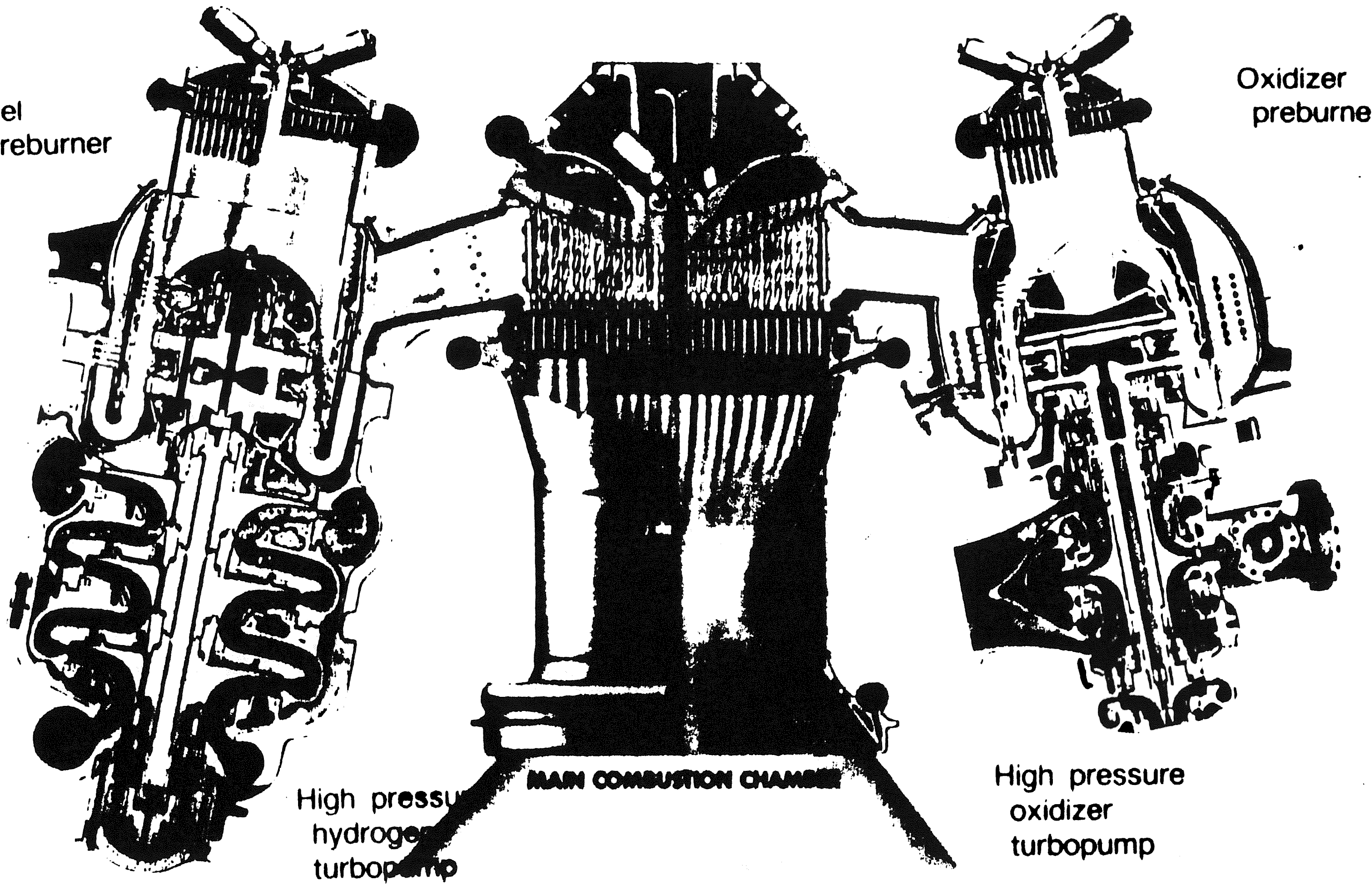
ROCKET COMBUSTION FUNDAMENTALS

- *Velocity Efficiency* (η_{c^*})
- *Fuel/Oxidizer Injection Methods*
- *Chamber Heat Transfer*
- *Low Frequency Instability*
- *High Frequency Instability*

JIM A. CLARK

ROCKET COMBUSTION SYSTEMS

Include Main Combustion Chamber And Preburners



SSME POWERHEAD

AV297817 850306 1547B

SELECTED LIQUID ROCKET PROPELLANTS

LIQUID OXIDIZERS

OXYGEN (O_2)
FLUORINE (F_2)

HYDROGEN PEROXIDE (H_2O_2)
CHLORINE TRIFLUORIDE (ClF_3)
NITRIC ACID (HNO_3)
NITROGEN TETROXIDE (N_2O_4)

LIQUID FUELS

HYDROGEN (H_2)
HYDROCARBONS

- METHANE (CH_4)
- ETHANE (C_2H_6)
- RP-1
- JP
- GASOLINE

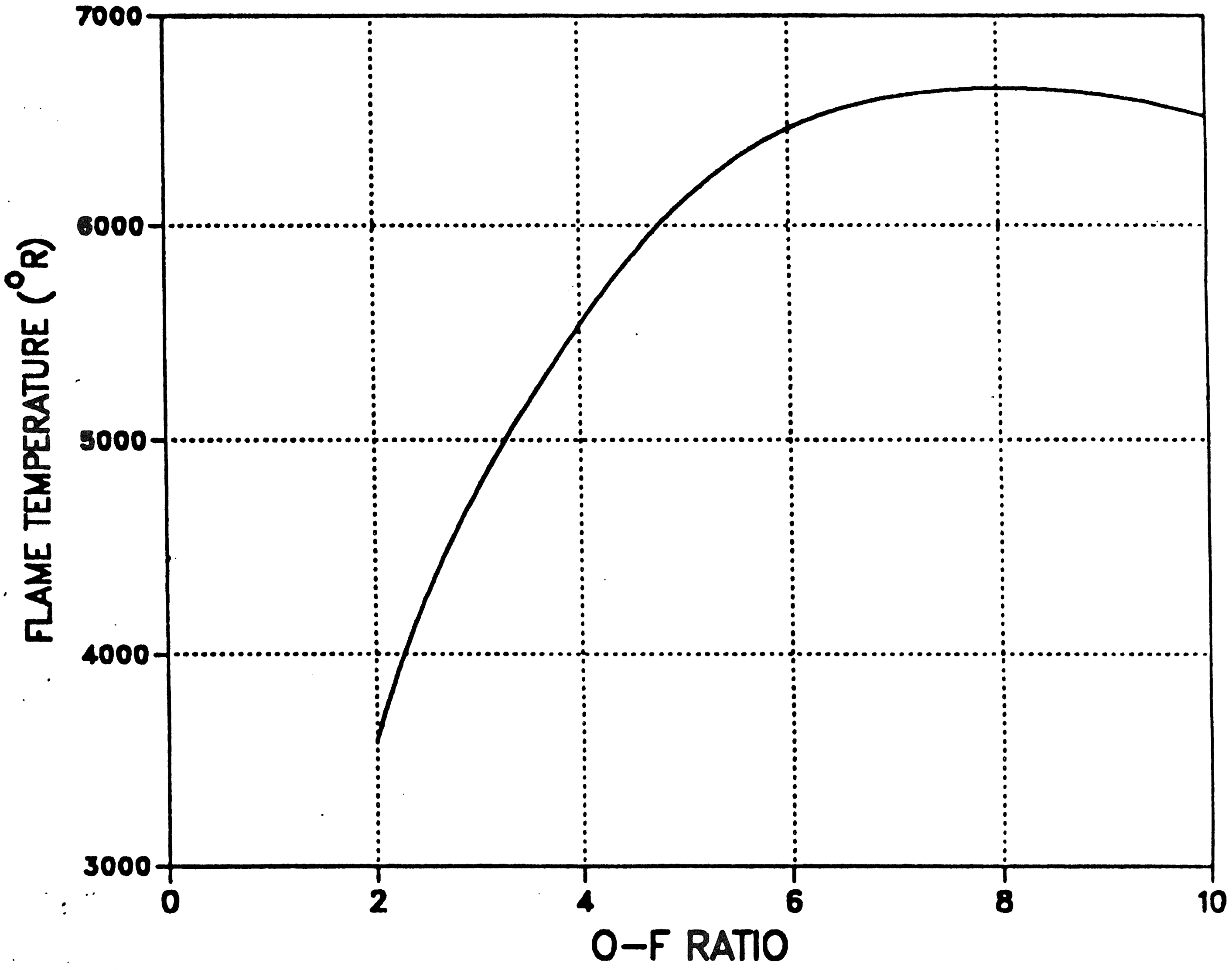
HYDRAZINE (N_2H_4)
ALCOHOLS
AMMONIA (NH_3)
UDHM

MONOPROPELLANTS

HYDRAZINE (N_2H_4)
HYDROGEN PEROXIDE-(H_2O_2)
NITROMETHANE (CH_3 NO_2)
ETHYLENE OXIDE

TYPICAL CRYOGENICS: LOX / H_2 , LOX / METHANE
TYPICAL STORABLES: N_2O_4/ UDMH & N_2H_4
TYPICAL HYPERGOLICS (SPONTANEOUSLY IGNITABLE): HYDRAZINE WITH AIR OR N_2O_4

HYDROGEN OXYGEN FLAME TEMPERATURE VS. O–F RATIO



η_{c^*} VS. η_c ACTUAL RELATIONSHIP

CHEMICAL EFFICIENCY $\eta_c = \frac{\Delta h_{actual}}{\Delta h_{ideal}} \approx \frac{\Delta T_{actual}}{\Delta T_{ideal}} \approx \frac{T_{2,\,actual}}{T_{2,\,ideal}}$

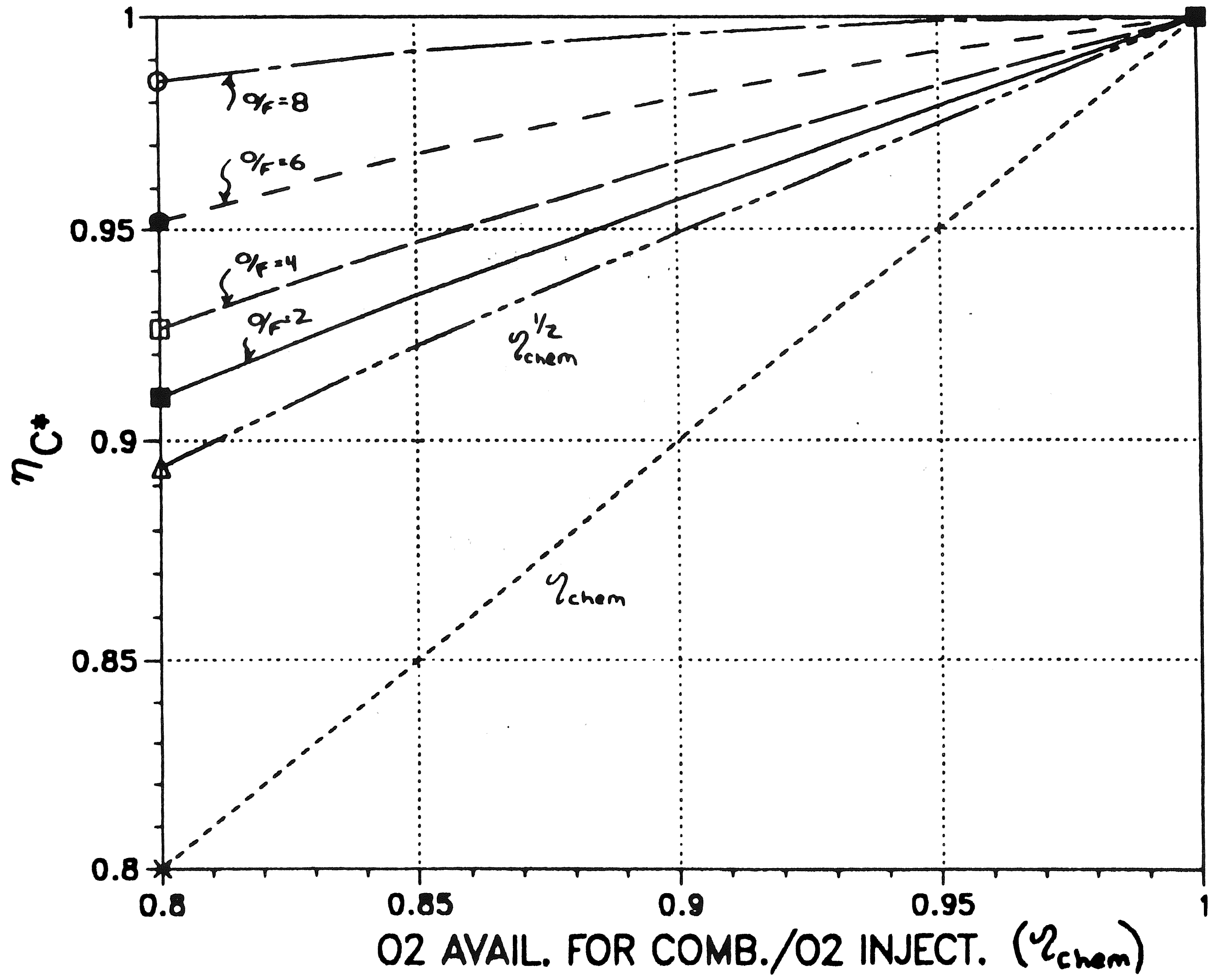
CHARACTERISTIC VELOCITY $C^{\cdot} = \frac{P_c A_t}{\dot{m}} \propto \sqrt{T_2}$

CHARACTERISTIC VELOCITY EFFICIENCY $\eta_{c^{\cdot}} = \frac{\sqrt{T_{2,\,actual}}}{\sqrt{T_{2,\,ideal}}}$

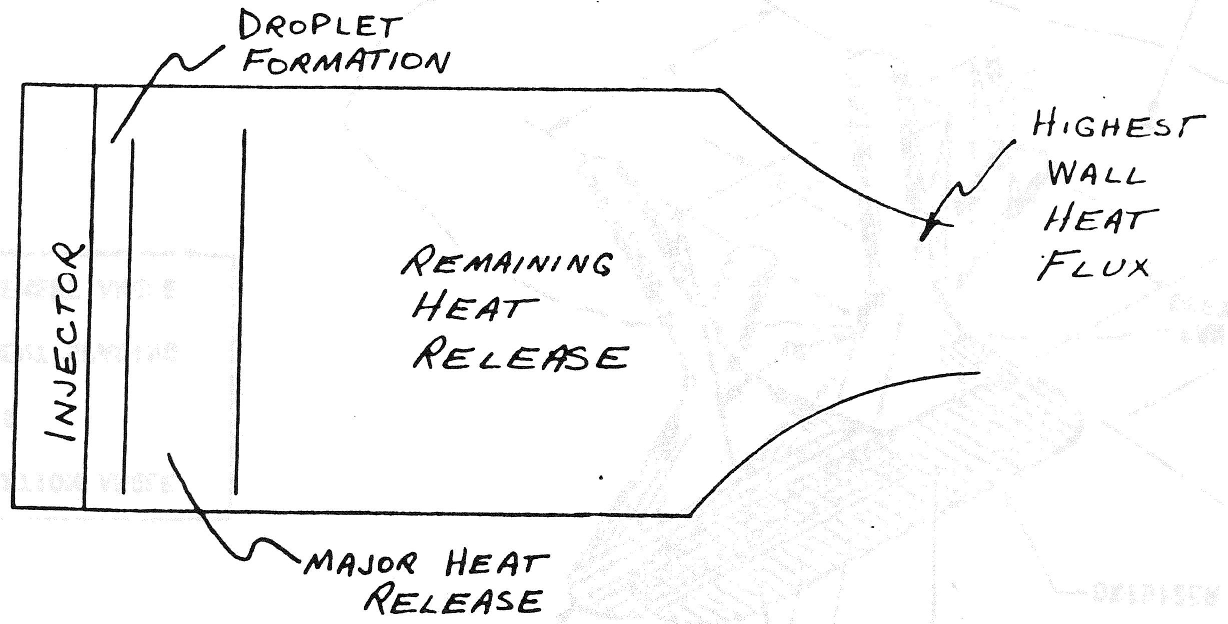
THUS $\eta_{c^{\cdot}} \approx \sqrt{\eta_c}$

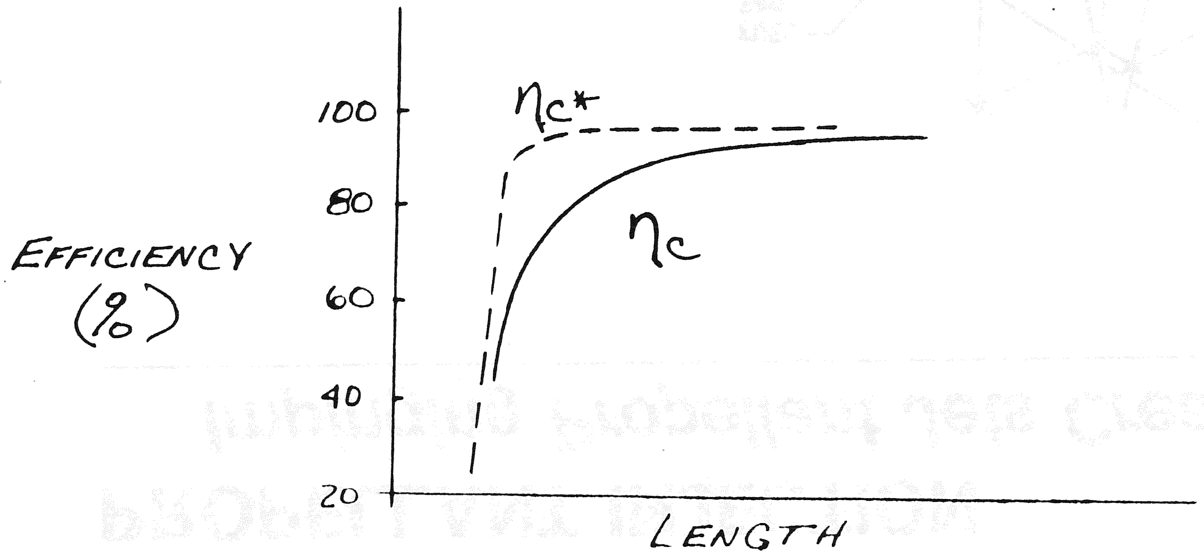
η_{C^*} VS. η_C ACTUAL RELATIONSHIP

$PC = 2300\ PSIA,\ TO2 = 195\,°\ R,\ TH2 = 175\,°\ R$



HEAT RELEASE PROFILE





PROPELLANT INJECTION

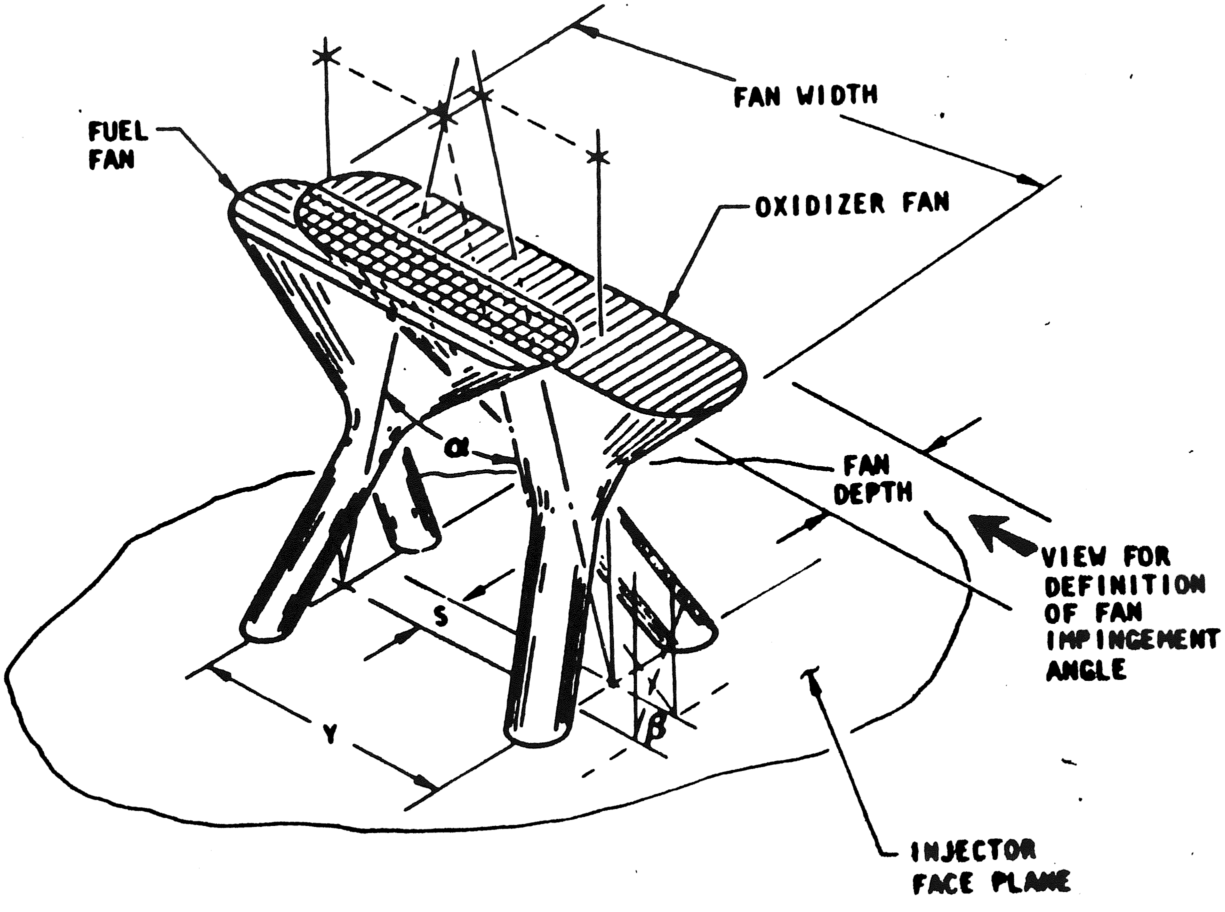
Impinging Propellant Jets Create a Spray Fan

α = FAN INCLINATION ANGLE

S = FAN SPACING

Y = INTRA-ELEMENT SPACING

β = FAN IMPINGEMENT ANGLE

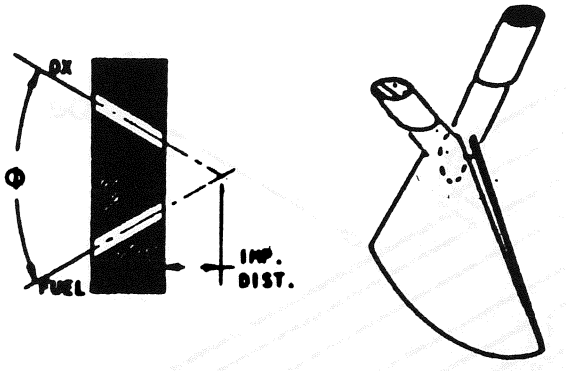


PROPELLANT INJECTION

Commonly Used Impinging Injection Elements

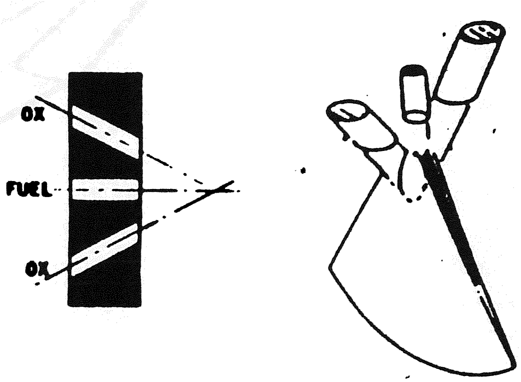
UNLIKE DOUBLET
(1 ON 1)

● ○



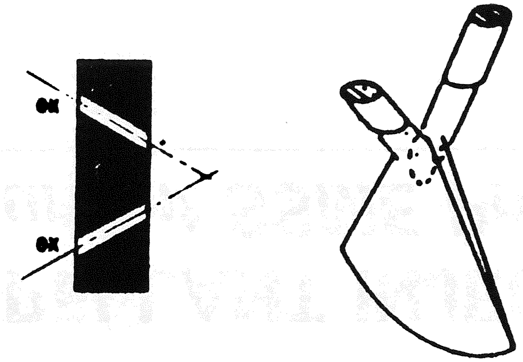
UNLIKE TRIPLET
(2 ON 1)

●○●



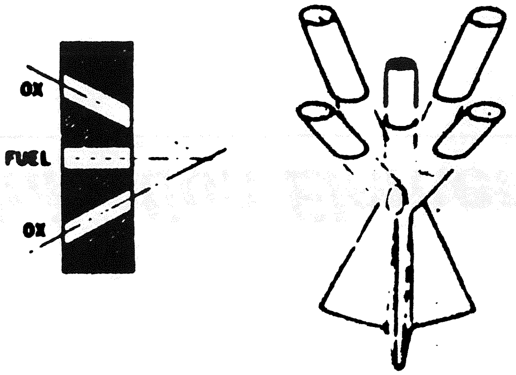
LIKE DOUBLET
(1 ON 1)

●● OR ○○



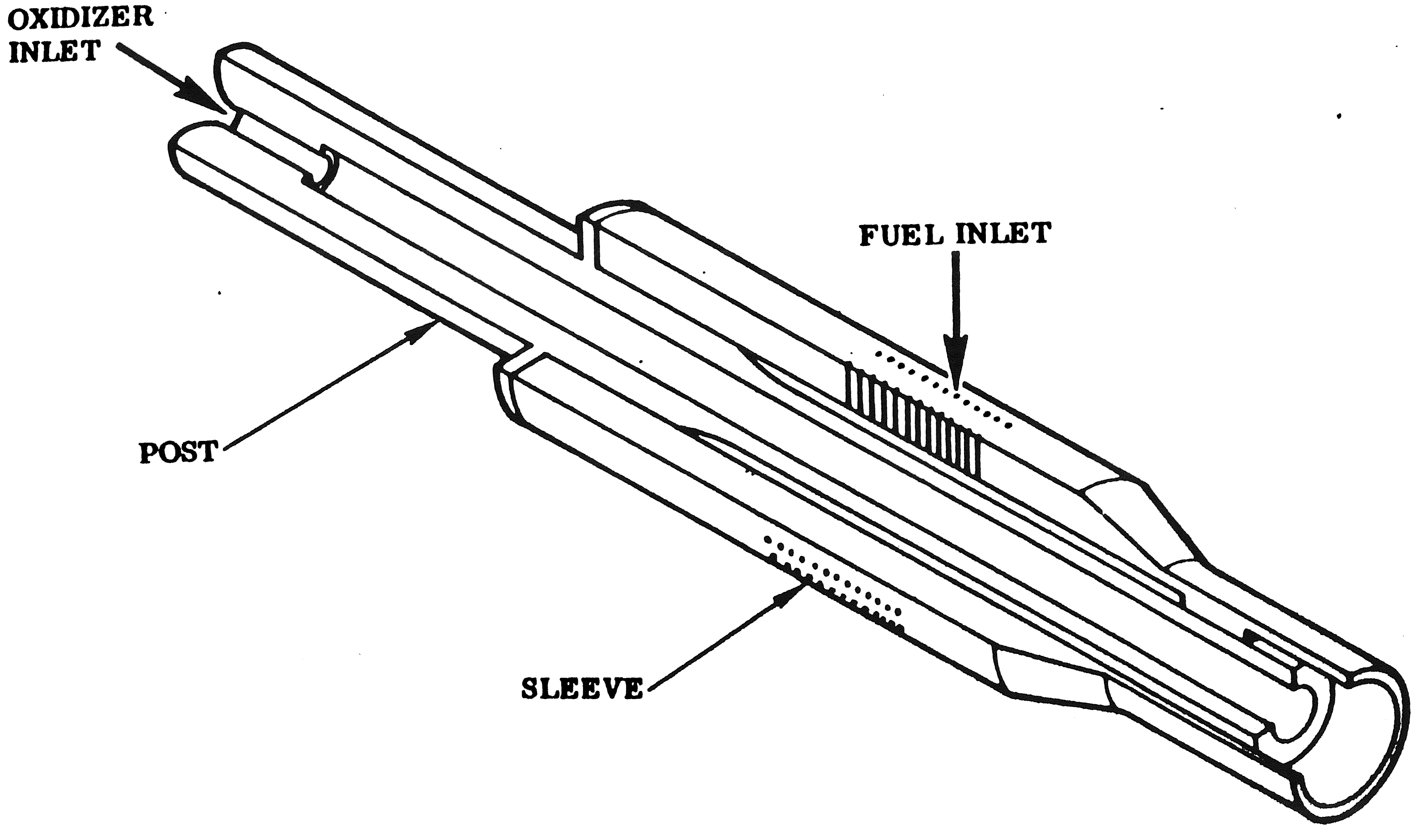
UNLIKE PENTAD
(4 ON 1)

●
●○●
●



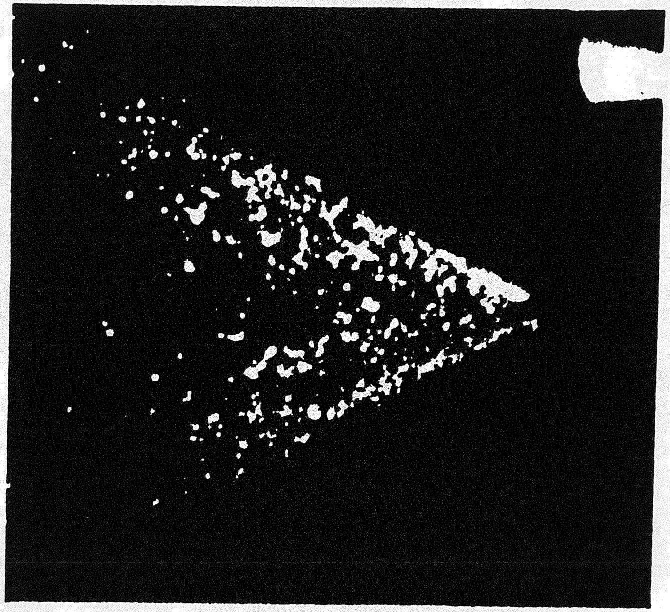
PROPELLANT INJECTION

Current SSME Preburner Coaxial Injection Element



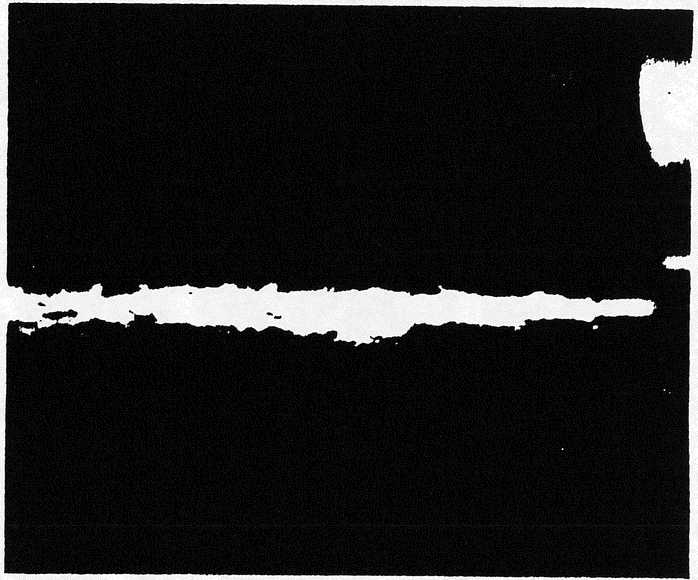
Injector Element Flow Characterization Tests

Simulated 109% RPL, Matched O/F Velocities



P&W BASELINE ELEMENT
"Oxidizer" Fluid - Mil-C-7024
Oxid Press, Pri. 41, Sec. 18
Oxid Flow, 211 PPH
"Fuel" Fluid - AIR
Fuel Press, 0.321 PSID

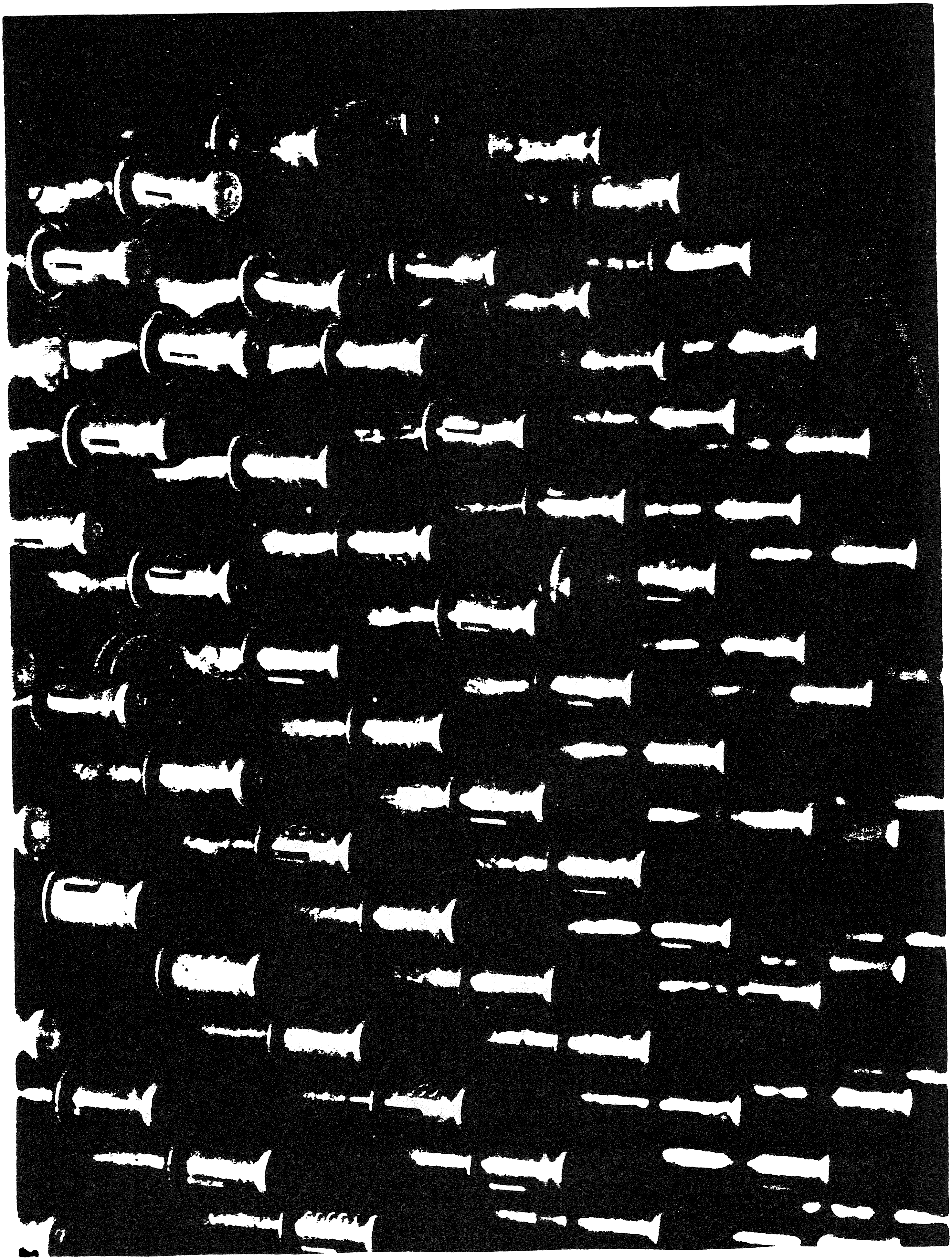
Tangential Oxidizer Entry



CONCENTRIC AXIAL FLOW ELEMENT
"Oxidizer" Fluid - Mil-C-7024
Oxid Press, Pri. 197, Sec. -
Oxid Flow, 314 PPH
"Fuel" Fluid - Air
Fuel Press, 1.20 PSID

Axial Oxidizer Entry

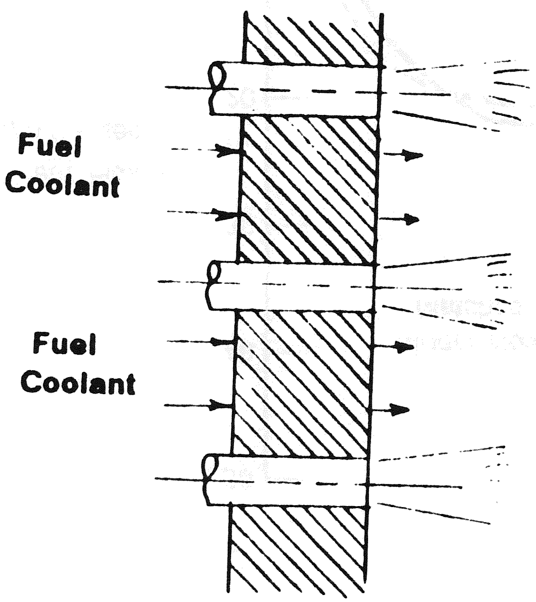
UP SUPERIÉURE OBEN おもて 1



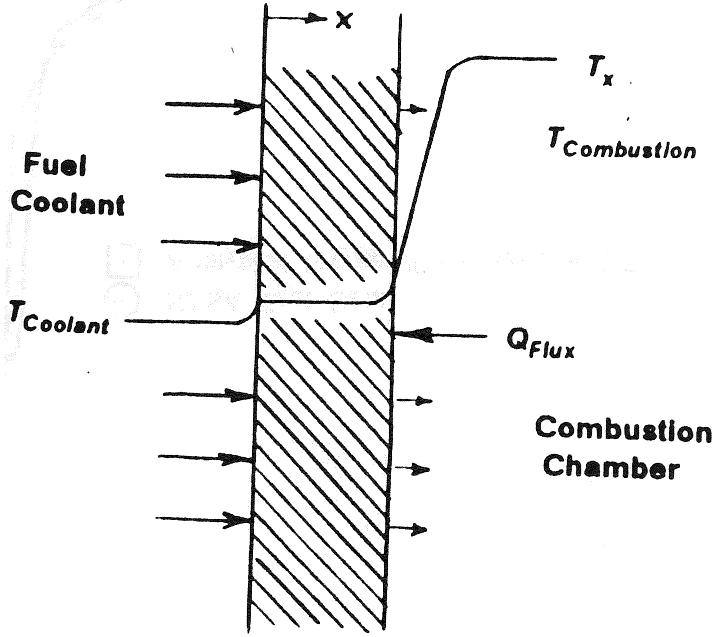
UP SUPERIÉURE OBEN おもて

COMBUSTION CHAMBER/INJECTOR COOLING

Transpiration Cooling Produces Negligible Temperature Gradient



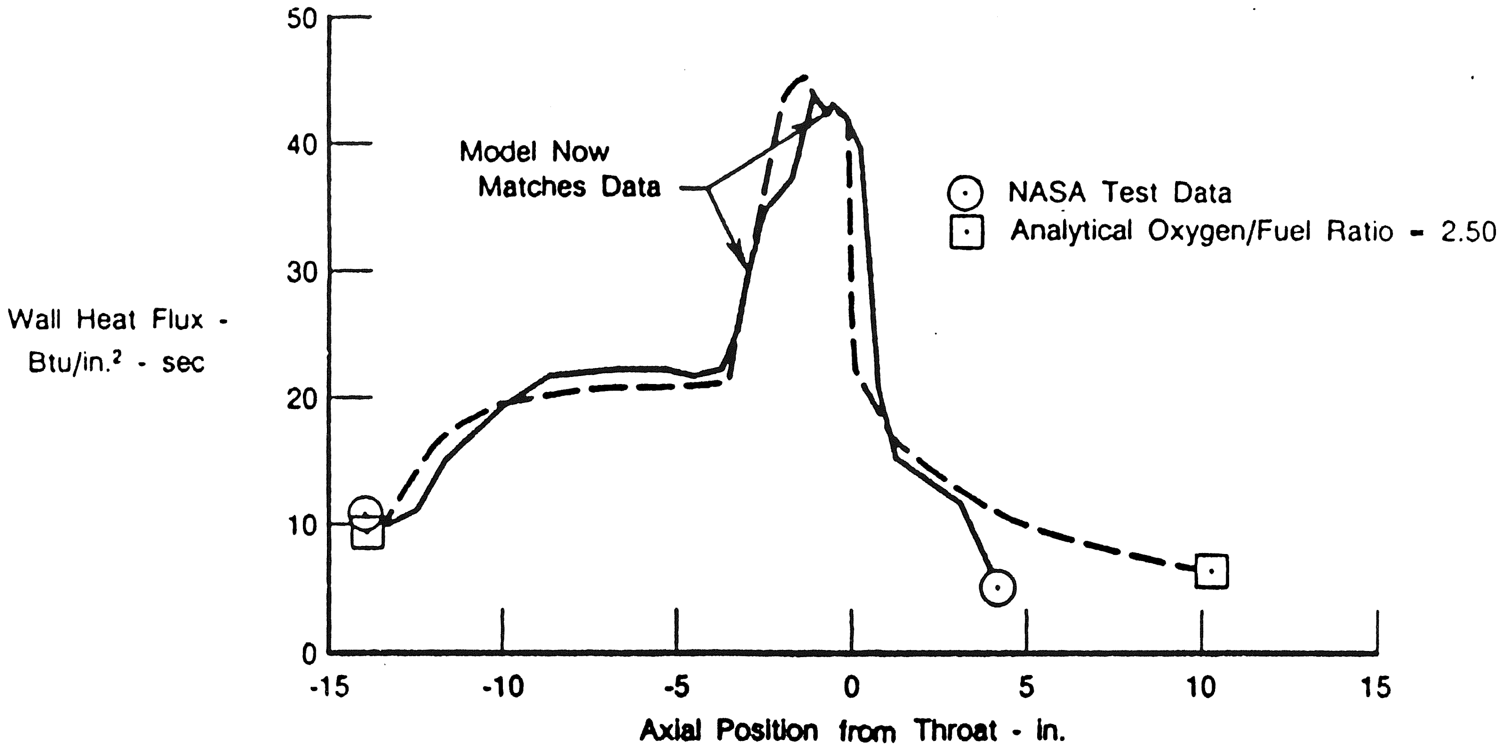
Porous
Injector
Faceplate
(Usually Rigimesh)



Porous
Injector
Faceplate

COMBUSTION CHAMBER COOLING

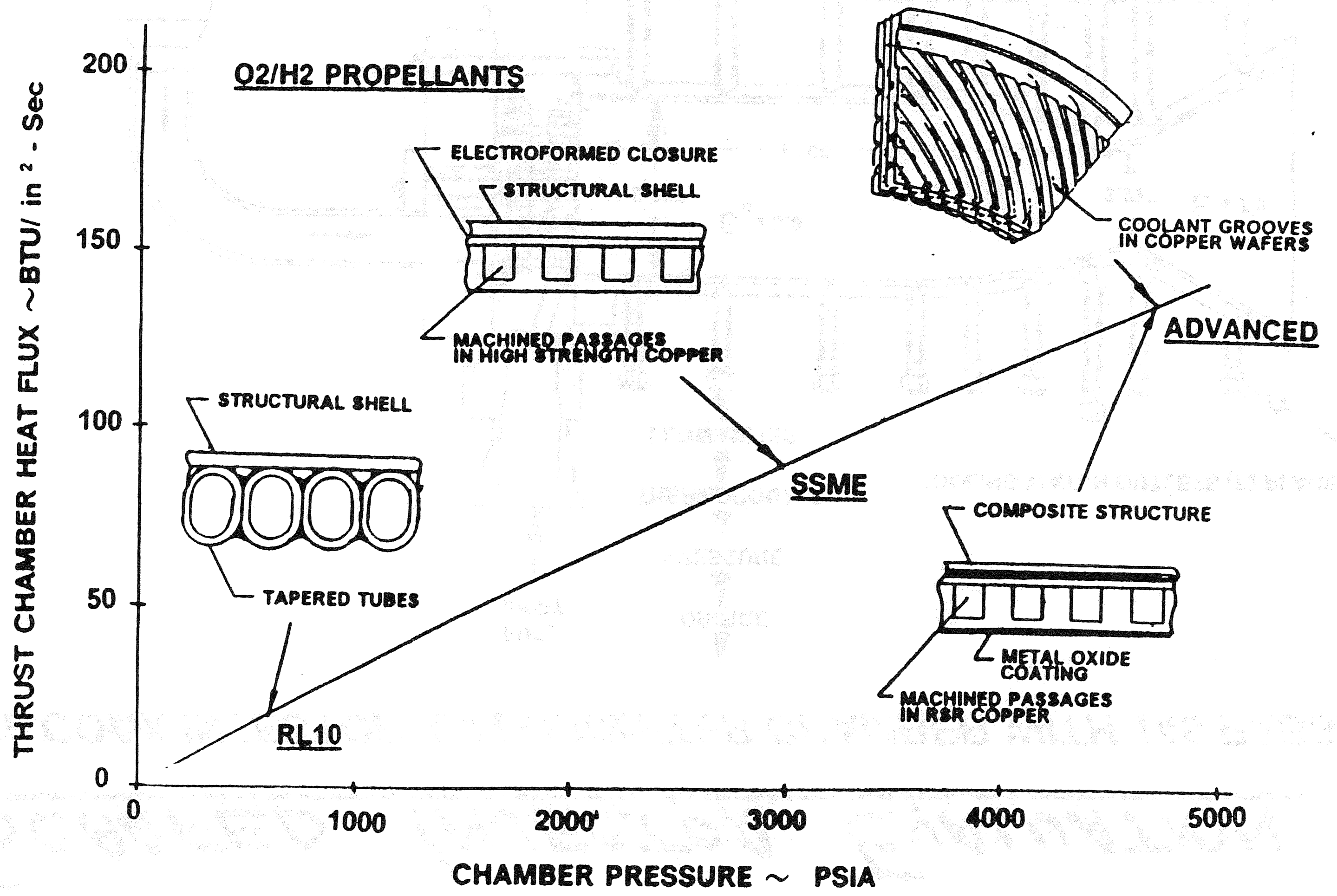
Wall Heat Flux Varies With Location



COMBUSTION CHAMBER COOLING

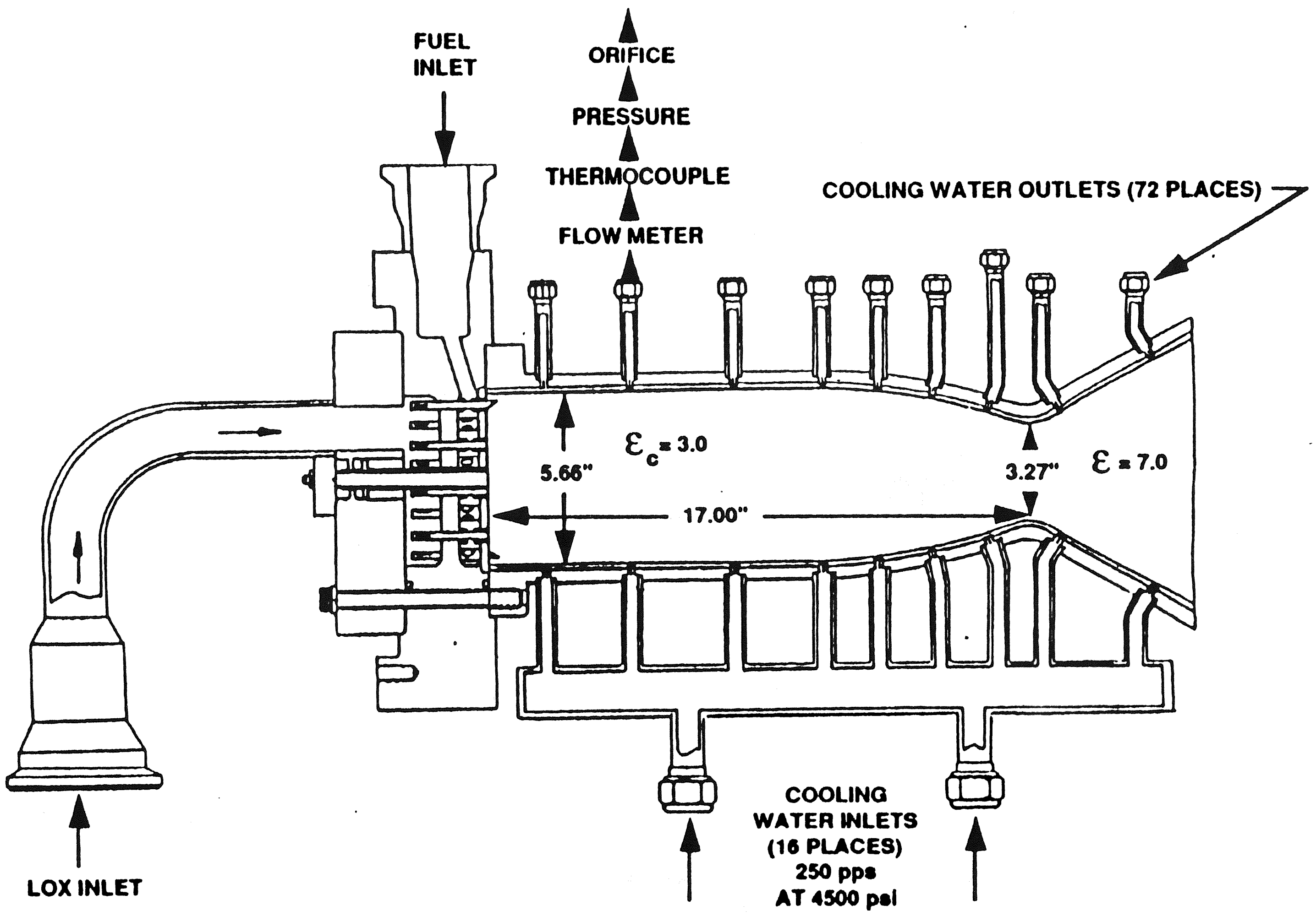
Advanced Cooling Schemes Required For Higher Pc

Thrust Chamber Cooling Driven by Chamber Pressure Requirements



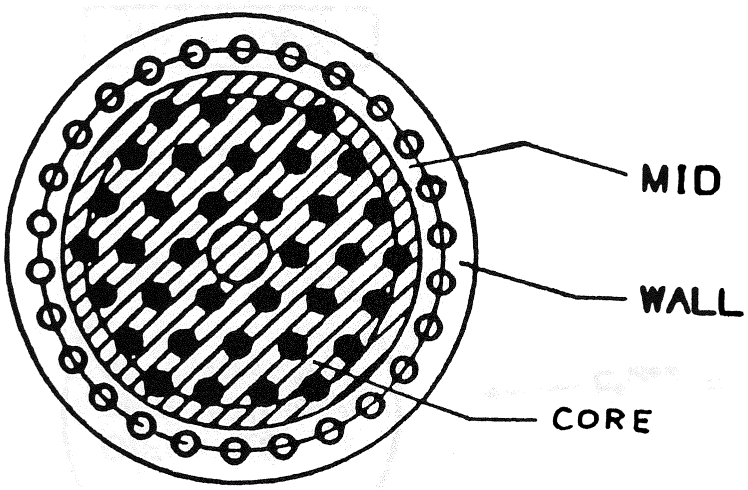
SCARFED INJECTOR EVALUATION

SWIRL COAX INJECTOR, CALORIMETER CHAMBER WITH 146 PASSAGES

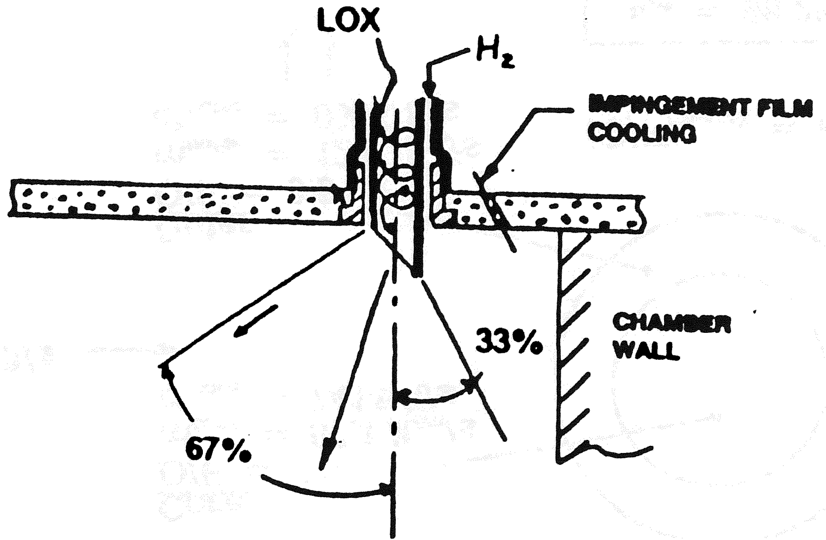


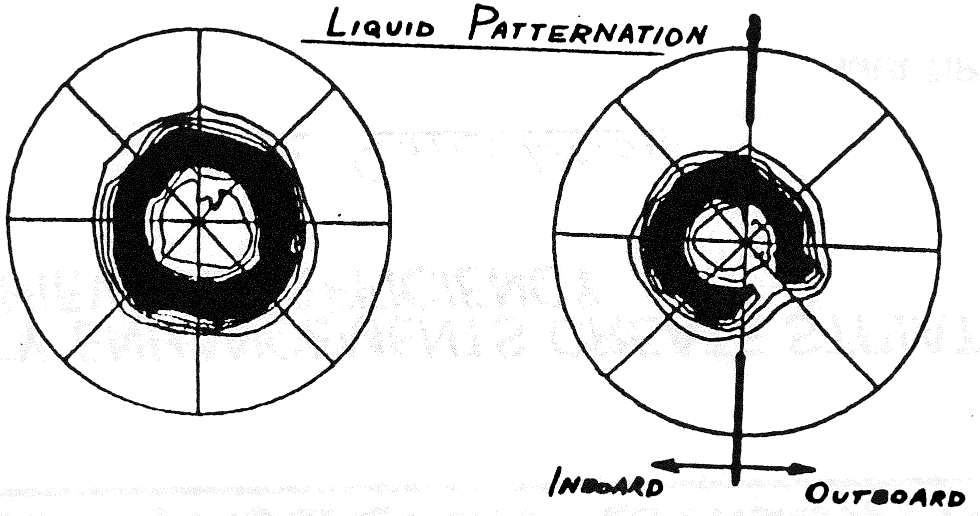
SCARFED INJECTOR EVALUATION

NORMAL AND SCARFED INJECTORS USED IN ALS SUBSCALE TESTS



INJECTOR FACE





CORE ELEMENT PATTERN

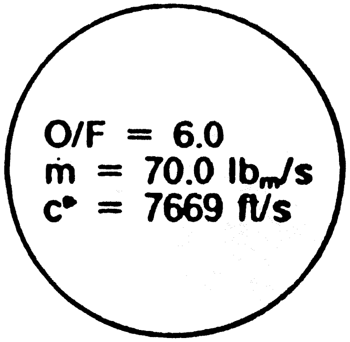
OUTER ROW
SCARFED ELEMENT PATTERN

SCARFED INJECTOR EVALUATION

WALL DURABILITY ENHANCEMENTS CREATE STRIATED FLOWS WHICH LIMIT ACHIEVABLE EFFICIENCY

Sample Calculation

SINGLE STREAMTUBE



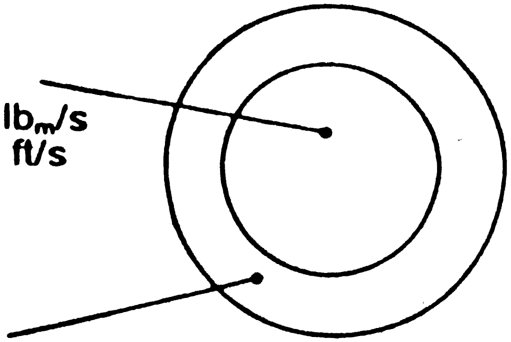
$O/F = 6.0$
$\dot{m} = 70.0\ lb_m/s$
$c^* = 7669\ ft/s$

$\longleftarrow c^*_{ideal} = 7669\ ft/s \longrightarrow$

$(O/F)_{overall} = 6.0$

$\boxed{\eta_{c^*} = 100\%}$

MULTIPLE STREAMTUBES



Core
$O/F = 7.0$
$\dot{m}_{core} = 57.1\ lb_m/s$
$c^*_{core} = 7414\ ft/s$

Outer
$O/F = 3.5$
$\dot{m}_{outer} = 12.9\ lb_m/s$
$c^*_{outer} = 8066\ ft/s$

$\Downarrow$

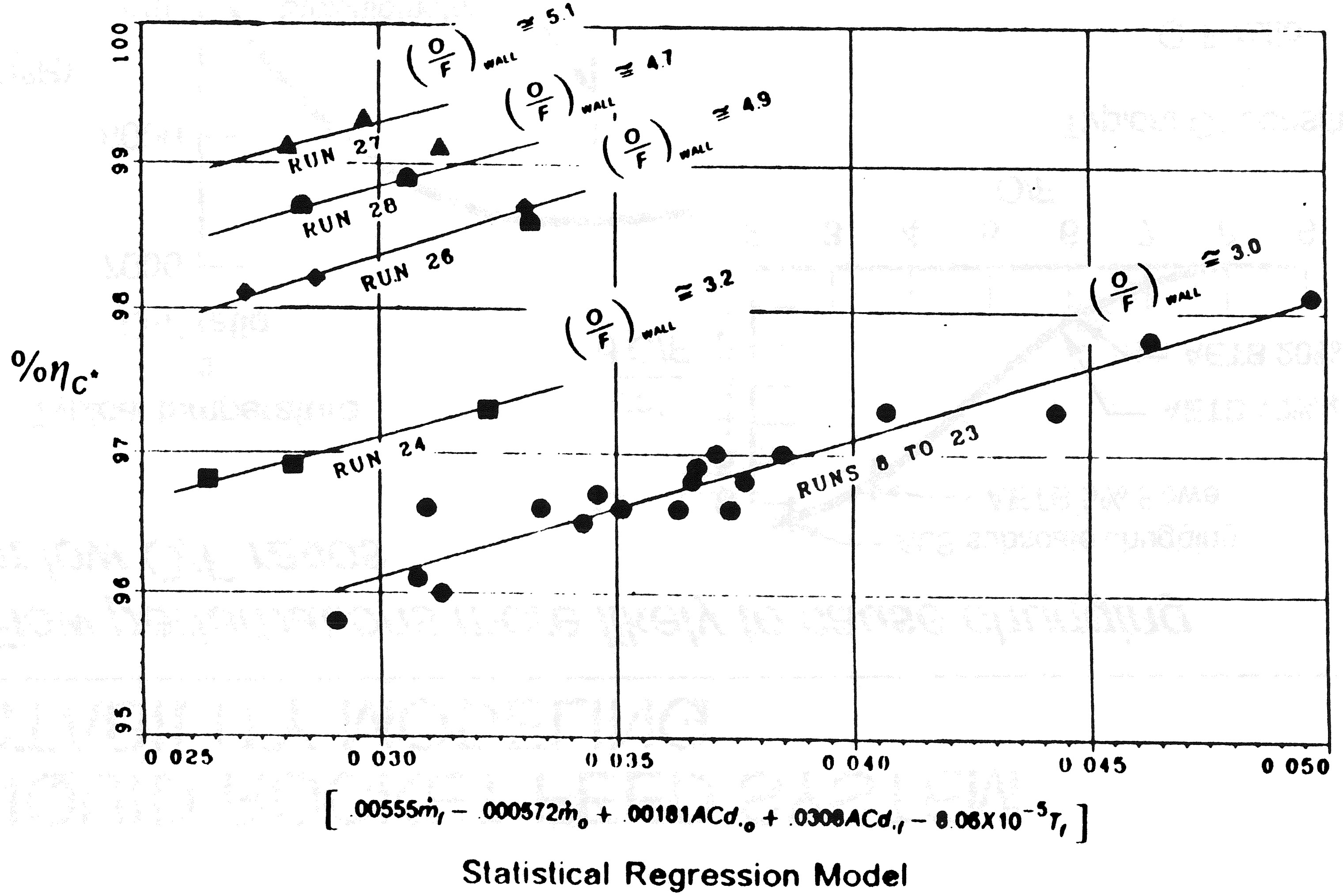
$c^* = 7534\ ft/s$

$(O/F)_{overall} = 6.0$

$\boxed{\eta_{c^*} = 98.2\%}$

SCARFED INJECTOR EVALUATION

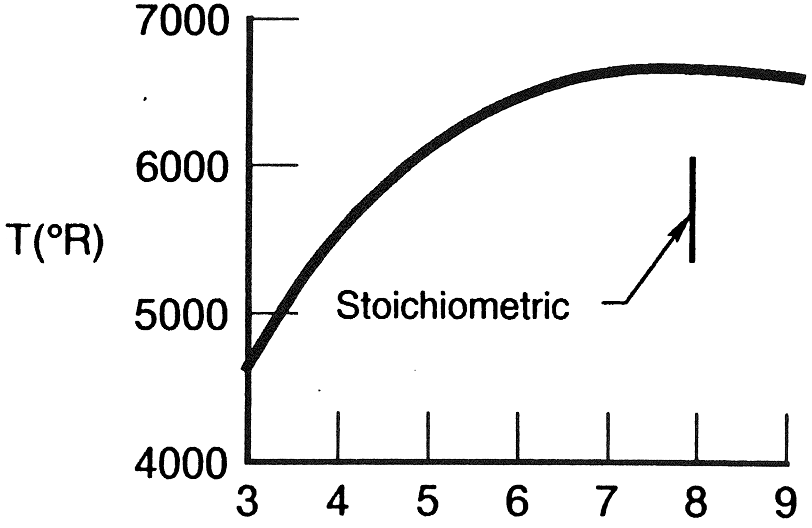
FOR A GIVEN WALL O/F, SUBSCALE DATA
CORRELATE WITH TEST MATRIX VARIABLES

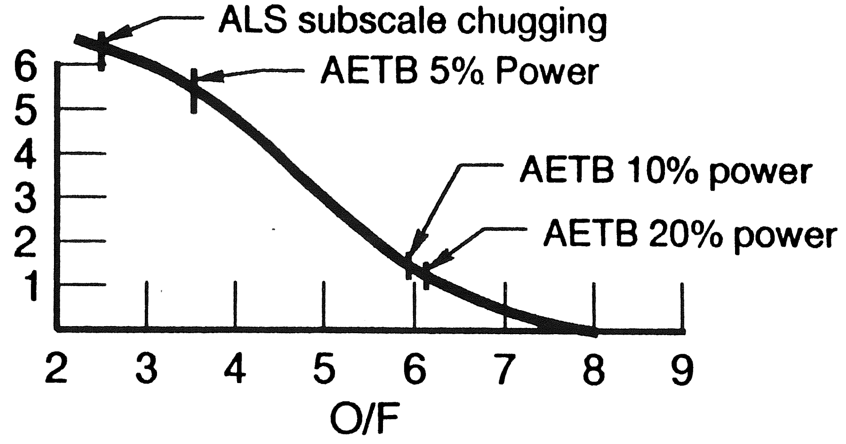


LIQUID ROCKET FEED SYSTEM STABILITY MODELING

Flow perturbations more likely to cause chugging at low O/F ratios

Typical temperature
vs
O-F ratio

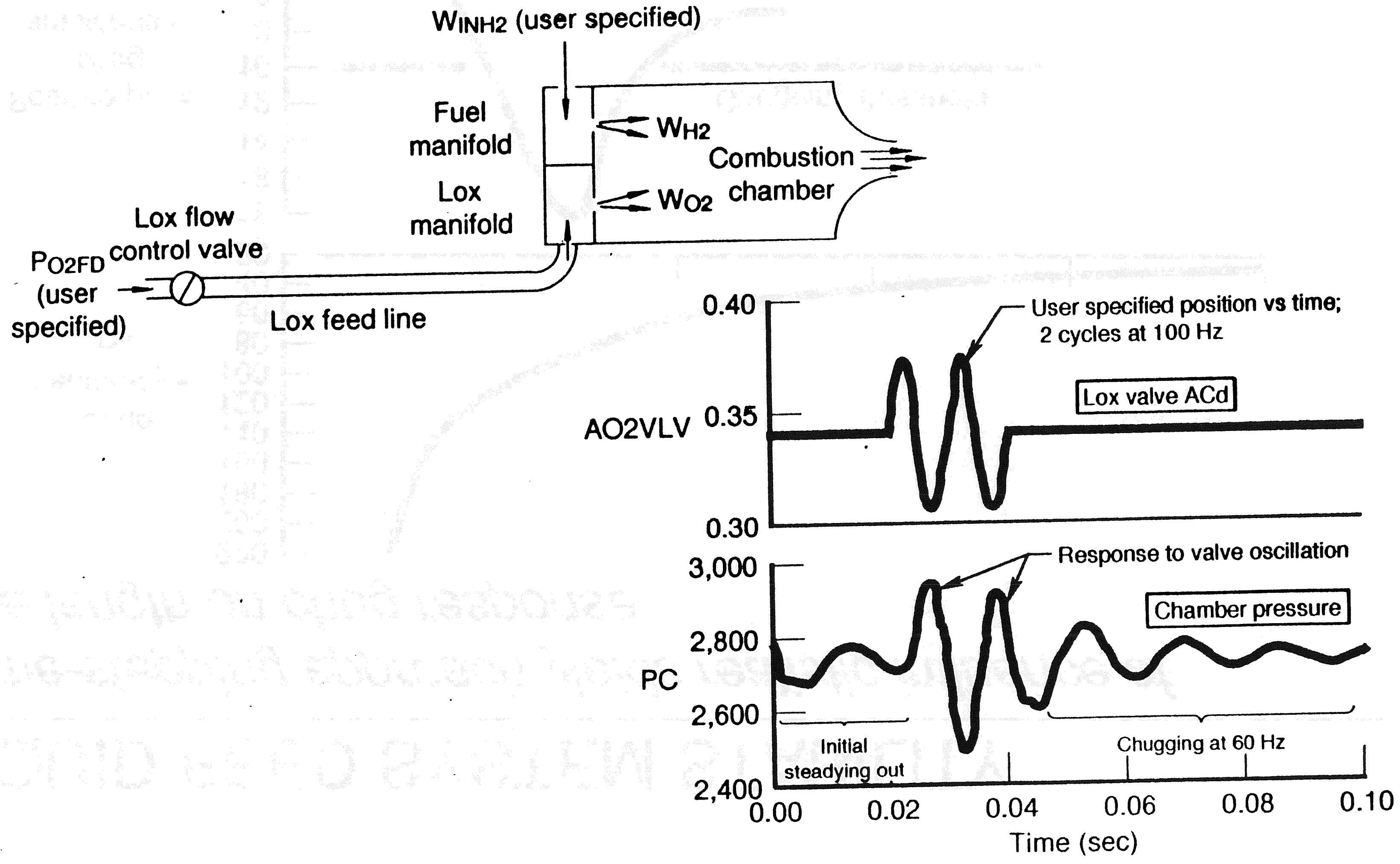




Typical C* sensitivity
to
O-F ratio

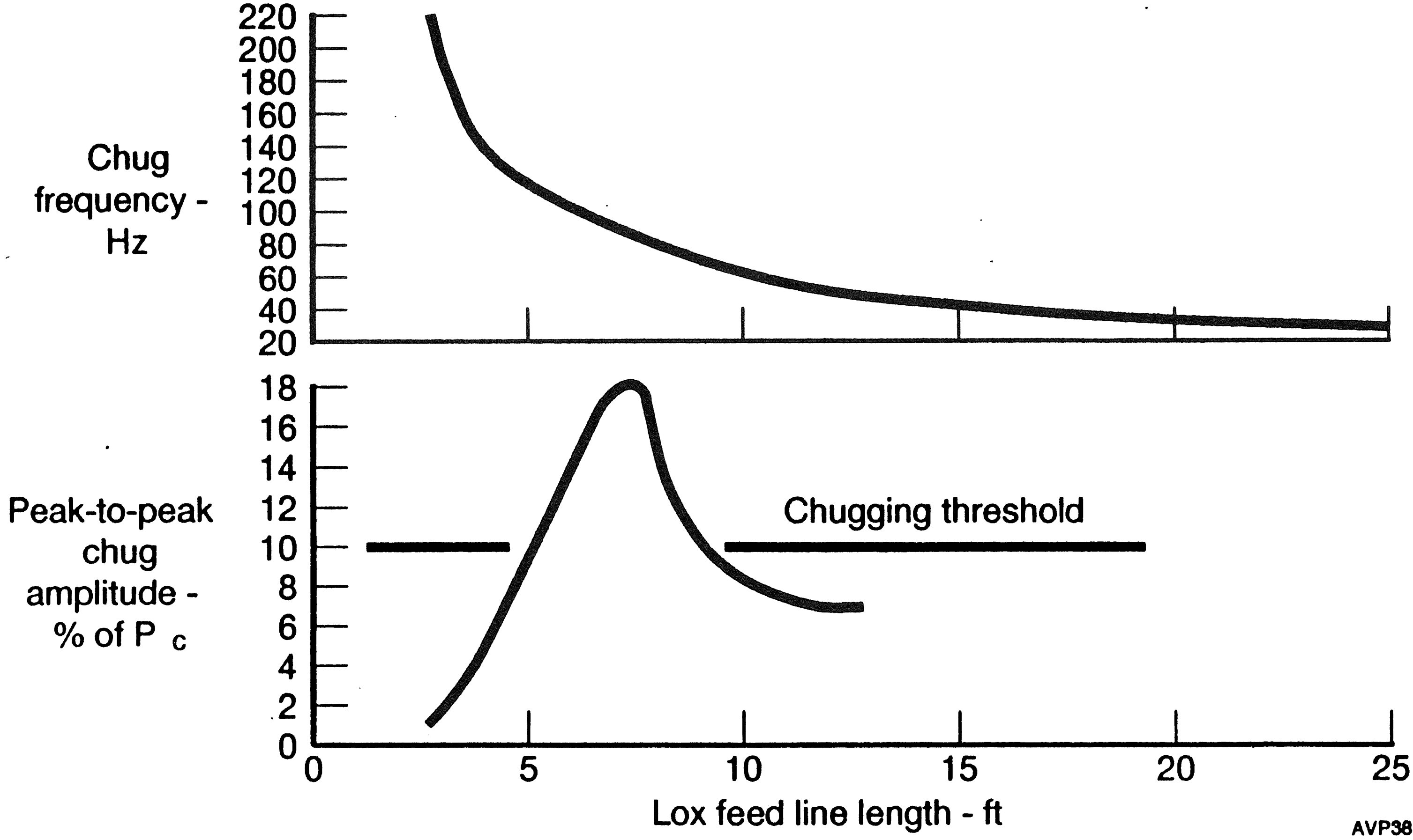
LIQUID FEED SYSTEM STABILITY MODELING

Early version of chugging model shows realistic response to oscillating valve



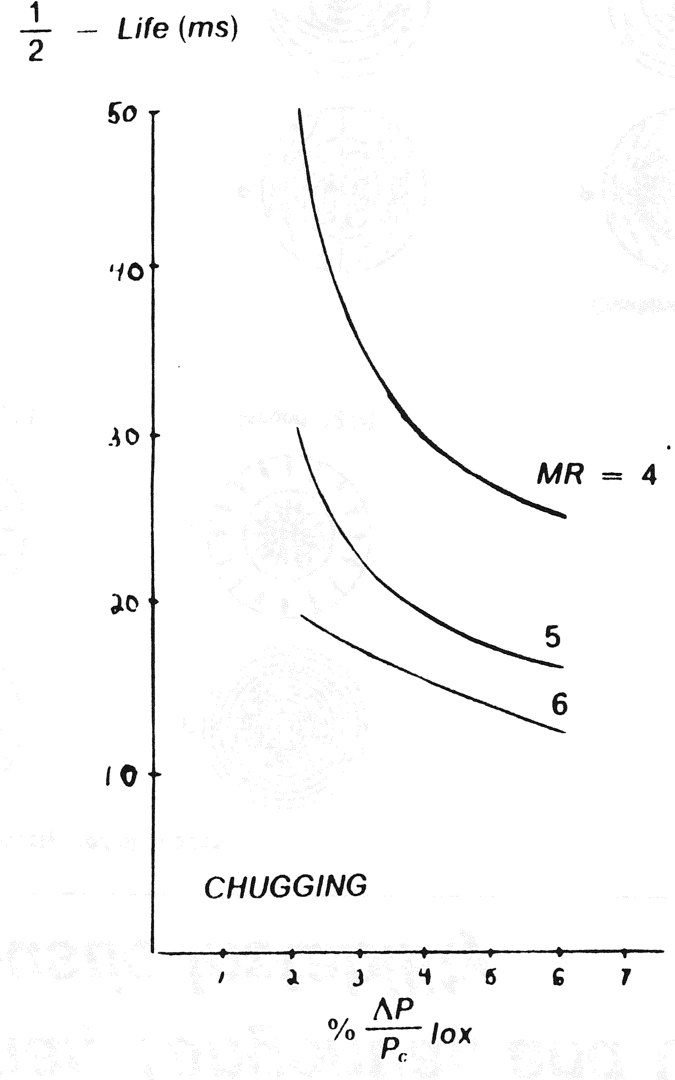
LIQUID FEED SYSTEM STABILITY

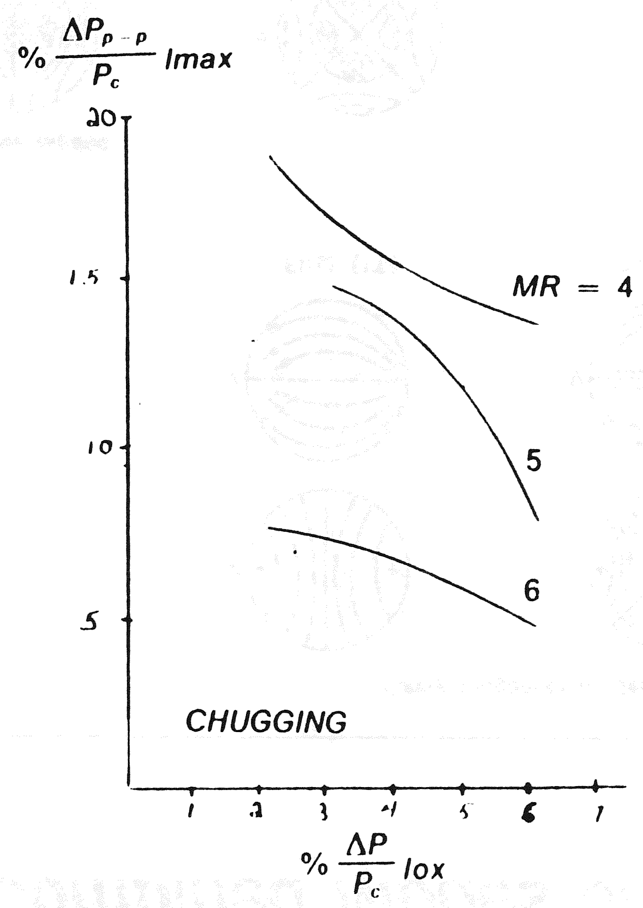
Time-stepping approach yields realistic influence of line length on chug response



AVP387237 912404

CHUGGING MODEL RESULTS





HIGH FREQUENCY COMBUSTION INSTABILITY

Radial, Tangential, and Combined Modes of Acoustic Instability

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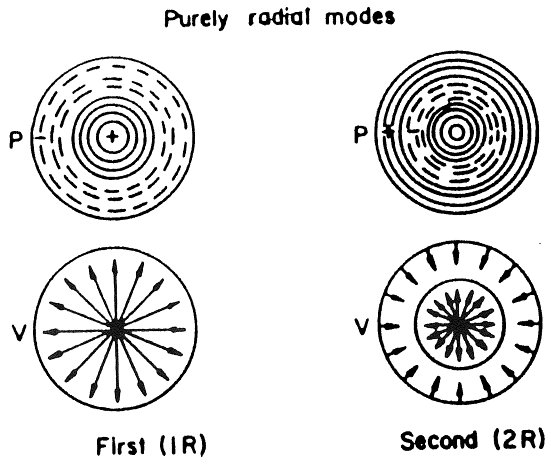

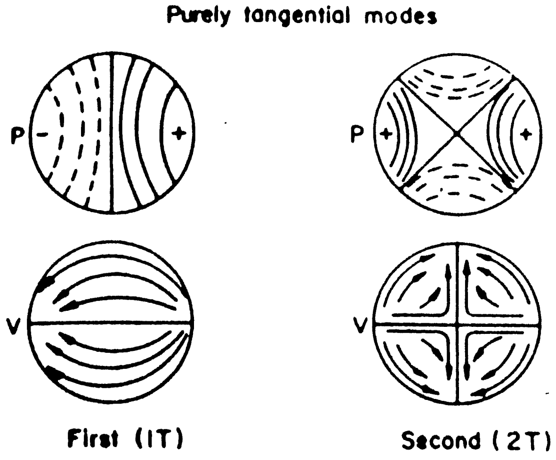

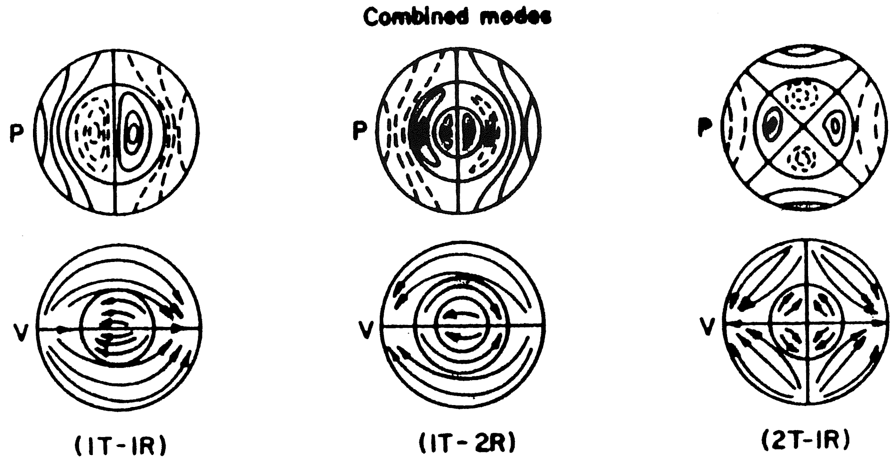

COMBUSTION INSTABILITY

Baffles on the Injector Face can Alleviate Acoustic Instability

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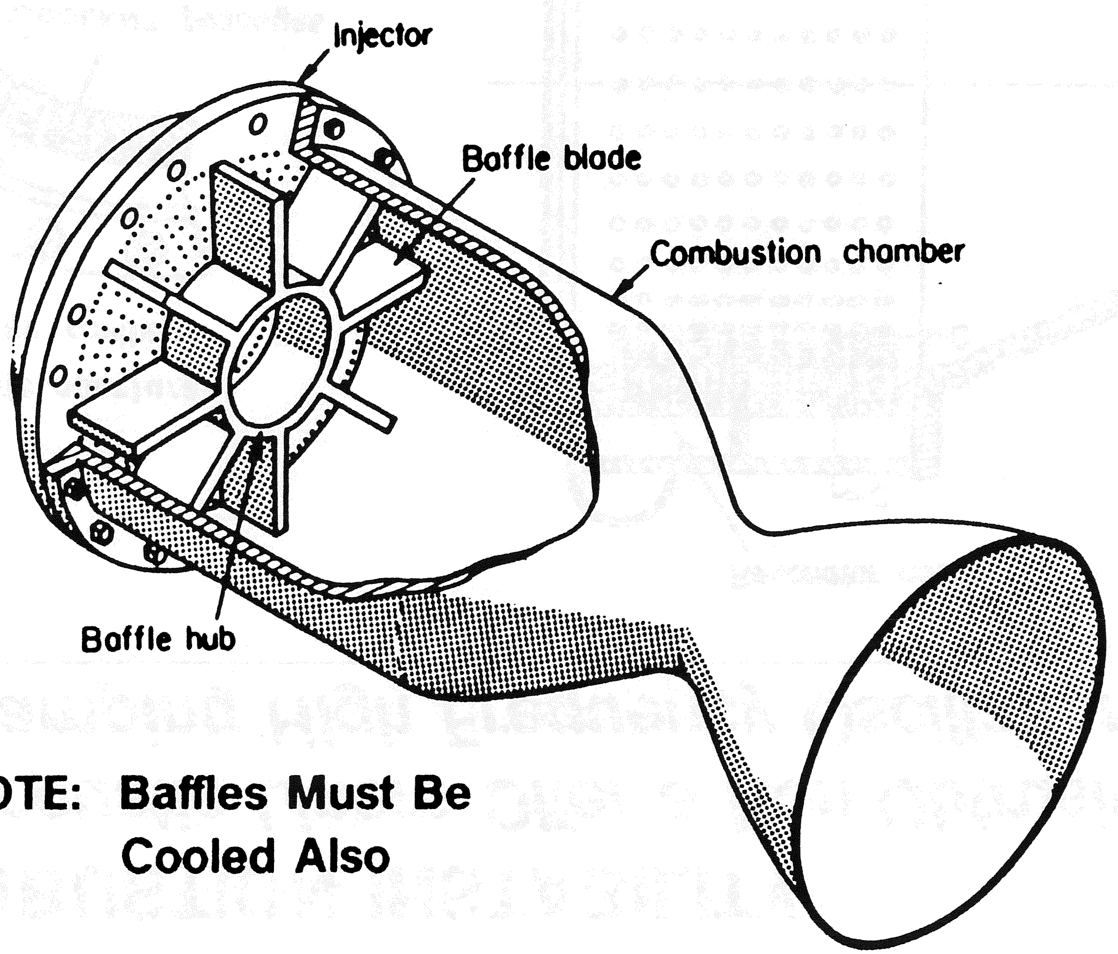
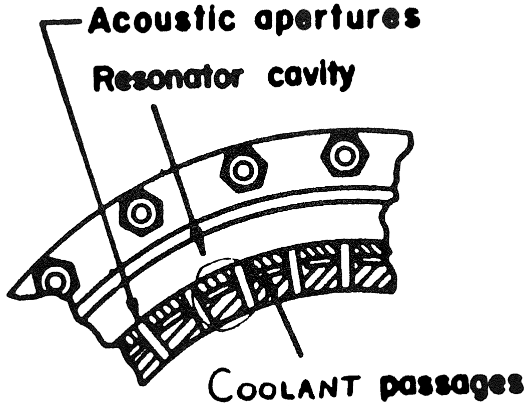


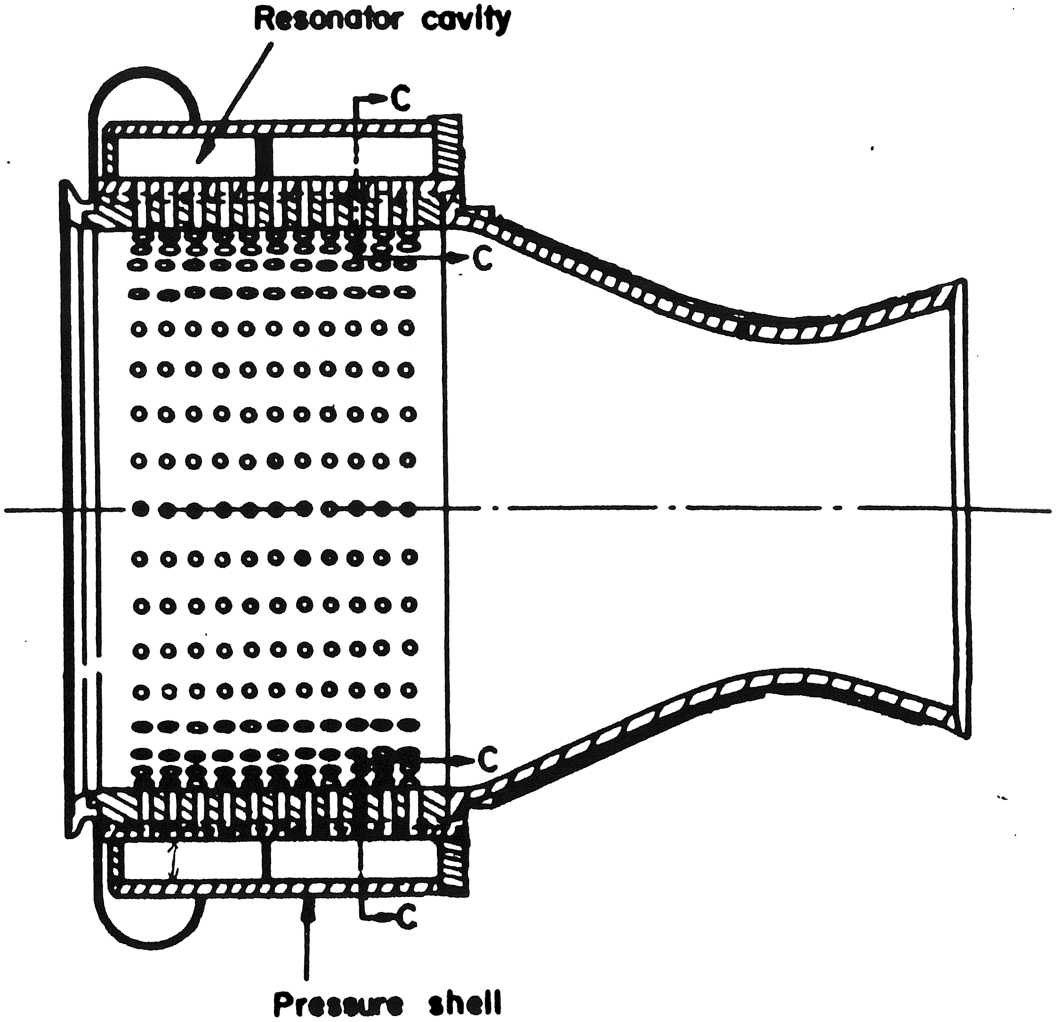
NOTE: Baffles Must Be Cooled Also

COMBUSTION INSTABILITY

Acoustic Liners Offer a Non-Obtrusive Means of Damping High Frequency Oscillations

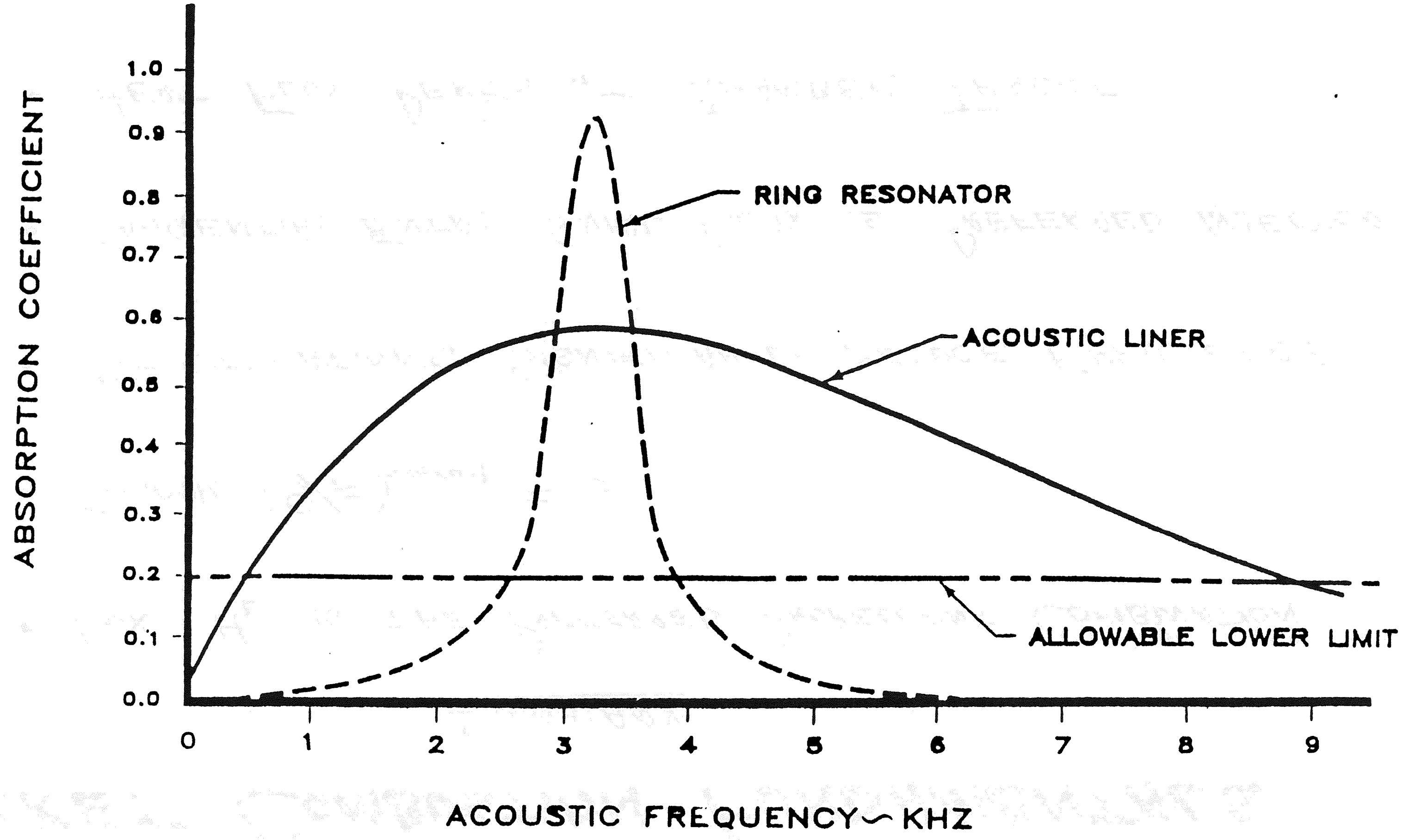


Section C-C



HIGH FREQUENCY COMBUSTION INSTABILITY

Absorptivity of Ring Resonator And Acoustic Liner



ROCKET COMBUSTION FUNDAMENTALS

SUMMARY

- LOX - H_2 IS THE PREFERRED PROPELLANT COMBINATION
- TYPICAL $(O/F)_{overall} = 6$
- O/F STRIATIONS SIGNIFICANTLY REDUCE EFFICIENCY
- TANGENTIAL ENTRY SWIRL COAX IS PREFERRED INJECTOR
- HEAT FLUX PEAKS AT CHAMBER THROAT
- REGENERATIVE COOLING FOR CHAMBER WALLS
  TRANSPIRATION COOLING FOR INJECTOR FACEPLATE
- CHUGGING GOVERNED BY $\frac{\Delta P_{INJ}}{P}$, O/F, FEED GEOMETRY
- ACOUSTIC INSTABILITY GOVERNED BY CHAMBER DIMENSIONS,
  LOX - POST DE-TUNING

ROCKET ENGINE

FUNDAMENTALS COURSE

Bearings & Seals

David A Haluck
Mechanical Component Analysis
X3274, M/S ~~713-15~~
715-91

PURPOSE

- To provide an overview on bearing and seal selection
- Point out unique aspects to cyrogenic applications

BEARINGS

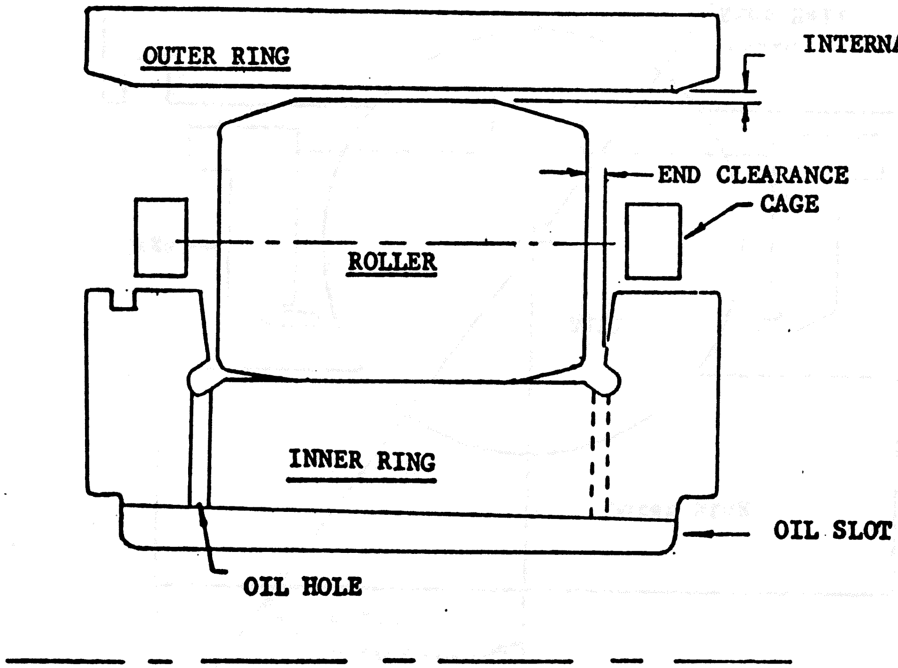
Bearing Purpose:

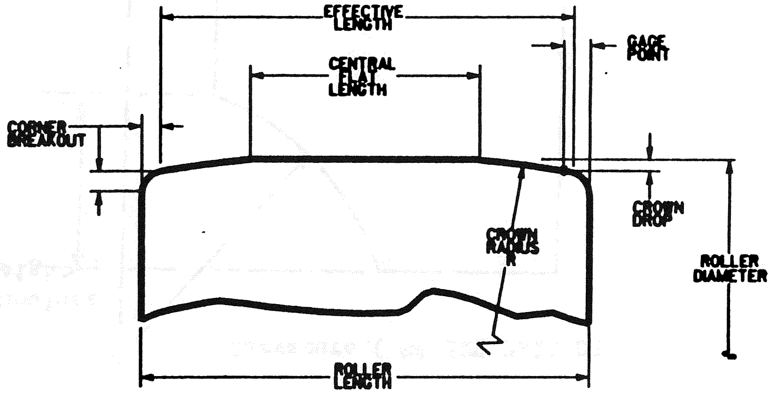
- o To provide rotor control by ensuring predictable Radial or axial movement while rotating
- o To minimize friction

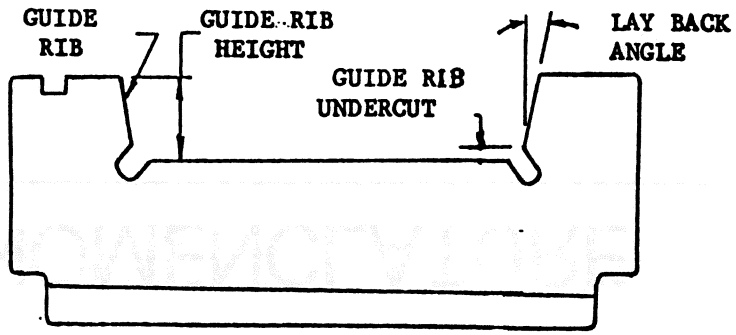
TYPES OF BEARINGS

- o Ball Bearings
  - Used in applications where axial and radial loads are applied
- o Roller Bearings
  - Used in applications where only radial load is applied

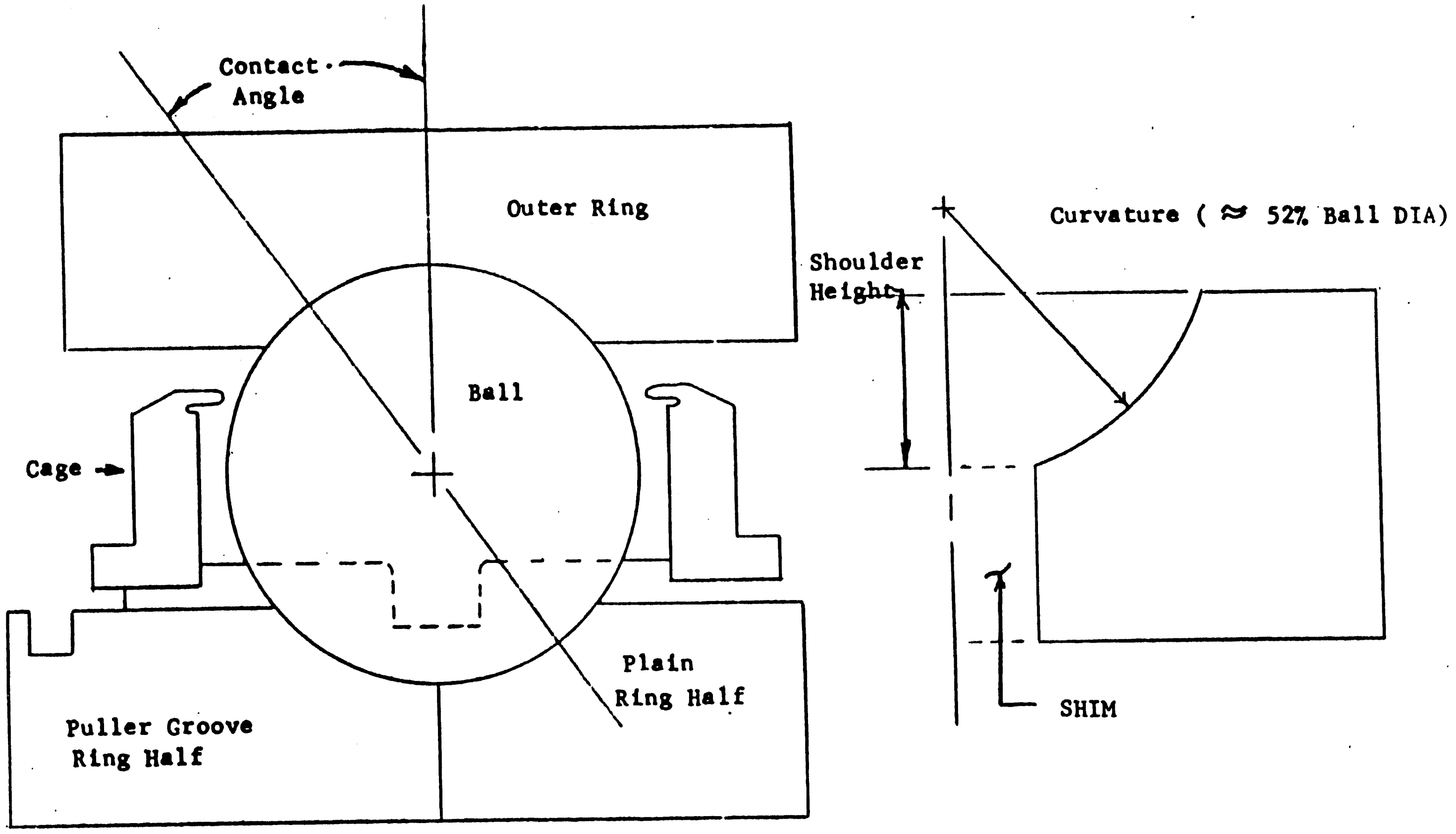
ROLLER BEARING NOMENCLATURE



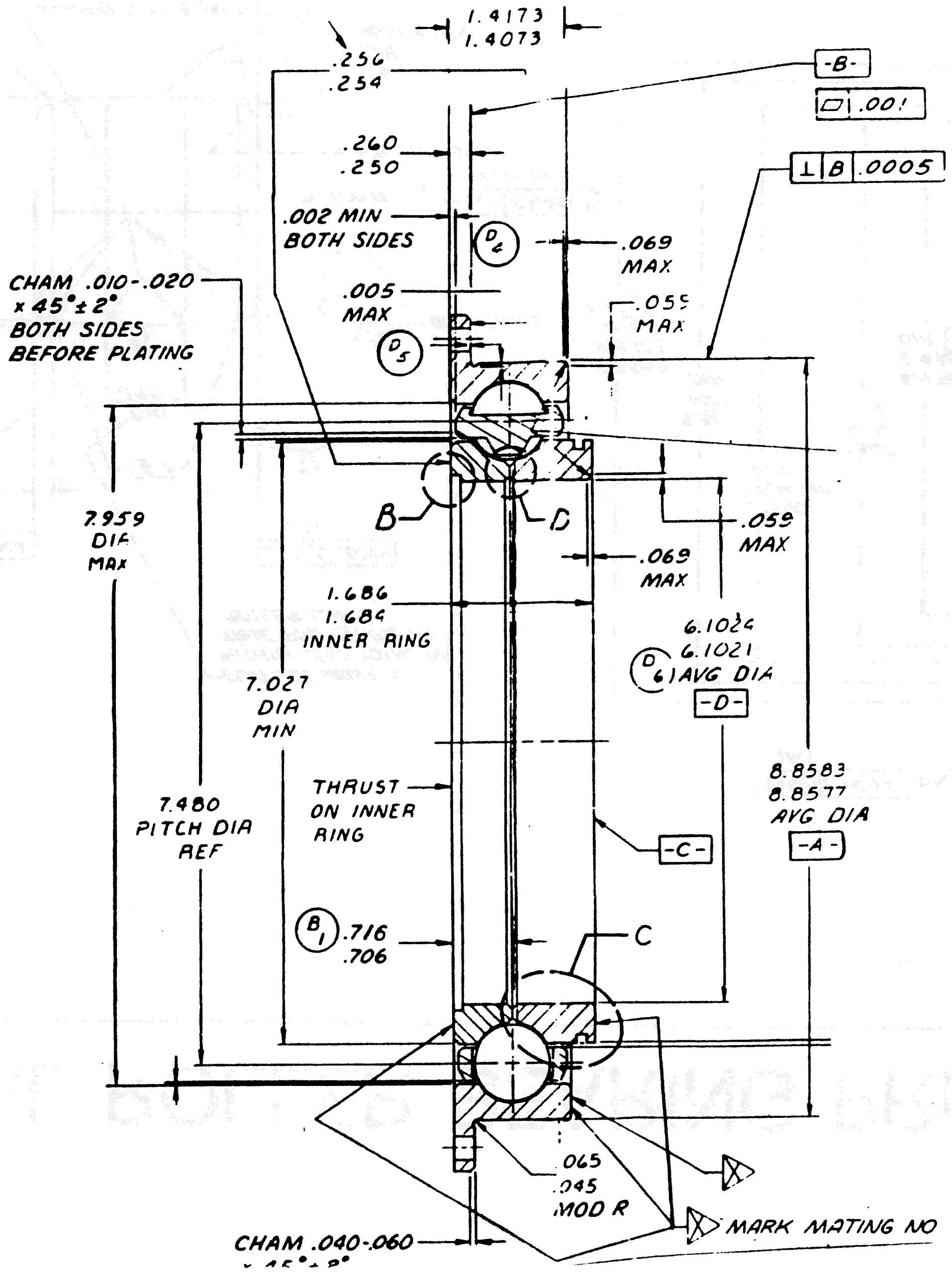




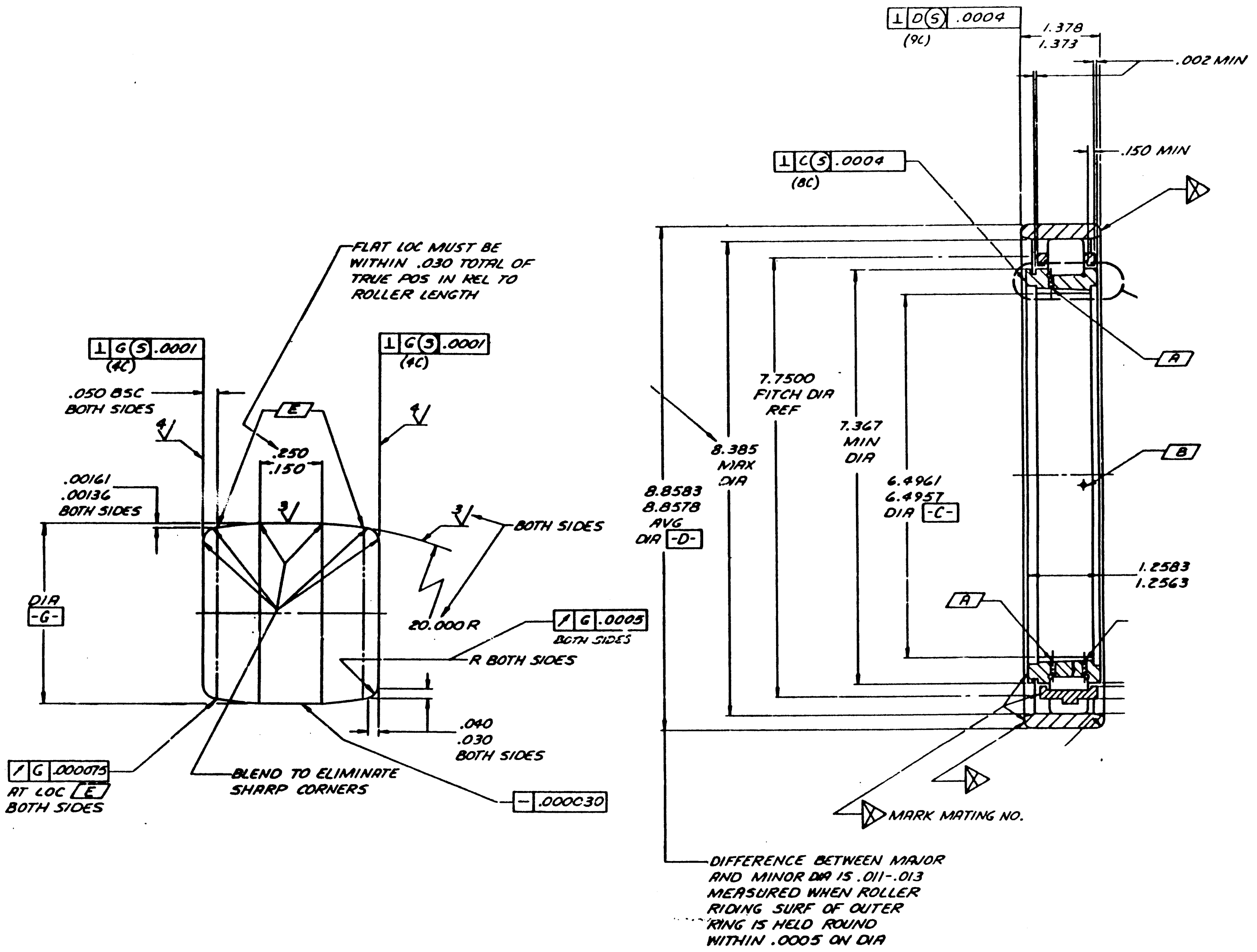
BALL BEARING NOMENCLATURE



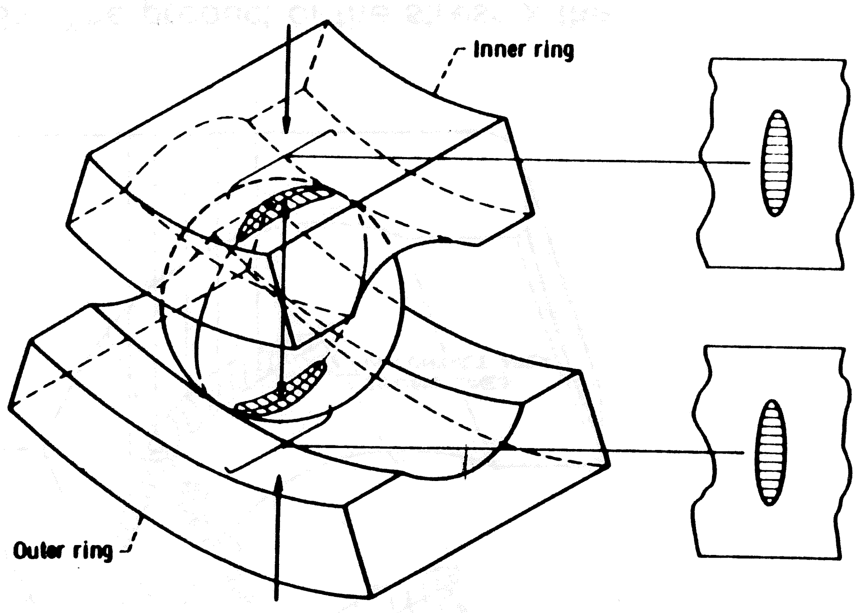
TYPICAL BALL BEARING PRINT



TYPICAL ROLLER BEARING PRINT

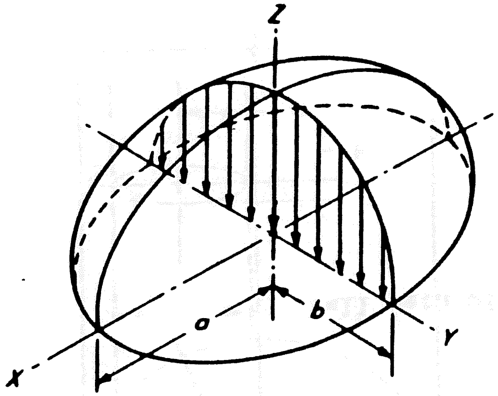


HOW DO THEY WORK ?



Load Deformation At The Ball/Race Contacts Forms An Ellipitical Pressure Area

- Ball to Race Interface

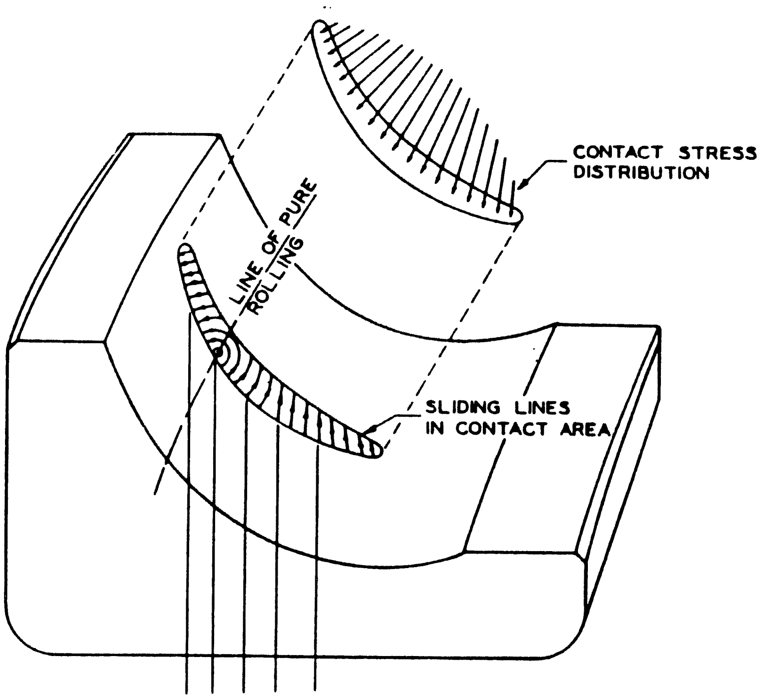


Hertz Stress

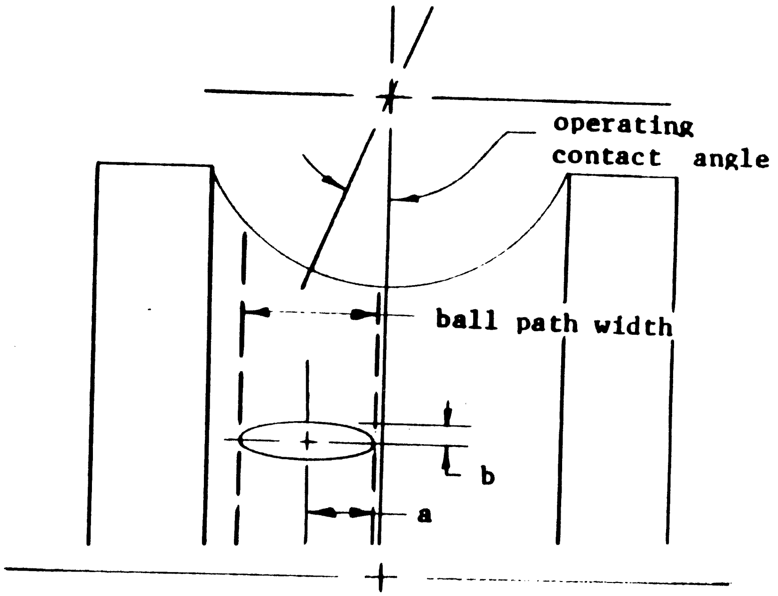
Mean Stress = P/A

Max Stress = 1.5x Mean

MORE ON CONTACT ELLIPSES

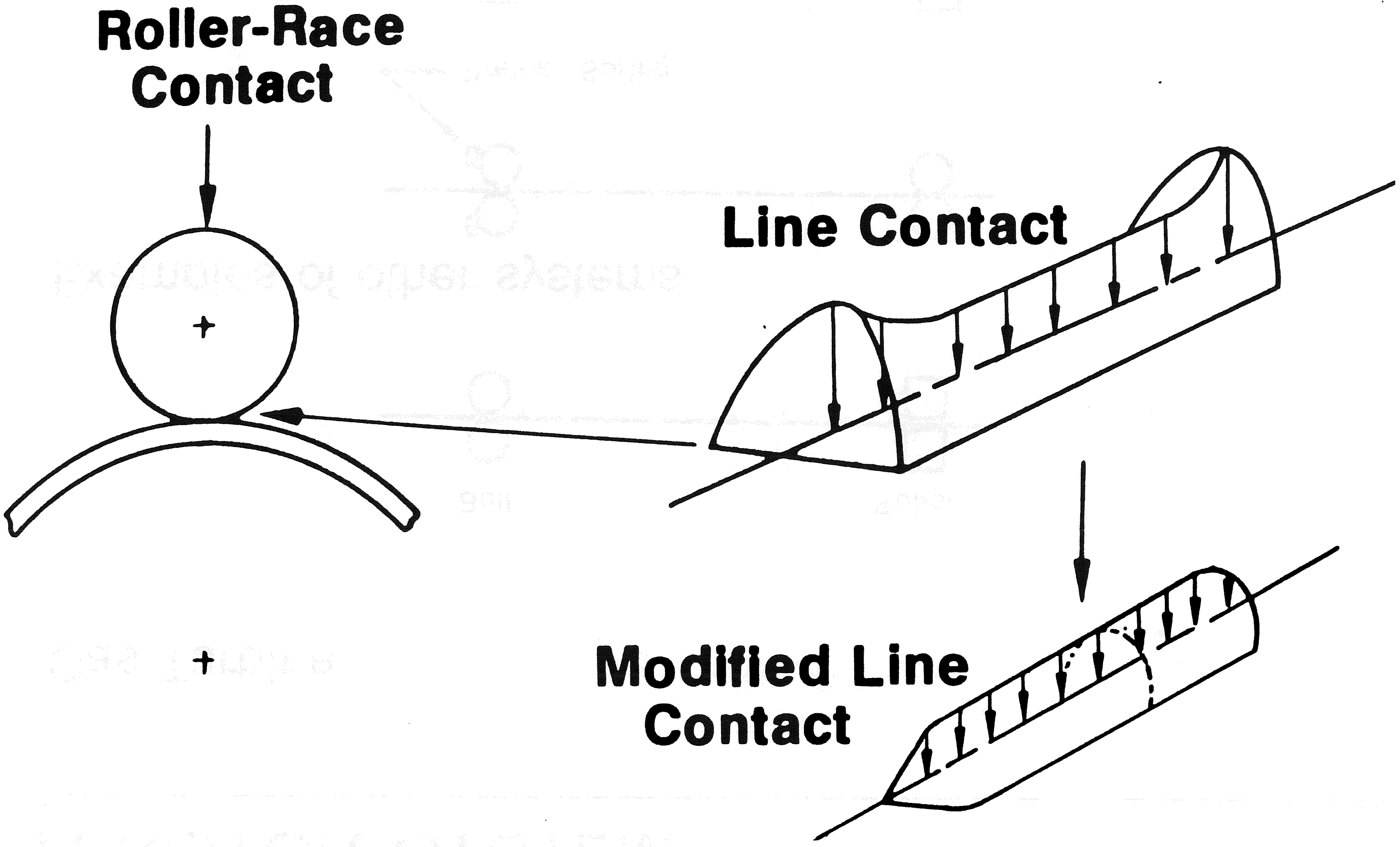


- SV. The product of the stress x the Spin Velocity. is a measure of the applications severity at the contact.



- Geometry of an angular contact bearing results in a Spin Component

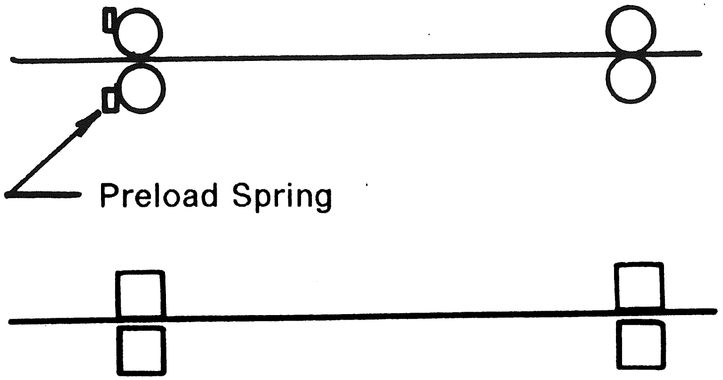
ROLLER BEARINGS HAVE LINE CONTACT



A ROTOR SYSTEM

Gas Turbine

Examples of other systems



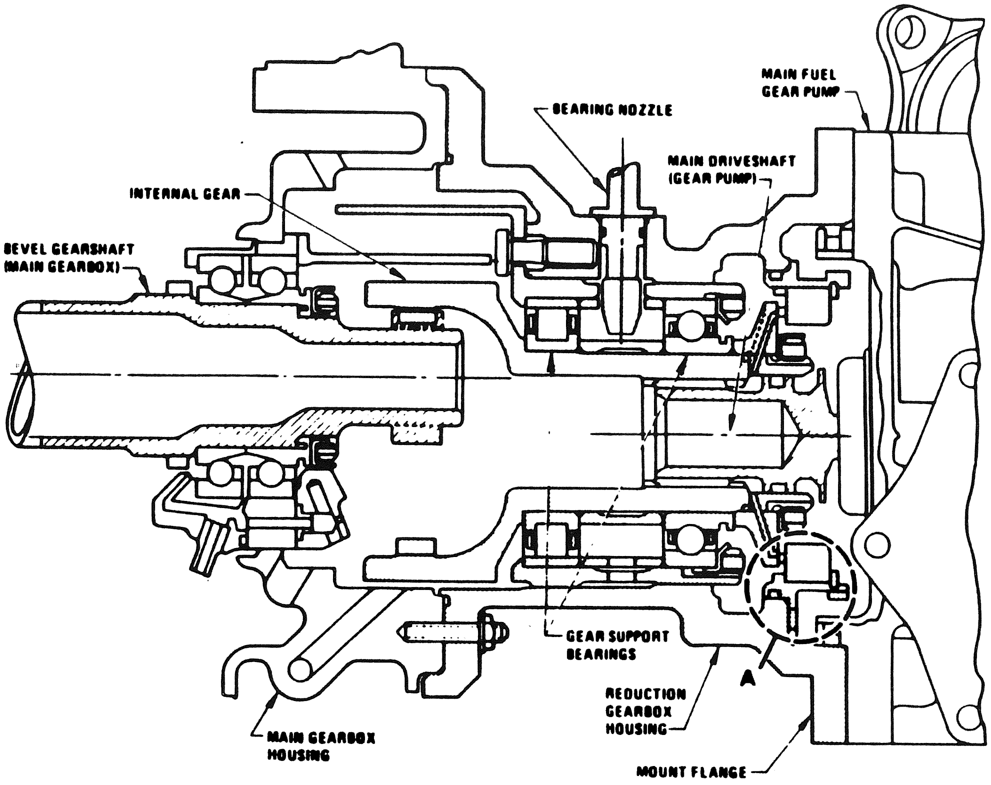
ROTOR SUPPORTS

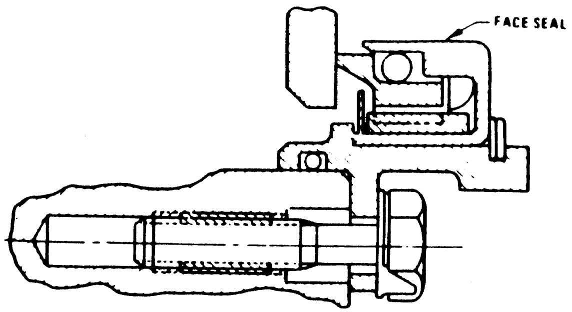
Types	Advantages	Disadvantages
Ball-Ball	o simple o Axial loads can be applied in both directions	o Not good for high radial loads o Not very stiff o Requires OR translation
Roller-Roller	o Very stiff o Good for high radial loads	o requires balance piston
Ball-Roller	o Axial and radial loads can be applied	o With a single ball preload is required and an initial wear surface o Requires a balance piston

LUBRICATION / COOLING

Purpose is to create a satisfactory environment for the bearing.

- **Cooling:** To carry away frictional heat, control temperature to acceptable level (i.e. material limits) and clearance control.

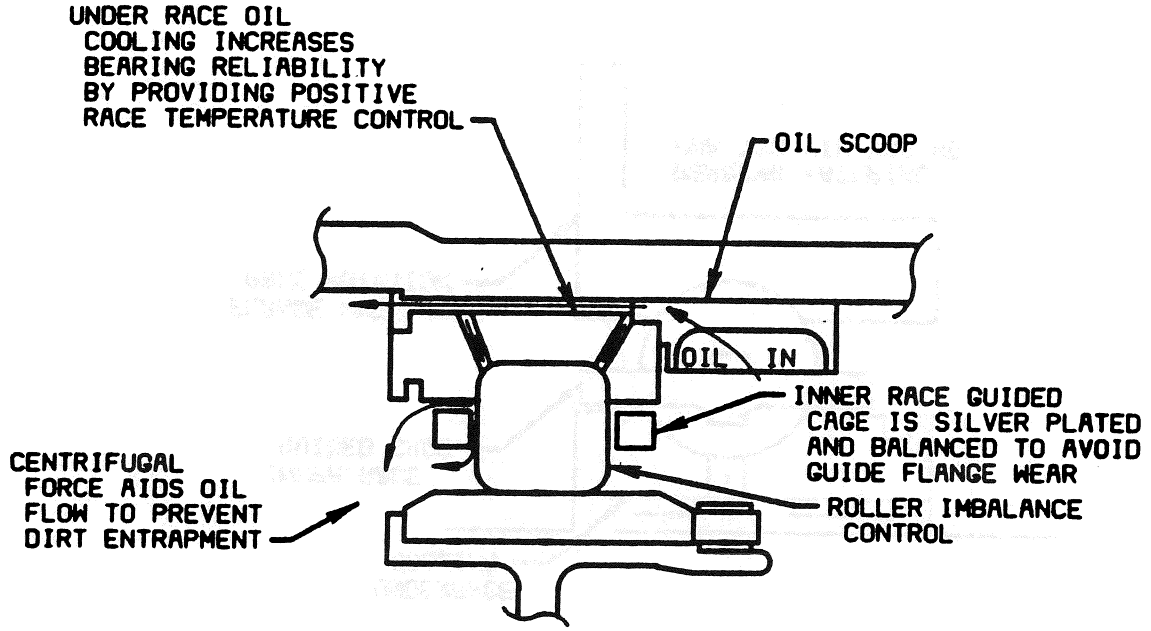




VIEW A

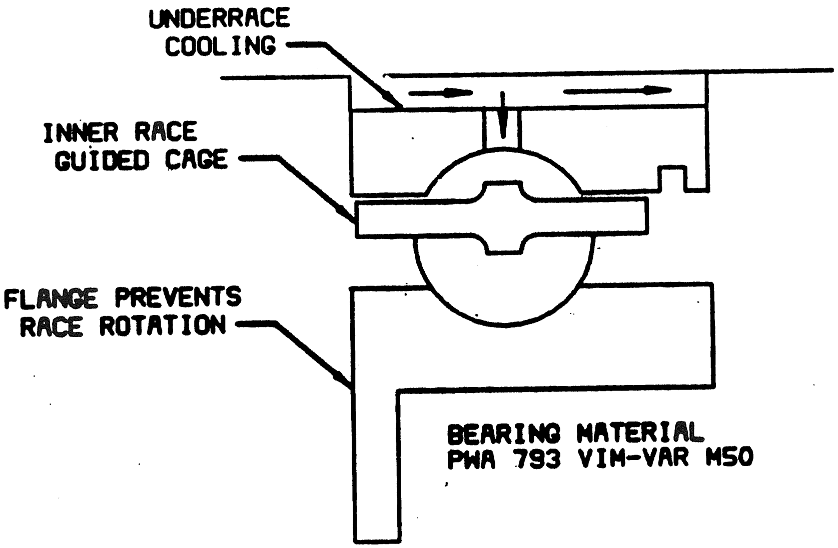
- **Lubrication:** Reduce friction (lowers heat generation) and reduce surface related stress.

LUBRICATION / COOLING



Under Race Scheme

LUBRICATION / COOLING



Under Race Scheme

LUBRICATION / COOLING

Cyrogenic Applications

- o Cooling by emersion in cryogen
- o Lubrication by transfer film from the cage

MATERIALS

	TYPICAL	CRYOGENIC
RACES	o M-50 (PWA 725, AMS 6490) o 52100 (PWA 723, AMS 6440)	o 440C (AMS 5618)
ROLLING ELEMENTS	o Same as above o SiN_2	o 440C
CAGE	o AMS 6414 with silver plate	o Ball Bearings - Bonze filled Teflon - Glass fibers filled with Teflon o Roller Bearings - Silicon Bronze with silver lead plate - Glass fibers filled with Teflon

FAILURE MODES

ALL SPALLS ARE NOT EQUAL

- o Rolling contact fatigue
- o Corrosion
- o Misassembly
- o Manufacturing or material related

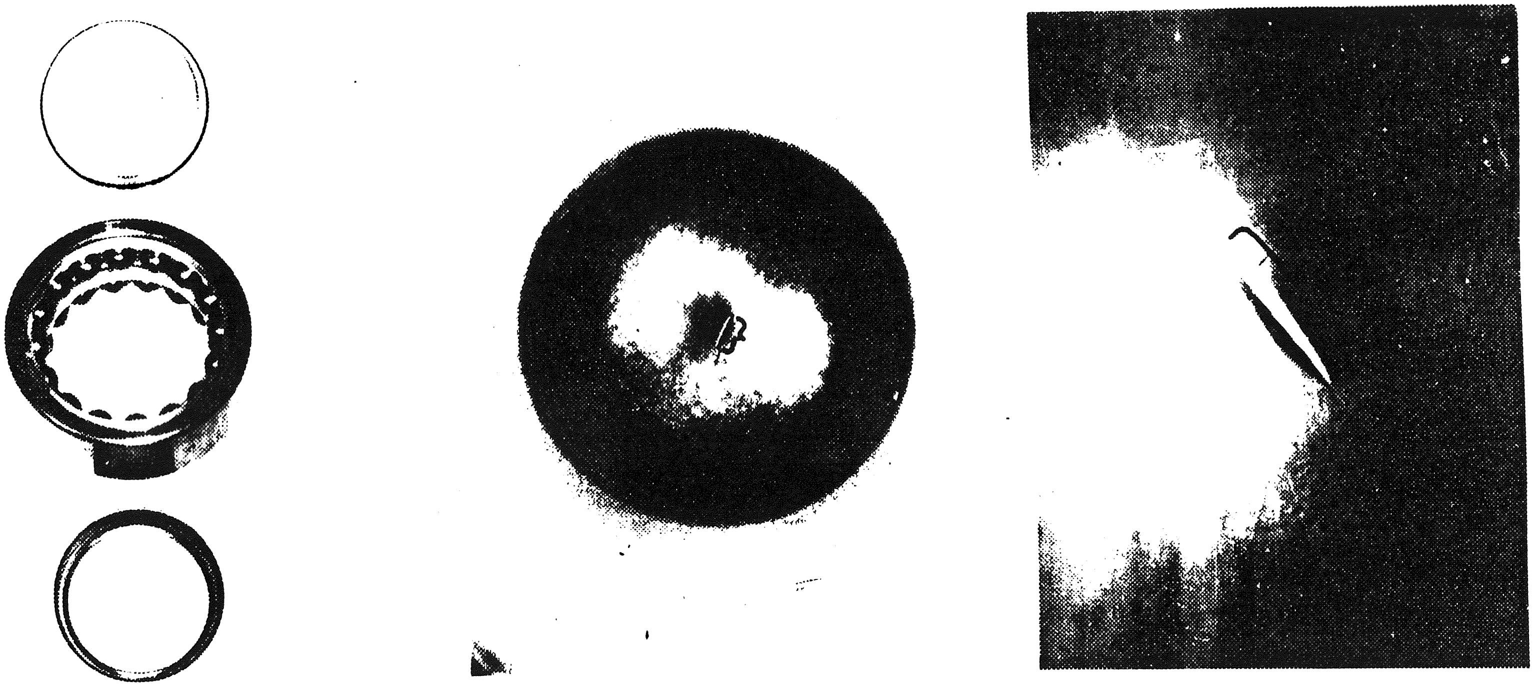
SKID DAMAGE

- o Skid
- o Wear

CYROGENIC APPLICATIONS

- o Cooling/Lubrication
- o Thrust overload
- o Roller stability
- o Wear

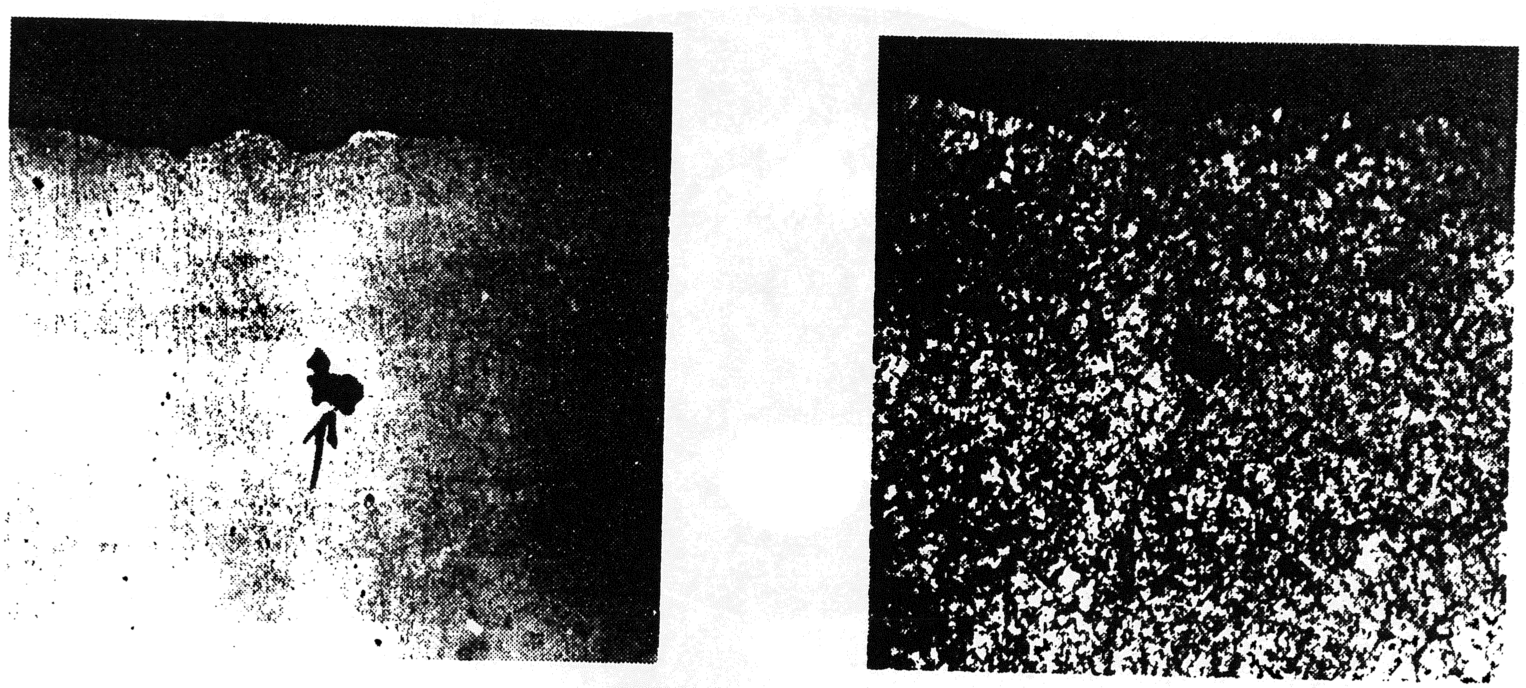
DENTS



F100 G/B PTO Shaft Duplex Ball Brg

AV 175498
793107
BT3-186

MATERIAL INCLUSION

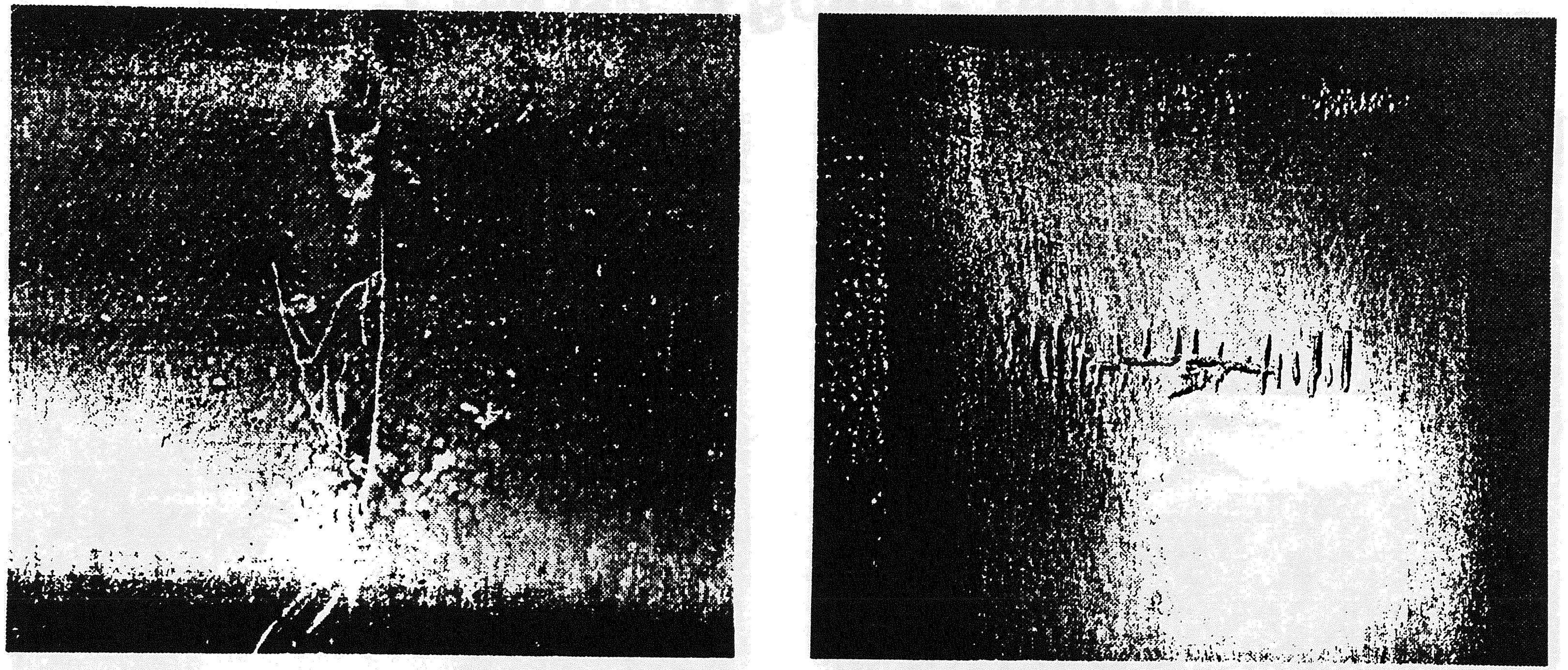


F100 No. 3 Ball

MATERIAL INCLUSION

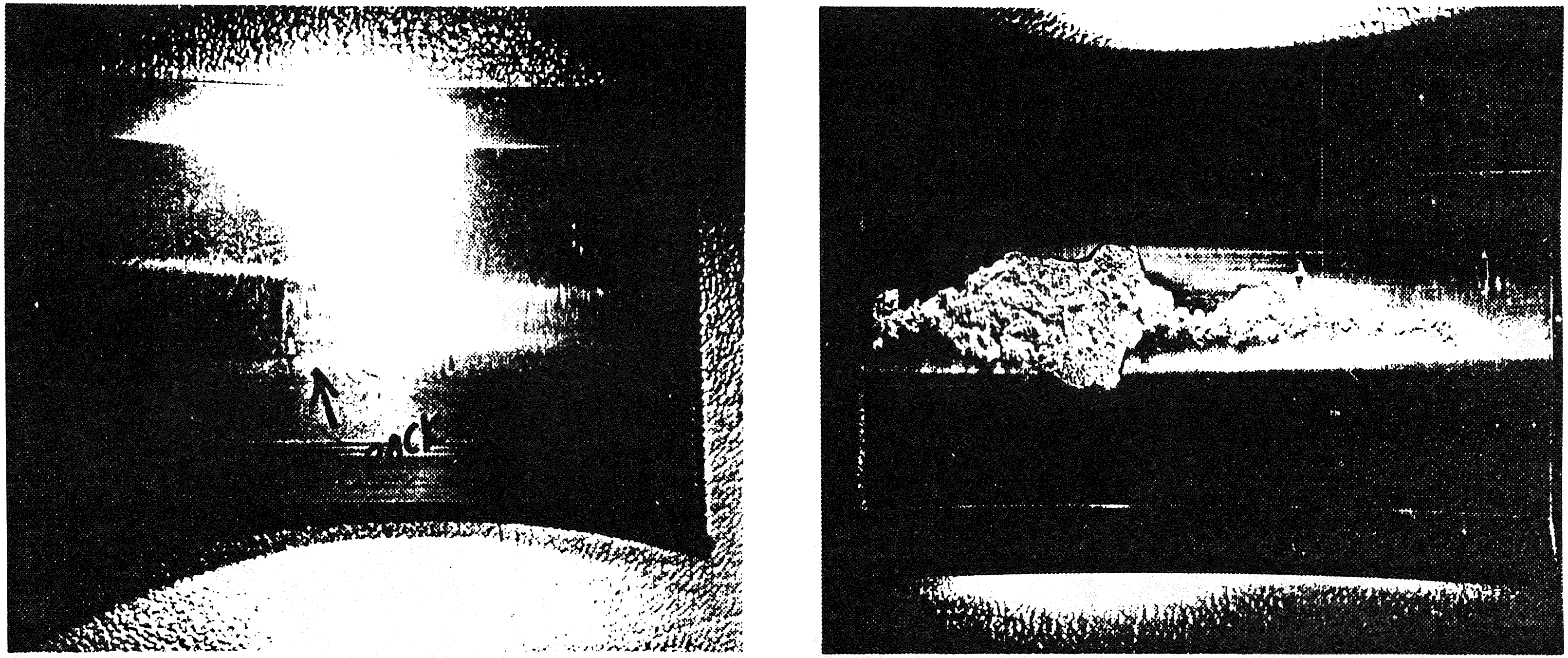
F100 No. 3 Ball

GRIND BURNS



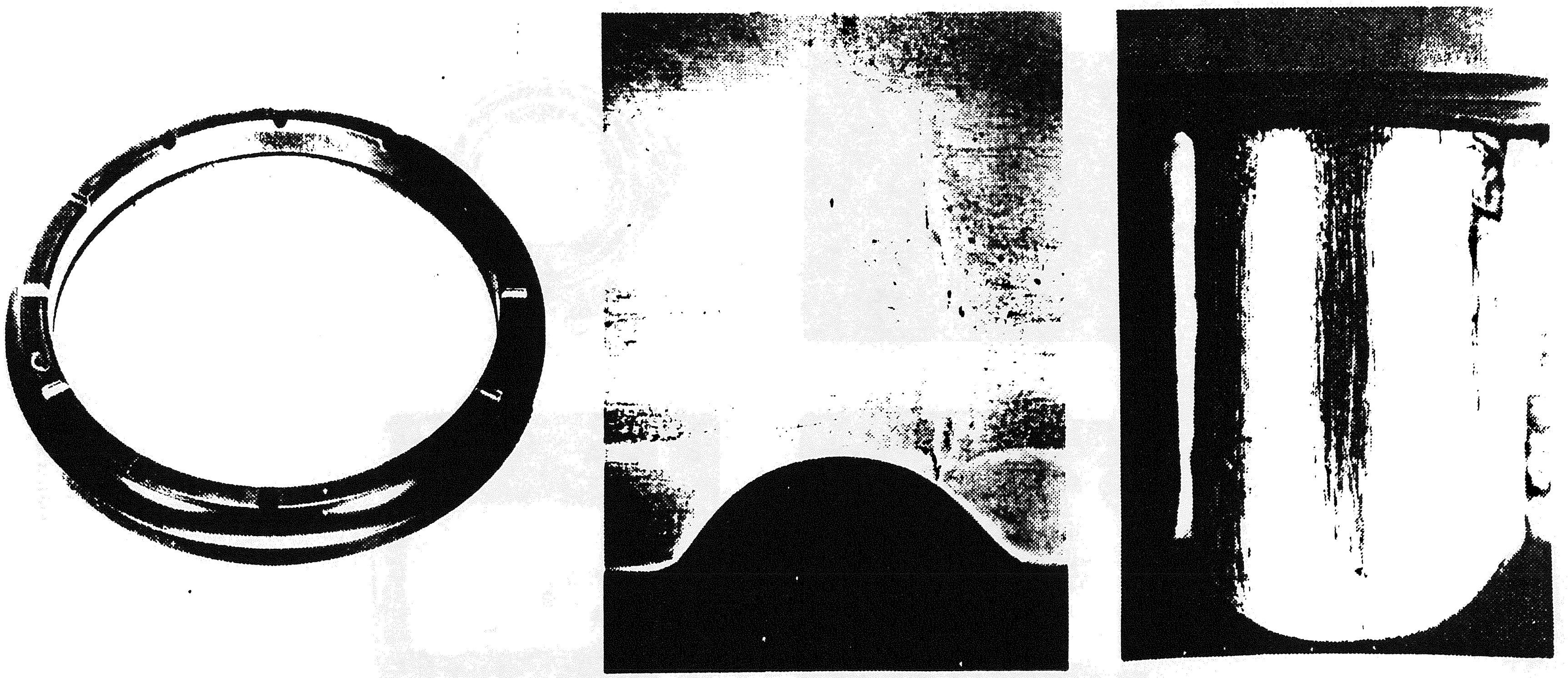
F100 No. 5 Roller - 1060 hr

GRIND BURNS



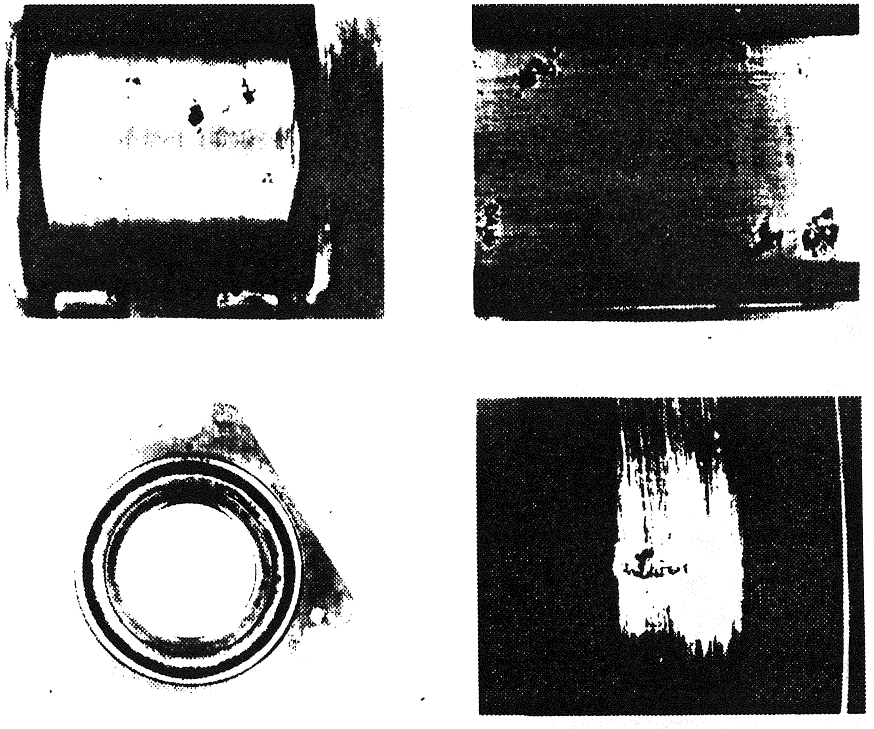
F100 No. 5 Roller - 1060 hr

GRIND BURNS



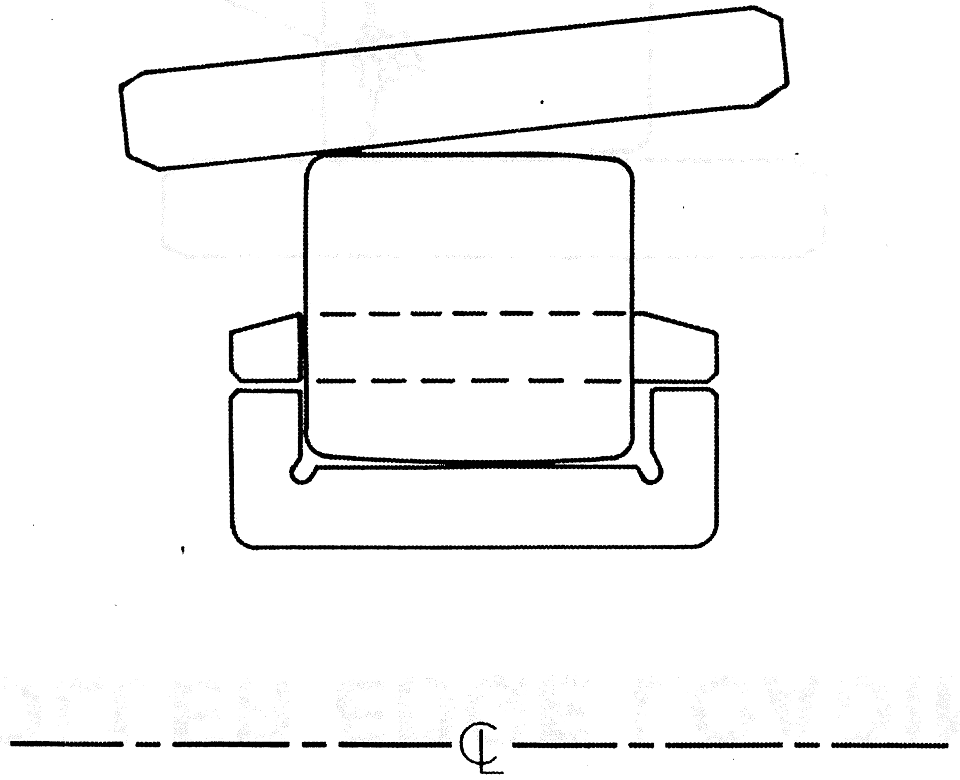
F100 Upper Tower Shaft Ball

SURFACE PITS

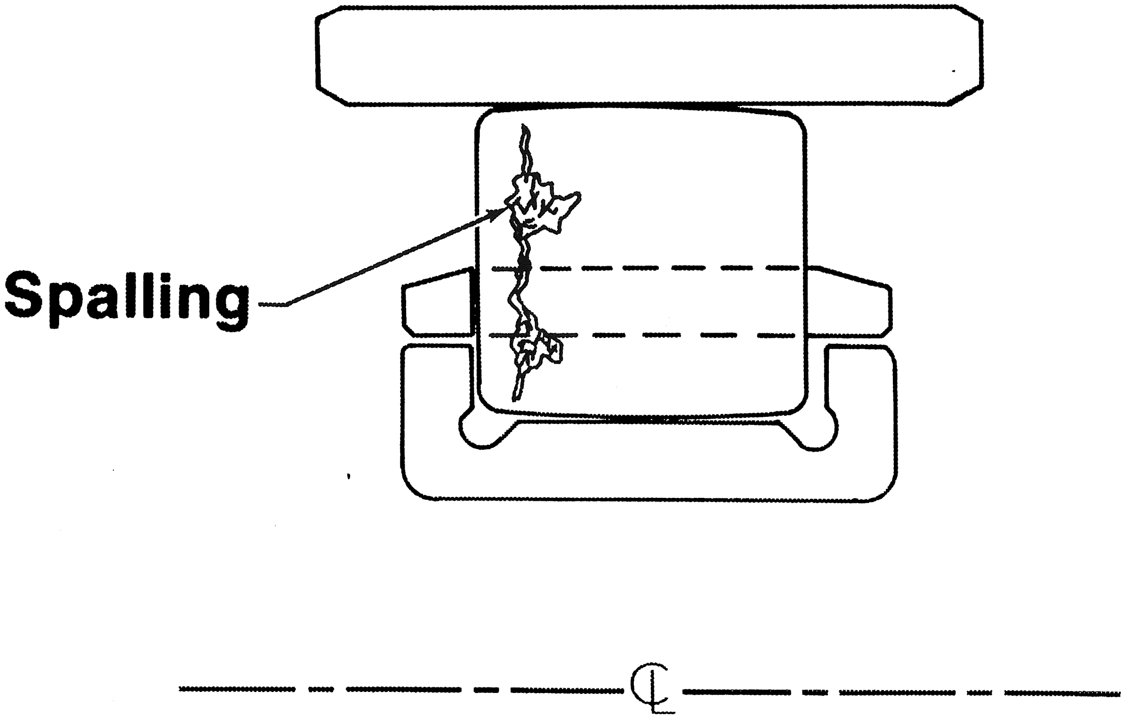


F100 G/B Alternator Shaft Roller - 1188 hr Operation

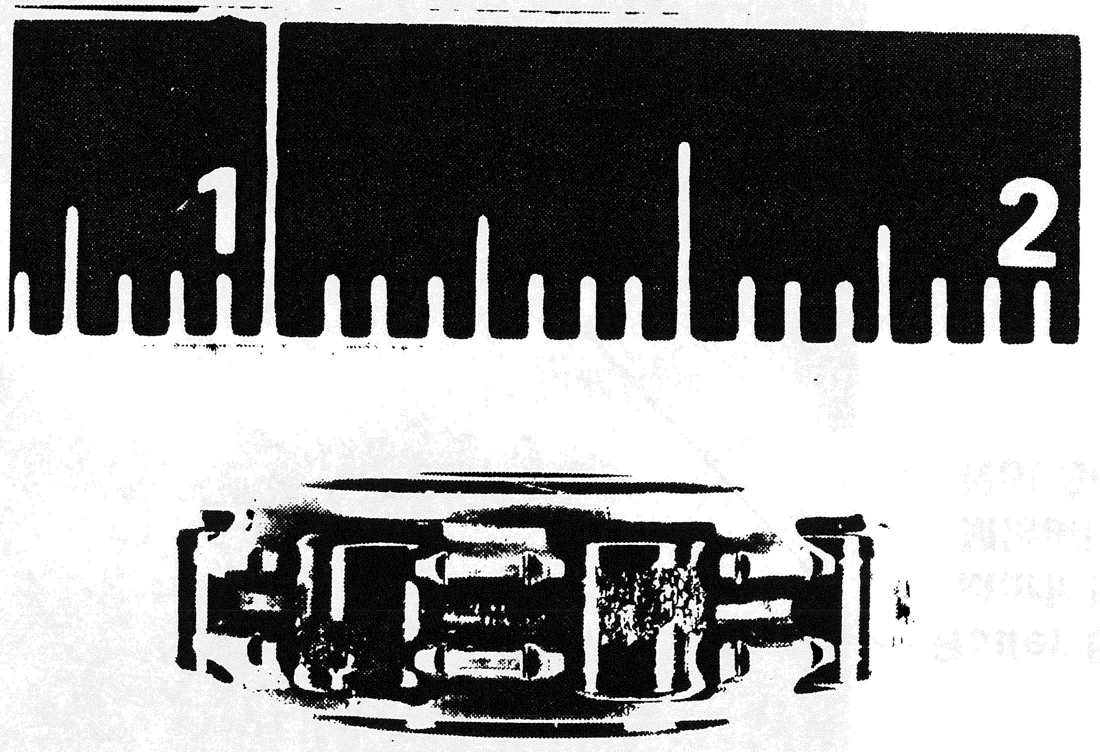
ROLLER EDGE LOADING



ROLLER EDGE LOADING

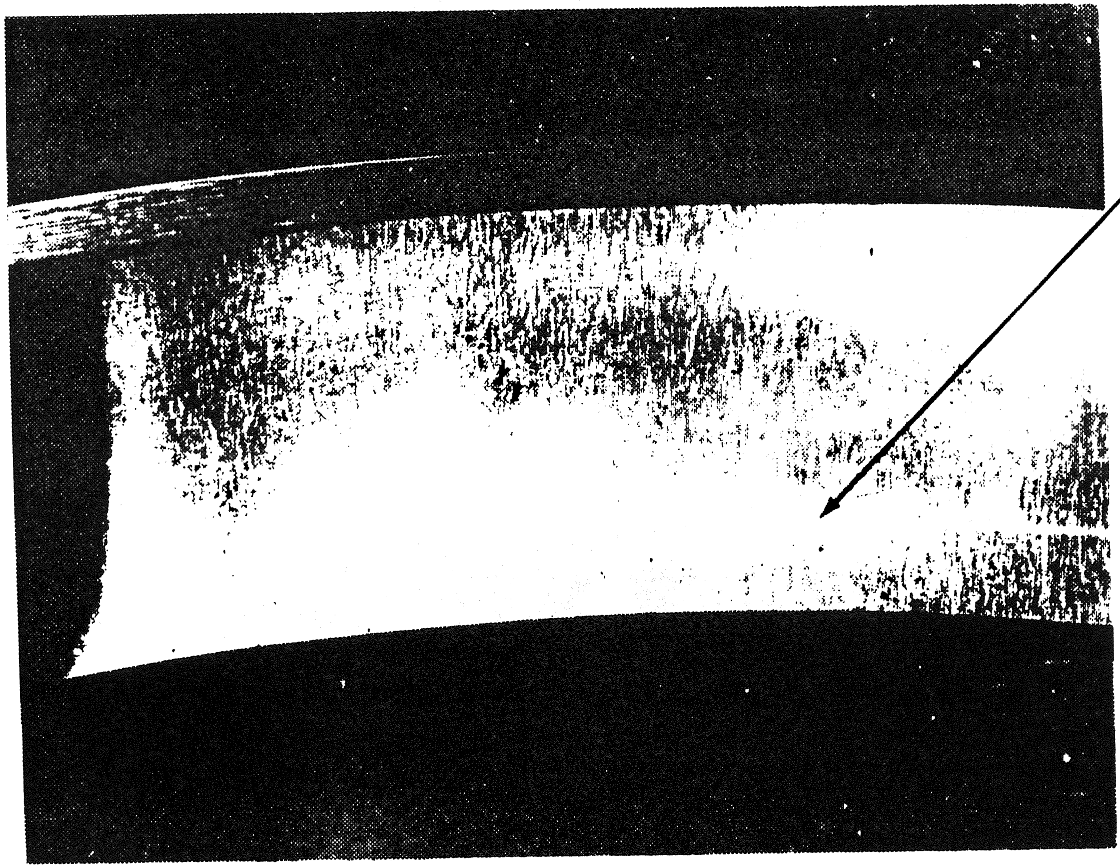


ROLLER EDGE LOADING



F100 G/B Oil Pump Idler Shaft Roller

ROLLER EDGE LOADING

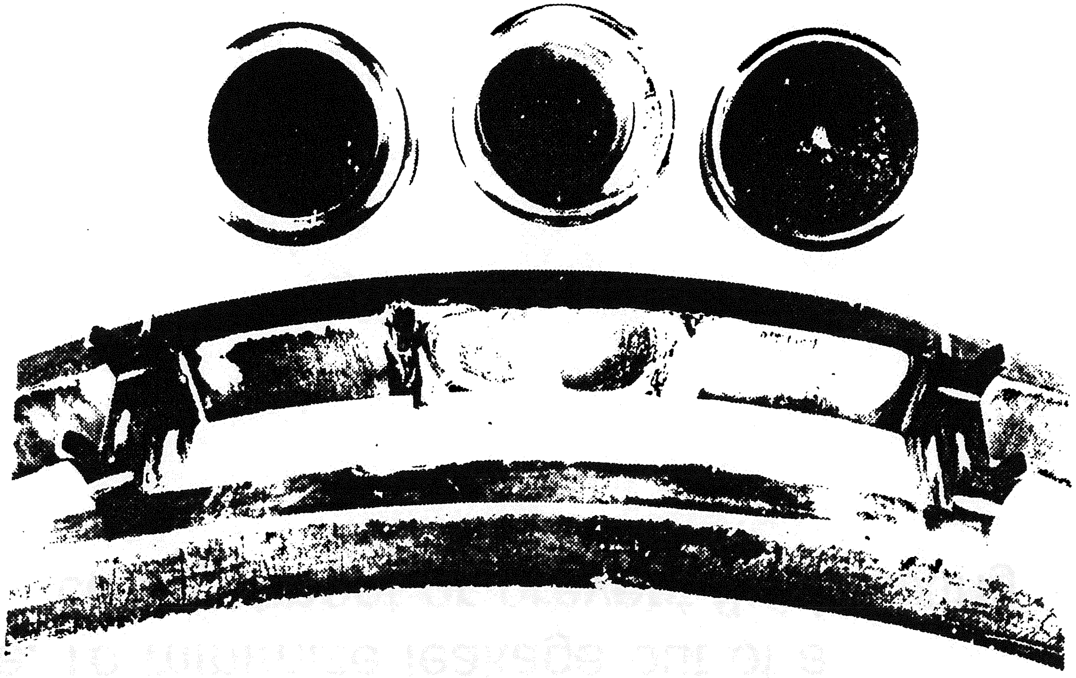


Roller Edge Load
Mark Due To
Misalignment -
Not Scoring

Orenda Rig

AV 175474
790106
BT3-239

WEAR

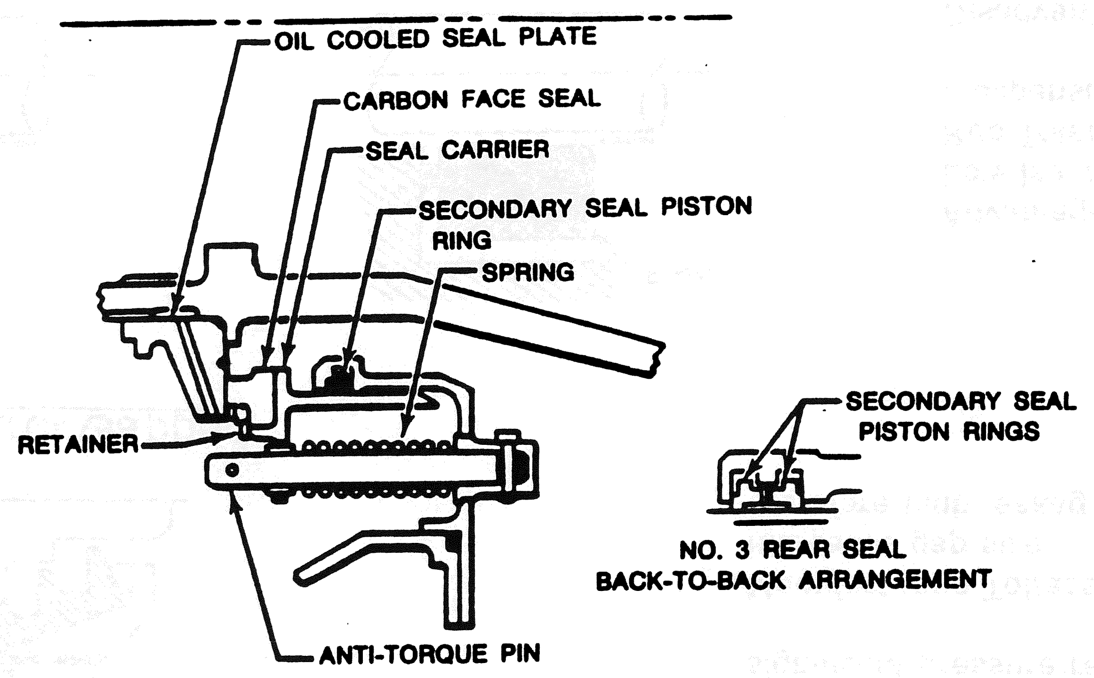


F100 No. 4 Roller Bearing Eccentric End Wear

SEALS

Purpose: To minimize leakage out of a compartment or prevent fluid mixing

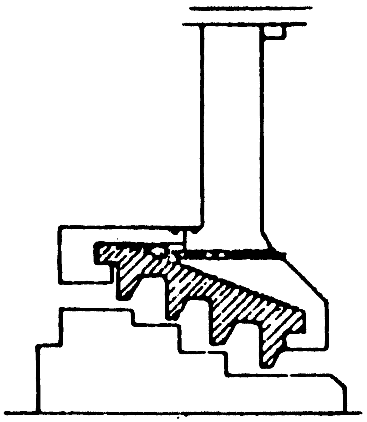
SEALS TYPES



Face Seals

Advantage:	Tight sealing capabilities
Disadvantage:	Space Requirements Speed limited to less than 450 surface FPS

SEALS TYPES

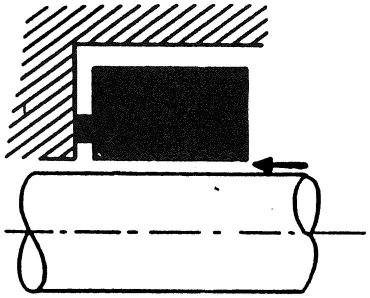


Labyrinth Seal

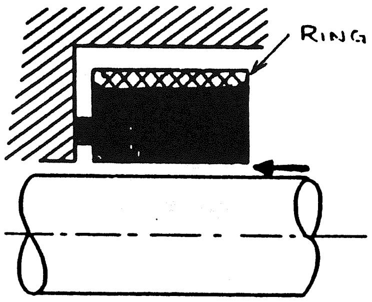
Advantage:	Durable Simple
Disadvantage:	Requires space to achieve significant pressure reduction Manufacturing Tol/assembly increases gap size therefore high leakages

Control Gapped

Constant Radius



Without Compressive Ring



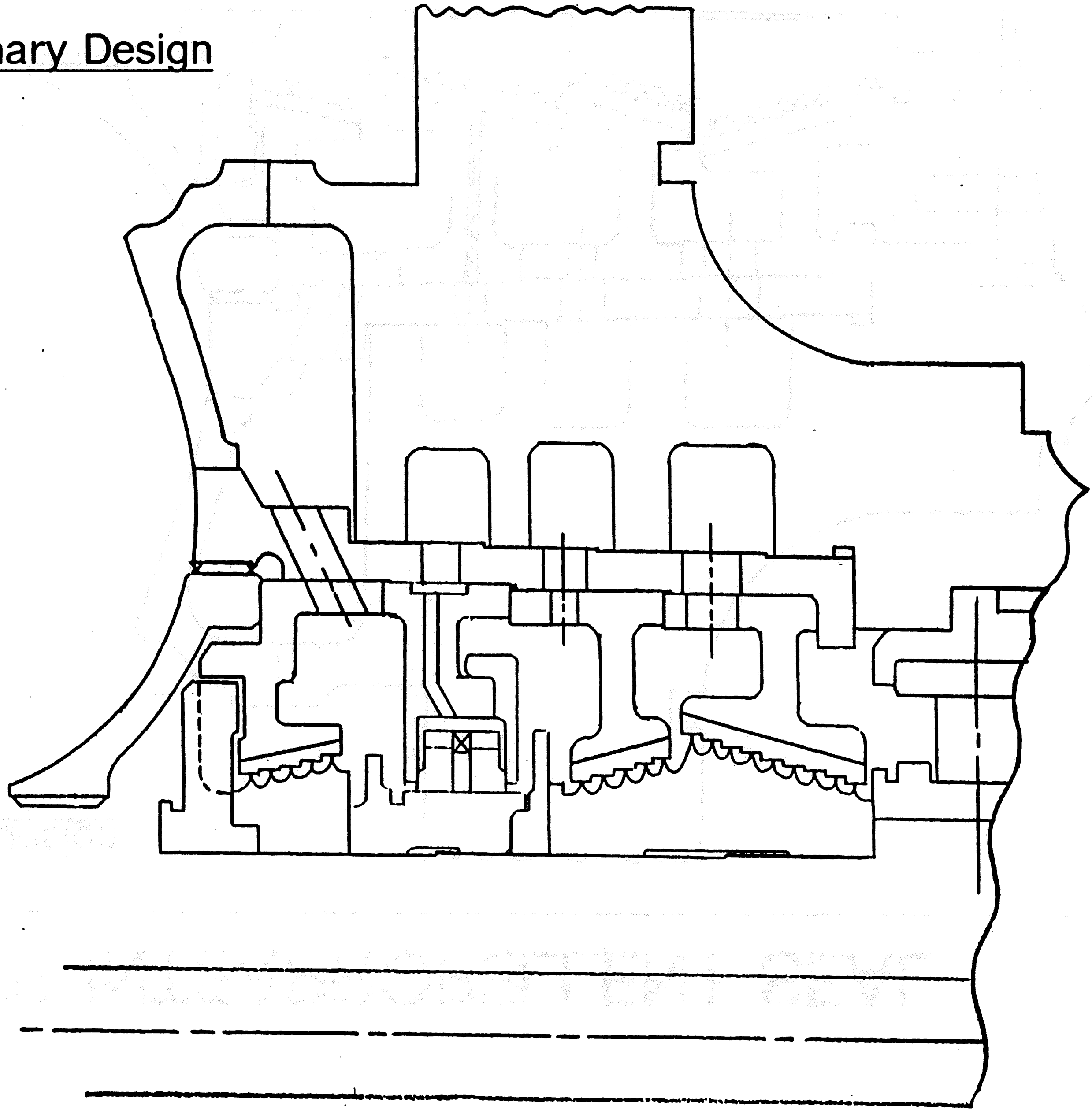
With Compressive Ring

Advantage:
Low leakages
Rub Tolerant*
Inexpensive

Disadvantage:
Durability

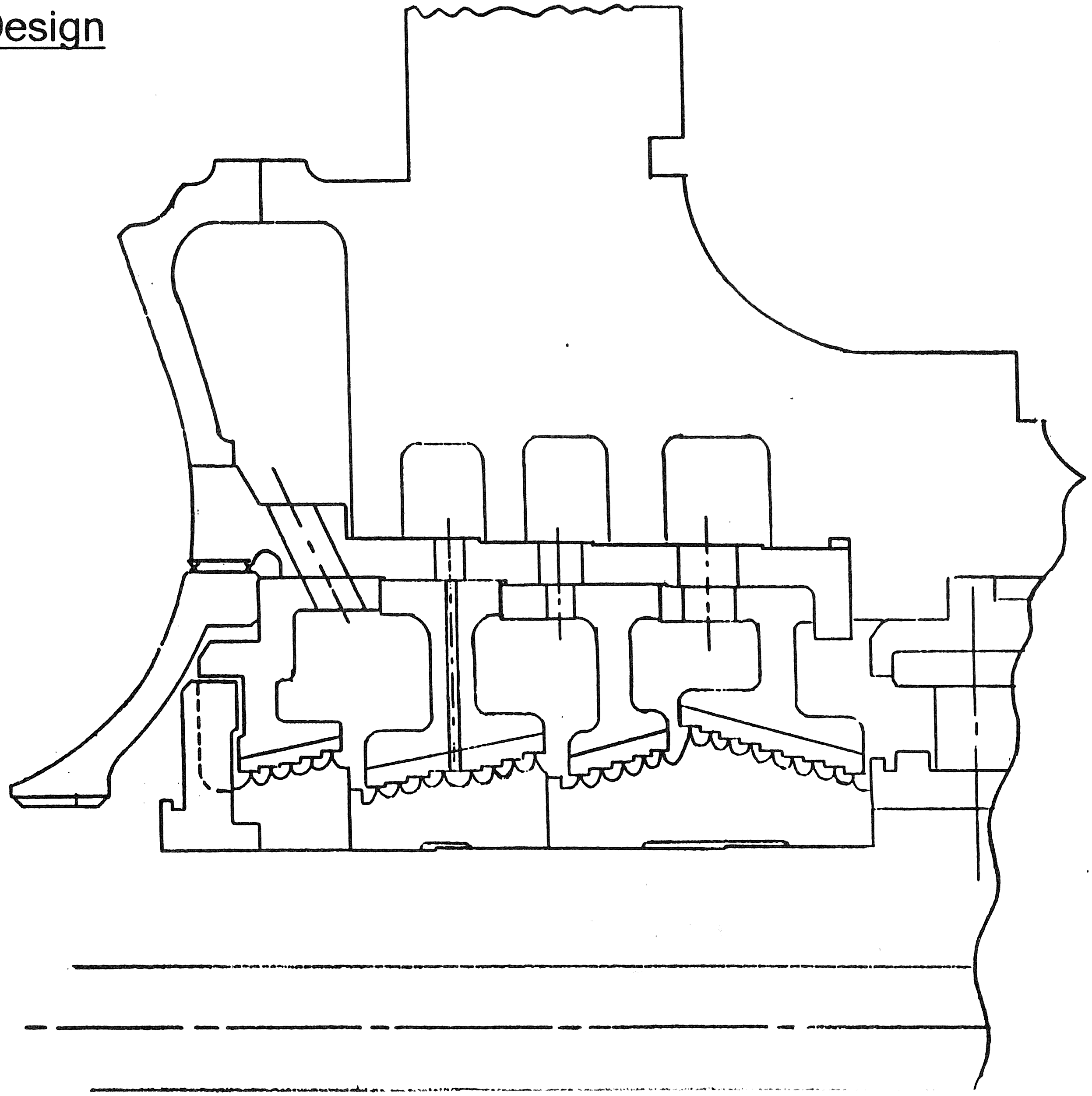
HPOTP INTERPROPELLENT SEAL

Preliminary Design



HPOTP INTERPROPELLENT SEAL

Back-up Design



MON 6/17
7:00 AM

INTRODUCTION TO ROCKET PLUMBING DESIGN AND SHIPPING EQUIPMENT

Al Palgon

ROCKET ENGINE
PLUMBING DESIGN

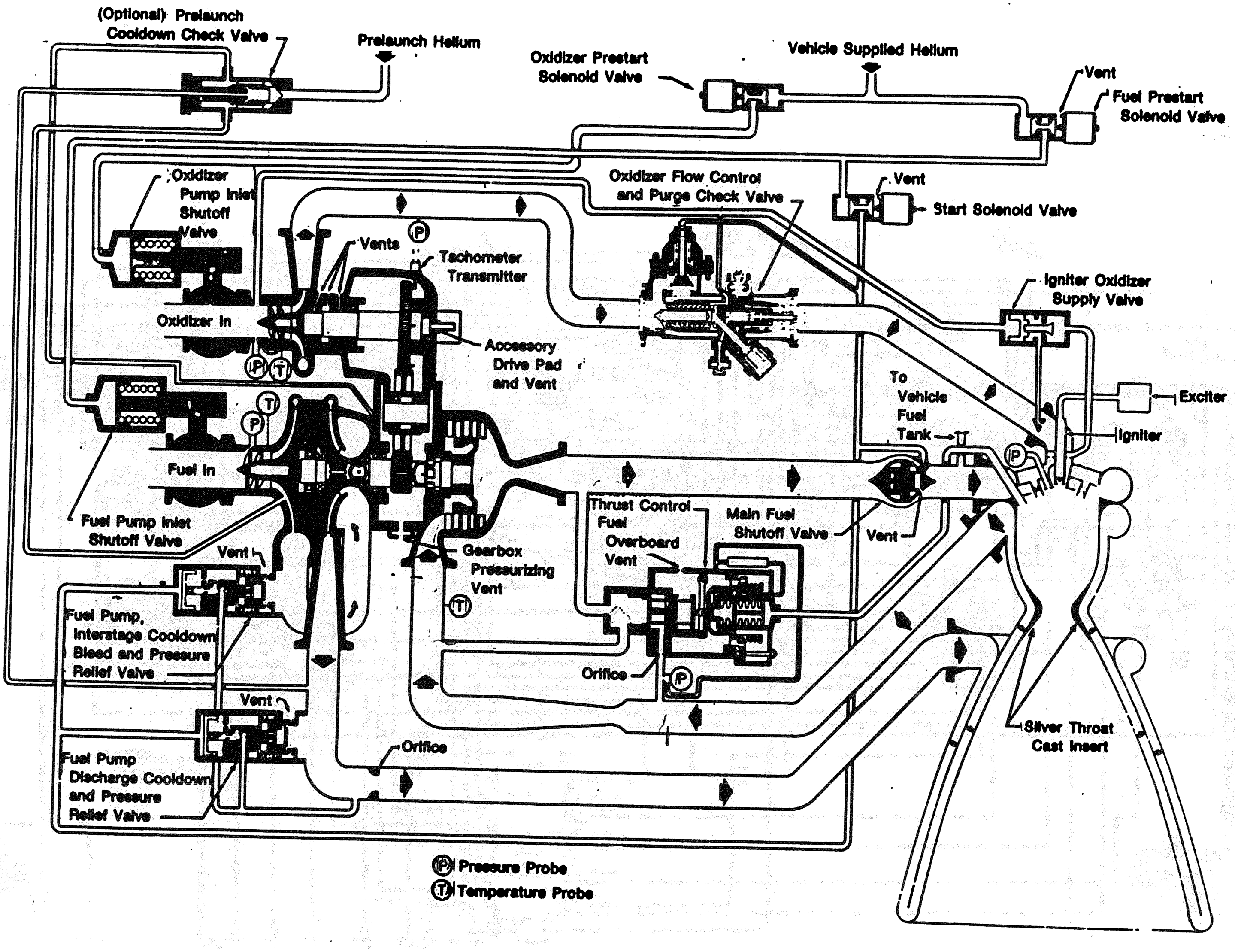
AKA – FLOW DUCTING

TYPES OF PLUMBING

- FIXED PROPELLANT FEED
- GIMBALLED (FLEXIBLE) PROPELLANT FEED
- PNEUMATIC
- INSTRUMENTATION
- HYDRAULIC
- (HOT GAS)

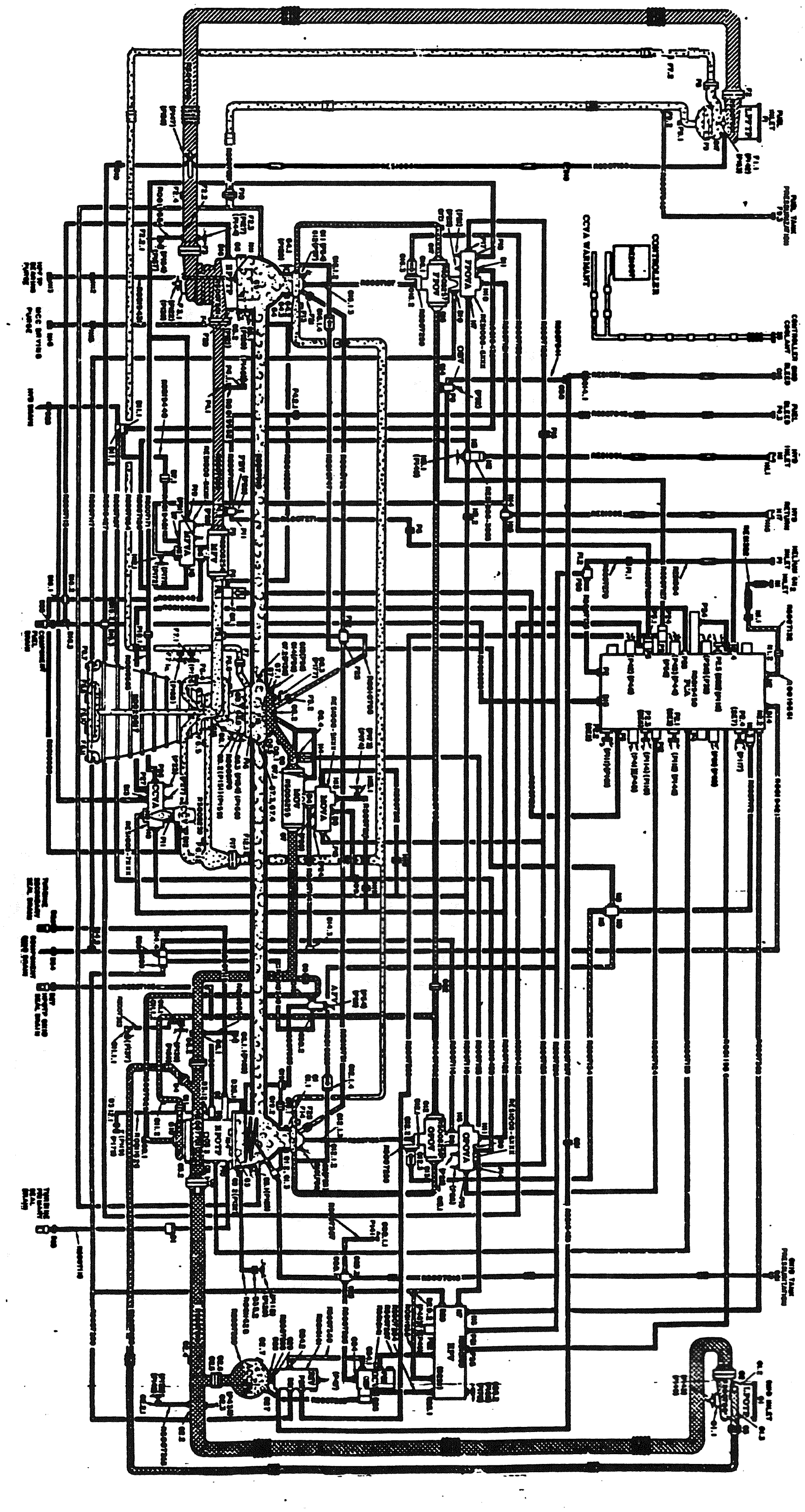
ENGINE FLUIDS

	TEMP. RANGE (°F)	USE
OXIDIZER - LOX	-300 to +200	COMBUSTION
FUEL - H_2, RP-1, METHANE	-420 to +200	COMBUSTION
PNEUMATICS - He, N_2	-350 to +400	PURGES, VALVE ACTUATION
HYDRAULICS	+40 to +240	VALVE ACTUATION



Propellant Flow Schematic for RL10A-3-3A Engine

FD 226267

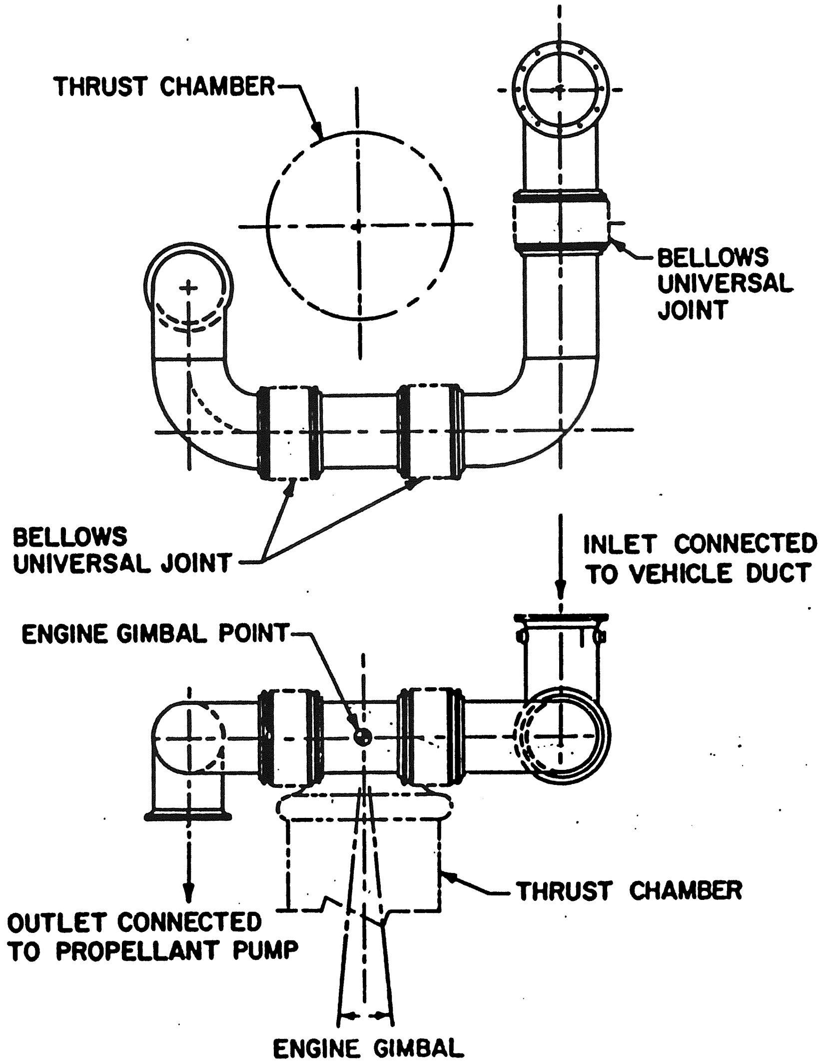


GIMBALLED PROPELLANT FEED PLUMBING

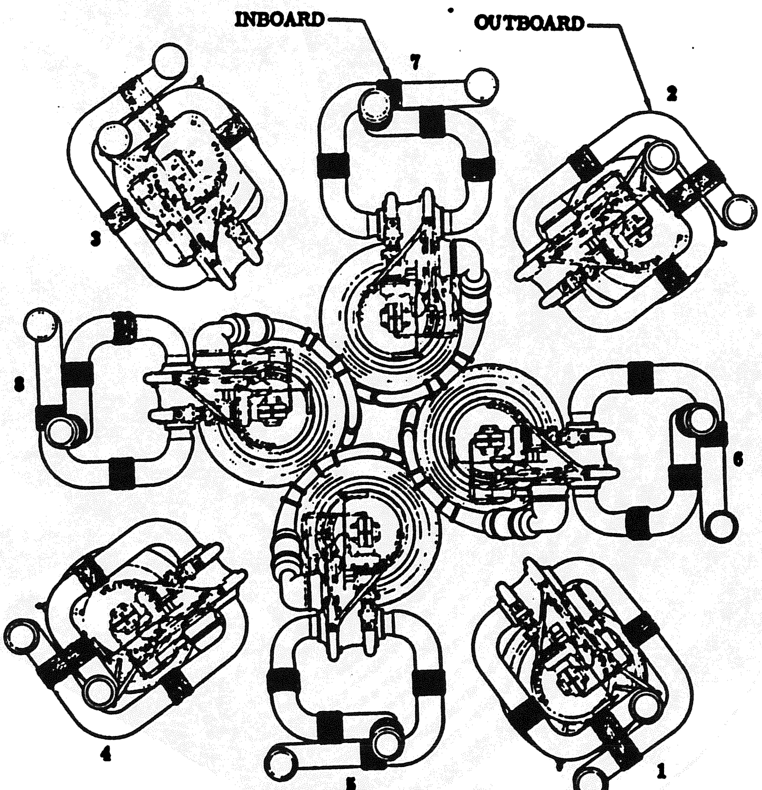
- REQUIRED TO PERMIT ENGINE TO GIMBAL (THRUST VECTORING)

- VEHICLE HARDWARE (CENTAUR)

- ENGINE HARDWARE (SSME)



ENGINE PROPELLANT INLET DUCT IDEAL FLEX JOINT ORIENTATION

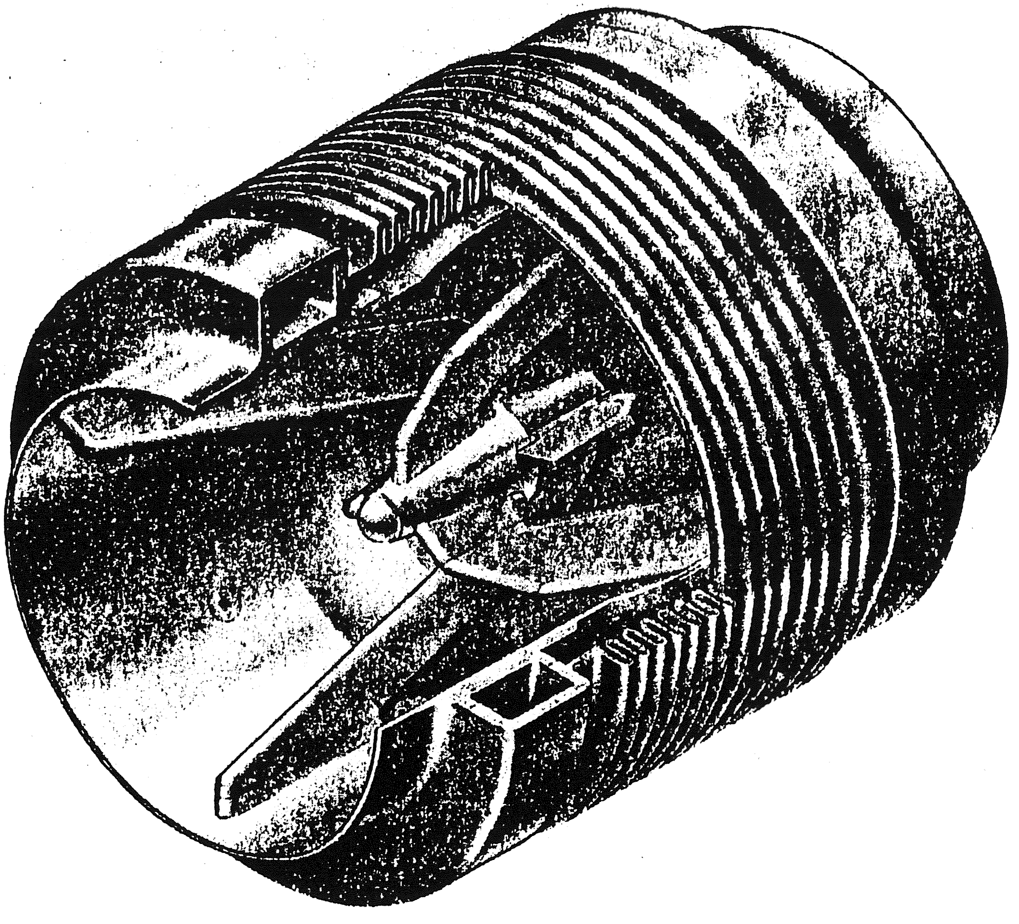


NOTE

**ENGINE ARRANGEMENT AS VIEWED
FROM FORWARD TO AFT, WITH
VEHICLE STAGE IN TRANSPORT
POSITION**

SATURN S-IB STAGE ENGINE CLUSTER

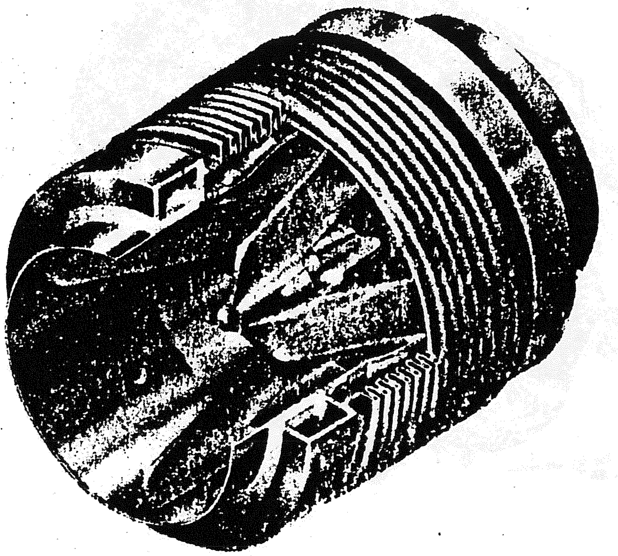
OXIDIZER PUMP DISCHARGE FLEX JOINT



LC305-290

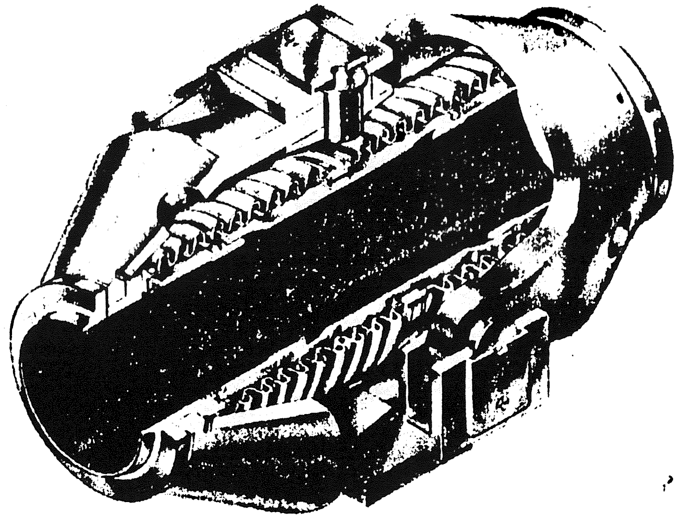
Rockwell International
Rocketdyne Division

INTERNALLY TIED FLEX JOINTS



OPERATING PRESSURE	300 TO 600 PSIA
TEMP.	-290F
I.D.	6.3 INCH LINER
ANGULAR DISPLACEMENT	±13°
LIFE	200 OPERATIONAL 1400 NON-OPERATIONAL FULL DEFLECTION CYCLES
MATERIAL	INCONEL 718 ARMCO 21-6-9
TYPICAL OF	LPOP DISCH.

EXTERNAL GIMBAL RING (LONG) FLEX JOINTS



OPERATING PRESSURE ....4000 TO 6000 PSIA

TEMP RANGE.................................... AMB. TO 670F

I.D. .... .75 TO 2.70 INCH LINER

ANGULAR DISPLACEMENT........................ ±13.25°

LIFE.. 200 OPERATIONAL
1400 NON-OPERATIONAL
FULL DEFLECTION CYCLES

MATERIAL.. INCONEL 718
INCONEL 903
TITANIUM

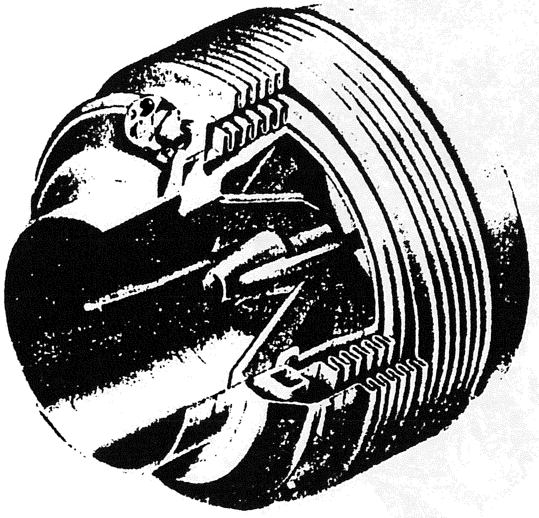
TYPICAL OF

- LPOP TURBINE DRIVE
- OXID TANK PRESSURANT
- LPFP TURBINE DRIVE
- LPFP TURBINE DISCH.

LC308-180A

Rockwell
International
Rocketdyne Division

INTERNALLY TIED WITH INSULATING JACKET FLEX JOINTS

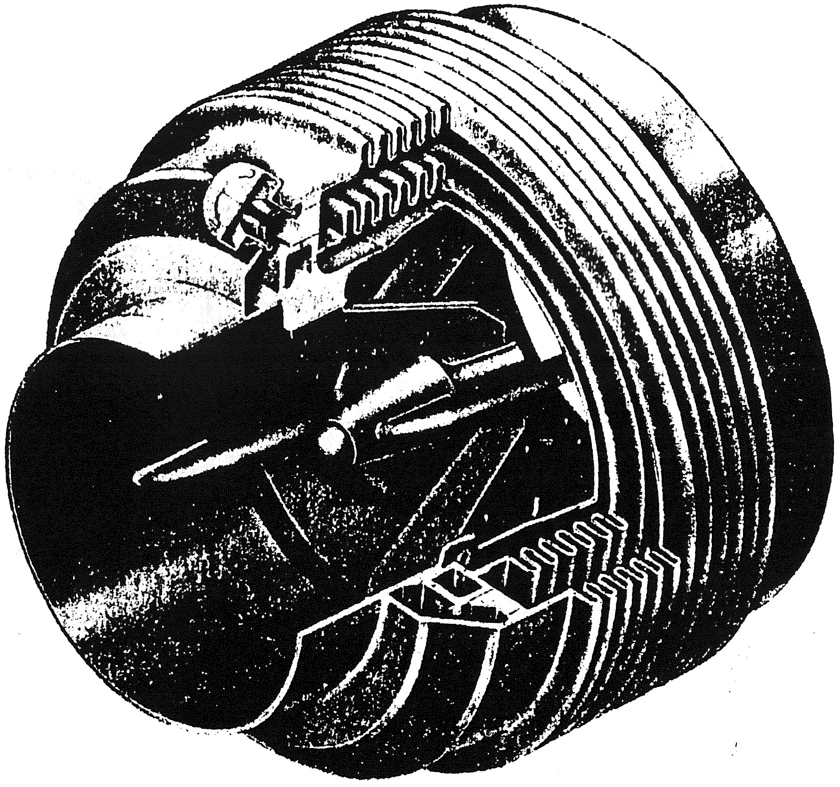


OPERATING PRESSURE	280 TO 300 PSIA
TEMP.	420F
I.D.	5.20 INCH LINER
ANGULAR DISPLACEMENT	±11.5°
LIFE	200 OPERATIONAL 1400 NON-OPERATIONAL FULL DEFLECTION CYCLES
MATERIAL	INCONEL 718 ARMCO 21-6-9
TYPICAL OF	LPFP DISCH.

LC308-183A

Rockwell International
Rocketdyne Division

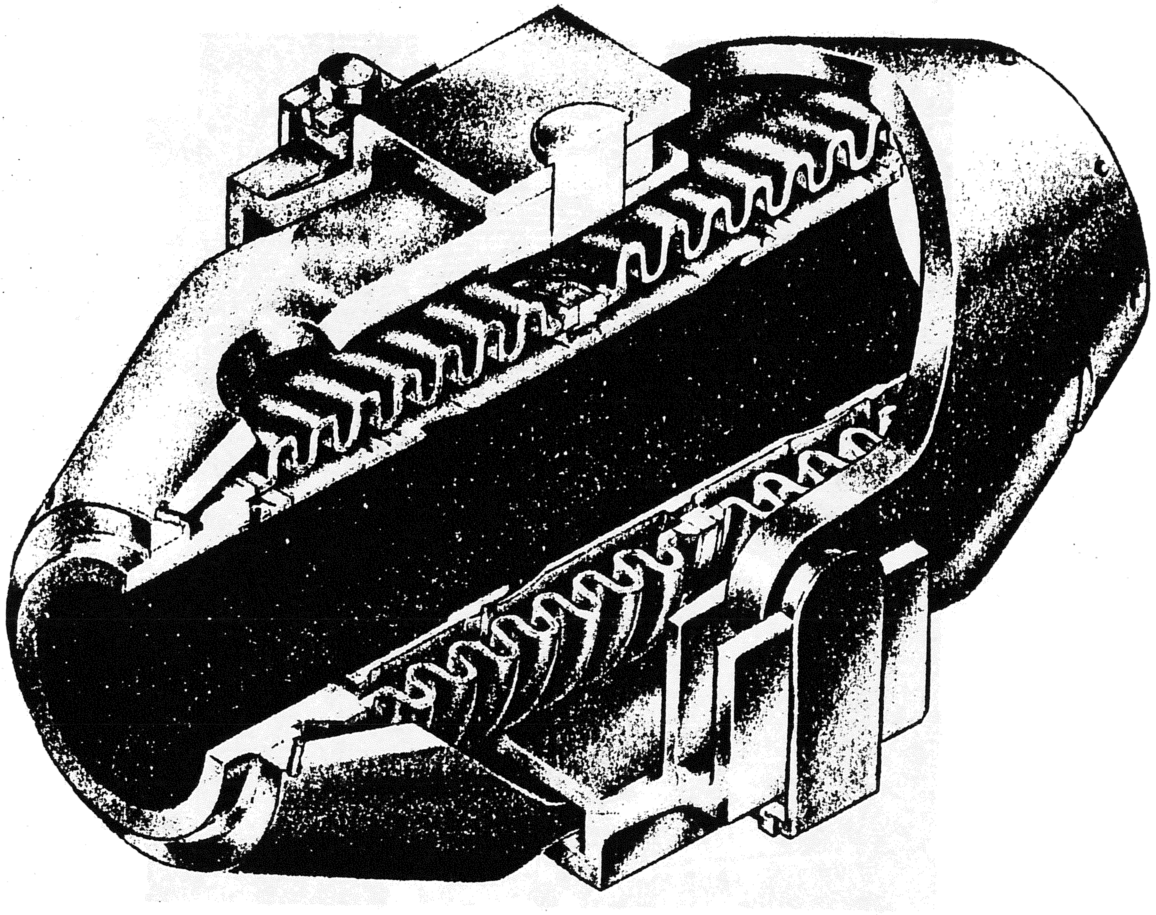
LPFP DISCHARGE FLEX JOINT

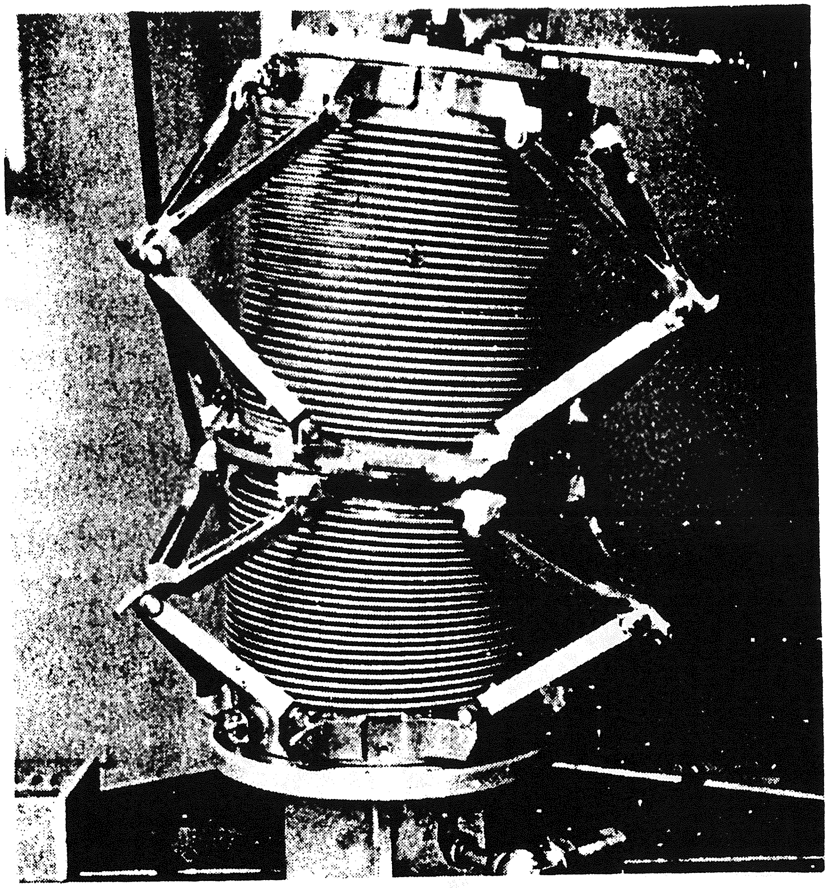


LC301-389A

Rockwell International
Rocketdyne Division

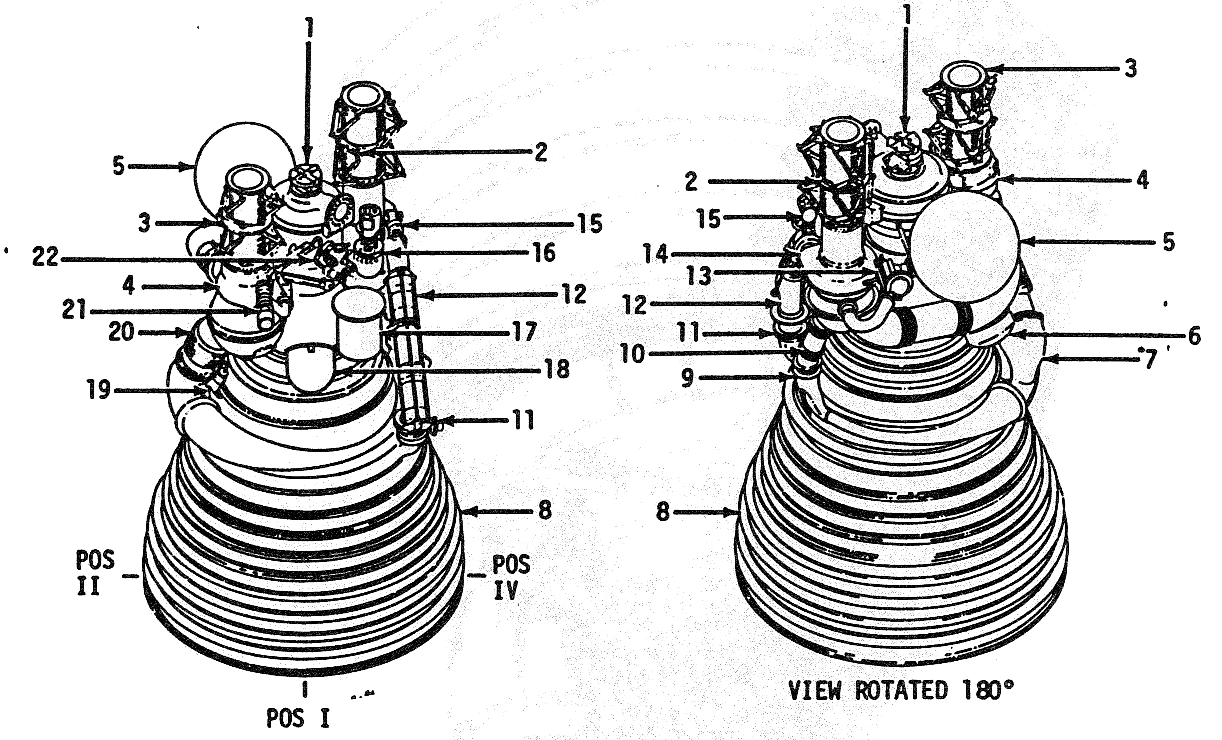
LPFP TURBINE DISCHARGE FLEX JOINT





Typical flexible propellant supply duct for a large turbopump-feed engine system. (Note: Duct is mounted in test fixture with fluid pressure connections.)

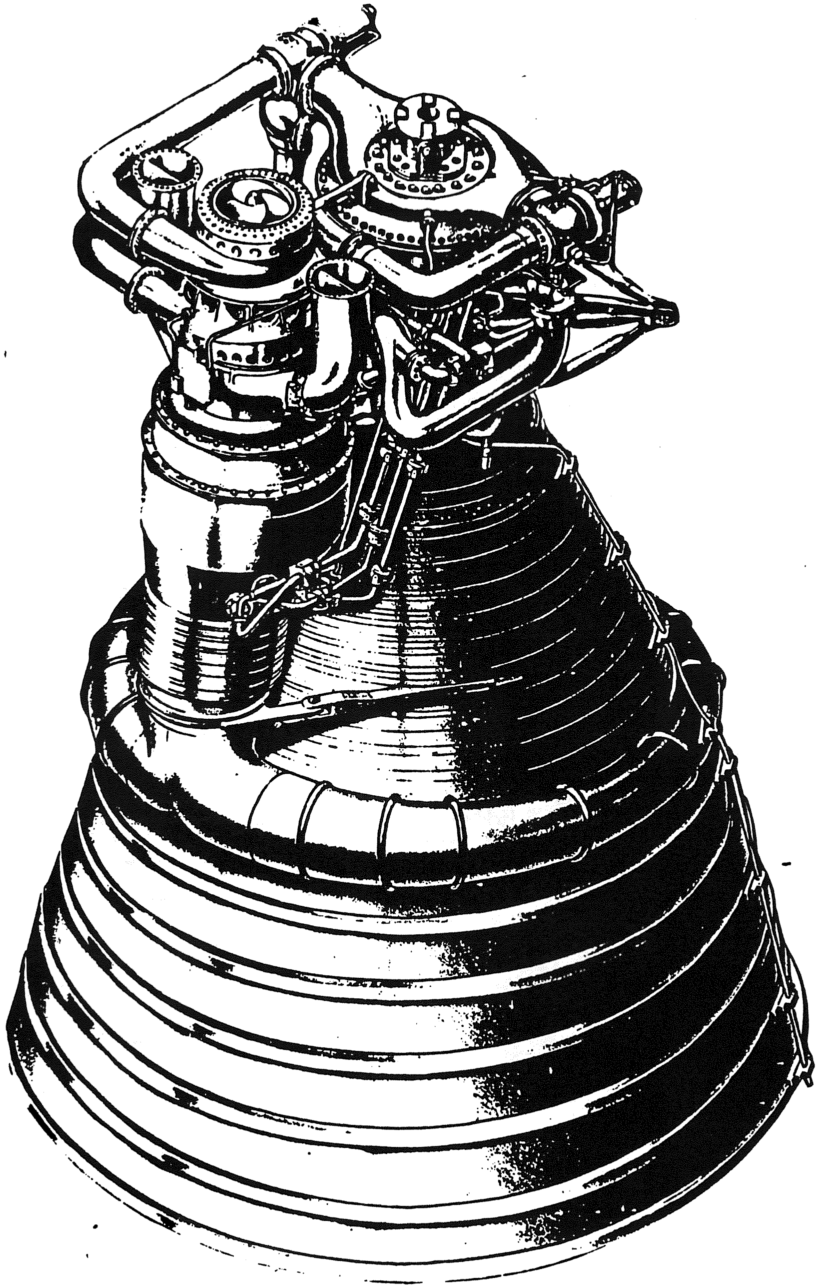
J-2 ROCKET ENGINE



VIEW ROTATED 180°

1. GIMBAL
2. FUEL INLET DUCT
3. OXIDIZER INLET DUCT
4. OXIDIZER TURBOPUMP
5. START TANK
6. AUXILIARY FLIGHT INSTRUMENTATION PACKAGE
7. EXHAUST MANIFOLD
8. THRUST CHAMBER
9. OXIDIZER TURBINE BYPASS VALVE
10. TURBINE BYPASS DUCT
11. MAIN FUEL VALVE
12. HIGH PRESSURE FUEL DUCT
13. START TANK DISCHARGE VALVE
14. FUEL TURBOPUMP
15. FUEL BLEED VALVE
16. GAS GENERATOR
17. ELECTRICAL CONTROL PACKAGE
18. PRIMARY FLIGHT INSTR. PACKAGE
19. ANTI-FLOOD CHECK VALVE
20. HEAT EXCHANGER
21. PROPELLANT UTILIZATION VALVE
22. PNEUMATIC CONTROL PACKAGE

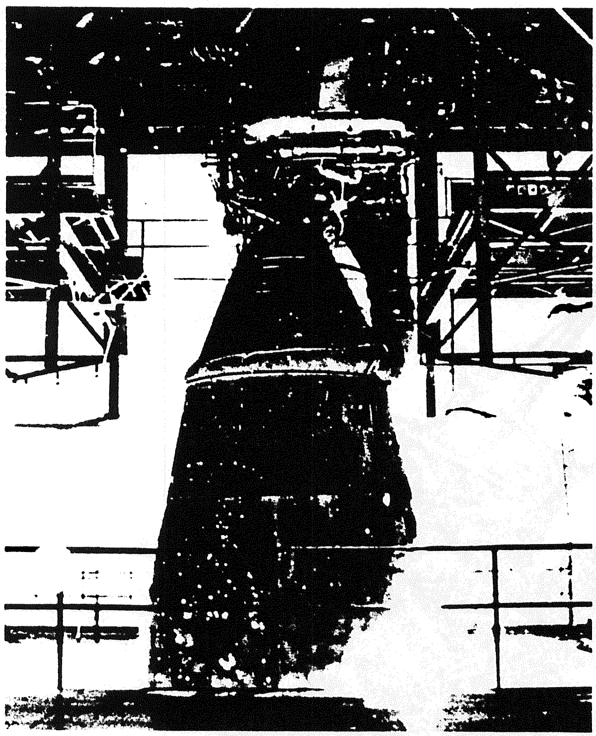
F-1



F-1

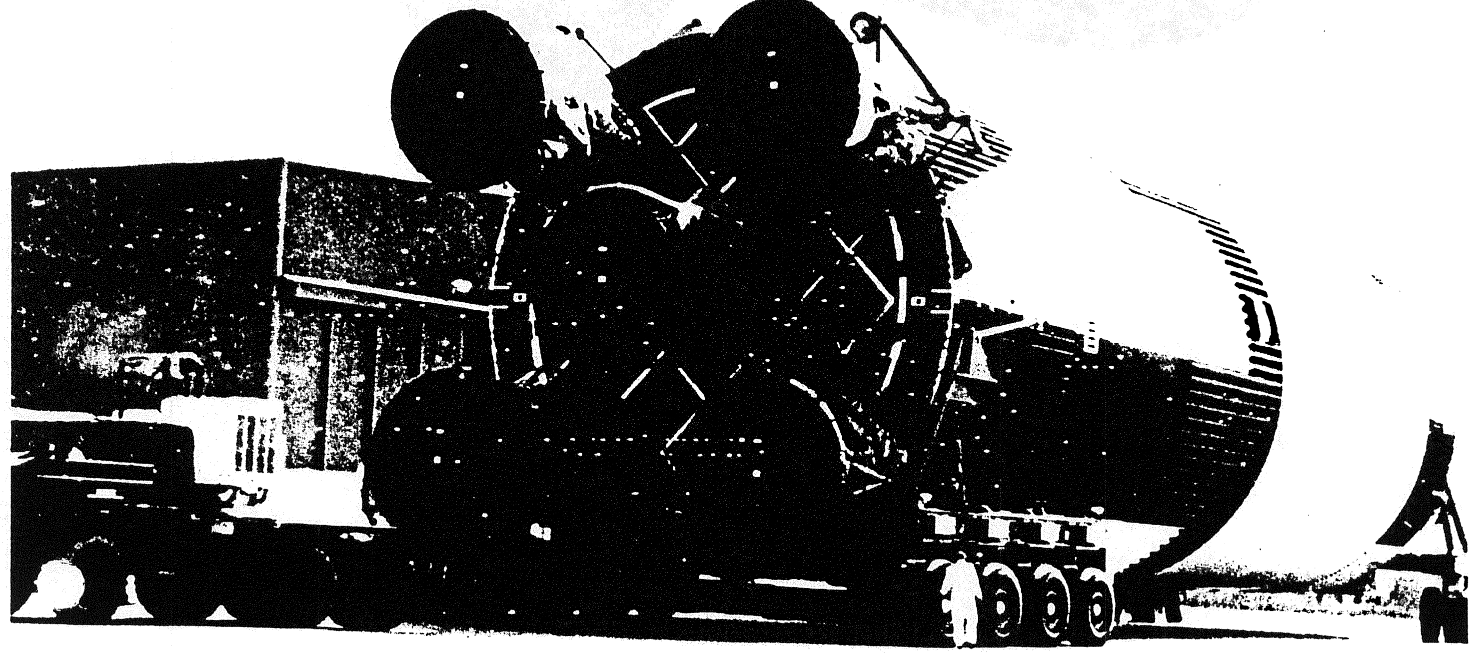
Rockwell International
Rocketdyne Division

Still the most powerful rocket engine ever built, the F-1's million-and-a-half pounds of thrust (quintupled in a cluster of five) lifted men to the moon atop the 363-foot Saturn vehicle. A single-start, fixed thrust engine, the F-1 is gimbaled and uses liquid oxygen as the oxidizer, while RP-1 (kerosene) is used as the fuel, the turbopump lubricant, and the control system fluid. A gas generator utilizing the same propellants drives the turbine, which is direct-coupled to the turbopump.



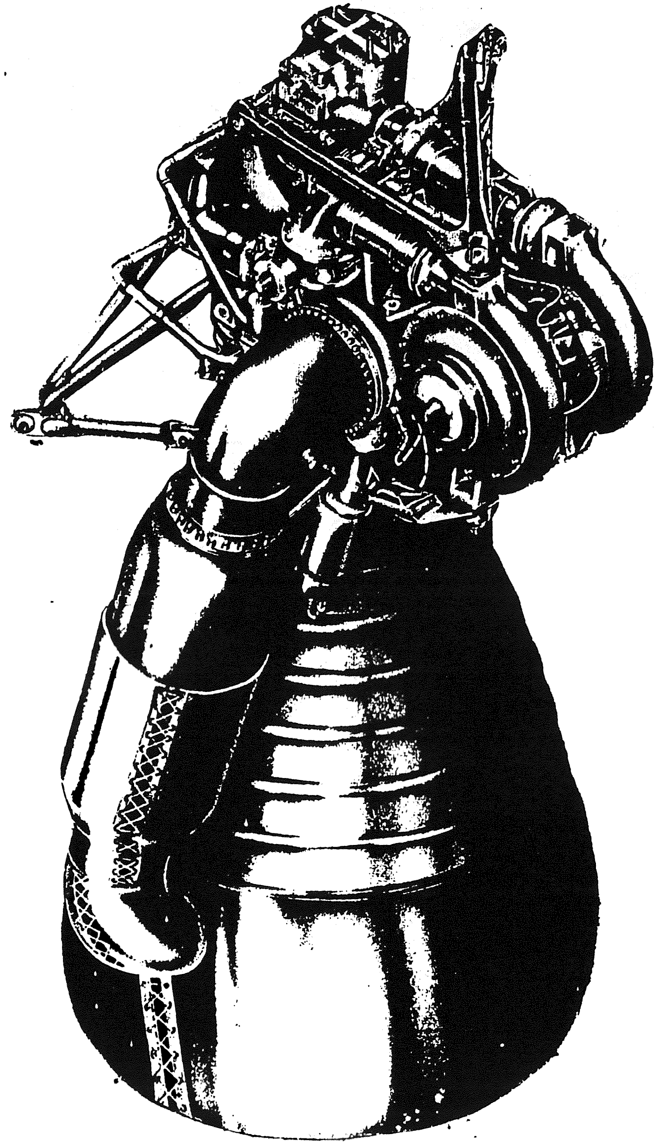
Specifications

Type:	Liquid-Propellant/Pump-Fed
Thrust:	1,522,000 lb
Propellants:	RP-1 (Kerosene)/Liquid Oxygen
Specific Impulse:	265 sec
Mixture Ratio (O/F):	2.27:1
Chamber Pressure:	982 psia
Area Ratio:	16:1
Weight (Flt. Config.):	18,616 lb
Dimensions:	220 in. long/144 in. wide



For more information contact: ELV Propulsion/Rockwell International/Rocketdyne Division/6633 Canoga Ave./
Canoga Park/CA/91303/(818)700-6027

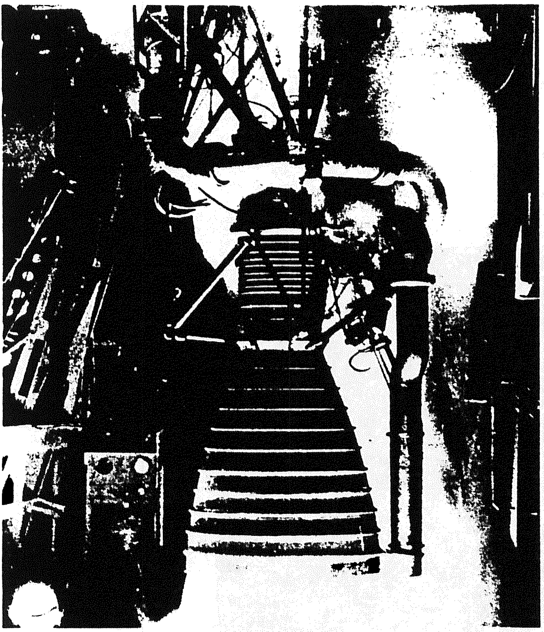
H-1



H-1

Rockwell International
Rocketdyne Division

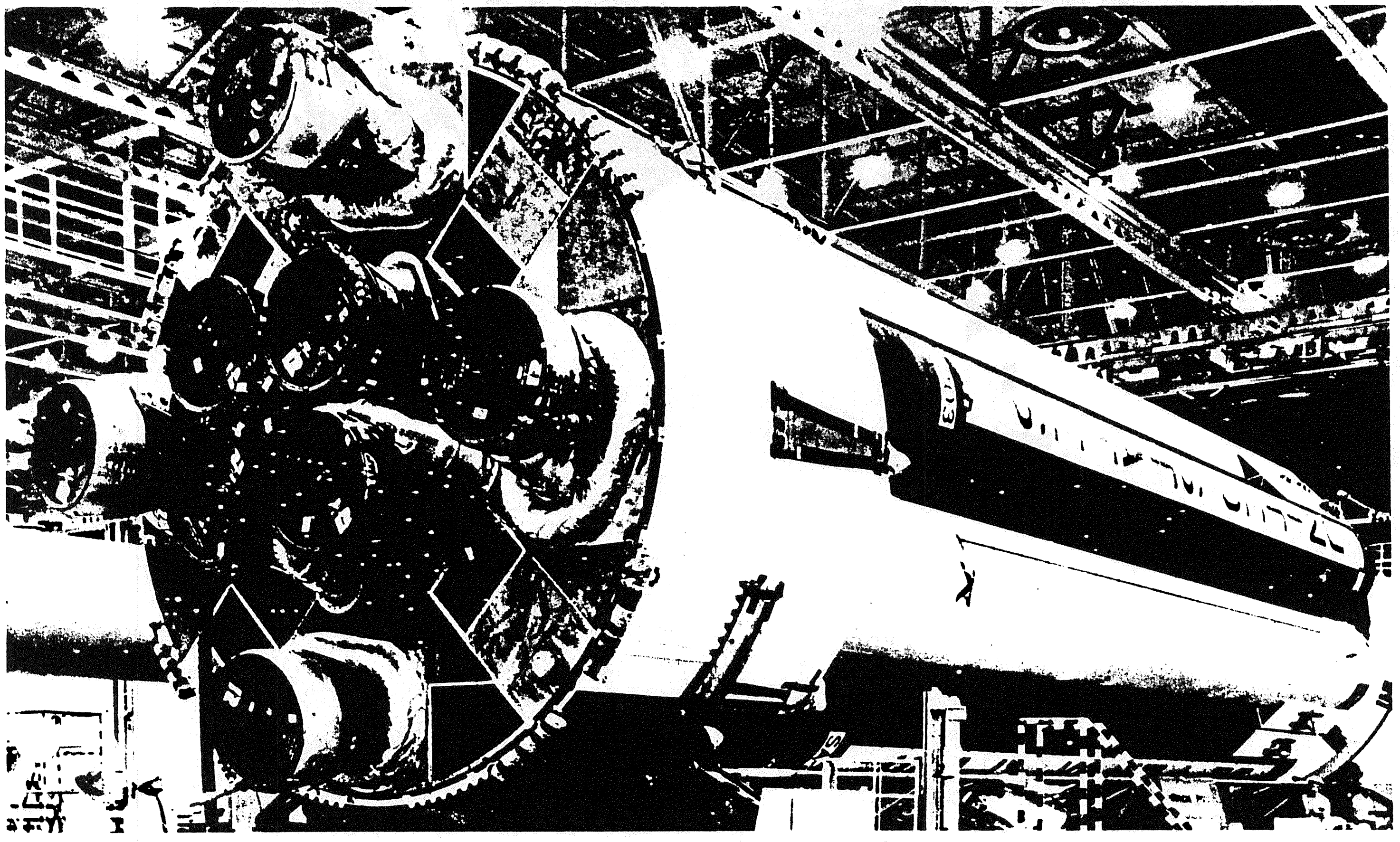
Initially designed for Apollo program application, the H-1 is a fixed-thrust, single-start gimbaled engine that employs a propellant system of RP-1 (kerosene) and liquid oxygen. Advances include a turbopump with a one-piece gearbox and fuel-additive lubrication, a solid propellant gas generator for start-up, propellant valve sequencing and hypergolic start-up in the thrust chamber.



H-1 Engine Test Firing

Specifications

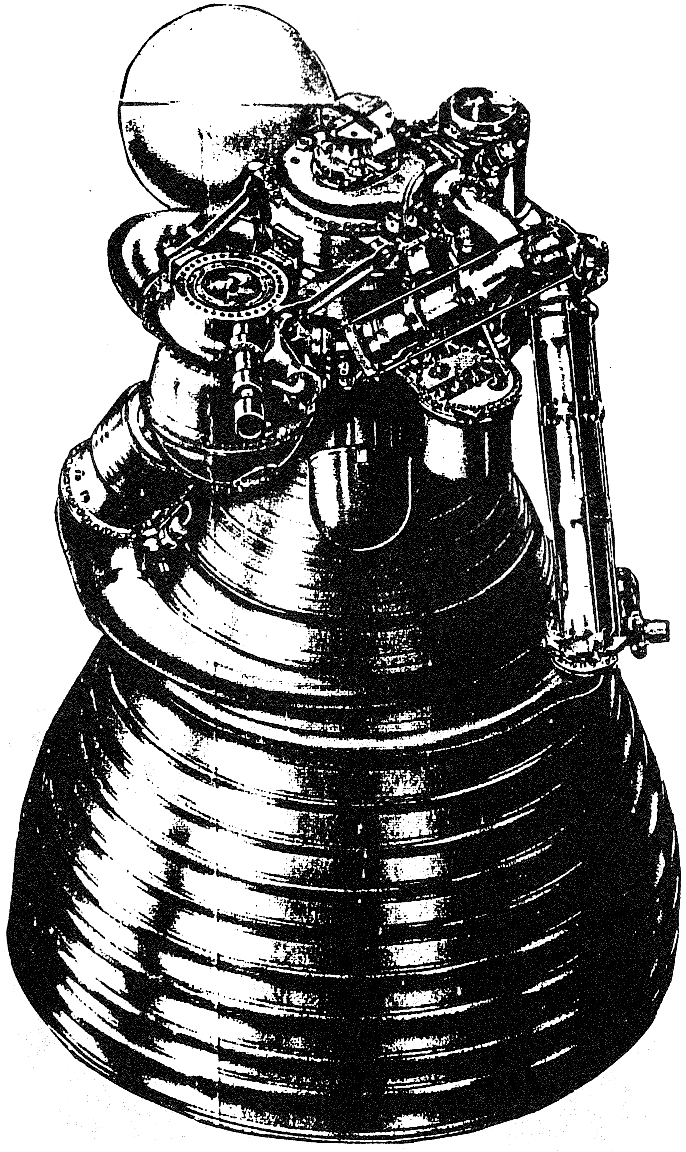
Type:	Liquid-Propellant, Pump-Fed
Thrust:	205,000 lb
Propellants:	RP-1 (Kerosene)/Liquid Oxygen
Specific Impulse:	263 sec
Mixture Ratio (O/F):	2.23:1
Chamber Pressure:	700 psia
Area Ratio:	8:1
Weight (Flt. Config.):	2,009 lb
Dimensions:	134 in. long/66 in. wide



Saturn IB First Stage With Eight H-1 Engines

For more information contact: ELV Propulsion/Rockwell International/Rocketdyne Division/6633 Canoga Ave./
Canoga Park/CA/91303/(818)700-6027

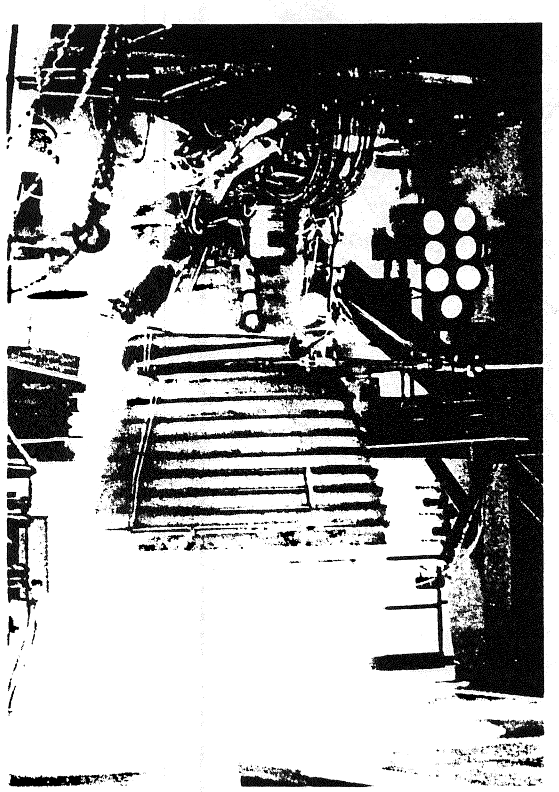
J-2



J-2

Rockwell International
Rocketdyne Division

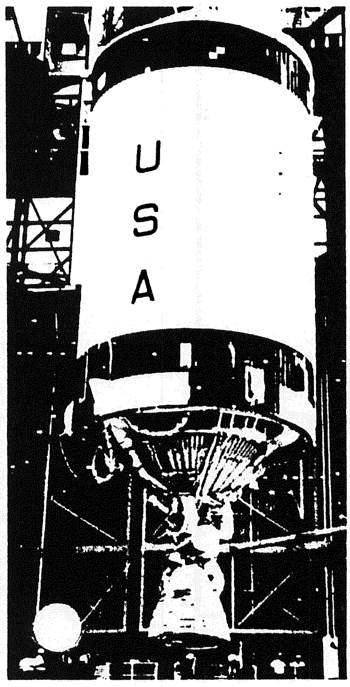
The J-2 provided highly reliable performance for both second and third stages of the Saturn vehicle in the Apollo program. The J-2 features independently driven pumps for both liquid oxygen and liquid hydrogen, a gas generator to supply hot gas to two turbines functioning in series, pneumatic and electrical control interlocks, altitude restart capability, and a propellant utilization system.



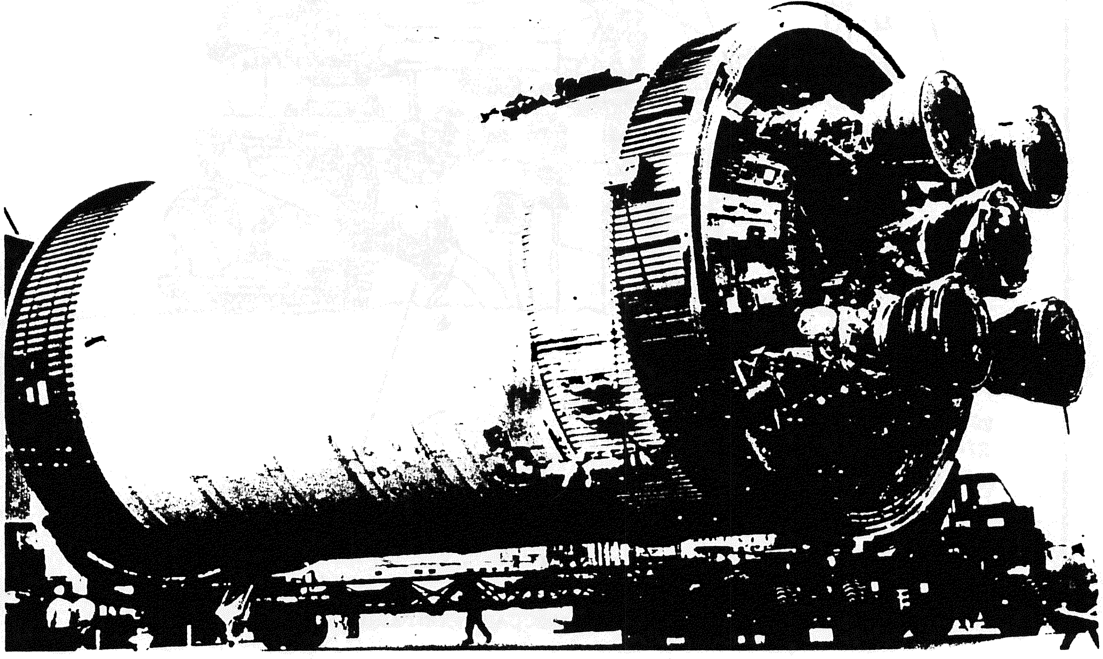
J-2 Engine Test Firing

Specifications

Type:	Liquid-Propellant/Pump-Fed
Thrust:	230,000 lb
Propellants:	Liquid Oxygen/Liquid Hydrogen
Specific Impulse:	425 sec
Mixture Ratio (O/F):	5.5:1
Chamber Pressure:	763 psia
Area Ratio:	27.5:1
Weight (Flt. Config.):	3,480 lb
Dimensions:	133 in. long/80.5 in. wide



Saturn V Third Stage With Single J-2 Engine



Saturn V Second Stage With Cluster of Five J-2 Engines

For more information contact: ELV Propulsion/Rockwell International/Rocketdyne Division/6633 Canoga Ave./
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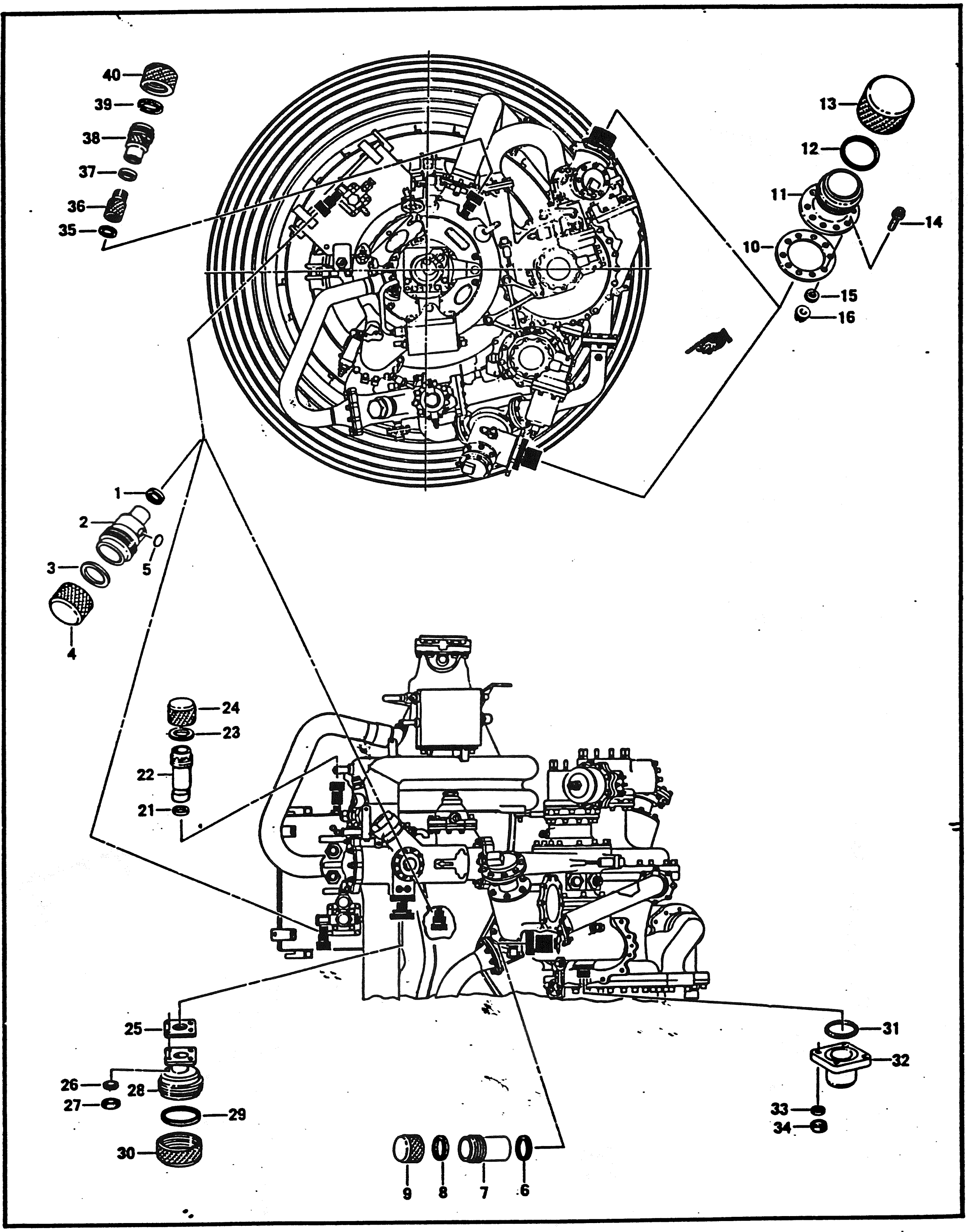
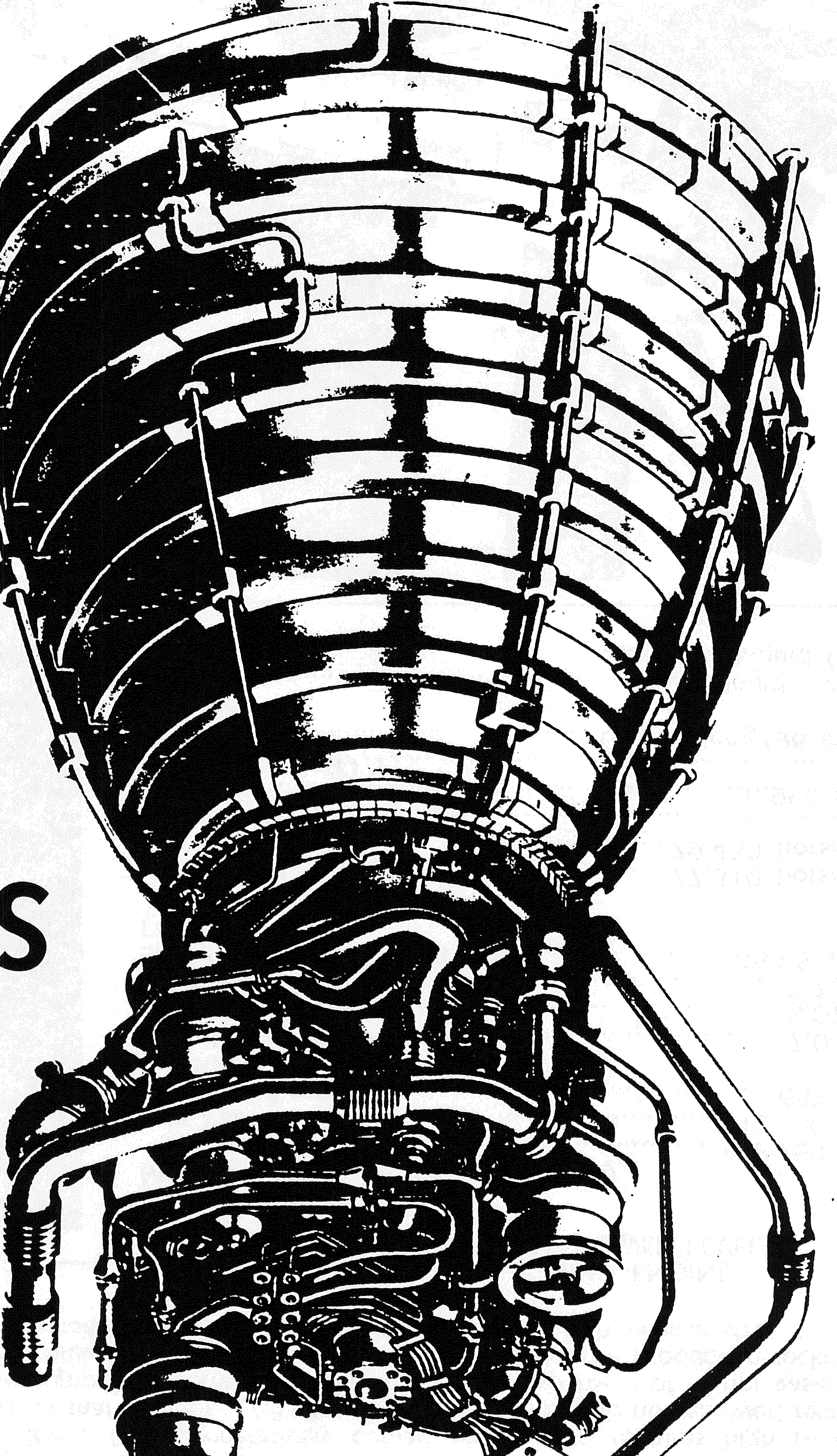
RL10 LIQUID ROCKET ENGINE



DESICCANT CONTAINER ASSEMBLIES



SSME

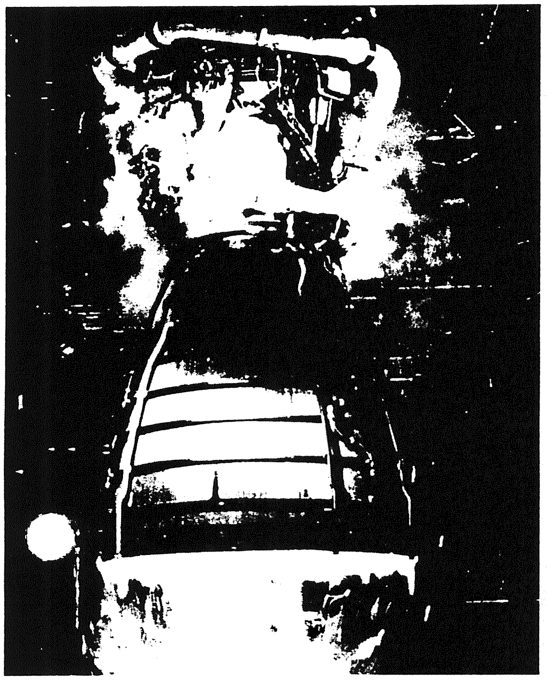
SSME

Rockwell International
Rocketdyne Division

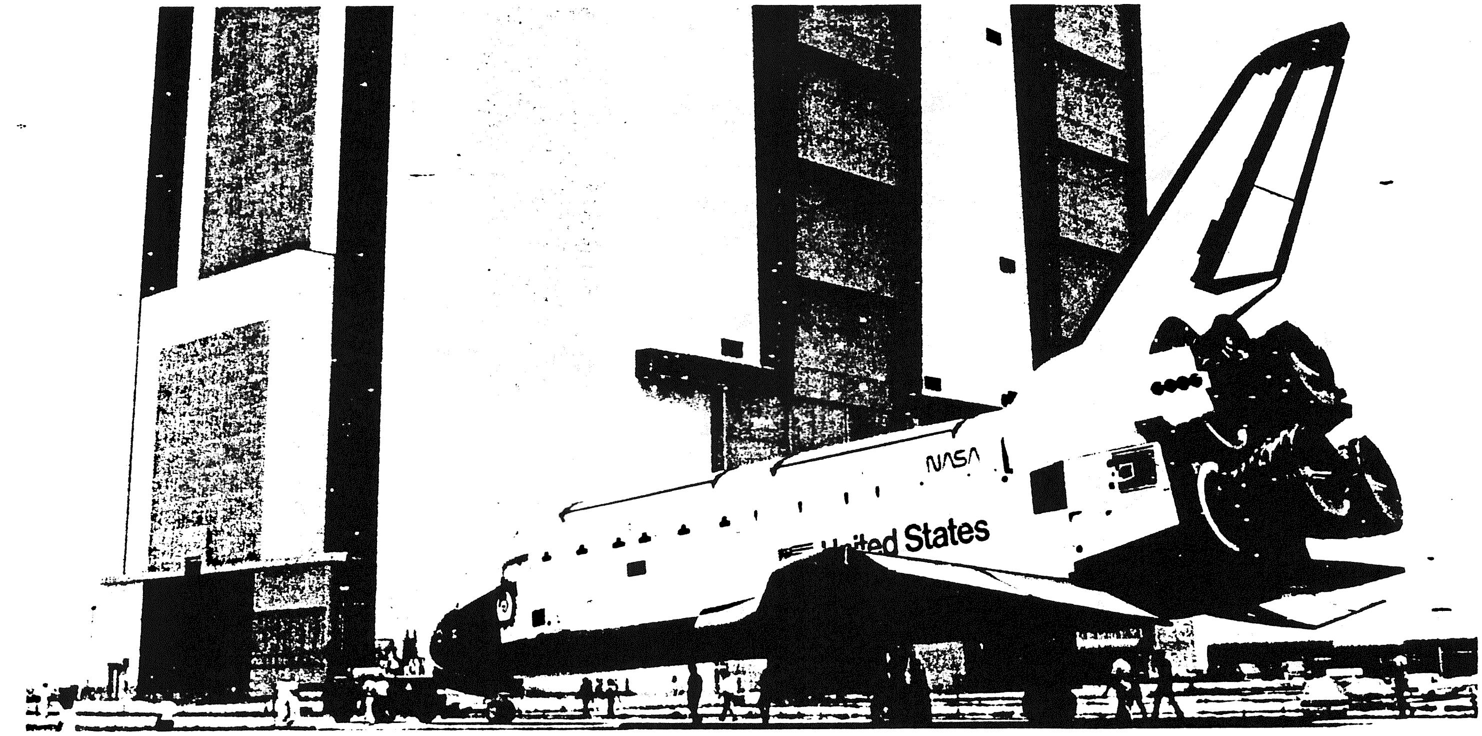
SPACE SHUTTLE MAIN ENGINE

The Space Shuttle Main Engine (SSME) was developed expressly for use on America's Space Shuttle. Using a mixture of liquid oxygen and liquid hydrogen, the SSME can attain a maximum thrust level (in vacuum) of 512,300 pounds at 109% power level. The regeneratively cooled engine also features high performance turbopumps for propellant and oxidizer that develop 77,310 horsepower and 29,430 horsepower, respectively. Ultra-high-pressure operation of the pumps and combustion chamber allows expansion of all hot gases through a high-area-ratio exhaust nozzle to achieve efficiencies never previously attained in a production rocket engine. These advantages allow a heavier payload to be carried without increasing launch vehicle size.

SPACE SHUTTLE MAIN ENGINE PERFORMANCE (FULL POWER LEVEL)



Maximum Thrust: (109% Power Level)	
At Sea Level	408,750 pounds
In Vacuum	512,300 pounds
Throttle Range	65%~109%
Pressures:	
Hydrogen Pump Discharge	7,040 psia
Oxygen Pump Discharge	8,070 psia
Chamber Pressure	3,260 psia
Specific Impulse (In Vacuum)	453.5 seconds
Power:	
High Pressure Pumps	
Hydrogen	77,310 horsepower
Oxygen	29,430 horsepower
Area Ratio	77.5:1
Weight:	6,990 pounds
Mixture Ratio (O/F)	6.0:1
Dimensions:	168 in. long/96 in. wide
Propellants:	
Fuel	Liquid Hydrogen
Oxidizer	Liquid Oxygen



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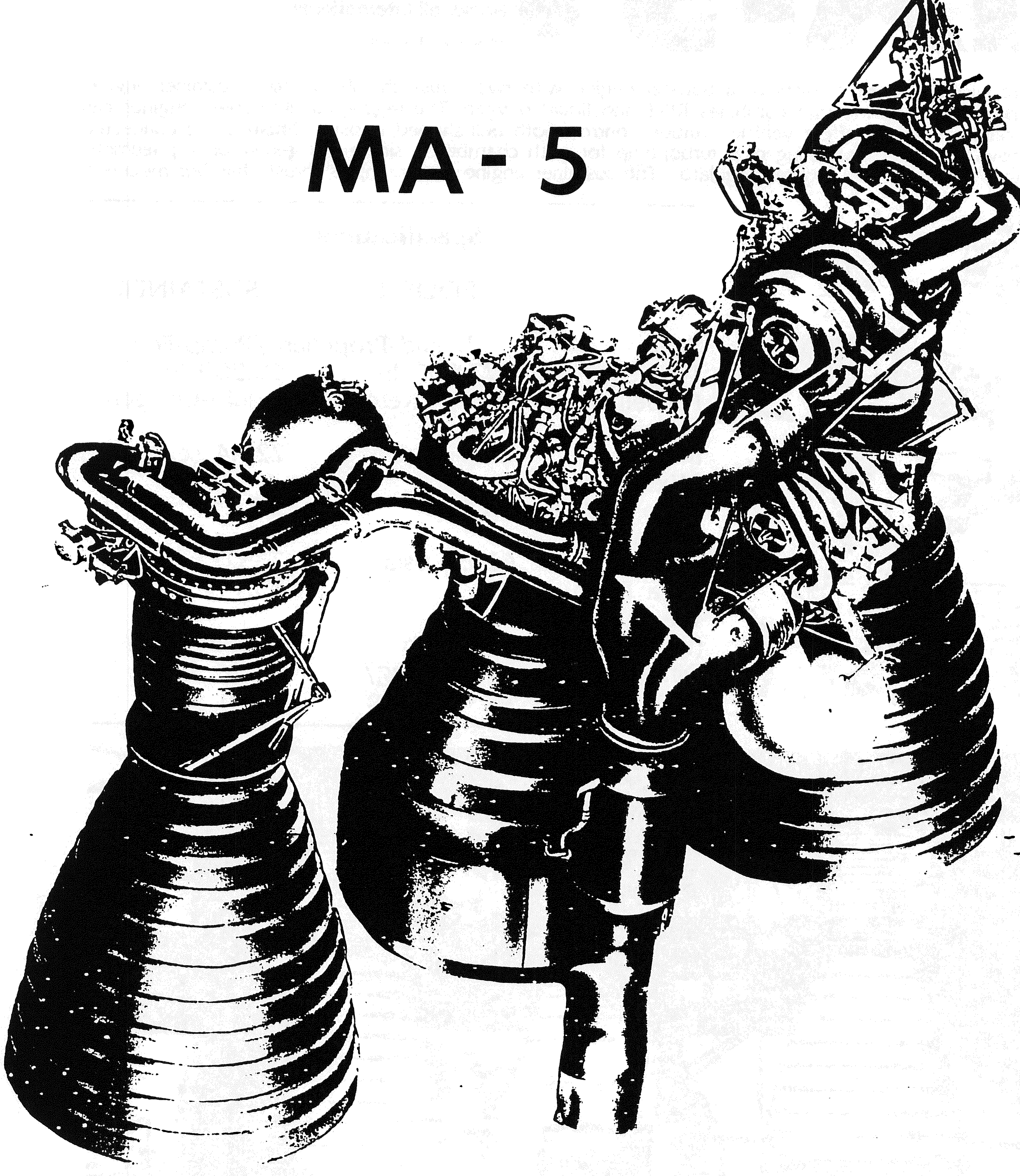
Pub. 571 M 80b New 1-87

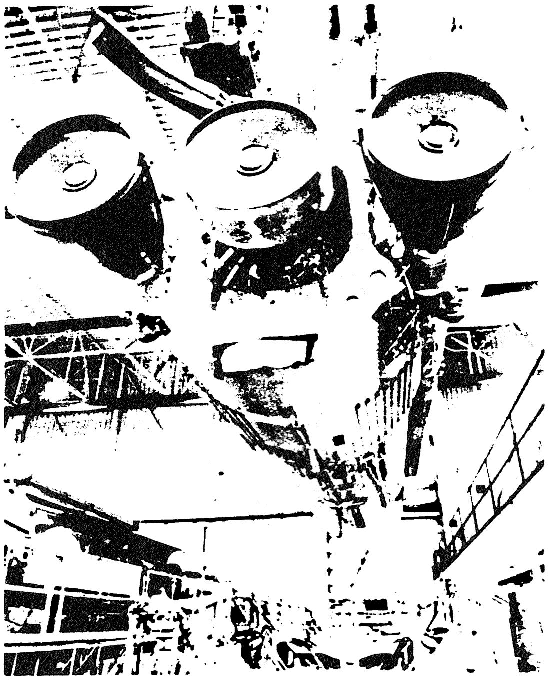
MA-5



MA-5

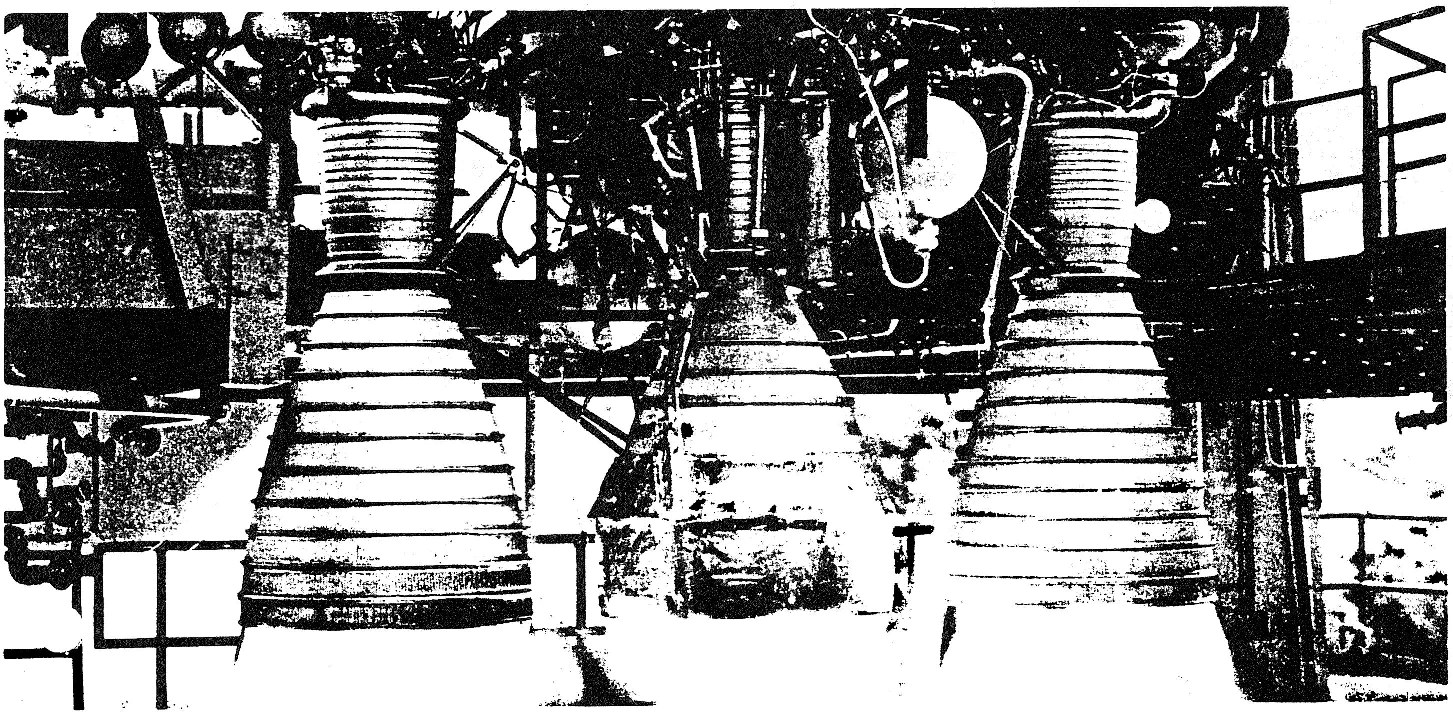
Rockwell International
Rocketdyne Division

The MA-5 propulsion system consists of a booster engine with two thrust chambers and a sustainer engine, all using a bipropellant scheme that combines RP-1 and liquid oxygen. The regeneratively cooled engines can be gimbaled during flight to effect vehicle attitude control. Both bell-shaped booster thrusters are connected to a central power package consisting of a turbopump for each chamber, a single gas generator, a pneumatic control package, and the liquid oxygen regulator. The sustainer engine turbopump is thrust chamber mounted.



Specifications

	BOOSTER	SUSTAINER
Type:	Liquid-Propellant/Pump-Fed	
Thrust:	377,500 lb	60,500 lb
Propellants:	RP-1 (Kerosene)/Liquid-Oxygen	
Specific Impulse:	259.1 sec	220.4 sec
Mixture Ratio (O/F):	2.25:1	2.27:1
Chamber Pressure:	639.0 psia	735.0 psia
Area Ratio:	8:1	25:1
Weight (Flt. Config.):	3,140 lb	1,035 lb
Dimensions:	97 in. long/ 47 in. wide	97 in. long/ 48 in. wide



For more information contact: ELV Propulsion/Rockwell International/Rocketdyne Division/6633 Canoga Ave./
Canoga Park/CA/91303/(818)700-6027

PLUMBING DESIGN CONSIDERATONS

- THERMAL SHRINKAGE (CYROGENIC)
- THERMAL INSULATION (FOR LIQUID HYDROGEN, -420 DEGREES F)
- INTERNAL PRESSURE AND DELTA P
- JOINT/FLANGE LEAKAGE
- FLUID VELOCITY
- MATERIAL
- ASSEMBLY TOLERANCES

FLUID CONNECTION AND SEAL CONFIGURATIONS

FLUID CONNECTIONS:

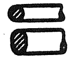
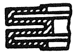
- 37 1/2 DEGREE CONE FITTINGS (SMALL SIZES < .375 DIA)
- FLANGED
- CANTILEVED FLANGE (HIGH PRESSURE)

FLUID CONNECTION AND SEAL CONFIGURATIONS

SEAL CATEGORIES:

- GASKETS
- METALLIC PRESSURE - ASSISTED
- PLASTIC SPRING-LOADED
- RADIAL METALLIC OR TOGGLE
- METALLIC BOSS

Chief Design Features of Commonly Used Static Seals

SEAL TYPE	CONFIGURATION OR TRADE NAME	MANUFACTURER	MATERIAL	COATING	GLAND DESIGN REQUIREMENTS					PRESSURE LIMIT WITHOUT SPECIAL GLAND OR SEAL DESIGN, PSI
					SURFACE FINISH, μ IN.	FLATNESS, IN./IN.	ALLOWABLE SEPARATION, IN.	UNIT SEALING LOAD, LBF/IN.	SEAL COMPRESSION CONTROL REQUIRED	
GASKET	O-RING 	VARIOUS	ELASTOMERS AND PLASTICS	NONE	32 to 64	0.004	0.004	5 to 200	YES	1500
	MOLDED SHAPES 	PARKER SEAL CO. STILLMAN RUBBER CO.	ELASTOMER MOLDED TO METAL BASE	NONE	32 to 64	0.003	0.003	30 to 50	NO	5000
	HOLLOW O-RING 	UNITED AIRCRAFT PRODUCTS, INC. ADVANCED PRODUCTS: D.S.D. MFG. CO.	VARIOUS METALS	TABLE II	8 to 32	0.0002	(a)	(a)	YES	5000
	SPIRAL WOUND 	JOHNS MANVILLE: FLEXITALLIC GASKET CO. GARLOCK, INC.	VARIOUS METALS AND FILLERS	NONE	64 to 125	0.0002	0.0005	(a)	YES	(b)
	FLAT 	VARIOUS	ELASTOMERS AND PLASTICS	NONE	32 to 64	0.010	0.010	(a)	(c)	(b)
	FLAT, PLAIN, AND GROOVED 	VARIOUS	VARIOUS METALS	NONE	8 to 32	0.001	NONE	(a)	(c)	(b)
	SOLID 	VARIOUS	VARIOUS METALS	NONE	32	0.001	NONE	(a)	(c)	15 000
METALLIC PRESSURE ASSISTED	U SEAL SPACER TYPE  GROOVE TYPE 	VARIOUS	INCONEL 718	TABLE II	32 (CIRCULAR LAY)	0.0005	0.008	200 to 300	SPACER TYPE: NO GROOVE TYPE: YES	10 000
	K SEAL 	HARRISON MFG. CO.	CRES	TABLE II	8 to 32	0.0002	-0.002	30 to 60	NO	3000
	HASKEL 	HASKEL ENGRG.	CRES	TABLE II	2 to 32	0.0005	0.002	30 to 100	NO	(b)
	HYDRODYNE 	DONALDSON CO.	VARIOUS METALS	TABLE II	8	0.0002	0.0002	70	YES	1000
METALLIC PRESSURE ASSISTED	V SEAL 	PARKER SEAL CO. HI-TEMP RINGS: TETRAFLUOR, INC.	VARIOUS METALS	TABLE II	16 to 32	0.0005	0.008	100 to 250	YES	(b)
	C SEAL 	PARKER SEAL CO.	INCONEL 718	TABLE II	32 to 64	0.0005	0.008	300 to 400	YES	(b)
	PRESSURE LOCK 	KOPPERS CO. NAVAN PRODUCTS	CRES	NONE	8 to 32	0.0015	0.002	220 to 560	YES	4000
SPRING LOADED	OMNISEAL 	AEROQUIP CORP.	TEFLON JACKET OVER FLAT STAINLESS HELICAL SPRING	NOT APPLICABLE	63	0.005	0.020	80 to 100	YES	1200
	RACO 	RACO MFG. CO.	TEFLON JACKET OVER STAINLESS FINGER SPRING	NOT APPLICABLE	32	0.005	0.015	80 to 100	YES	1200
	CREAVY 	HASKEL ENGRG.	TEFLON TUBE OVER STAINLESS STEEL COIL SPRING	NOT APPLICABLE	32	0.005	0.010	80 to 100	YES	1600
RADIAL OR TOGGLE	BOBBIN 	VARIOUS	CRES	NICKEL	32	0.020	0.005	800	YES	(b)
	NUCOSEAL 	NATIONAL UTILITIES	CRES OR ALUMINUM ALLOY	NONE	64	0.005	0.005	30 to 80	YES	(b)
	CONOSEAL 	AEROQUIP CORP.	CRES OR ALUMINUM ALLOY	NONE	32	0.020	0.020	800 to 600	YES	(b)
	GAMAH 	GAMAH CORP.	CRES OR ALUMINUM ALLOY	NONE	32	0.010	0.005	800	YES	(b)
METALLIC BOSS	K-SEAL 	HARRISON MFG. CO.	CRES	TABLE II	16 to 32	0.001	0.0005	(a)	NO	(b)
	NATORQ 	NAVAN PRODUCTS	CRES	SILVER AND NICKEL	32	0.001	0.0005	1600	NO	(b)

(a) TOTAL LOAD APPLIED SHOULD NOT CRUSH GASKET
(b) PRESSURE LIMIT DEPENDS ON SEAL DIAMETER AND CROSS SECTION (REFER TO MANUFACTURERS' RECOMMENDATIONS)
(c) DEPENDENT ON BASE MATERIAL AND SEAL CROSS-SECTIONAL CONFIGURATION (SEE TABLES 6.3.3.2.b, c, & d and 6.3.3.4c IN REF. 9)

Basic Characteristics of Materials Used as Platings or Coatings on Static Seals (adptd. from ref. 47)

Status	Plating or Coating	Useful Temperature Range, °F	Hardness		Remarks
			Brinell or as noted	Rockwell	
Operational	Copper	–452 to +1900	Vickers 41 to 220	< 97 R_B	Suitable for vacuum applications; resistant to fluorine and certain other corrosive chemicals.
	Gold	–452 to +1850	65 to 125	26 to 77 R_B	Similar to silver but somewhat better resistance to certain corrosive fluids; very wide temperature range but higher in cost than silver.
	Kel-F	–452 to +300	Rockwell R 80 to 85	80 to 85 R_R	Similar to Teflon, but more resilient and plastic at low temperatures; generally higher in cost than Teflon.
	*Rhodium	–452 to +3500	400 to 800	43 to 72 R_C	Similar to silver, but useful at much higher temperatures; higher in cost than either silver or gold.
	Silver	–452 to +1650	50 to 150	88 R_B	Excellent general purpose plating for high-temperature use, but generally less suitable for cryogenic temperatures than gold or Teflon; excellent chemical and radiation resistance.
	Teflon (FEP)	–452 to +400	Shore D 59	55 R_D	Similar to Teflon (TFE), but somewhat softer and denser; useful high-temperature limit lower than that of Teflon (TFE).
	Teflon (TFE)	–452 to +500	Shore D 52	75 to 95 R_J	Excellent coating for applications up to +500°F; excellent chemical resistance; particularly suitable for cryogenic applications.
Research and Development	Aluminum	–452 to +900	Vickers 30 to 90	< 56 R_B	Compatible with most oxidizers and fuels, and particularly suitable for use with liquid and gaseous fluorine; however, it is very difficult (and costly) to obtain quality of plating required.
	Indium	–452 to +300	Soft	NC	Very soft plating, limited to moderate temperatures; suitable for cryogenic applications.
	Lead	–65 to +450	5	NC	Very soft plating with limited temperature range; excellent radiation resistance.
	Nickel (soft)	–452 to +2500	140 to 200	76 to 93 R_B	Withstands high temperatures but slightly softer and less ductile than other platings.
	Platinum	–452 to +3100	Knoop 280 to 290	NC	Withstands higher temperatures than other platings; normally limited to use with ultra-high-temperature base metals such as TZM.
	Tin (pure)	–32 to +350	5	NC	Very ductile, but very limited temperature range; used only with a few corrosive chemicals.

*Used primarily as flash coating to prevent sticking

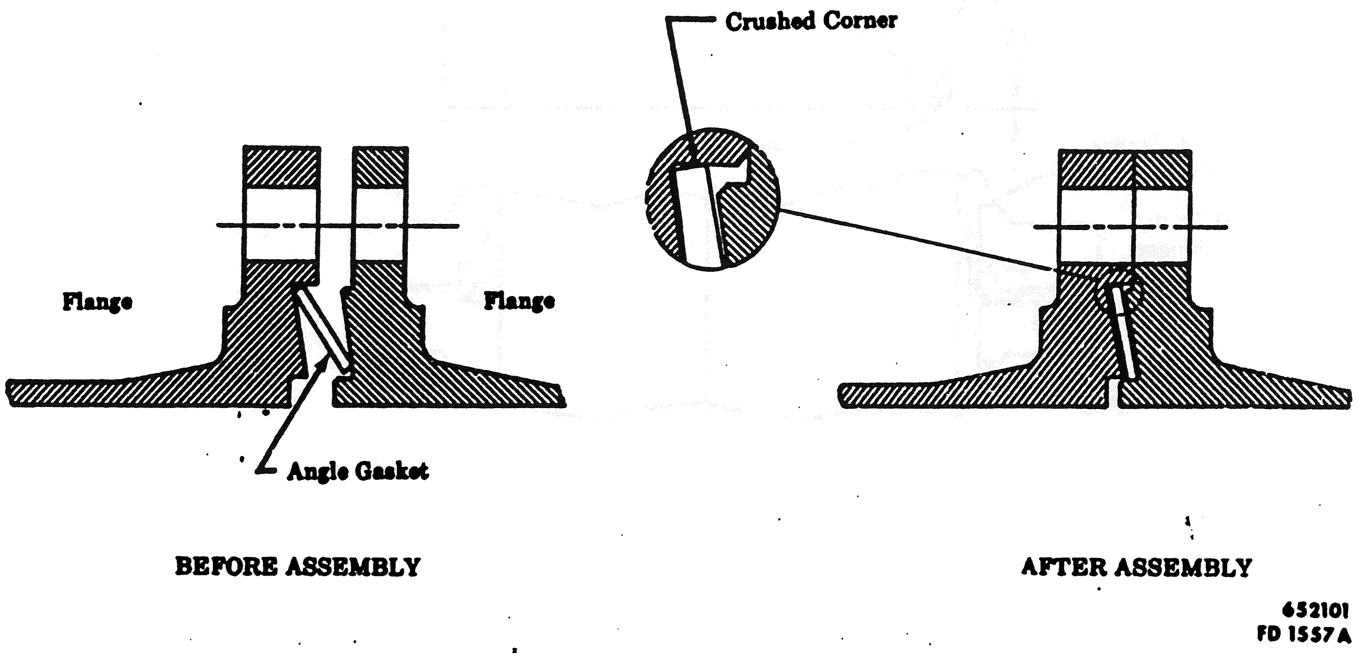
NC = no correlation to Rockwell scales.

TZM = molybdenum alloy containing 0.5Ti-0.08Zr-0.3C.

FLUID CONNECTION AND SEAL CONFIGURATION

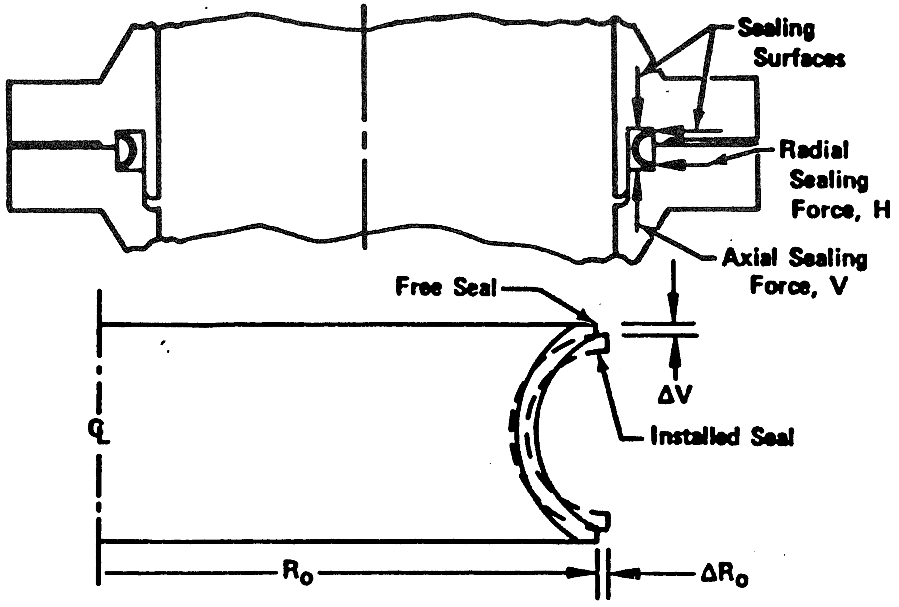
PW SEAL EXPERIENCE:

- ANGLE GASKET
- ALUMINUM FLAT GASKET, TEFLON COATED
- METAL "O" - RING
- CONICAL SEAL - VOISHAN (RL10 GREEN RUN ONLY)
- TOROIDAL SEAL (PW SSME PROPOSAL)
- OMEGA & "E" SEALS (NOW USED ON ATD PUMPS)



RL-10 Propellant Pipe Sealing Method

TOROIDAL SEGMENT SEAL

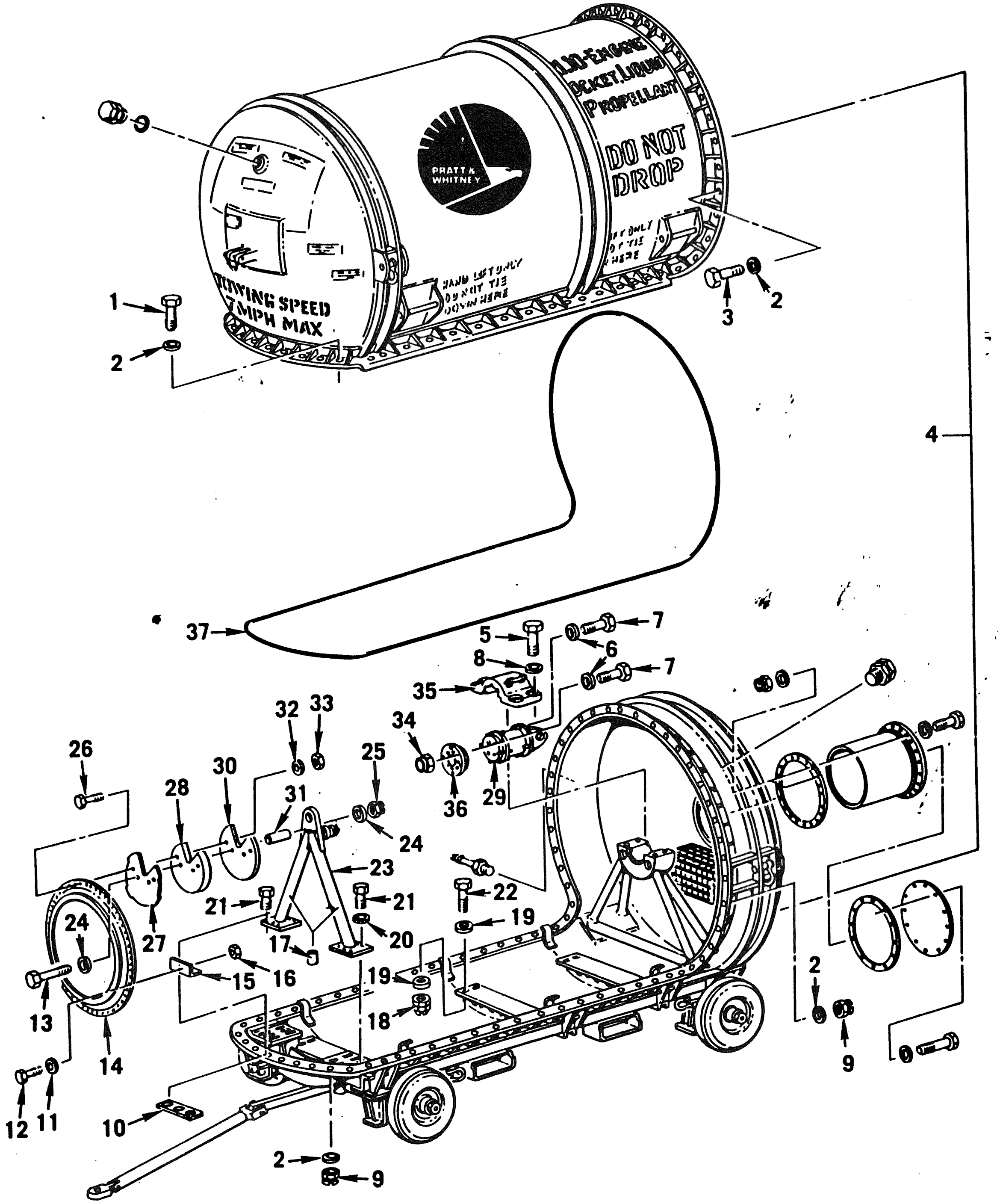


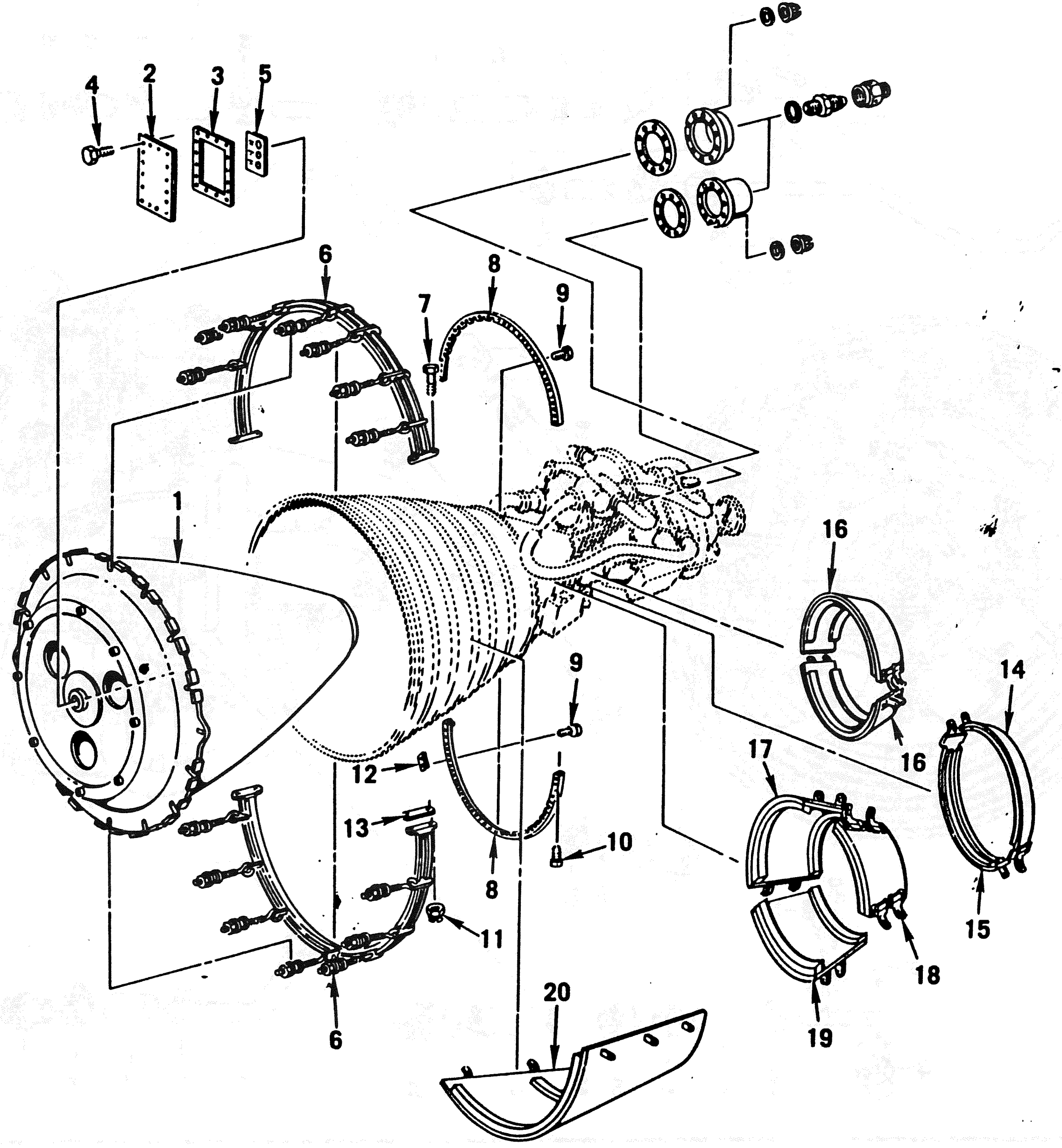
FD 37247

Sealing Surfaces on Typical Installation

ROCKET ENGINE
HANDLING AND SHIPPING
EQUIPMENT

Engine Shipping Container

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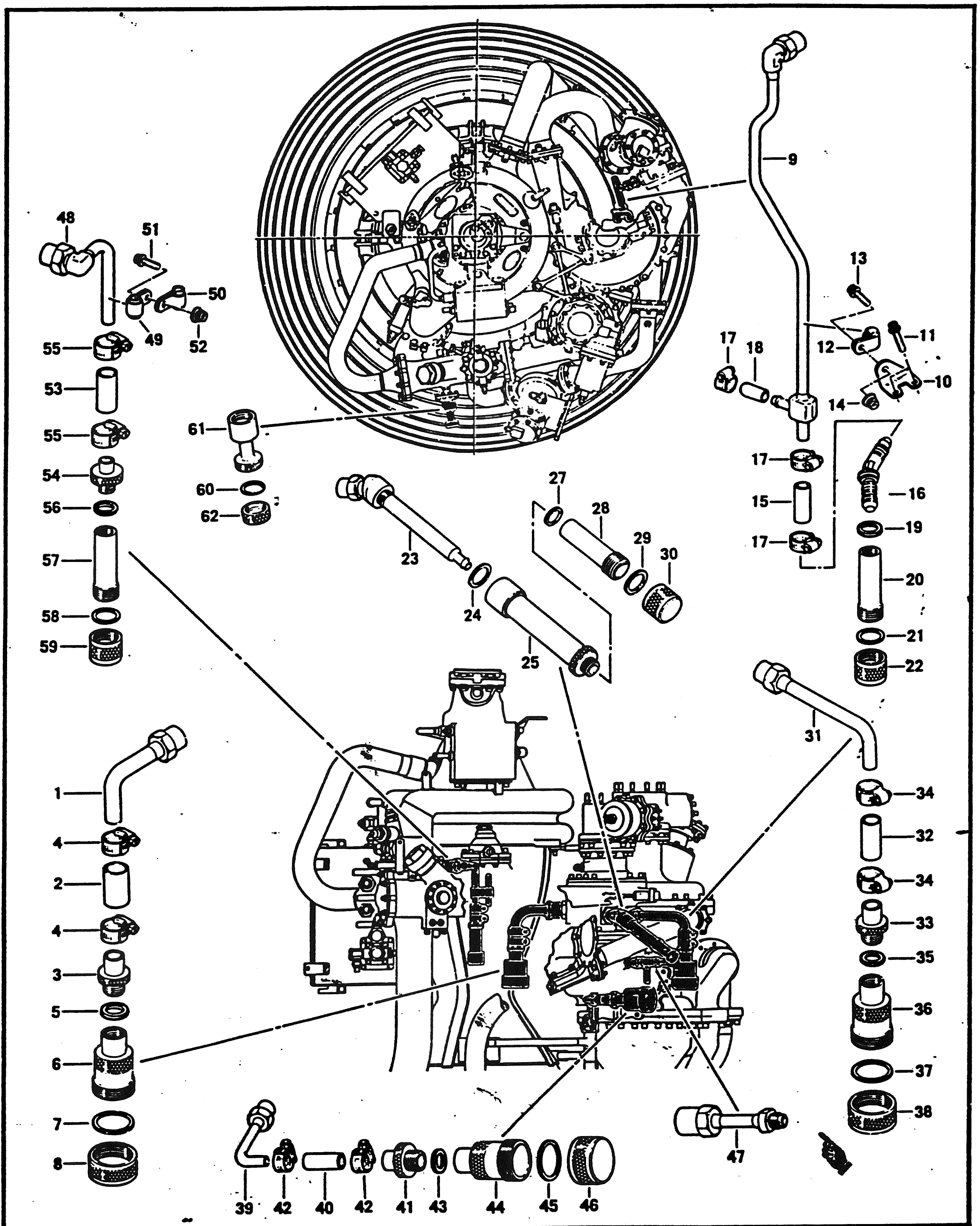
RL10 LIQUID ROCKET ENGINE



DESICCANT CONTAINER ASSEMBLIES

ROCKET PLUMBING DESIGN COURSE

Reference:

1) Liquid Rocket Disconnects, Couplings, Fittings,
   Fixed Joints, and Seals, --- NASA SP-8119

2) Liquid Rocket Lines, Bellows, Flexable Hoses,
   and Filters --- NASA SP-8123

3) Design of Liquid Propellant Rocket Engines,
   by: Huzel and Huang --- NASA SP-125

4) SSME High Pressure Plumbing Design
   Procedure --- FTDM -- 340 & 341

INTRODUCTION TO

ROCKET ENGINE

CONTROLS AND VALVES

Al Palgon

ROCKET ENGINE MECHANICAL CONTROLS AND VALVES

COURSE OBJECTIVE:

- TO HELP YOU LEARN LOGICAL STEPS FOR GENERATION OF ROCKET ENGINE CONTROL COMPONENT DESIGNS SO YOU CAN AVOID NON-PRODUCTIVE EFFORTS AND CREATE QUALITY DESIGNS EVERY TIME.

BASIC OUTLINE:

- FLUID CONTROL REQUIREMENTS
- VALVE CONCEPT SELECTION
- ACTUATORS
- SEALS
- POSITION FEEDBACK CONCEPTS
- ROCKET/TURBINE ENGINE CONTROLS COMPARISONS

FLUID CONTROL REQUIREMENTS

TRANSIENT STUDIES PROVIDE DYNAMIC REQUIREMENTS

- PRECONDITIONING (PRESTART) ANALYTICAL DATA
- START SEQUENCE/FLOW CONTROL DATA
- ACCELERATION CONTROL SCHEDULES AND SEQUENCES
- STEADY STATE DYNAMIC LIMITS
- DECELERATION SCHEDULES AND SEQUENCES
- SHUTDOWN/VENT/SAFING REQUIREMENTS

FLUID CONTROL REQUIREMENTS

STEADY-STATE CYCLE STUDIES AND SYSTEM SPECIFICATION PROVIDE BASIC DESIGN INFORMATION

- STEADY STATE ENGINE CHARACTERISTICS (ALL POINTS)
- VALVE AND CONTROL LOCATIONS AND FLUIDS DATA
- VALVE ACTUATION MEDIUM CHARACTERISTICS
- ALLOWABLE LEAKAGES
- LIFE REQUIREMENTS
- SAFETY REQUIREMENTS

VALVE CONCEPT SELECTION

CONCEPTS MUST BE BASED ON BOTH *REQUIREMENTS* AND *SPECIFIC CHARACTERISTICS*

- VALVE TYPE SELECTION VIA TRADE STUDY
- EACH TYPE HAS ADVANTAGES AND DISADVANTAGES
- TWO-POSITION (ON/OFF) VALVE CHARACTERISTICS NOT NECESSARILY THE SAME AS FOR CONTROL VALVES
- HISTORICAL DATA/STUDIES IMPORTANT
- *ALL* REQUIREMENTS NEED CONSIDERATION (RESPONSE, WEAR, LIFE, PRECISION, ACTUATION POWER, FLOW PATH ΔP, OVERBOARD LEAKAGE, ETC.)
- UNIQUE APPLICATIONS DEMAND UNIQUE SOLUTIONS.

VALVE CONCEPT SELECTION

TRADE STUDIES ARE TOOLS FOR OPTIMUM CONCEPT SELECTION - FIRST TIME

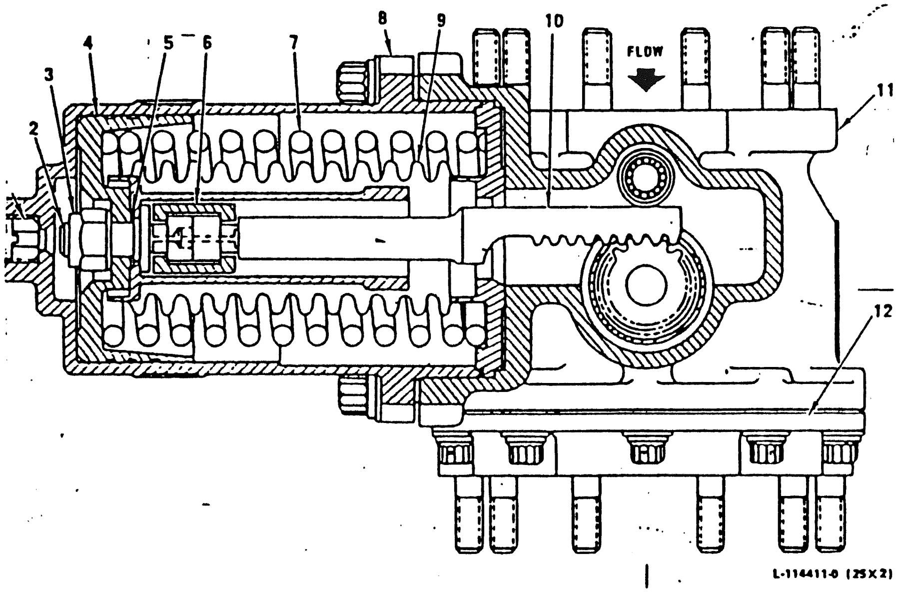
- VALVE TYPE
- ACTUATOR TYPE
- MAJOR SEAL ELEMENTS
- VALVE DRIVE MECHANISM
- MATERIALS
- LUBRICANTS
- COATINGS

VALVE CONCEPT SELECTION

BALL VALVES HAVE UNIQUE CHARACTERISTICS

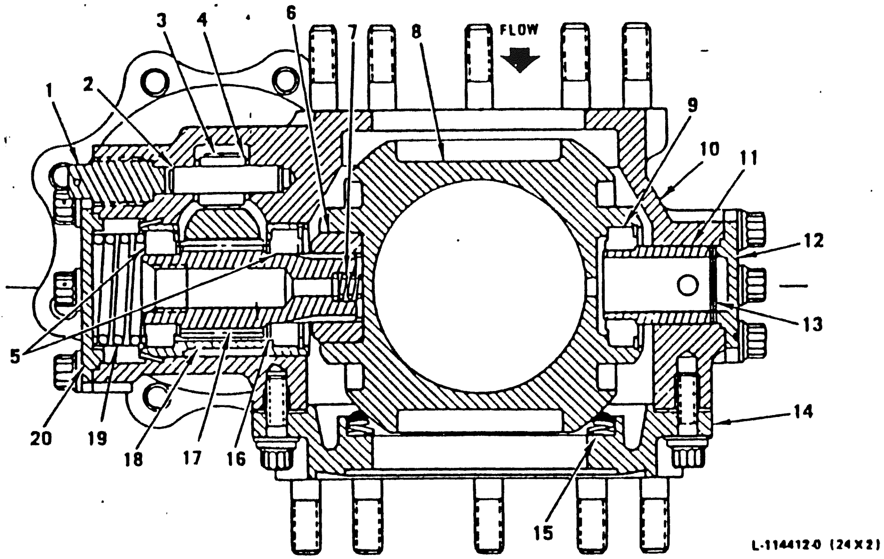
- VERY LOW PARASITIC LOSS
- MODERATE ACTUATION FORCE
- GOOD ELEMENT SEALING
- DUAL SEALS POSSIBLE
- SHORT STROKE
- AREA VS STROKE SIMILAR TO EXPONENTIAL CURVE
- RELATIVELY HEAVY AND/OR EXPENSIVE

BALL SHUTOFF VALVE



1. Orifice plug
2. Piston stud
3. Self-locking nut
4. Piston
5. Gasket
6. Split collar
7. Spring
8. Cylinder
9. Bellows assembly
10. Rack
11. Body assembly
12. Flange assembly

Figure 2-7. Fuel and Oxidizer Inlet Shutoff Valves - Closed Position (Actuator View)



1. Plug
2. Roller pin
3. Roller bearing
4. Spacer
5. Ball bearing
6. Drive insert
7. Spring
8. Ball
9. Ball bearing
10. Body assembly
11. Pivot
12. Pivot cap
13. Shim
14. Flange assembly
15. Seal assembly
16. Spacer
17. Gear
18. Sleeve.
19. Spring
20. Cap

Figure 2-6. Fuel and Oxidizer Inlet Shutoff Valves - Closed Position

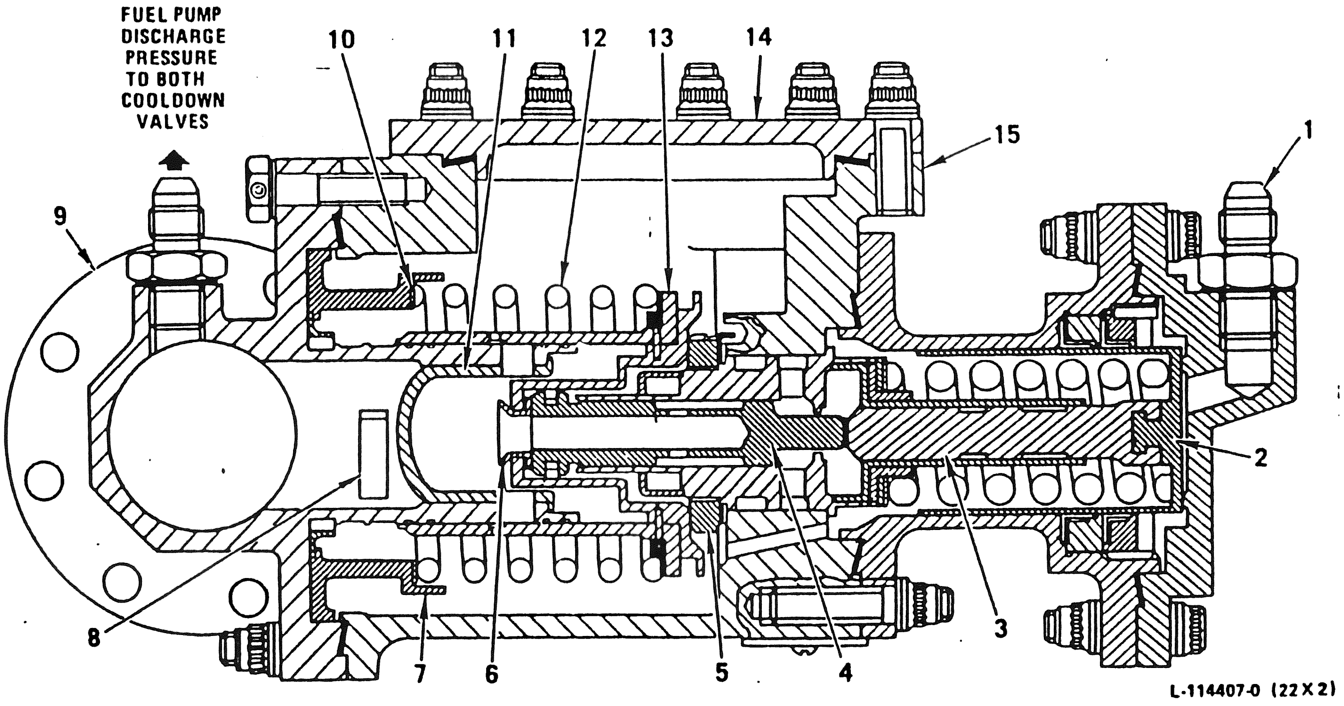
FEATURES: SPRING (BELLEVILLE) -LOADED SHUTOFF SEAL
QUICK-ACTING PISTON/BELLOWS ACTUATOR
RACK & PINION BALL DRIVE
BEARING-SUPPORTED BALL

VALVE CONCEPT SELECTION

SLEEVE VALVES OFFER FLEXIBILITY

- VARIABLE AREA VS STROKE
- VARIABLE ACTUATION FORCE
- HIGH PARASITIC LOSS
- POOR ELEMENT SEALING
- RELATIVELY HEAVY AND/OR EXPENSIVE
- DISTORTION-SUSCEPTIBLE

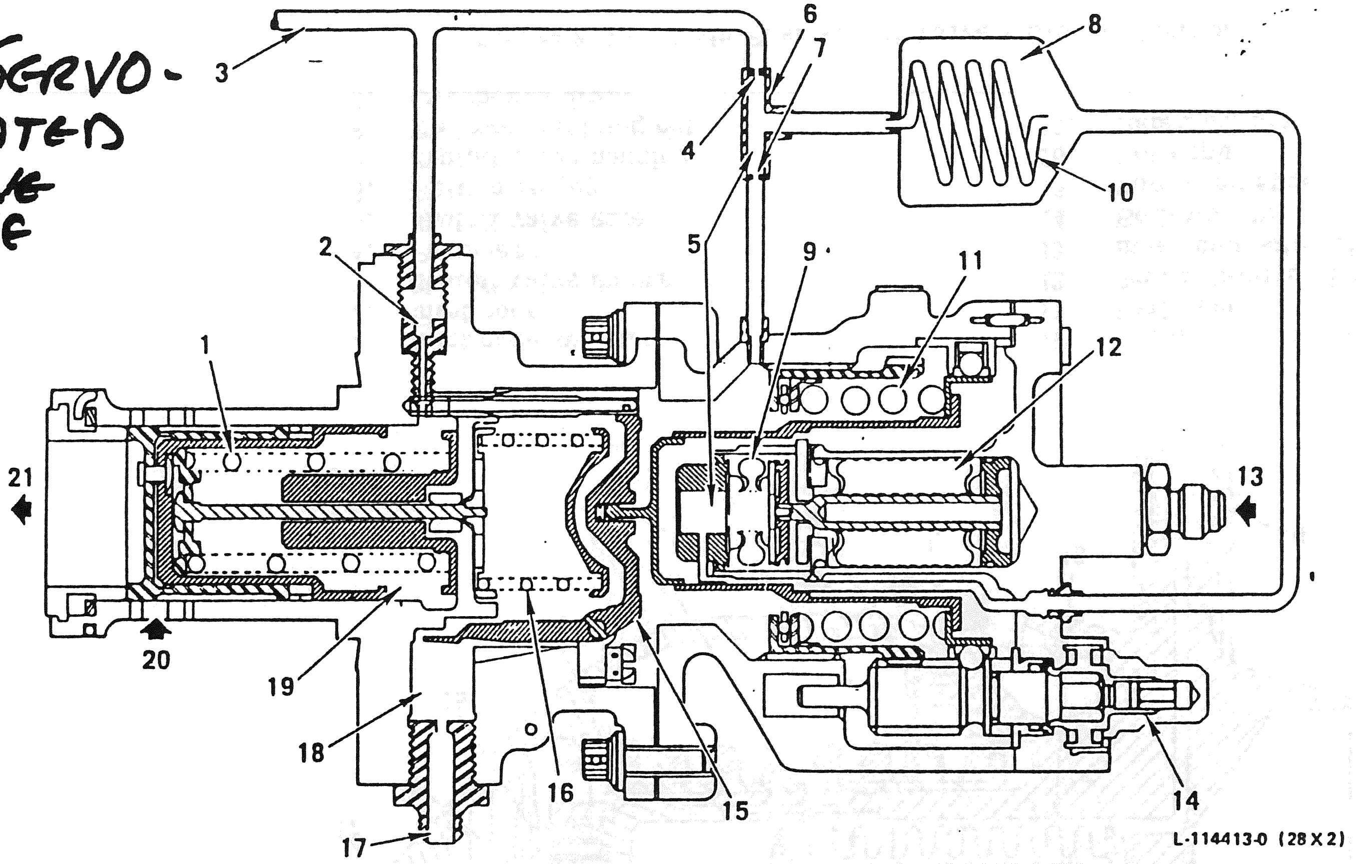
LH_2 SLEEVE VALVE



1. Helium inlet connector
2. Outer actuating piston
3. Inner actuating piston
4. Fuel piston
5. Nut
6. Shim retainer
7. Spring seat
8. Cooldown valve port
9. Lower housing
10. Shim
11. Plug
12. Helical spring
13. Valve assembly
14. Cover
15. Upper housing

Figure 2-12. Fuel Pump Discharge Cooldown Valve - Open Position

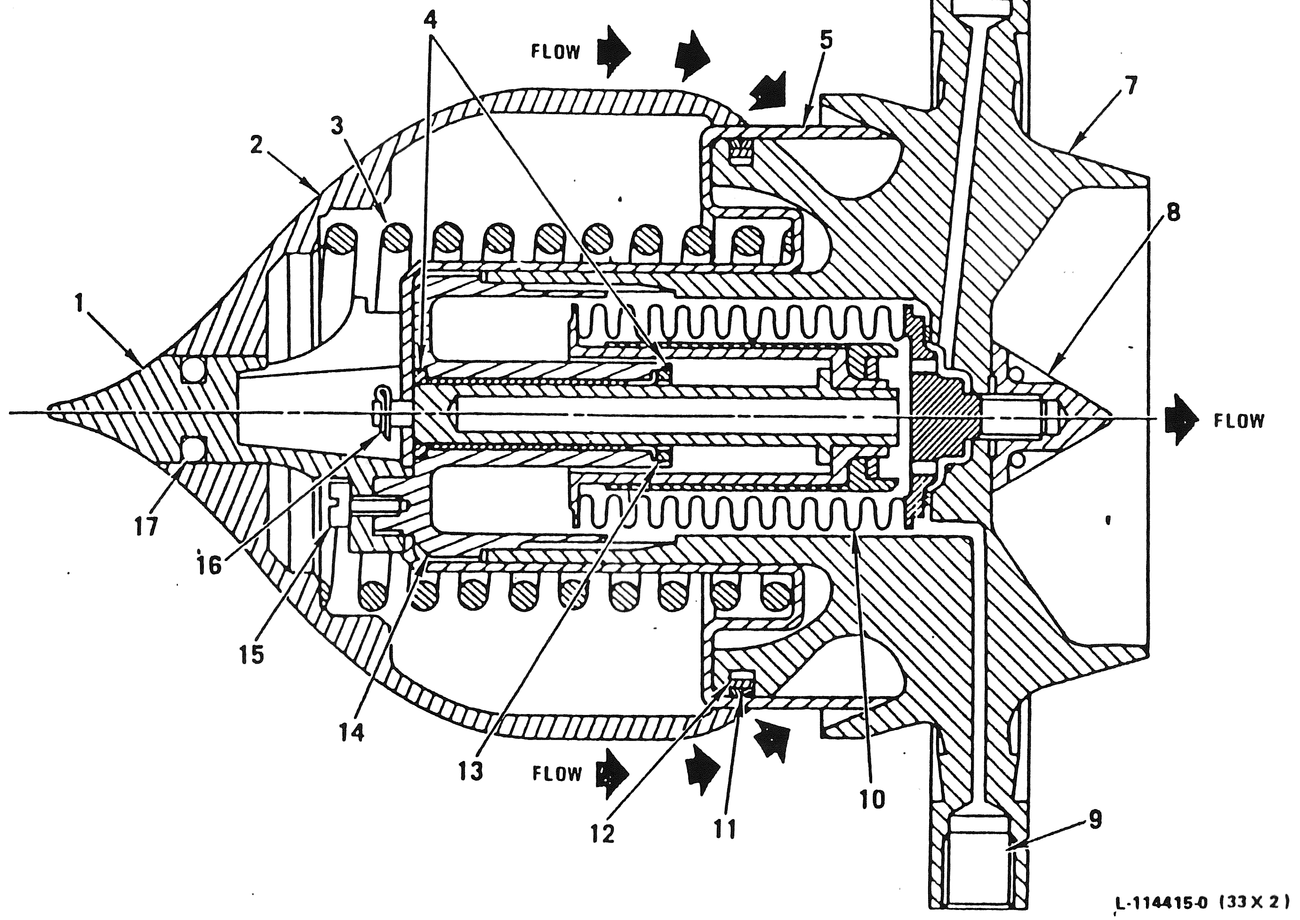
GH2 SERVO-
ACTUATED
SLEEVE
VALVE



1. Bypass valve spring
2. Servo supply orifice
3. Servo supply pressure
4. Inlet orifice
5. Reference pressure
6. Orifice-adapter
7. Discharge orifice
8. Pneumatic reset
9. Reference bellows
10. Hypodermic tube
11. Reference spring
12. Motor bellows
13. Chamber pressure
14. Ground thrust adjustment
15. Servo lever
16. Feedback spring
17. Overboard vent
18. Body pressure
19. Servo chamber pressure
20. Turbine inlet
21. Turbine discharge

Figure 2-9. Thrust Control Valve (Functional View)

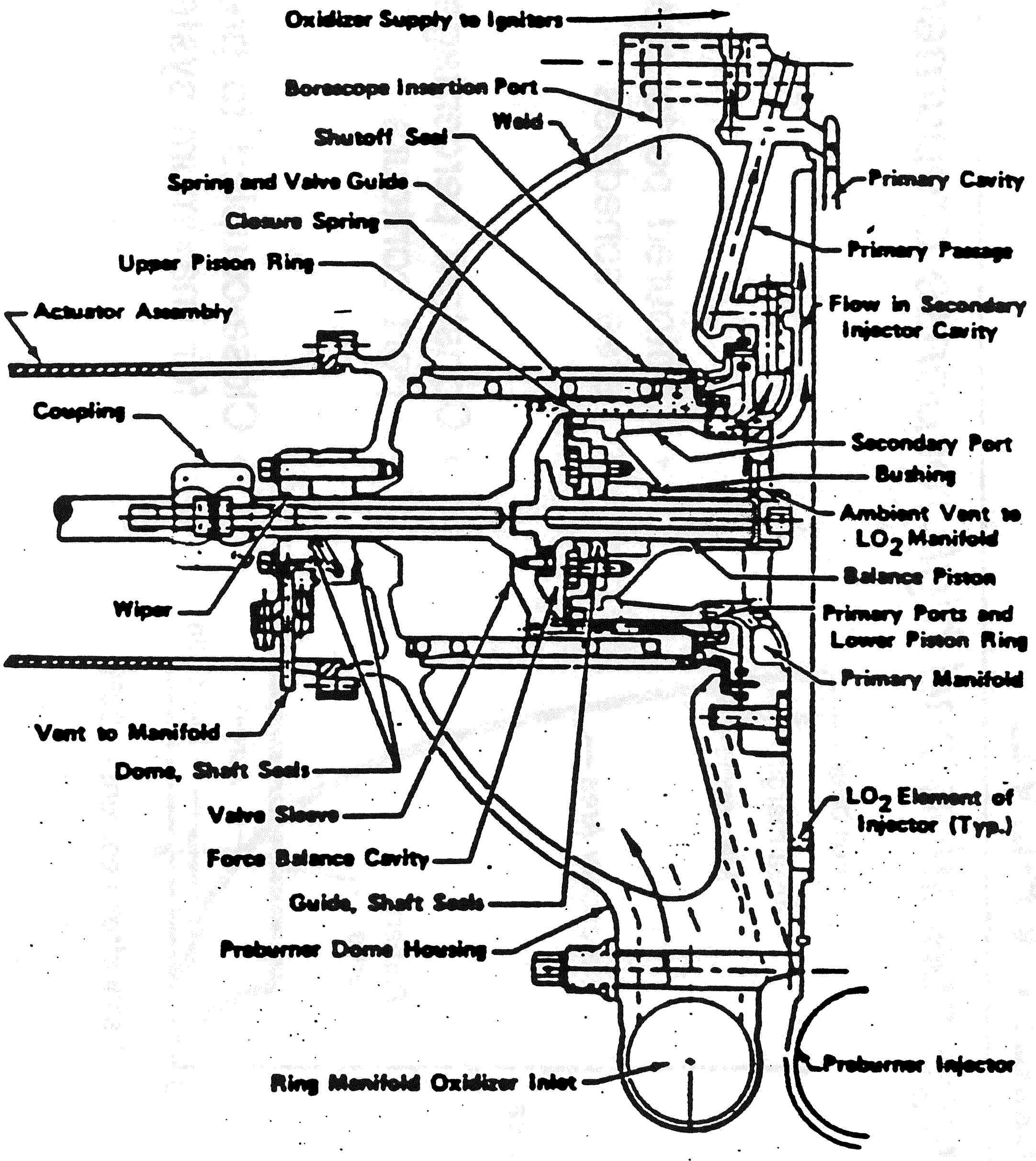
GH2 SLEEVE VALVE



1. Inlet cone support
2. Inlet cone
3. Shutoff valve spring
4. Retainers
5. Shutoff valve gate
6. Helium supply
7. Shutoff valve housing
8. Bellows retaining nut
9. Overboard drain
10. Bellows assembly
11. Seal rings
12. Seal retaining ring
13. Inner and outer shaft seals
14. Bellows cap
15. Inlet cone support screw
16. Cotterpin
17. Retaining pin

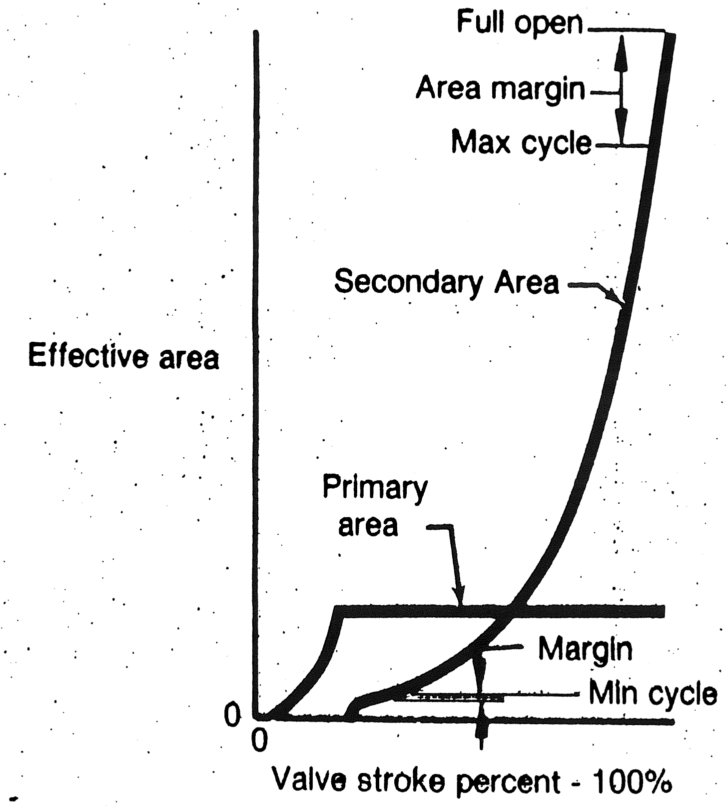
Figure 2-10. Main Fuel Shutoff Valve - Closed Position

LO2 SLEEVE VALVE



IMPROVED SSME PREBURNER OXIDIZER VALVE

Two-stage operation and high response preburner control



- Contoured ports provide desired area schedules
- Constant percentage error port contours
- Close-coupled to preburner injector for maximum system response

AV283025 841408 8892B

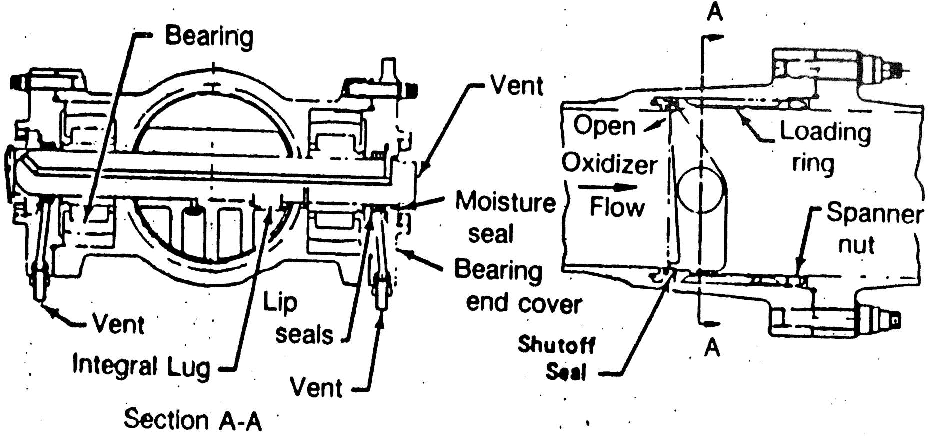
VALVE CONCEPT SELECTION

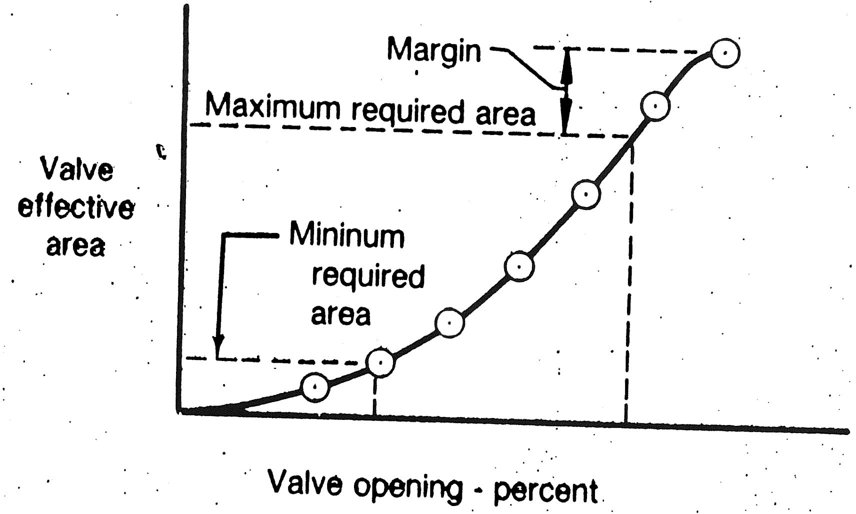
BUTTERFLY VALVES OFFER BEST COMPROMISE SOMETIMES

- LOW PARASITIC LOSS
- LOW ACTUATION FORCE POSSIBLE
- FAIR ELEMENT SEALING
- VARIABLE STROKE
- EQUAL PERCENTAGE AREA VS STROKE
- LOW TO MODERATE WEIGHT/COST

IMPROVED SSME MAIN OXIDIZER VALVE

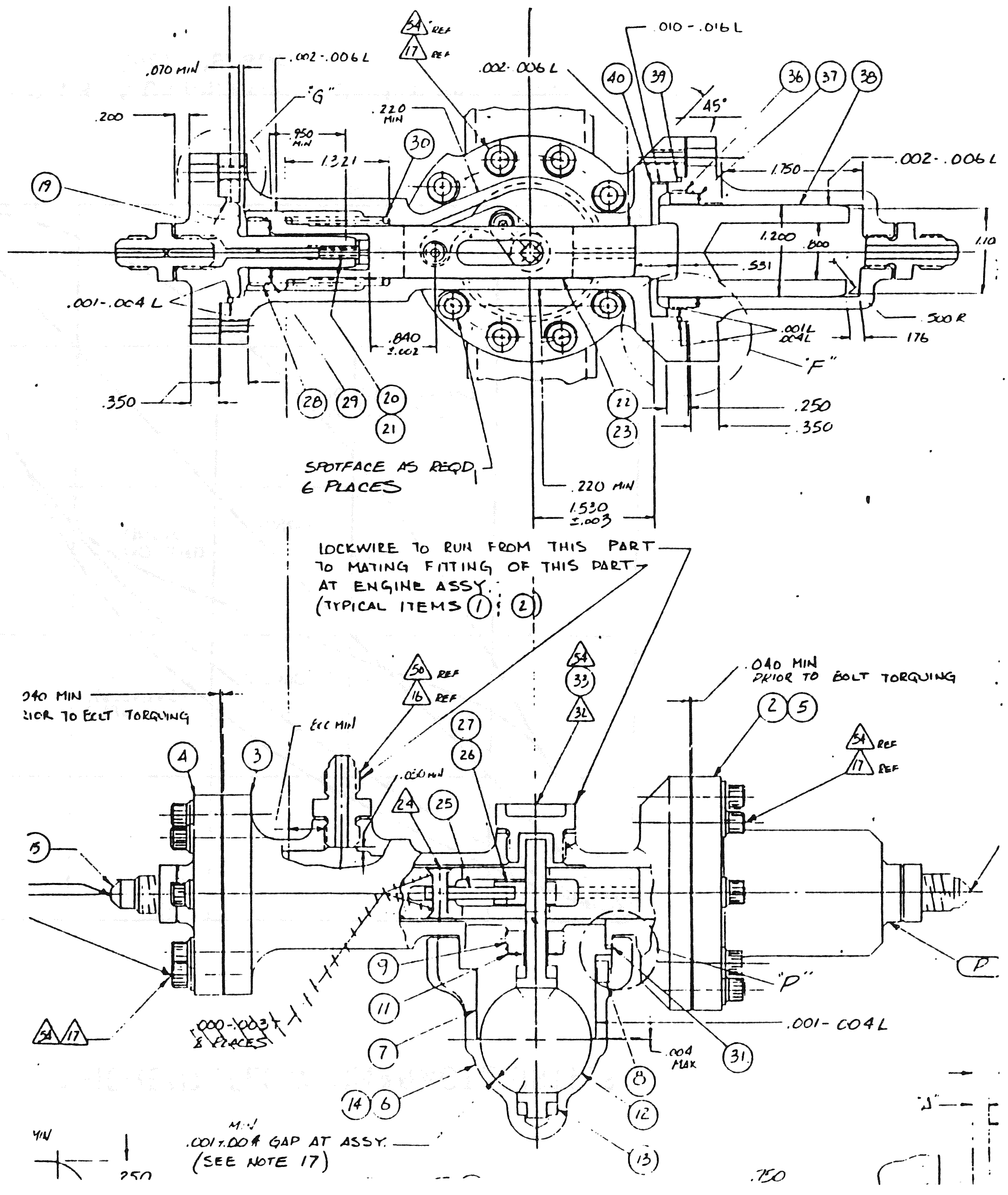
XLR129 butterfly valve design updated for SSME





- Contoured butterfly disc reduces torque
- Equal percentage flow control characteristic
- Integral with line
- Close-coupled to main burner injector
- Shutoff, shaft seal and pressure cycle endurance tested
- 46 firings, 753 sec

HELIUM/LINE PRESSURE-ACTUATED BUTTERFLY VALVE



INHERENT FLOW CHARACTERISTICS

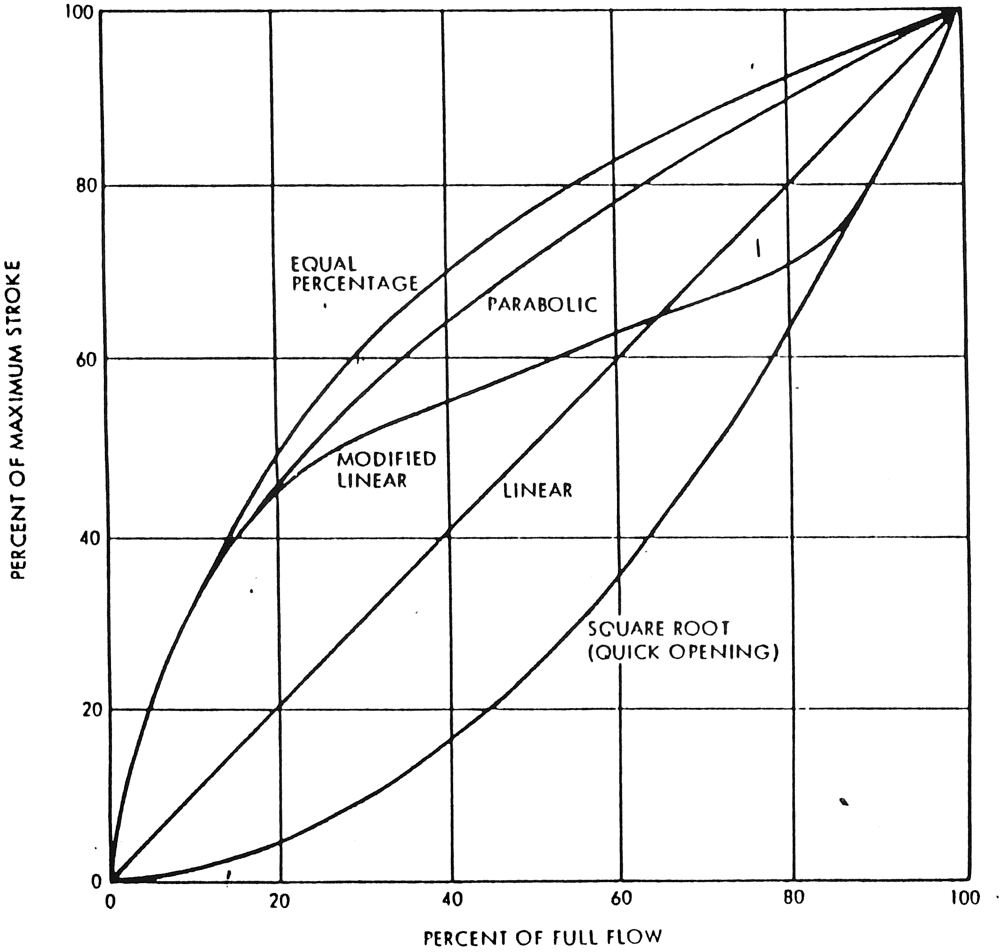


Figure 5.3.3. A Comparison of Control Valve Flow Characteristics

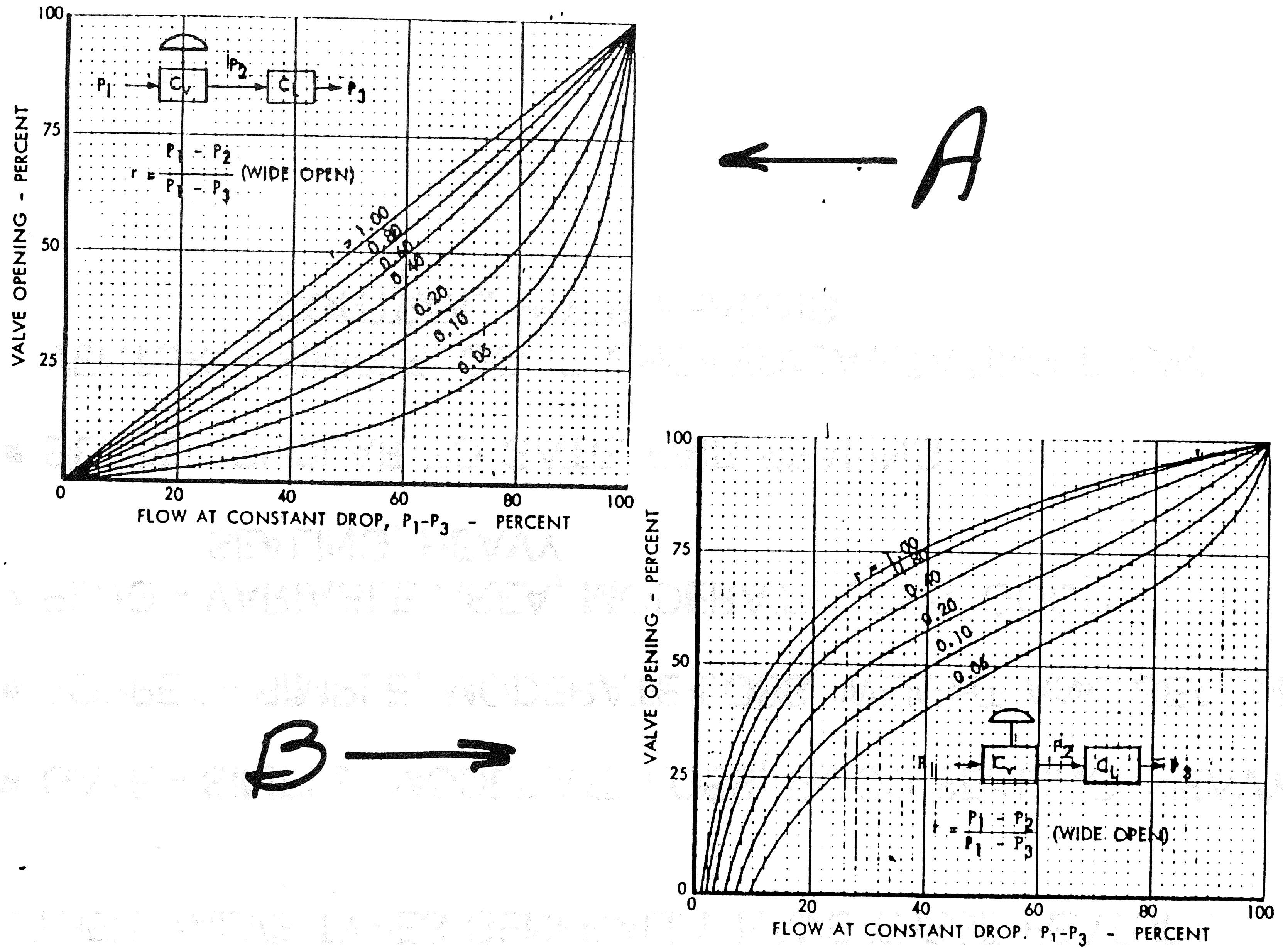


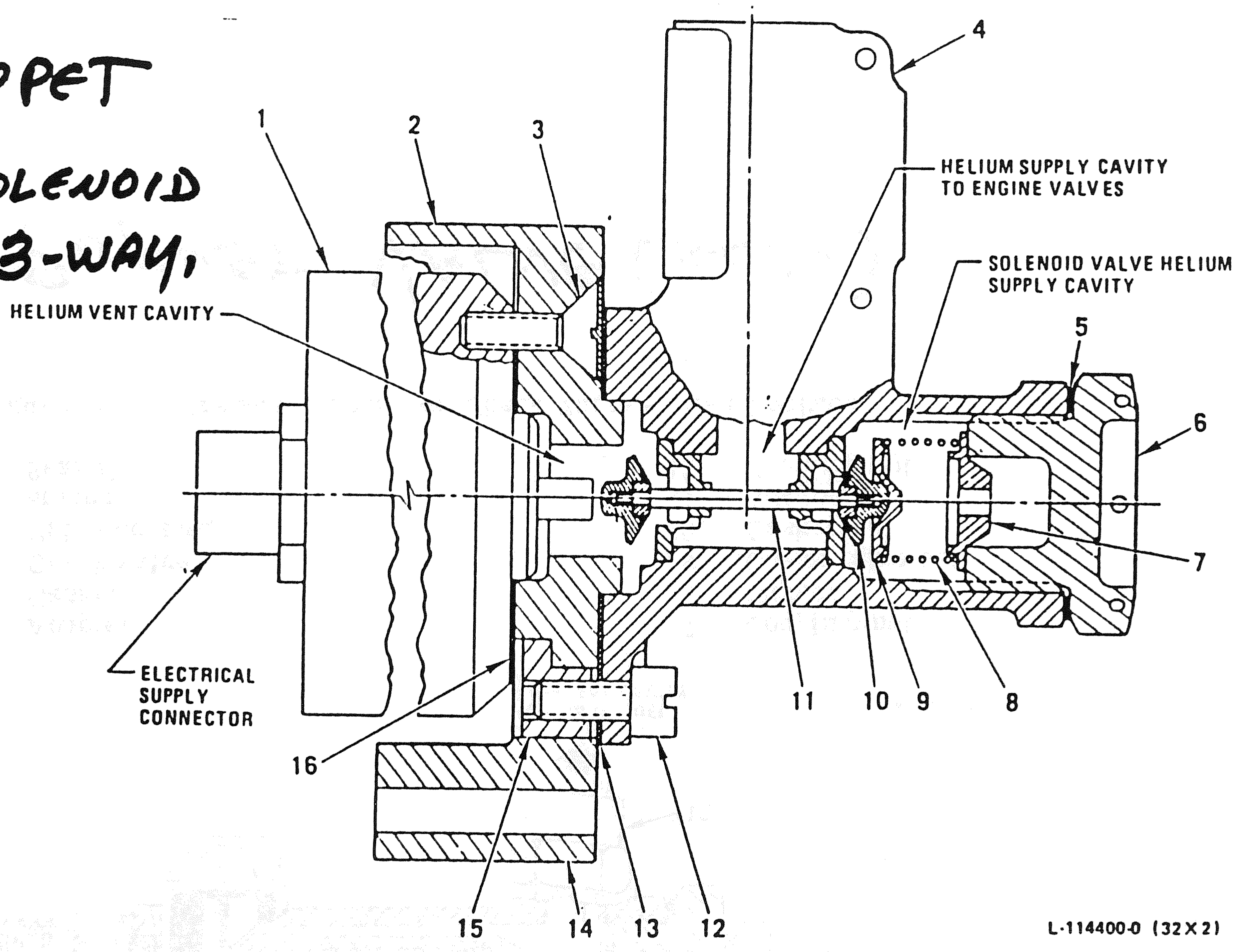
Figure 5.3.4a,b. Effective Flow Characteristics (a) Linear Valve (b) Equal Percentage Valve

VALVE CONCEPT SELECTION

OTHER VALVE TYPES GENERALLY HAVE THESE FEATURES

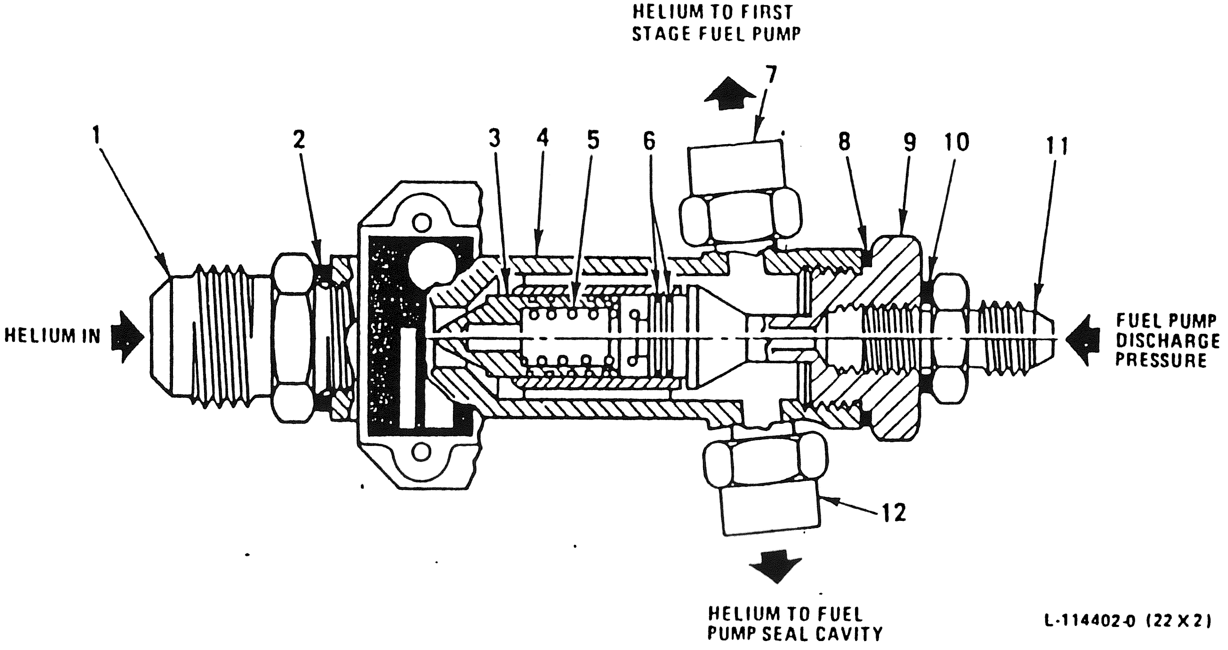
- GATE - SIMPLE, MODERATE LOSS, GOOD SEALING, HEAVY
- POPPET - SIMPLE, MODERATE LOSS, WEIGHT AND SEALING
- PLUG - VARIABLE AREA, MODERATE LOSS, GOOD SEALING, HEAVY
- BLADE - SIMILAR TO GATE, FAIR SEALING
- VENTURI - SIMPLE, GOOD CHOKED/CAVITATING FLOW CONTROL, POOR SEALING

HELIUM POPPET
VALVE-SOLENOID
ACTUATED, 3-WAY,
2 POSITION.



1. Solenoid coil
2. Solenoid assembly
3. Screw
4. Solenoid valve housing
5. Gasket
6. Spring retaining plug
7. Spring rear retainer
8. Solenoid valve spring
9. Spring front retainer
10. Solenoid valve seal assembly
11. Solenoid valve guide
12. Screw
13. Solenoid valve gasket
14. Solenoid spacer
15. Spacer insert
16. Spacer gasket

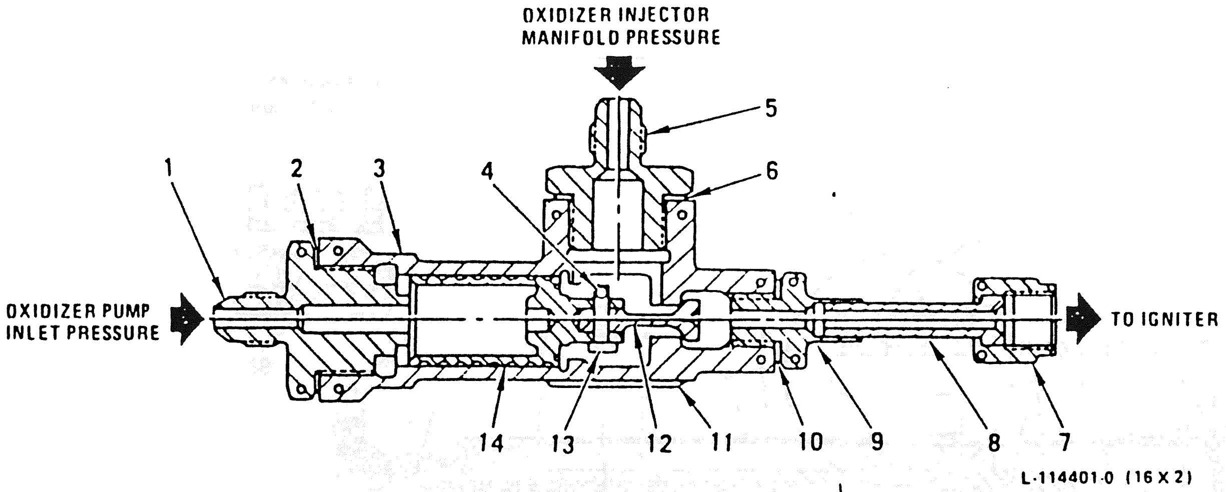
Figure 2-16. Prestart and Start Solenoid Valves - Closed Position



1. Adapter
2. Gasket
3. Check valve
4. Valve housing
5. Spring
6. Seal ring
7. Coupling nut
8. Gasket
9. Check valve seat
10. Gasket
11. Adapter
12. Coupling nut

Figure 2-18. Prelaunch Cooldown Check Valve - Open Position

POPPET VALVE (RL10)



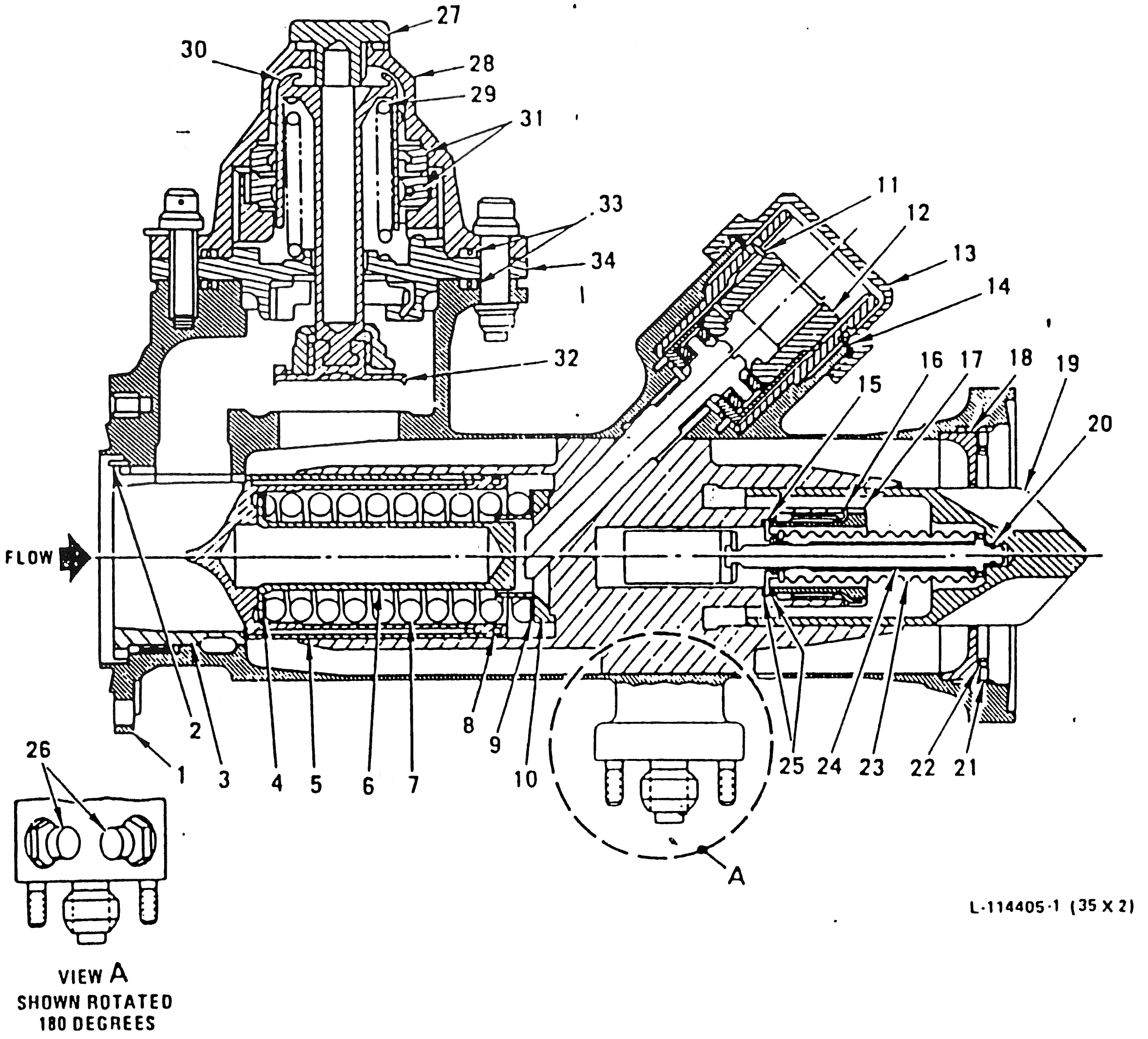
1. Tube reducer
2. Gasket
3. Housing
4. Cotterpin
5. Tube connector
6. Gasket
7. Tube nut
8. Tube
9. Tube connector
10. Gasket
11. Data plate
12. Valve
13. Pin
14. Piston

Figure 2-17. Igniter Oxidizer Supply Valve - Closed Position

POPPET VALVE (RL10)

RL10 LIQUID ROCKET ENGINE

LO2 POPPET/CONTOURED-PLUG VALVE



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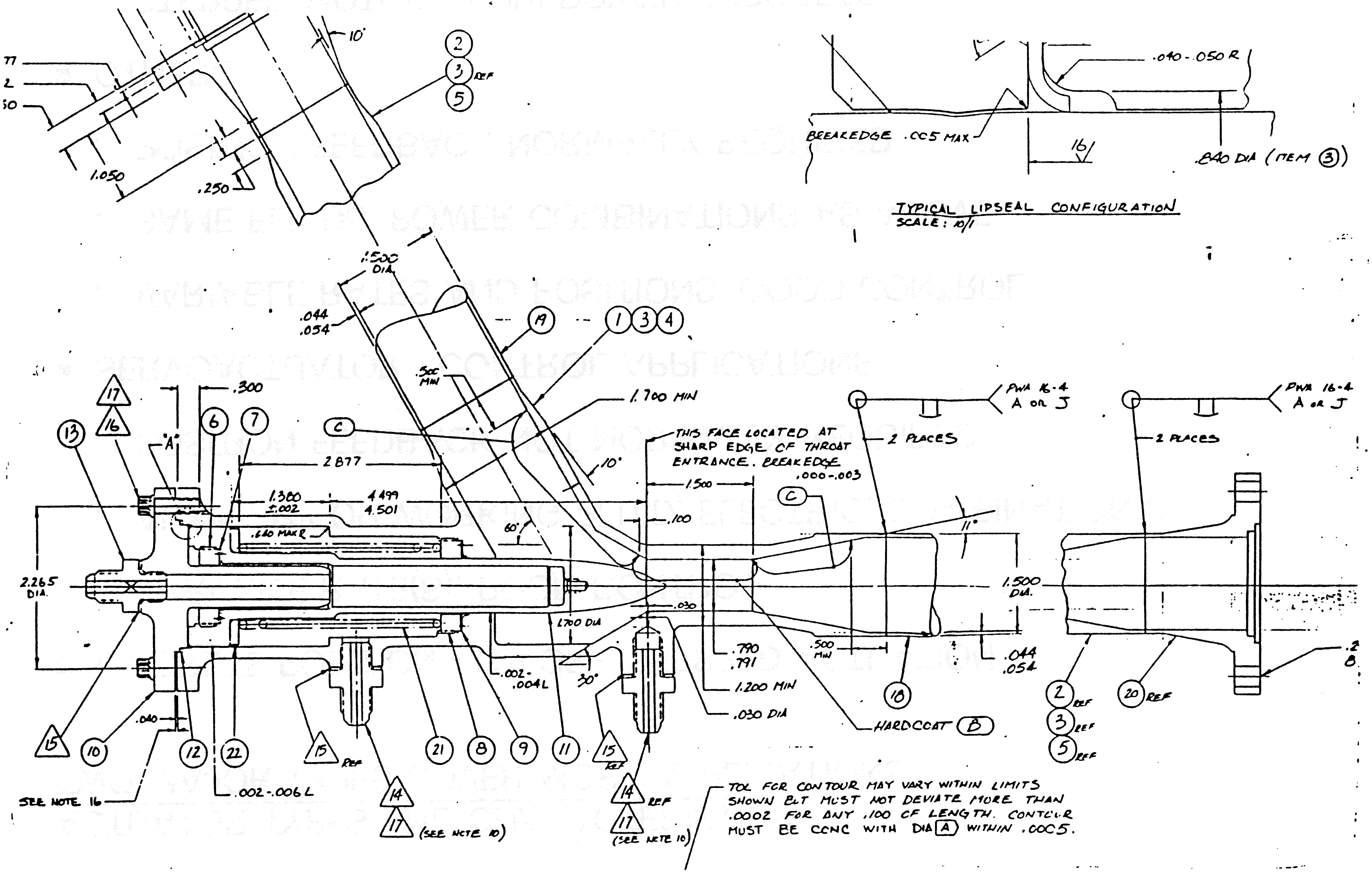
LH₂ CAVITATING VENTURI VALVE
(RL10-IIB ENGINE)



TYPICAL LIPSEAL CONFIGURATION
SCALE: 10/1

TOL FOR CONTOUR MAY VARY WITHIN LIMITS SHOWN BUT MUST NOT DEVIATE MORE THAN .0002 FOR ANY .100 OF LENGTH. CONTOUR MUST BE CONC WITH DIA $\boxed{A}$ WITHIN .0005.

ACTUATOR TYPES AND CHARACTERISTICS

TWO MAJOR TYPES COVER MOST APPLICATIONS

- DISCRETE POSITION - ON/OFF, STEPPED ACTUATION
    - VARIABLE RATE(S), POOR CONTROL
    - AUXILIARY OR WORKING FLUID, ELECTRIC, COMBINATION(S)
    - POSITION FEEDBACK NOT NORMALLY REQUIRED.
- SERVOACTUATOR - CONTROL APPLICATIONS
    - VARIABLE RATES AND POSITIONS, GOOD CONTROL
    - SAME FLUIDS, POWER COMBINATIONS AS ABOVE
    - POSITION FEEDBACK NORMALLY REQUIRED
- OTHERS
    - STEPPER MOTOR - LOW POWER, DISCRETE
    - GEARED ELECTRIC - LOW TO MODERATE POWER, VARIABLE

POSITION FEEDBACK CONCEPTS

ALL VALVE DESIGNS SHOULD PROVIDE FOR POSITION MEASUREMENTS

- CONTROL ELEMENT POSITION MEASUREMENT
- SIMPLE INSTALLATION, REMOVAL
- COMPATIBILITY WITH ENVIRONMENT, OPERATION
  - POTENTIOMETER - WIRE WOUND OR STRIP
  - LINEAR VARIABLE DISPLACEMENT TRANSFORMER (LVDT), ROTARY (RVDT)
  - RESOLVER
  - RESPONSE, RESOLUTION, THERMALS, FLUIDS, PRESSURES

SEAL CONCEPT SELECTION

EACH APPLICATION MAY REQUIRE SEPARATE TRADE STUDIES

- CONVENTIONAL VS EXOTIC MATERIALS
- CLAMPED VS GLAND TYPE RETAINER
- LEAKAGE VS LIFE
- ELASTOMERIC VS "SUPERFINISH"
- THROWAWAY VS REUSEABLE
- RESISTANCE TO CONTAMINATION, THERMAL TRANSIENTS, MOISTURE
- COMPATIBILITY WITH FLUIDS, ENVIRONMENT

BASIC DESIGN CONSIDERATIONS

ALL ELEMENTS SHOULD BE OPTIMIZED PRIOR TO LAYOUT COMPLETION

- MATERIALS, COATINGS, FINISHES, FABRICATION PROCESSES
- DEFLECTIONS, THERMAL COMPATIBILITY, STRESS AND LCF MARGINS
- RIG TESTS FOR CONFIDENCE, WHERE NEEDED
- WEIGHT
- PRODUCIBILITY AND INSPECTABILITY
- COST

ROCKET ENGINE CONTROLS AND VALVES SUMMARY

PLANNING AND ATTENTION TO DETAIL PRODUCE 1ST CLASS RESULTS.

1. KNOW SYSTEM AND SPECIFICATION REQUIREMENTS
2. SEARCH THE HISTORY
3. PAY ATTENTION TO SYSTEM PECULIARITIES
4. MAKE AS MANY TRADES AS YOU CAN
5. MAKE SURE YOUR DESIGN MEETS WEIGHT, COST, PRODUCIBILITY REQUIREMENTS

ROCKET/TURBINE CONTROLS COMPARISONS

ROCKET CONTROLS REQUIREMENTS ARE GENERALLY TOUGHER

- WORKING FLUIDS TEMPERATURE/PRESSURE EXTREMES GREATER
- MATERIALS CHOICES FEWER
- LUBRICANTS POORER, FEWER
- PRESSURE/TEMPERATURE TRANSIENTS FASTER
- SEALING MORE DIFFICULT, CRITICAL
- VIBRATION, ACOUSTICS WORSE
- CONTROLS PRECISION REQUIREMENTS TOUGHER