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NASA Contractor Report 174752

Study of Advanced Fuel System Concepts for Commercial Aircraft and Engines

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W. D. Byers, H. W. Fogg,
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Lockheed-California Company
Burbank, California

Contract NAS3-23271
January 1983

NASA
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FOREWORD

This is the final report of a study made under Contract NAS3-23271 for NASA-Lewis Research Center, Cleveland, Ohio. Mr. Steve Cohen at Lewis Research Center was Project Manager for the study. The report presents results of work performed during the 13-month period, from December 1981 through December 1982.

The Lockheed-California Company was the prime contractor to NASA and the study was managed by G. Daniel Brewer in the Commercial Advanced Design Division at Burbank, California. A portion of the work was subcontracted to Ergo-Tech, Inc., Dr. Jose Chirivella, President.

Lockheed wishes to acknowledge the technical advice of Exxon Research and Engineering Company and Union Oil of California as well as contributions in the form of illustrative material by Rolls-Royce, Limited.

The following individuals were principal contributors to this work:

<u>Lockheed-California Company</u>

Mr. E. F. Versaw	- Project Engineer
Mr. W. D. Byers	- Fuel System Analysis
Mr. D. E. Hanks	- Thermodynamics
Mr. H. W. Fogg	- Airplane Performance Analysis

<u>Ergo-Tech, Inc.</u>

Dr. Jose Chirivella	- Fuel property definition, analysis of systems to accommodate low thermal stability fuels, and general consulting.

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STUDY OF ADVANCED FUEL SYSTEM CONCEPTS FOR
COMMERCIAL AIRCRAFT AND ENGINES

E. F. Versaw, G. D. Brewer, W. D. Byers,
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SUMMARY

The specification for Jet A, the fuel currently used worldwide by commercial jet aircraft, ASTM D 1655-81, has evolved over a period of years. It represents a good match between cost of producing it from high quality crudes which have been readily available up to the recent past, and meeting the requirements for high performance aircraft and engines with minimum maintenance.

Since the oil embargo of 1973-74, it has become apparent that high quality crude may not be readily available in the foreseeable future. This situation will stem from two factors; the declining quantity of a finite resource, and the fact that a significant percentage of the world's crude oil is controlled by a politically unstable cartel. Accordingly, it is to be expected that increasing quantities of the Jet A of the future will be produced from crude of lesser quality, or from a synthetic crude derived from coal, shale, or tar sands. The question thus arises, should the present specification for Jet A be modified to relax certain of the properties in order to increase the yield, decrease the cost, and minimize the energy required to refine either the present or future grade crudes?

The subject study was undertaken to address this question insofar as it pertains to the effect such changes might have on the airframe and engine fuel system of a typical modern commercial jet transport. Specifically, the objectives of the study were to:

- Identify credible values for specific properties of jet fuel which can be considered realistic candidates for relaxation from the present specification,

- evolve advanced fuel system designs which will permit use of the relaxed property fuels,

- evaluate the performance of the candidate advanced fuel systems and the relaxed property fuels in a typical transport aircraft.

The following table lists values of the properties of jet fuel selected to represent relaxation considered feasible and realistic in the sense that they would simplify refining requirements from both present and probable future grades of crude oil. For comparison, values of these properties as presently required by specification ASTM D 1655-81 are also shown.

Fuel Property	ASTM Specification	Candidate Relaxed Property Value
Freeze Point, min, °C (°F)	-40 (-40)	-20 (-4)
Thermal Stability, max, °C (°F) (JFTOT Breakpoint Temp.)	260 (500)	204 (400)
Viscosity, min, mm^2/s (cSt)		
at -23.3°C (-10°F)	12 (12)	--
at -17.8°C (0°F)	--	15 (15)
Aromatic Content, % by Vol.	20 - 25	35
Lubricity, WSD, mm	--	0.45

The L-1011-500 TriStar commercial transport was used as a basis for study of design concepts for the airframe and engine fuel systems which could accommodate fuels with the relaxed properties. It was found that the unmodified baseline aircraft would be unable to use a fuel with the suggested high freeze point in commercial service. For example, fuel would freeze in the wing tanks on a long range flight at normal cruise altitudes. Also, on a -49°C day (the cold day environment accepted for this study), an aircraft which is forced to remain on the ground for a lengthy period after being fueled could find a significant fraction of its fuel load unpumpable. Some form of thermal protection or heat addition will be necessary. Similarly, an unmodified aircraft would be unable to use a fuel with the postulated low thermal stability on either standard or high temperature days. A method of cooling the fuel in critical engine fuel system components is required.

Suggested ranges for relaxation of the other fuel properties, i.e., viscosity, aromatic content, and lubricity, do not require changes which would be reflected to any significant degree in modifications of either aircraft weight or specific fuel consumption; therefore, aircraft performance would not be affected. Increased viscosity would affect the pumping power required. It would have negligible affect on engine power. Higher aromatic content would require modification of some seals and gaskets to use materials better able to resist softening or swelling; such material substitutions are readily available. The selected lubricity specification would not require any change in fuel system design; corrosion inhibitors currently used in Jet A fuel would provide acceptable lubricity characteristics.

To accommodate the specified higher freeze point and lower thermal stability property changes, several designs were studied, from which three preferred candidate fuel system concepts were envolved. System A uses electrical heating elements applied to the lower surfaces of the aircraft fuel tanks to keep the high freeze point fuel from freezing. It also has provision for supplying warm bleed air to heat fuel lines and critical components of the fuel system of each engine and the auxiliary power unit which may be subject to freeze-up, e.g., when these units are shut down in flight. To accommodate a fuel with low thermal stability, peak temperatures in the engine fuel system are reduced to levels which are compatible with a JFTOT rating of 204°C by using fan air, tanked fuel, and/or fuel being pumped to the engine as heat sinks for engine oil cooling, and by using a variable displacement pump to reduce the high pressure pump heat rejection.

Candidate Systems B and C incorporate all of these features from System A and, in addition, make use of insulation in various areas of the fuel tanks to decrease heat loss and minimize the requirement for adding heat to prevent fuel freeze-up. System B has 3.175 mm (0.125 in) thick insulation on the lower surfaces, including stringers, in all wing tanks. System C has the same type insulation on the upper surfaces of the outboard tanks, in addition to the lower surfaces of all tanks. In both systems, the heating elements are applied on the fuel side of the insulation layer on the bottom of the tanks. No heating elements are used over the insulation on the upper outboard tank surfaces in System C. The insulation suggested is polysulfide filled to 50 percent by volume with hollow borosilicate glass spheres which average approximately 80 microns diameter.

For fuel costs near present values, i.e., approximately $1.00/gal., System A was found to provide the lowest direct operating cost. However, with only a 27 percent increase in fuel cost, System B achieves parity in DOC and thereafter, as fuel costs continue to increase, System B would show increasing cost advantage. System B is therefore recommended as the most attractive fuel system candidate for the long term if changes such as those herein postulated are made to the jet fuel specification.

It is emphasized that there are no current plans in the industry to implement changes in the fuel specification. This preliminary study was conducted to explore potential technology requirements and performance trade-offs in the event future considerations indicate such relaxation is necessary. Much more work is needed, both experimental and analytical, before firm conclusions can be reached and final recommendations made.

SYMBOLS AND ABBREVIATIONS

A	ampere
a, abs	absolute
A/C, Ac	aircraft
acc	accelerate
ADV	advanced
API	American Petroleum Institute
APU	auxiliary power unit
ASTM	American Society for Testing and Materials
BOCM	Ball-On-Cylinder Machine
Btu	British thermal unit
c	centi
C	specific heat
°C	Celsius
CAL	calorie
CRC	Coordinating Research Council
cS, cST, cSt	centistokes

SYMBOLS AND ABBREVIATIONS

D	diameter
dc	direct current
decc	decelerate
deg	degree
DOC	direct operating cost
E	activation energy
e.g.	for example
eng.	engine
ERBS	Experimental Referee Broad-Specification
°F	Fahrenheit
FAA	Federal Aviation Administration
FCOC	fuel-cooled oil cooler
fpm	feet per minute
FSED	full scale engineering development
ft	feet
g	gram or gauge
gal	gallon (U.S.)
GCMS	gas chromatography - mass spectrometry
gen.	generator
h	hour or film heat transfer coefficient
HE	high energy
Hg	mercury
HP	high pressure or horsepower
hr	hour
i.e.	that is
in.	inch
IP	intermediate pressure
ISA	international standard atmosphere
J	joule
Jet A, A-1, B	Designation for Commercial Aviation Jet Fuels
JFTOT	Jet Fuel Thermal Oxidation Tester
JP-4	Designation for a Military Aviation Jet Fuel

SYMBOLS AND ABBREVIATIONS

K	kelvin
k	kilo or thermal conductivity
KCAS	knots calibrated airspeed
kt	knot
kVA	kilo volt-amperes
L, l	liter, length
lb	pound
lbm	pound mass
LP	low pressure
M	Mach number or mega
m	meter or milli
max	maximum
min	minute or minimum
mol	mole
N	Newton
NASA	National Aeronautics and Space Administration
n.mi.	nautical mile
NMR	nuclear magnetic resonance
No.	number
Nu	Nusselt number
OEW	operating empty weight
P	pressure
p	pico
Pa	pascal
Pr	Prandtl number
psi	pounds per square inch
Qobs	percent swell of a polymeric elastomer in a solvent
R-C	resistor-capacitor
Ref.	reference
RFP	Request for Proposal
Rn	Reynolds number
rpm	revolutions per minute

SYMBOLS AND ABBREVIATIONS

S	Sieman
s	second
sec	second
SFC	specific fuel consumption
SL, S.L.	sea level
SmCo	Samarium Cobalt
SMD	Sauter mean diameter
sp gr	specific gravity
Std	standard
sys, syst	system
SW	switch
T	temperature
TDR	tube deposit rating
temp	temperature
TOGW	takeoff gross weight
U.S.	United States
V	velocity or volt
visc	viscosity
Vol, vol	volume
vs.	versus
W	weight, weight flow rate or watt
W/O	without
WSD	Wear Scar Diameter
wt	weight
yr	year

SYMBOLS AND ABBREVIATIONS

SUBSCRIPTS

Related To Fluid Flow

aw	adiabatic wall
f	fuel
H	hydraulic
o	stagnation
p	pressure

Related to Solubility

o	solvent
p	polymer

GREEK SYMBOLS

Δ	difference (used as a prefix)
δ	ratio of static pressure to sea level static pressure or solubility parameter
μ	micro or absolute viscosity
ν	kinematic viscosity
ρ	density
σ	surface tension
Ω	ohm

1. INTRODUCTION

This is the final report describing a study conducted by the Lockheed-California Company for the NASA-Lewis Research Center to assess the impact on a commercial jet transport aircraft of using fuels which have relaxed property limits, relative to the current commercial jet fuel. This study is part of an overall program being conducted by NASA to provide the technological data base needed in the event it becomes necessary to make changes in aviation fuel properties.

The fuel currently used by the commercial aviation industry is derived from high quality crude oil. It meets a specification developed jointly by the engine manufacturers and the fuel producers following many years of laboratory research and operational experience. The resulting fuel has contributed importantly to the outstanding record of performance and operational characteristics of commercial transport aircraft. Until the oil embargo of 1973-74 the crude oils from which these fuels were refined were readily available throughout the world, at reasonable cost. This is no longer true, however, inasmuch as costs have increased tremendously and there is a declining quantity of high quality crude available on the market. The producers of jet fuels will increasingly be forced to consider the use of lower quality crudes as well as synthetic crudes obtained from coal, shale, and tar sands. The problem in doing this is that it requires costly changes to be made in the refining process in order to produce a jet fuel which meets the current specification. One means of minimizing this increased cost is to relax certain of the required fuel properties. If this can be accomplished it can also increase the yield of jet fuel obtainable from the existing high quality crudes. The question is, which fuel properties can be relaxed without compromising the performance and operational characteristics of the engine, or the safety of the aircraft, recognizing that the recent decline in availability of high quality crudes has already resulted in a reduction of the margins which had previously existed between delivered and specification fuel properties.

The objectives of the study were: to identify credible values for specific properties of jet fuel which could be considered realistic candidates for relaxation, to evolve advanced fuel system designs for commercial aircraft and engines which would permit use of the relaxed property fuels, and to compare the performance of a modern commercial transport aircraft using these advanced fuel systems and the relaxed fuel property limits with that of the baseline aircraft using current specification fuel. The study was limited to any system, subsystem, or component that is involved in the containment, delivery, or control of the fuel to the engine combustor. It thus was limited to delivery through the combustor fuel injection nozzles and did not include the combustion process itself.

The methodology of the study is outlined in Section 2, Technical Approach. Data used as input are identified in Section 3. Fuel properties are discussed in Section 4, which includes a listing of the property limits selected for relaxation. Section 5 presents an analysis of the effect the specified relaxation of fuel properties would have on the baseline aircraft, and Section 6 describes advanced fuel system component designs which will permit the satisfactory use of fuel with the candidate relaxed properties in the subject aircraft. Section 7 then provides a description and the results of an evaluation of candidate fuel system concepts which were evolved to accommodate a hypothetical fuel combining the relaxed properties in the reference aircraft. Based on results of this analysis, recommendations are presented in Section 8.

2. TECHNICAL APPROACH

The technical approach followed in this analytical study was predicated upon satisfying a set of guidelines and requirements established early in the program. These guidelines and requirements, together with the overall approach used in performing the analysis, are outlined in the following paragraphs.

2.1 Guidelines and Requirements

Reference Aircraft - The L-1011-500 commercial transport aircraft was selected to serve as the baseline vehicle for evaluating candidate advanced fuel system concepts in connection with the use of potential relaxed property fuels.

Flight Requirements - Evaluate aircraft performance in each of the following flight durations:

- Short range, duration $\leq$ 2 hours.
  Use 926 km (500 n.mi.) range.

- Medium range, duration between 2 and 6 hours.
  Use 3704 km (2000 n.mi.) range.

- Long range, duration $\geq$ 6 hours.
  Use 9260 km (5000 n.mi.) range.

Temperature Conditions - Evaluate aircraft performance for each of the following temperature conditions:

- Standard day normal atmosphere

- A standard high temperature atmosphere.

- A special low temperature atmosphere selected to represent a one day per year worst case condition.

Candidate Fuels - Fuels to be considered in the study were limited to hydrocarbons which would result from relaxation of property limits currently specified in ASTM D 1655-81. Other types of fuels such as hydrogen, methane, alcohols, and metal or carbon slurries were not included.

Fuel System Limits - The study was limited to consideration of any system, subsystem, or component that is involved in the containment, delivery, or control of the fuel to the engine combustor. It was thus limited to delivery of the fuel through the combustor fuel injection nozzles and did not include the combustion process itself.

Evaluation Requirements - Compare the flight performance of the reference aircraft using ASTM D 1655-81 Jet A kerosene as a baseline with four versions of its fuel system using postulated relaxed fuel properties. The four versions include the unmodified reference aircraft fuel system and three candidate advanced fuel system concepts designed to permit use of the relaxed fuel properties.

Evaluation Criteria - Evaluate the performance of the reference aircraft using the following criteria:

- Aircraft weight variation for constant range missions (see Flight Requirements)

  Gross Takeoff Weight
  Operating Empty Weight
  Block Fuel Weight
  Payload Weight

- Cost (manufacturing, maintenance, operations) as functions of fuel cost
- Safety
- Support requirements
- Complexity

2.2 Overall Approach

The study was conducted using analytical methods which have largely been substantiated by, or correlated with, experimental data; however, in some cases new methods had to be developed for which no experimental results exist.

The study effort was divided into three separate tasks. A block diagram showing the task breakdown and the interrelationships between them is provided in figure 1. A more detailed listing of the task breakdown is as follows:

Task I Baseline Aircraft Design and Input Data

- Compile pertinent data on the L-1011-500 long-range commercial transport aircraft.
- Identify flight parameters and fuel flow conditions for the L-1011-500 aircraft for short, medium, and long-range flights, and for hot, cold, and standard atmospheric conditions.
- Identify candidate values for fuel properties which are relaxed from those of the current specification.
- Evaluate performance of the baseline aircraft using the selected fuel properties.

Task II Conceptual Fuel System Designs

- Develop design concepts of advanced fuel system components and subsystems.
- Establish designs of three candidate fuel systems for the baseline aircraft which are capable of using hypothetical fuels with the relaxed properties.

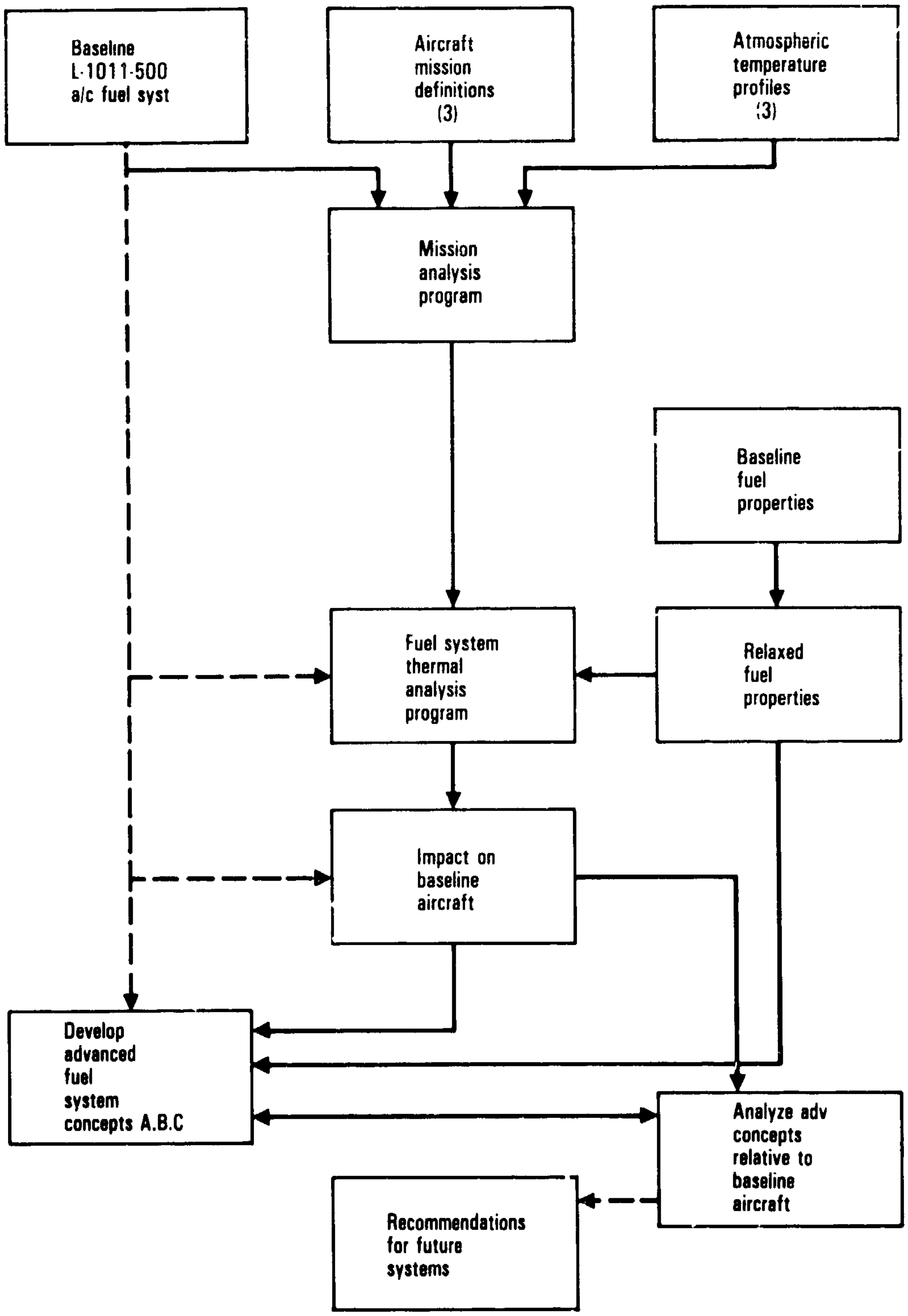


Figure 1 - Advanced fuel system concepts technical approach.

Task III Performance Evaluation of Candidate Fuel Systems

- Determine performance of each candidate fuel system in the modified baseline aircraft using the relaxed property fuels. Evaluate for short, medium, and long range flights, and for nominal and extreme temperature conditions. Select the preferred system.

- Provide recommendations.

3. BASELINE AIRCRAFT

3.1 Aircraft Description

The baseline aircraft selected for evaluating the candidate fuel systems is the Lockheed L-1011-500 shown in figure 2. A summary of its design characteristics is shown in table 1. The L-1011 is typical of current wide-body aircraft used in both domestic and international air routes. Versions of the L-1011 are used in short and medium-range applications as well as long-range applications which require added fuel in center section tanks.

The major impact of fuel property changes will be on the aircraft and engine fuel systems including fuel tanks, fuel supply systems, fuel metering systems, and the associated materials which are in contact with the fuel. Consequently, the following sections will describe only those systems which are directly affected by the fuel property changes.

3.1.1 Aircraft fuel system.

3.1.1.1 Aircraft fuel tank arrangement: The aircraft fuel tank arrangement, shown schematically in figure 3, includes four engine fuel feed tanks, all located in the wing, which function as a three-tank system. Tanks are numbered from left to right, 2L, 1, 3 and 2R. The 2L and 2R Tanks have an inboard and outboard compartment. Although any tank can supply fuel to any engine, the No. 1 Tank normally supplies fuel to the No. 1 engine, the 2L and 2R Tanks to No. 2 engine, and the No. 3 Tank the No. 3 engine. In addition, two auxiliary fuel tanks, designated 1A and 3A located in the aft three bays of the wing center section box beam, replenish fuel depleted from Tanks 1 and 3 as required.

The 2L and 2R Tanks each have an inboard and outboard compartment separated by a solid bulkhead. For structural reasons during flight, approximately 3856 kg (8500 lb) of fuel are retained in each outboard compartment as long as possible. A transfer line connects the outboard compartment to the inboard compartment surge box. When fuel in the surge box drops below approximately 454 kg (1000 lb), a float level control valve opens and fuel transfers by gravity to the surge box rendering the outboard compartment fuel available for usage.

Fuel transfer from the auxiliary Tanks (1A and 3A) is accomplished by means of ejector pumps. Motive flow for the ejector pumps is taken from the main discharge of the booster pump in Tanks 1 or 3.

3.1.1.2 Fuel management: The aircraft fuel storage tanks were initially sized to provide each engine with essentially the same available fuel quantity for operation on a normal tank to engine feed system selection. However, with the addition of fuel stored in the center section bays for long range flights, fuel management is utilized to sustain the basic tank to engine feed system principle and to allow the wing bending moments to remain within their design criteria. This is accomplished as follows:

Taxi, takeoff and initial climb mission segments are performed with tank to engine fuel feed. When the total fuel depletion approaches 6350 kg (14,000 lb), the

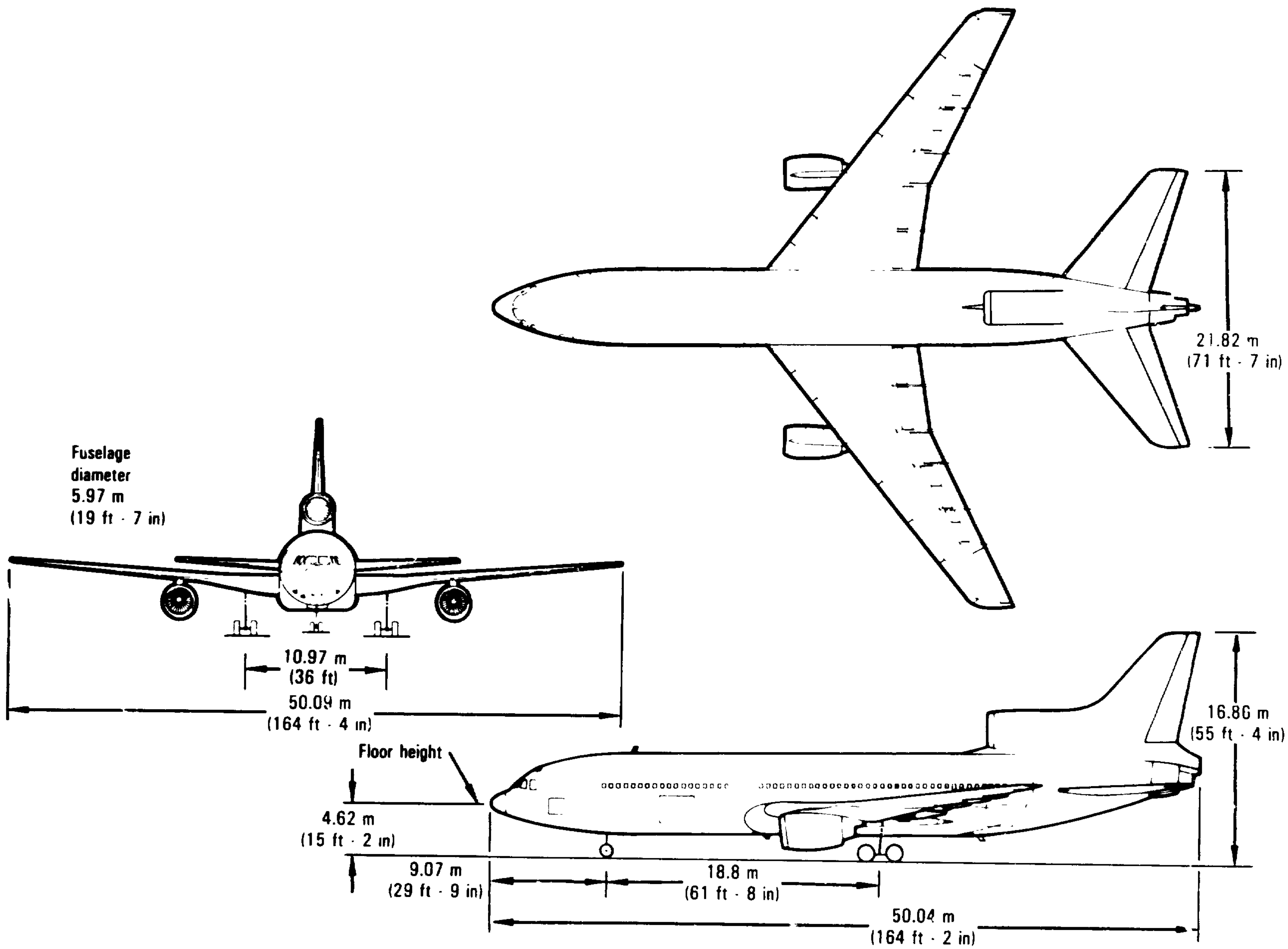


Figure 2 - L-1011-500 baseline aircraft general arrangement.

TABLE 1 - BASELINE AIRCRAFT DESIGN SUMMARY

	S.I. Units	U.S. Units
Wing		
Area	329.0 m^2	(3,541 ft^2)
Ref. Area	321.1 m^2	(3,456 ft^2)
1/4 Chord Sweep	35^o	
Aspect Ratio	7.62	
Horizontal Tail		
Area	119.1 m^2	(1,282 ft^2)
Sweep	35^o	
Vertical Tail		
Area	51.1 m^2	(550 ft^2)
Passenger Capacity	242	
Design Weights		
MAX Takeoff	231 293 kg	(510,000 lb)
MAX Landing	166 922 kg	(368,000 lb)
MAX Zero Fuel	[illegible] 314 kg	(338,000 lb)
Operating Empty	1[illegible] 307 kg	(245,3[illegible]0 lb)
Fuel Capacity	96 900 kg	(213,640 lb)
Engine		
SL Static Thrust	222 410 N	(50,000 lb)
Takeoff Flat Rating	29°C	(84^oF)
Airplane Performance		
Takeoff Field Length at SL 29°C (84°F)	2760 m	(9,060 ft)
Landing Field Length	2070 m	(6,790 ft)

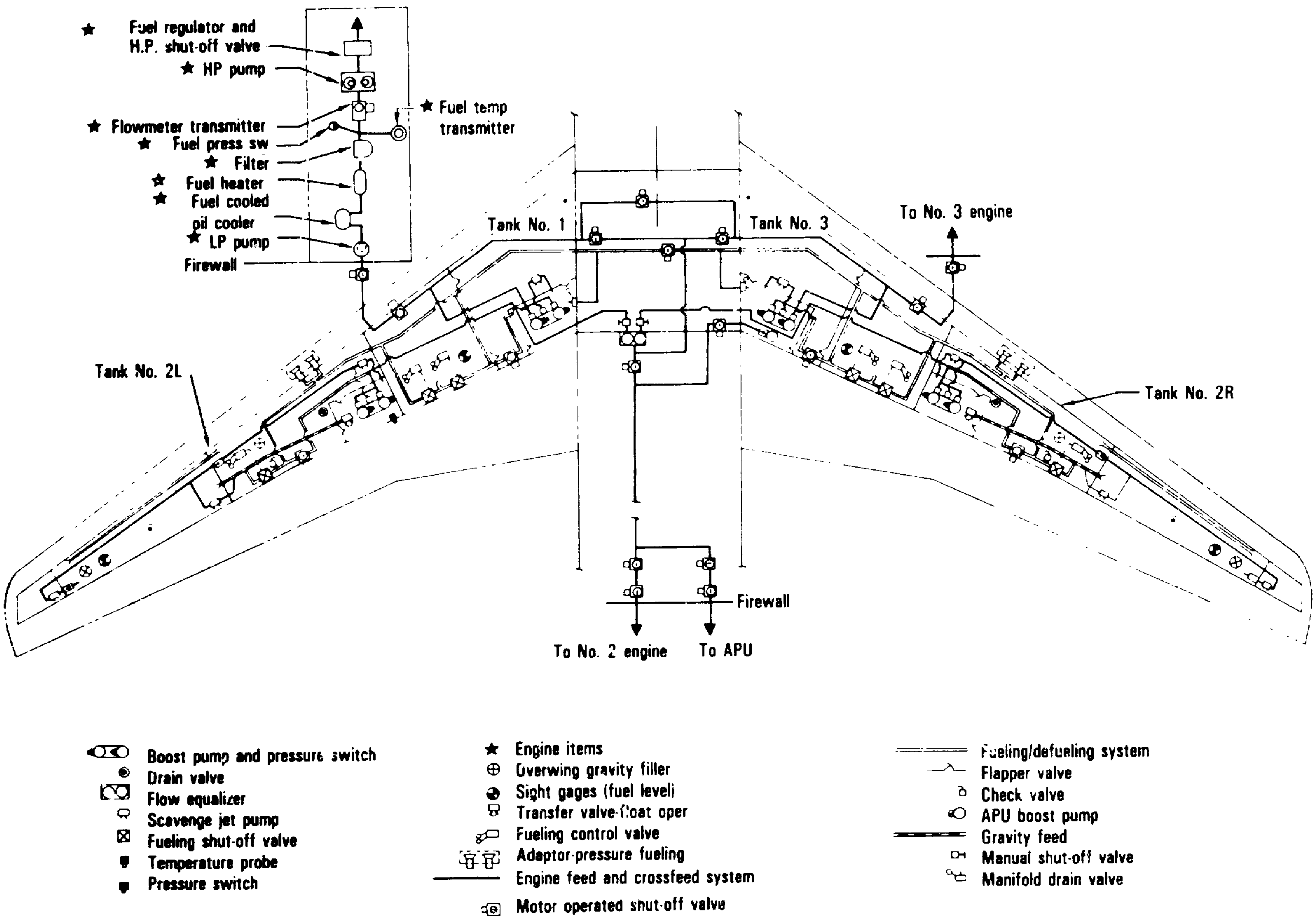


Figure 3 - Baseline aircraft fuel system.

crew initiates crossfeeding to all three engines from Tanks 1 and 3 only. Cross-feeding is continued until fuel is depleted in Tanks 1A and 3A, and fuel quantities in Tank 1, and Tank 3 and the sum of the fuel remaining in Tanks 2R and 2L are equal.

3.1.1.3 Engine feed system: The No. 1 and No. 3 Tanks each contain two identical ac motor-driven boost pumps and check valves, a dc motor-actuated tank shutoff valve located within the engine pylon upstream of the firewall, and the interconnecting plumbing to the interface with the engine.

The feed system for the No. 2 (aft fuselage) engine consists of; a) two pumps identical to those in the No. 1 and No. 3 Tanks in each of the two outboard Tanks (2L and 2R, b) a manually operated shutoff valve for each line where the lines exit from each tank at the wing root, c) a flow equalizer which equalizes the two tributary flows, d) a dc motor-actuated isolation valve located at the aft wall of the center section, e) two dc motor-actuated emergency (firewall) shutoff valves located upstream of the firewall, and f) the interconnecting plumbing to the engine interface. All fuel lines in the wing are contained inside the tanks. The No. 2 engine feed line is enclosed in a shrouded tube under the cabin floor within the pressurized compartment.

A cross-feed system connects to the normal tank-to-engine feed lines through appropriate dc motor-actuated shutoff valves and lines so that fuel can be supplied from any of the three tank systems to any engine.

To assure fuel availability to the tank boost pumps during various airplane attitudes and reduced fuel tank capacities, each tank contains a 454 kg (1000 lb) surge box reservoir, maintained full by scavenge ejector pumps.

3.1.1.4 Auxiliary power unit (APU) feed system: Fuel for the APU is supplied from Tanks 2L/2R by means of a common feed line with the No. 2 engine. A branch from the common line feeds directly to the APU interface through two emergency (firewall) motor-actuated shutoff valves.

3.1.1.5 Refueling system: A pressure fueling system with two fueling stations (outboard of each wing engine nacelle) is used to fuel the airplane. Each station has two 2-1/2 inch diameter standard type D-1 adapters suitable for accepting hoses from ground support refueling equipment. The right side station contains all of the gages and switches necessary to control and monitor the complete fueling operation. All tanks can be fueled from one station or the other, or from both stations simultaneously. A dc motor-actuated shutoff valve is located in the cross ship fueling manifold so that the left and right sides of the system are isolated from each other during normal fueling from both stations simultaneously. Dual type electrically-operated shutoff valves are used to each tank. Fuel level, dual float control pilot valves located at the full tank quantity level automatically operate the shutoff valves to prevent overfilling the fuel tanks.

3.1.1.6 Jettison system: Fuel jettison in flight is accomplished by means of the fuel tank boost pumps. During jettison, the boost pumps also feed the engines at the necessary fuel flow rate demanded for flight operational conditions. Fuel exits overboard through a dump mast which is located well outboard in the wing trailing edge. To prevent jettisoning fuel below 10 886 kg (24,000 lb) of airplane fuel, low level thermistors are installed in the tanks to shut off the jettison flow.

3.1.1.7 Vent systems: An open vent system is provided for all fuel tanks. Two vent outlets in each tank are required to insure communication to the ullage space for various aircraft attitudes. The aft vent outlet incorporates a float-operated vent valve which closes to avoid spillage out the vents during climb and opens during descent to allow venting at attitudes in which fuel covers the open forward outlet.

3.1.1.8 Scavenge system: The scavenge system consists of a series of jet pumps using motive fuel flow under pressure from the fuel tank boost pumps to induce a secondary flow from low points in the fuel tanks. The intent is to remove fuel and free water by scavenging through the secondary lines and delivering it to the surge boxes where it is pumped to the engines and consumed. The system works in parallel with the surge box wall-mounted flapper check valves for supplying fuel to the boost pumps.

3.1.2 Engine fuel system. - The engine fuel system is shown schematically in figure 4. It consists of a low pressure fuel pump which receives fuel from the aircraft fuel tank boost pumps and delivers fuel through a low pressure fuel filter to the low pressure side of a fuel-cooled oil cooler. The fuel then passes through a high pressure fuel pump and is delivered through the high pressure side of the fuel-cooled oil cooler and fuel regulators to the fuel injectors in the engine combustors. A brief description of the major components of the system follows.

3.1.2.1 Fuel pumps and spill valve: The LP pump is a centrifugal design and has been sized to provide the best matching between RPM and fuel flow rate during takeoff and climb.

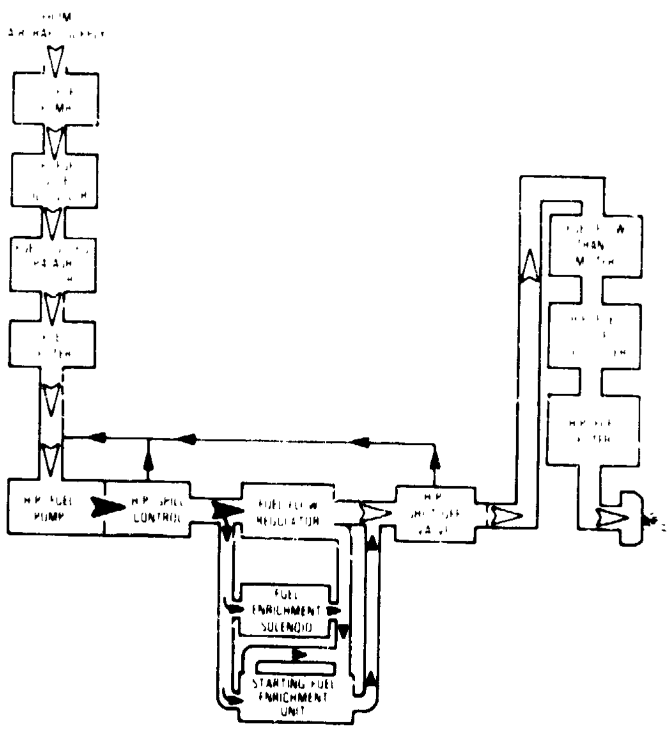


Figure 4 - Baseline engine fuel system schematic.

The HP pump is of the gear type with the plate ends being lubricated by the fuel. Its operating point has been selected to provide an optimum match between speed and volumetric flow rate at takeoff and climb conditions. At low power levels, the delivered fuel flow rate is much in excess of what is required by the engine fuel metering system, and a HP bypass conduit is provided to spill the excess fuel. The amount of bypass fuel is controlled by the spill valve assembly under the control of the fuel metering system. The spill valve assembly is integrated with the combined pump unit.

3.1.2.2 LP fuel filter: This unit has the primary function to filter the debris washed down from the wing tank and prevent it from invading the fuel metering system and other small passages. In those cases in which severe pressure drops may exceed the operational limits of the fuel system, a bypass mechanism overrides the fuel filter.

3.1.2.3 Fuel cooled oil cooler: The fuel-cooled oil cooler exchanges heat between the fuel and the engine scavenge oil. It serves two purposes: to heat the fuel at cold fuel conditions and to cool the oil at hot oil conditions.

3.1.2.4 Fuel flow regulators: The Main Fuel Flow Regulator controls the fuel pressure supplied to the fuel injectors directly and through the Starting Fuel Regulator and Cold Day Enrichment Valve it supplies added fuel during normal and cold day engine starts, respectively.

3.1.2.5 High pressure shutoff valve: This valve has a dual function: a) it interrupts completely the fuel supply to the burner system when the engine is shut down; b) after shutoff, it allows the fuel located in the hot region system components (manifold, distribution valves, and injectors) to be drained by means of a separate line to the cool fan case region, and delivered to a small tank.

3.1.2.6 Fuel manifold distribution valves and spray nozzles: The fuel manifold distribution valves and pigtail conduits to the injectors are shown in figure 5. The distribution valves are passive and control the fuel fraction through them by means of a biased spring-loaded poppet. There are six distribution valves with each valve distributing the fuel to three injectors. The location of the spray nozzles and distribution valves can be seen in figure 6. Spray nozzles Nos. 8 and 12 are provided with ignitor plugs for engine start.

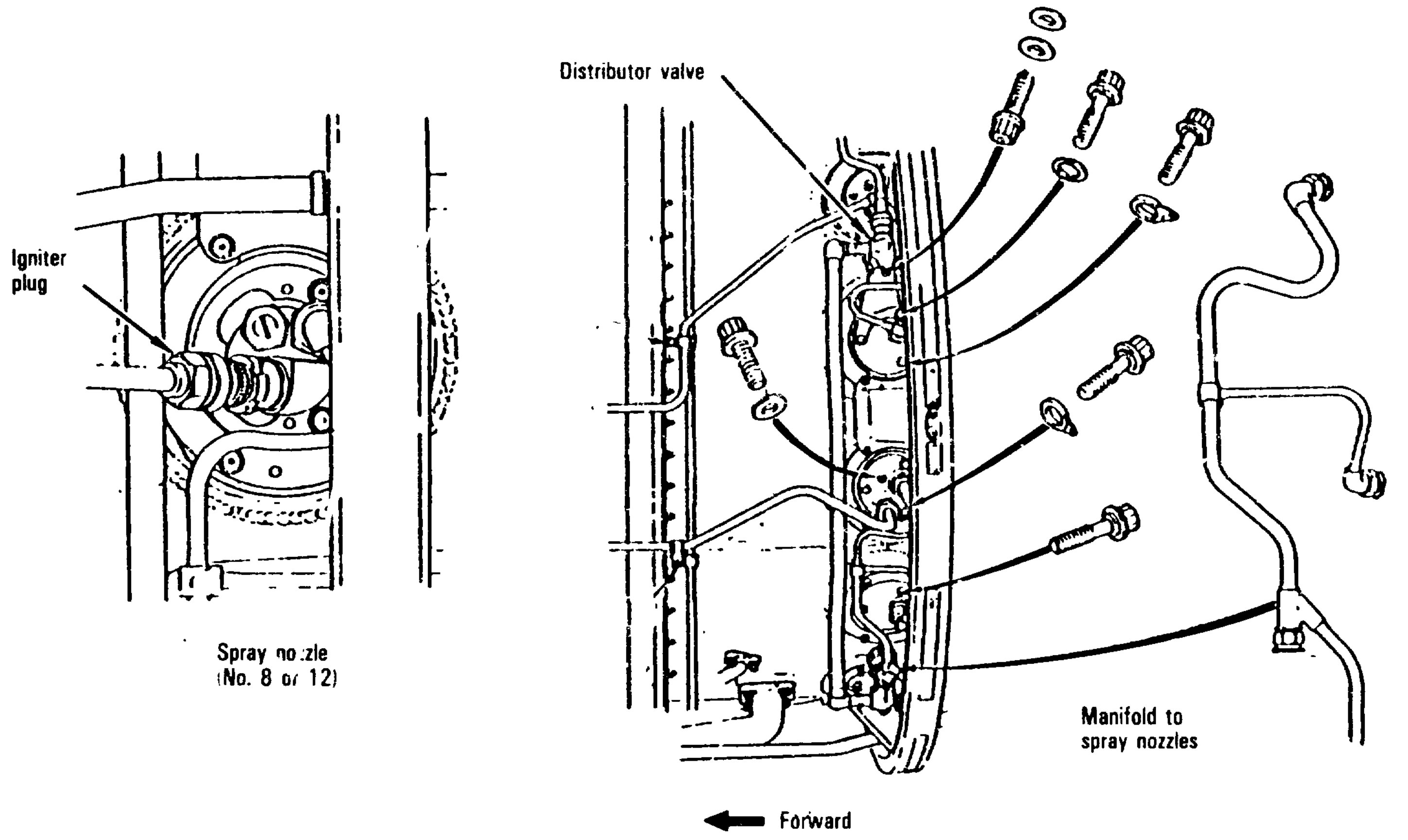


Figure 5 - Fuel spray connections and manifolding.

Figure provided through the courtesy of Rolls-Royce, Limited.

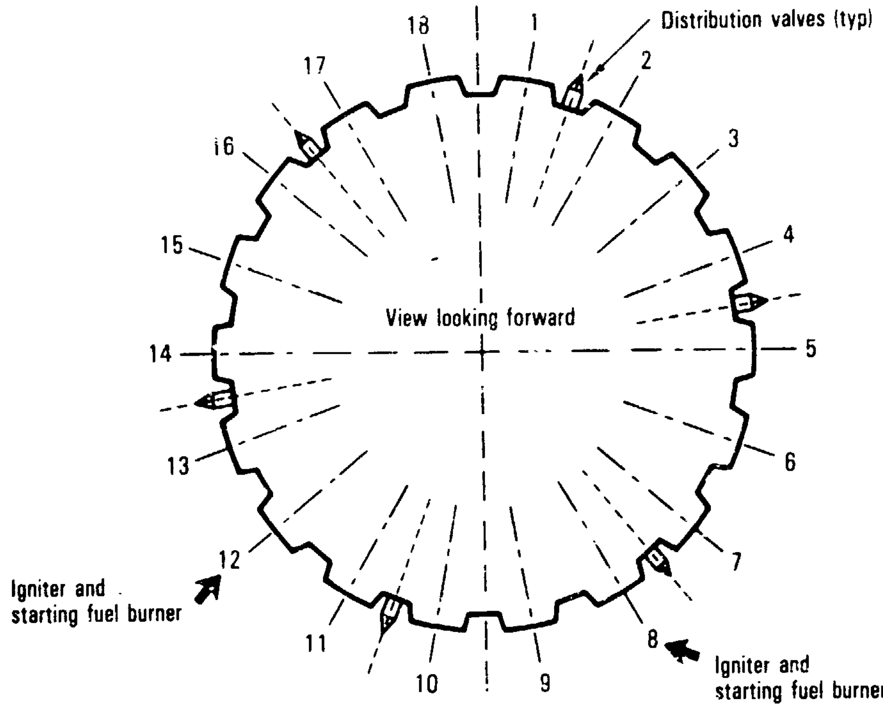


Figure 6 - Burner and fuel distribution valve positions

Figure provided through the courtesy of Rolls-Royce, Limited.

3.2 Flight and Temperature Profile Descriptions

3.2.1 Payload/range requirements. - A payload range curve for the L-1011-500 is presented in figure 7. This type of presentation shows the limiting values for an aircraft in a particular configuration flying under a particular set of conditions. The figure shows, for example, that maximum payload is limited to 42 000 kg and that maximum range at that payload is obtained when the airplane takes off at the maximum allowable takeoff gross weight. Payload is traded for fuel as the operating point moves down the maximum takeoff gross weight line until the maximum fuel capacity is reached. From this point down to zero payload, the fuel capacity is the limiting factor and the range increases as the takeoff and thus the mission weight is reduced.

With selection of the L-1011-500 as the baseline aircraft, typical missions that fit the study requirements for flight duration were defined as follows:

a) short range, (less than 2 hours) = 926 km (500 n.mi.)
b) medium range, (2 - 6 hours) = 3704 km (2000 n.mi.)
c) long range, (greater than 6 hours) = 9260 km (5000 n.mi.)

These ranges are consistent with operational missions that an airline might schedule for the L-1011-500. In service an airplane is not scheduled at its design range and payload for each of its flights but will typically fly at shorter ranges

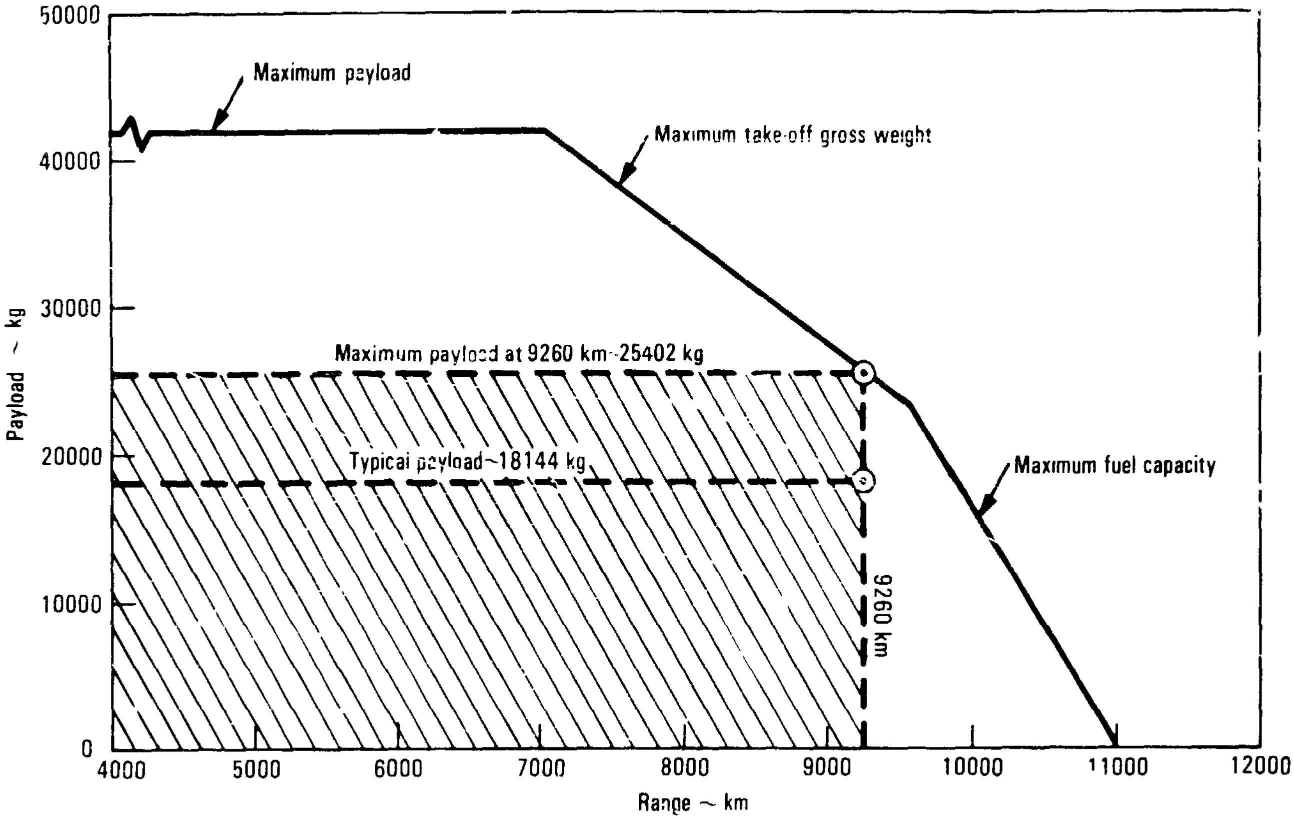


Figure 7 - Baseline aircraft payload/range - hot day (ISA + 34°C)

and reduced capacity, as for example the shaded area of figure 7. During 1981, two operators of the L-1011-500, reported average ranges of 6612 km (3570 n.mi.) and 7084 km (3825 n.mi.), respectively.

A payload of 18 144 kg (40,000 lb) was selected as the typical operational payload for this study. This represents passenger load factors in the range from 60 to 70 percent and cargo loads between 4536 and 2268 kg (10,000 and 5000 lb). The airlines mentioned above had load factors of 63 percent and 74 percent for the L-1011-500 for the reporting period. The total industry average load factors for the same period of 1981 and the available reporting quarters of 1982, averaged slightly lower (Ref. 1). The selected payload of 18 144 kg (40,000 lb) therefore encompasses the payloads being realized.

To allow for the full capacity case, calculations were also made with 100 percent passenger load factor and 2268 kg (5000 lb) of cargo. This was accomplished on the extreme hot day and used to determine the operational capability of the airplane with the fuel system changes.

3.2.2 Flight profile. - Flight profiles were selected to simulate properly airline operation. Thus, Federal Aviation Agency rules were considered as well as practical operational limitations. A consistent set of ground rules were used for each of the temperature environments, differences occurring only when dictated by engine limitations. The flight profiles selected consist of four major segments: takeoff, climb to altitude, step cruise and descent. On the standard and extreme cold days they are identical insofar as altitudes and speeds are concerned but on the extreme hot day the altitudes vary due to thrust limitations on the engines.

On the extreme cold and standard days climb is made with Normal Climb (Maximum Cruise) power at a calibrated airspeed of 165 m/sec (320 kt)/Mach 0.82 to the nearest odd pressure altitude below that for a W/ of approximately 0.86 x 10^6 kg (1.9 x 10^6 lb), with a maximum pressure altitude of 11 887 m (39,000 ft). This value of weight over ambient pressure ratio has been determined from previous Flight Management System studies to represent the best altitude at which to initiate cruise from an optimum cruise standpoint. On the extreme hot day, climb is made with Maximum Climb power at a calibrated airspeed of 165 m/sec (320 kt)/Mach 0.82 to the highest pressure altitude at which the aircraft can still fly with Maximum Cruise Power, again with a maximum pressure altitude of 11 887 m (39,000 ft). A minimum of 91.4 m/min (300 fpm) rate of climb capability is maintained throughout the climb segments.

A Mach 0.82 cruise, at partial power, is then initiated at the end of the climb segment and continues until the specific air range (km/kg fuel) is greater at an altitude 1219 m (4000 ft) higher. This procedure is continued to achieve the desired mission range. This step cruise operation is used to approximate cruise at optimum specific air range and is consistent with airline operation when Air Traffic Control designates the available altitudes.

The descents from altitude for the mission profiles are done at cabin pressure limited rates of descent. A cabin limited rate of descent is defined as the rate at which the total time to descend is equivalent to the time required to pump the pressure in the cabin up to the ambient pressure at the end of descent. Upper portions of the descent are often limited by the maximum cabin pressure differential. In these cases an idle power setting would bring the aircraft down faster than the limiting pump rate could bring the pressure up in the cabin and the cabin pressure differential would exceed its limit. For this reason, the first segment of some of the high altitude descents require partial power.

Reserve fuels were calculated and are included in the missions. Domestic rules were used for the 926 and 3704 km (500 and 2000 n.mi.) missions and International rules for the 9260 km (5000 n.mi.) mission. All of the reserves were calculated for a 370 km (200 n.mi.) alternate range. The cruise portion of the flight to an alternate airport for extreme cold and standard atmospheres was flown at 9144 m (30,000 ft); however, on the extreme hot day mission a 6096 m (20,000 ft) cruise altitude was used due to thrust limitations.

Domestic Reserves are calculated using the following flight profile segments:

1) Missed approach, climb to 457 m (1500 ft)
2) Climb to 3048 m (10,000 ft) at a calibrated airspeed of 129 m/s (250 kt)
3) Accelerate to a calibrated airspeed of 154 m/s (300 kt)
4) Climb to cruise altitude at a calibrated airspeed of 154 m/s (300 kt)
5) Cruise at optimum Mach
6) Descend to 3048 m (10,000 ft) at a calibrated airspeed of 154 m/s (300 kt)
7) Decelerate to a calibrated airspeed of 129 m/s (250 kt)
8) Descend to Sea Level
9) Include a 45 minute hold maintaining the fuel flow at the end of cruise

International Reserve calculations are broken down into two parts: Part I is the contingency fuel which is 10 percent of the total flight time at the fuel flow at the end of the last cruise segment, and Part II which is a flight profile broken down into the following segments:

1) Missed approach, climb to 457 m (1500 ft)
2) Climb to 3048 m (10,000 ft) at a calibrated airspeed of 129 m/s (250 kt)
3) Accelerate to a calibrated airspeed of 154 m/s (300 kt)
4) Climb to cruise altitude at a calibrated airspeed of 154 m/s (300 kt)
5) Cruise at optimum Mach
6) Descend to 3048 m (10,000 ft) at a calibrated airspeed of 154 m/s (300 kt)
7) Decelerate to a calibrated airspeed of 129 m/s (250 kt)
8) Descend to Sea Level

3.2.3 Temperature-altitude profiles. - For this study three different atmospheric days were used for analysis purposes. These were a standard day, an extreme cold day and an extreme hot day (figure 8).

3.2.3.1 Standard day: The standard day temperature - altitude profile is defined to be that set forth in the 1962 U.S. Standard Atmosphere tables.

3.2.3.2 Cold day: The cold day temperature-altitude profile was developed to realistically represent an extreme cold day environment. World extremes of temperature have been compiled into MIL-STD-210B, "Climatic Extremes for Military Equipment". These extreme cold day ambient temperatures may be expected to occur once in 10, 15 or 20 years, depending upon the lengths of record from which they were obtained but represent approximately zero probability basis for a given year. A more

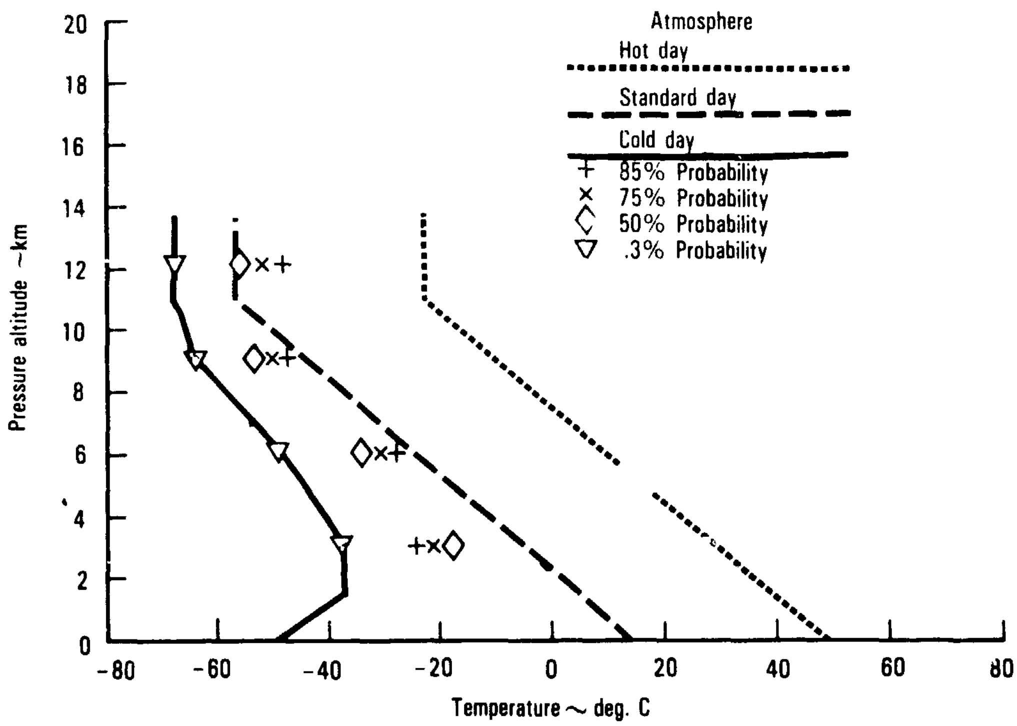


Figure 8 - Altitude/temperature profiles

realistic method relating extreme cold ambient temperatures at altitude to global location was formulated in a recent NASA study (ref. 2). This study showed the extreme cold temperatures that might be encountered by aircraft during a flight with an annual 0.3 percent probability of occurrence (one day a year).

The one day a year percent probability of occurrence, translates into an extreme cold temperature exposure time of 1.8 minutes for the 9260 km (5000 n. mi.) cold day mission. The time duration at the minimum ambient temperature is very short relative to the total time of the flight and thus has little effect on the the fuel temperature. An aircraft flying the route for a month would have the equivalent of 30, 1.8 minute extreme cold temperature exposures, or one 54 minute period per month. A time period of one hour was assumed in this study which allows a reasonable amount of fuel tank exposure time to the extreme cold temperature.

The minimum fuel temperature is calculated by determining the heat transfer to the air adjacent to the wing skin and the time of exposure. The temperature of this air is a function of static air temperature, aircraft speed, and the percent of stagnation temperature rise recovered in the boundary layer air. The static or ambient temperatures for the cold day temperature-altitude profile were developed from statistical temperature information derived from actual worldwide temperature recordings to 16 154 m (53,000 ft) (ref. 3). Temperature data were compiled and computed to report a mean route temperature with 50, 75, and 85 percent probability of occurrence; i.e., temperatures which are not expected to be exceeded 50, 75, and 85 percent of the time.

To determine the probability of temperatures which are not expected to be exceeded, an inverse normal integral function (Gaussian distribution) was utilized. The one day a year, 0.3 percent probability, was input to this distribution to determine the standard normal variable. The standard normal variable for 0.3 percent and the temperature and standard normal variables associated with 50, 75, and 85 percent probabilities are plotted to extrapolate a temperature for the one day a year, 0.3 percent probability.

The cold day temperature-altitude profile is shown in figure 8. These altitudes, temperatures, and probabilities were utilized in the cruise portion of the flight to define the ambient temperature through which the aircraft flies. This ambient temperature profile for the 9260 km (5000 n.mi.) cold day mission (figure 9) represents the worst case extreme cold day temperature environment used to predict the fuel tank temperatures in this study.

3.2.3.3 Hot day: The hot day temperature - altitude profile follows the hot day environmental operating envelope of the L-1011-500. This is the maximum temperature day for which the environmental control system of the L-1011-500 remains within its operating design limits.

3.3 Baseline Aircraft Performance

Performance of the reference aircraft was determined on the selected flight profiles for each of the mission ranges and for the three temperature profiles; this was accomplished using the Lockheed Aircraft Mission Analysis Program. The results have been summarized as time histories of altitude, Mach number, ambient temperature and fuel quantity in each of the tanks while flying at a constant airspeed of Mach 0.82. Each of these variables were calculated for all nine mission range-atmosphere combinations for use in determining fuel tank temperatures. However, only selected values are shown for each of the combinations.

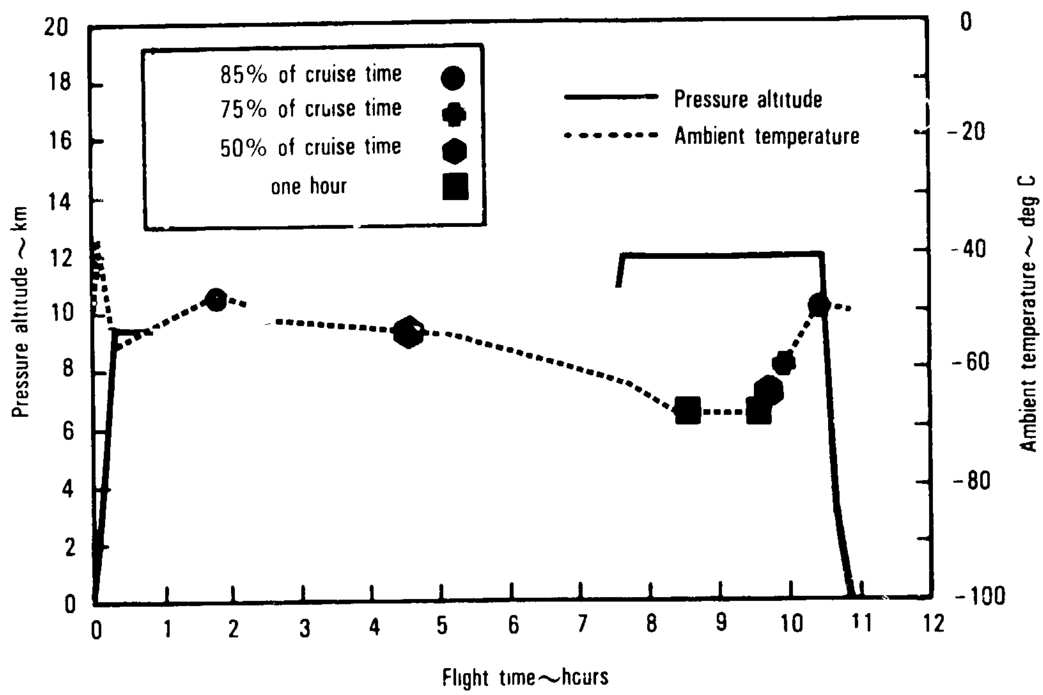


Figure 9 Cold day flight profile, 9260 km (500 n.mi.) mission

3.3.1 9260 km (5000 n.mi.) mission. - The entire set of data for the critical mission, 9260 km (5000 n.mi., on the cold day, are presented for illustration in figures 10 through 12. Figure 10 shows that for optimum specific range the airplane cruises at altitudes of 9449, 10 668, and 11 887 m (31,000, 35,000, and 39,000 ft). The ambient temperature that the aircraft operates in during the flight is presented in figure 11. During the majority of the cruise the ambient temperature is in the range from -73 to -51°C. Fuel quantities in each of the four tanks are shown in figure 12. From this figure it can be seen that all of the fuel in Tank 1A is used before burning any significant amount of fuel from any of the other three tanks. When Tank 1A is depleted, fuel is used from Tank 1 and shortly afterward from Tank 2 Inner. When Tank 2 Inner reaches the 454 kg (1000 lb) level, this level is maintained by transfer from Tank 2 Outer. Tanks 1 and 2 Outer are then used until the end of the flight.

For the remaining mission range-atmosphere combinations, only the altitude and ambient temperature time histories are presented. The flight profile on the standard day uses the same cruise altitudes as the extreme cold day and the altitude time history is therefore similar, as shown in figure 10. The corresponding ambient temperature time history for standard day is presented in figure 11. On the extreme hot day, however, figure 10 shows that the cruise altitudes vary, due to thrust limitations, and the flight profile is at lower altitudes. The cruise altitudes for the extreme hot day are 8839, 10 058, and 11 278 m (29,000, 33,000, and 37,000 ft). The ambient temperature time history for this flight profile and atmosphere is also shown in figure 11.

3.3.2 3704 km (2000 n.mi.) mission. - The flight profiles for the medium range mission are similar for the standard and extreme cold days and different for the extreme hot day. On the standard and extreme cold days the aircraft climbs to a cruise altitude of 11 887 m (39,000 ft) and remains there until the descent to the

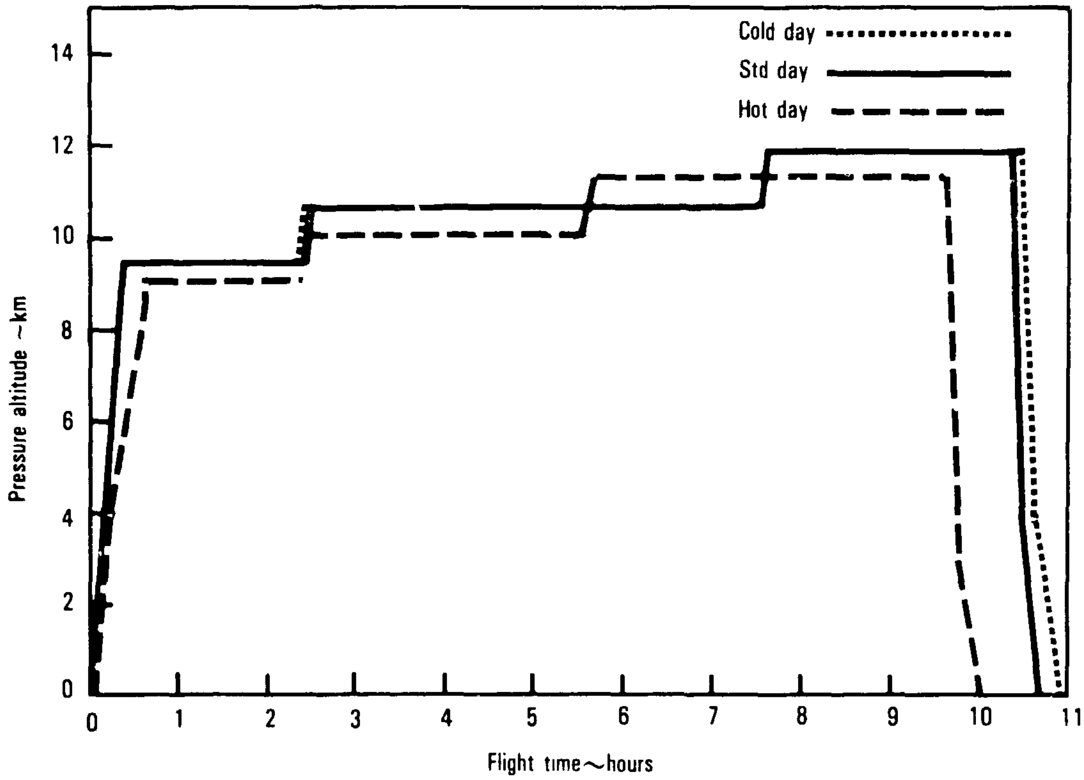


Figure 10 - Flight profiles, 9260 km (5000 n.mi) mission.

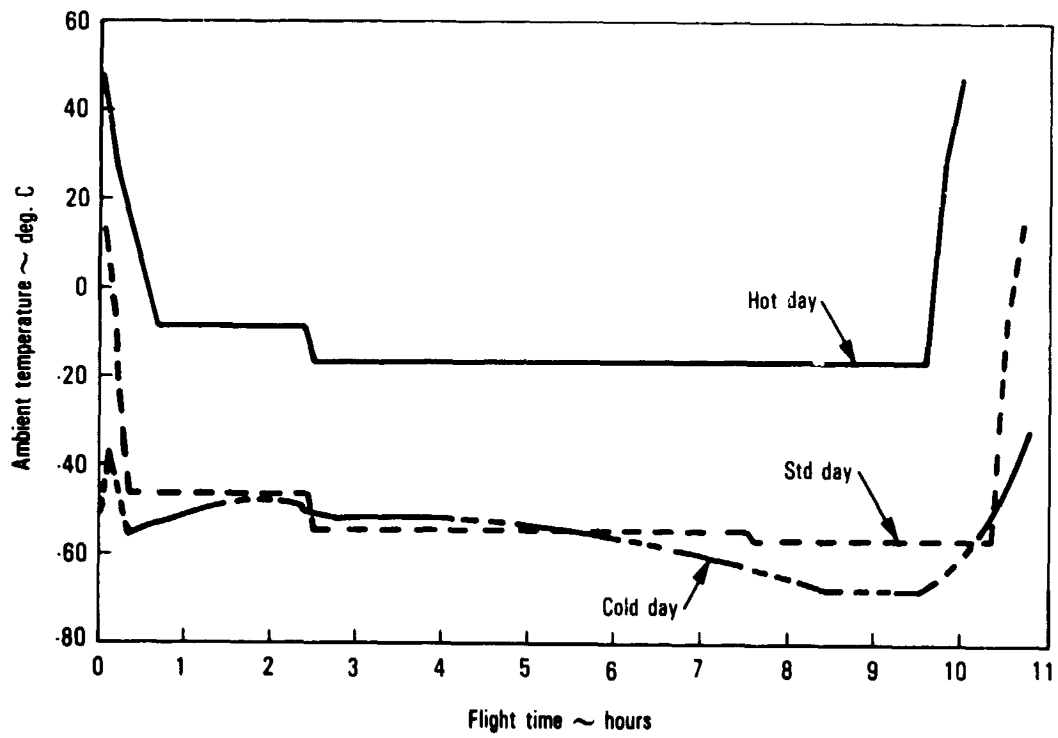


Figure 11 - Ambient temperature profile, 9260 km (500 n.mi.) mission.

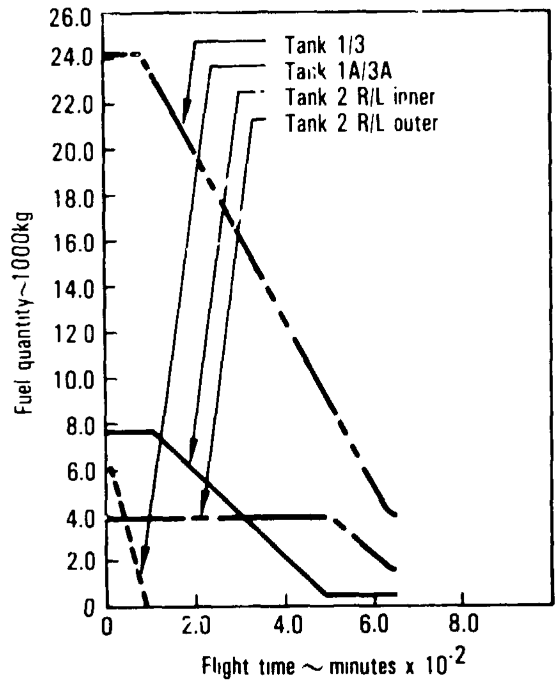


Figure 12 - Cold day fuel quantity, 9260 km (5000 n.mi.) mission.

destination. However, on the extreme hot day the aircraft climbs to an initial cruise altitude of 10 668 m (35,000 ft) where it remains for a period of time and then to a final cruise altitude of 11 887 m (39,000 ft) where it remains until descending to the destination.

In figure 13 the flight profiles are shown as plots of pressure altitude vs flight time for the extreme cold, standard and hot days. The ambient temperatures associated with these altitude-time profiles are presented in figure 14. The mission fuel tank quantities are shown in figure 15 for the cold day only since the standard and hot day fuel tank quantities are not critical in this study.

3.3.3 926 km (500 n.mi.) mission. - In the short range mission, less than two hours, a flight profile containing only one cruise segment is used. The aircraft climbs to the cruise altitude, cruises, and descends to the destination. The same cruise altitude of 11 887 m (39,000 ft) was used for all three temperature profiles. In figure 16 the flight profiles are shown as plots of pressure altitude vs flight time for the extreme cold, standard and hot days. The ambient temperatures associated with these altitude-time profiles are presented in figure 17 and the cold day mission fuel tank quantities in figure 18.

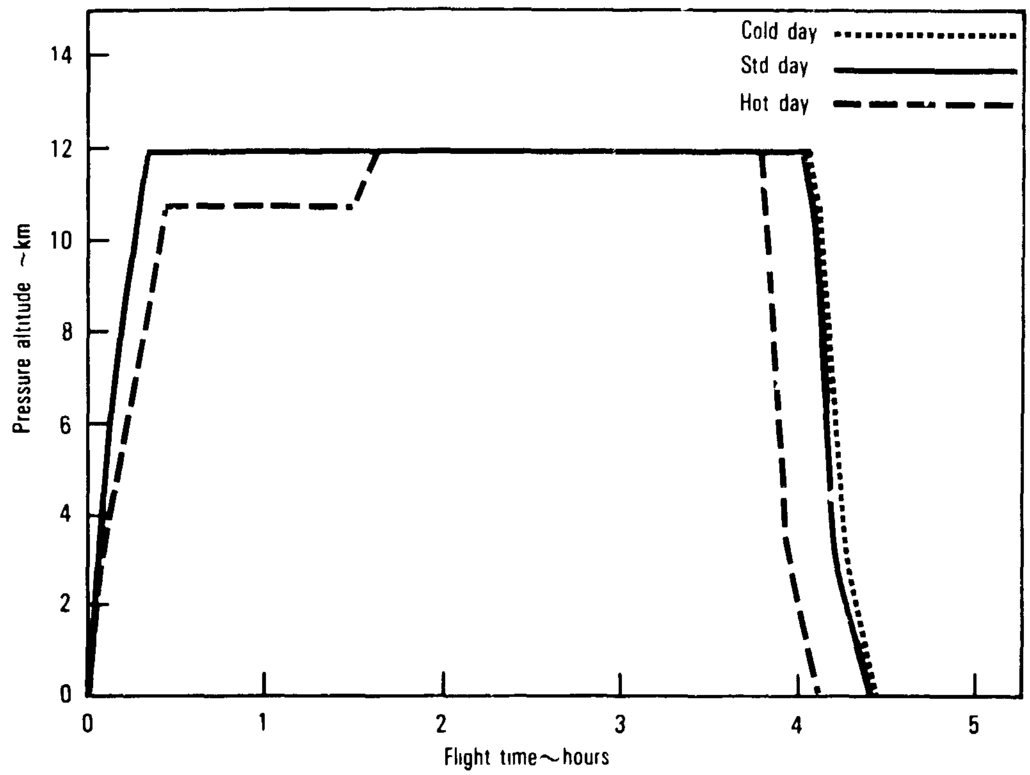


Figure 13 - Flight profiles, 3704 km (2000 n.mi.) mission.

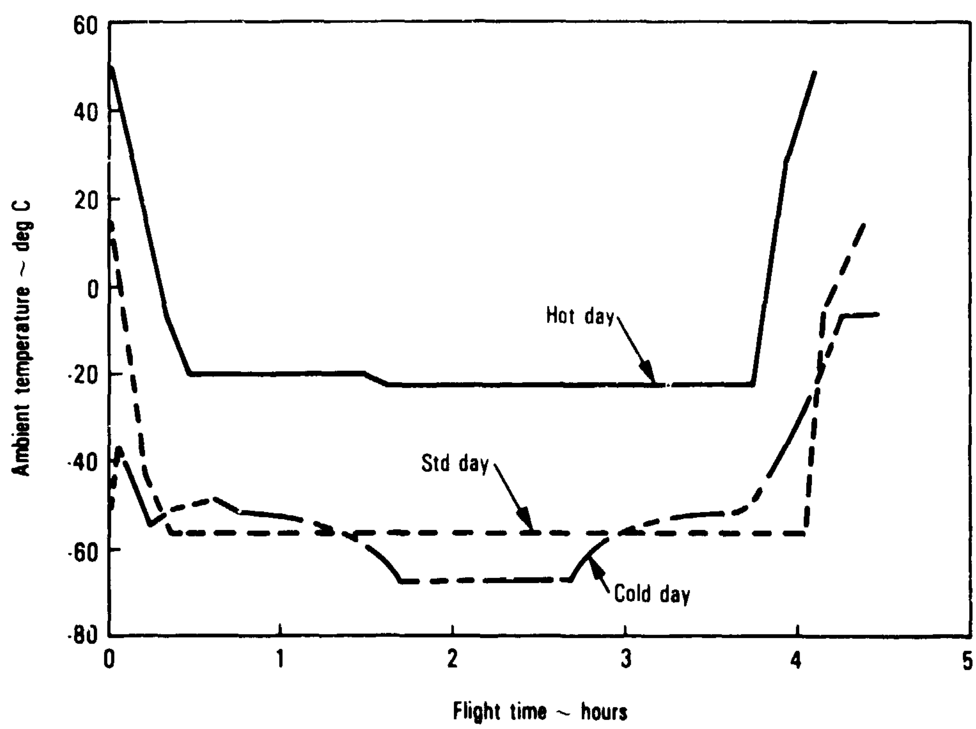


Figure 14 - Ambient temperature profiles, 3704 km (2000 n.mi.) mission.

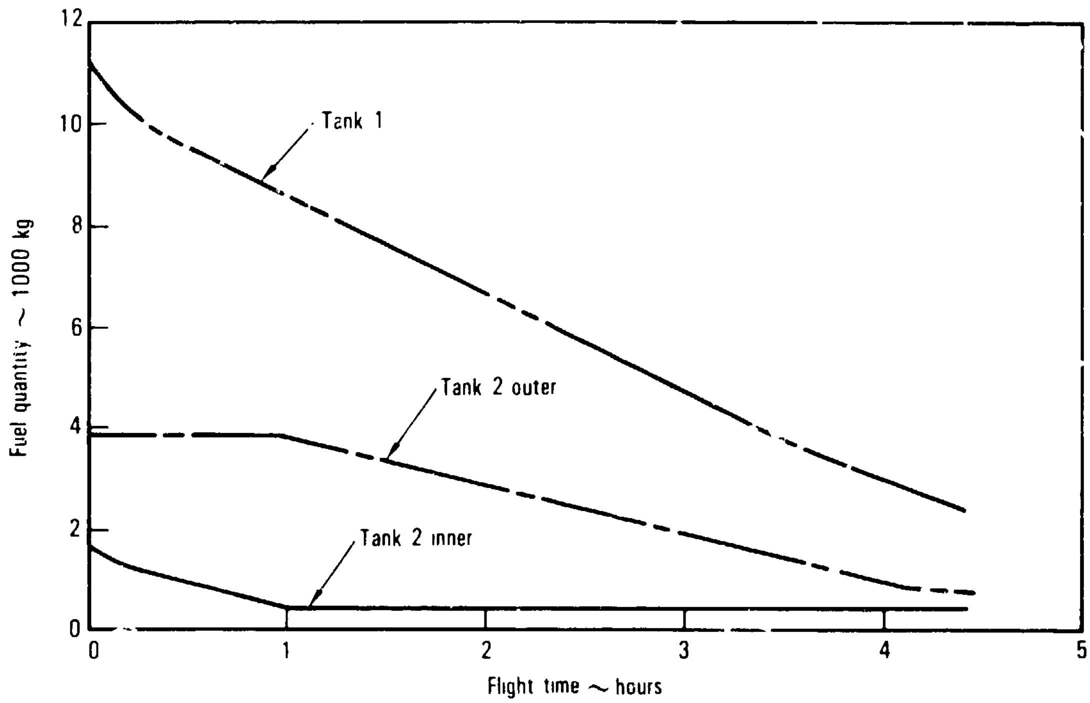


Figure 15 - Cold day fuel quantity, 3704 km (2000 n.mi.) mission.

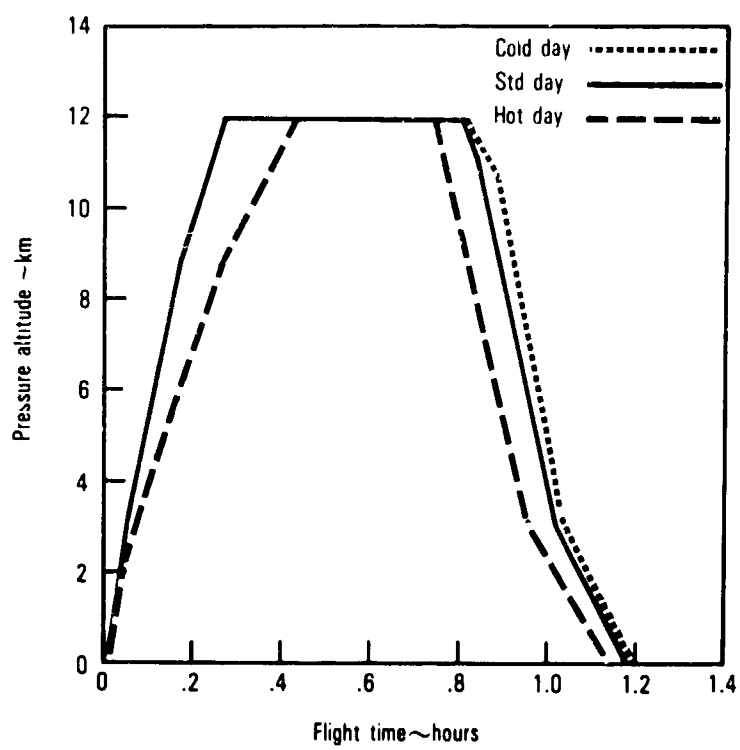


Figure 16 - Flight profiles, 926 km (500 n.mi.) mission.

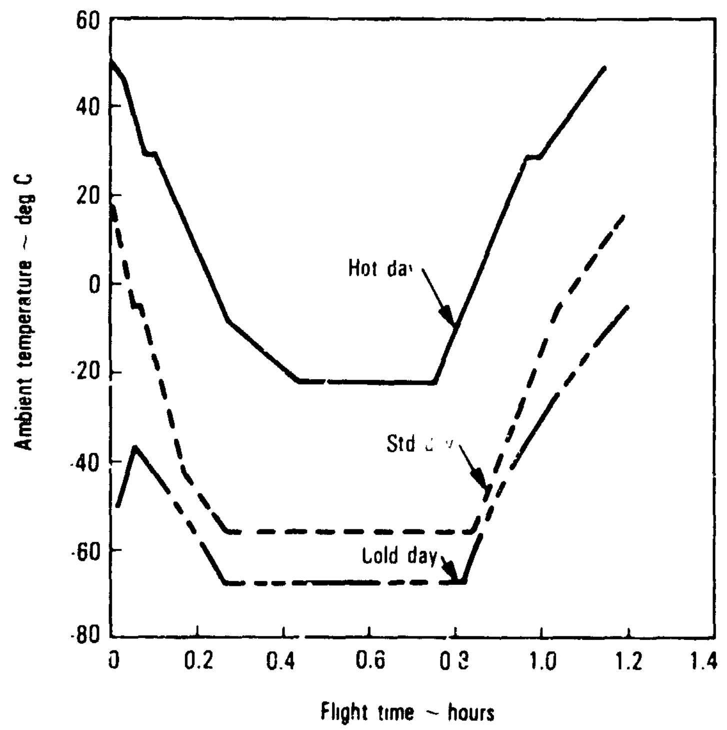


Figure 17 - Ambient temperature profiles, 926 km (500 n.mi.) mission.

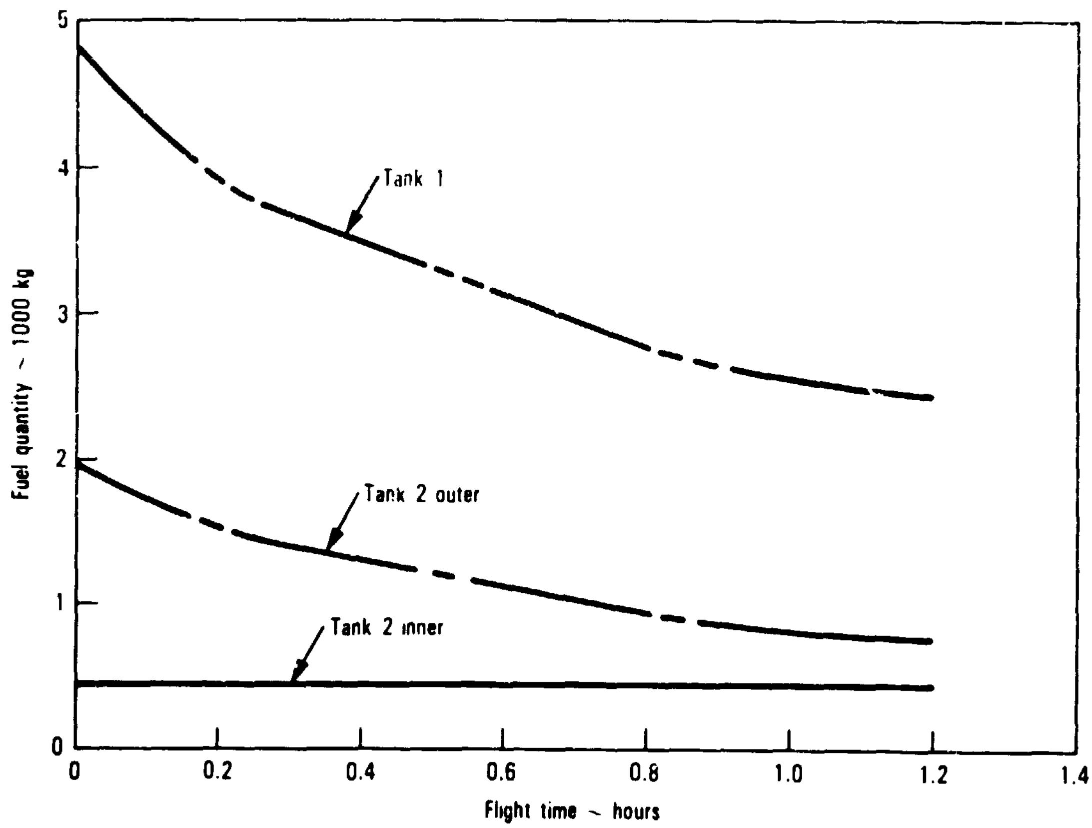


Figure 18 - Cold day fuel quantity, 926 km (500 n.mi.) mission.

4. FUEL PROPERTIES

4.1 ASTM Specification

Current commercial aircraft use jet fuels whose properties are within an envelope of limiting values. These limits, currently established by ASTM specification D 1655-81 for Jet A, Jet A-1, and Jet B, are shown in table 2.

The airframe and engine manufacturers have also established materials selection and component design criteria, which assure a highly reliable aircraft able to operate within reasonable economic margins, as long as the jet fuel properties are kept within specifications. When significant problems arise in fuel supply or utilization, the related specification limits are carefully examined in order to decide whether they should be relaxed (supply problem) or tightened (utilization problem) for coping with temporary or permanent situations. For example, in the past years there has been a trend to increase the aromatics content beyond the normally allowed maximum limit of 20 percent in volume. As can be seen in table 2, aromatics content is presently permitted up to 25 percent, provided the supplier notifies the purchaser within 90 days, or if other reporting conditions mutually agreeable to both parties are made. The allowance was only temporary and was subject to further approval by 1982. If the experience accumulated during the property relaxation period demonstrates that it is safe and reasonable to operate under such conditions, this relaxation can become permanent.

In the last ten years, however, there have been clear signals that during periods of fuel shortage, the availability of jet fuel may be jeopardized because of competition from other important sectors of the fuel market, such as diesel fuels and heating oils. Studies have been conducted which show that in the future, jet fuels will most probably have to be produced from heavier petroleum fractions with increasing participation of shale oil and coal syncrude blends. The back end of these distillates would necessitate additional processing in the refinery in order to meet present specifications. This would undoubtedly translate into higher fuel costs, with the corresponding impact on direct operating costs.

An alternate approach under study by NASA and DoD agencies proposes a re-examination of the present specifications in view of the advancements which have been introduced in the last 30 years in airframe and engine technology, coupled with a deeper understanding of the behavior of fuels in fuel systems and engines. An assessment on how far the specification limits could be relaxed, while still operating the aircraft within safe and economic limits, could result in a greater availability and significant energy and cost savings.

4.2 Candidate Fuel Property Changes

Since the Arab oil embargo the cost of jet fuels has almost quadrupled and under special circumstances, such as severe winter weather, fuel procurement actions have been hampered. These developments have motivated the oil industry to turn their interest to non-petroleum resources for the production of fuels for jet aircraft and other uses. In the military, these concerns were raised early in the middle 1970's when a series of feasibility studies were conducted to assess the potential contribution of oil shale, coal, and tar sands as raw materials for augmenting the supplies

TABLE 2 - DETAILED REQUIREMENTS OF AVIATION TURBINE FUELS D 1655-81 SPECIFICATIONS

Property	Jet A or Jet A 1	Jet B	ASTM Test Method[B]
Acidity, total max mg KOH g	0 1		D974 or D3242
Aromatics vol, max %	20[C]	20[C]	D1319
Sulfur mercaptan,[D] wt max %	0 003	0 003	D3227
Sulfur total wt max %	0 3	0 3	D1266 or D1552 or D2622
Distillation temperature °C (°F)			
10% recovered max temp	204 4 (400)		D86
20% recovered max temp		143 3 (290)	
50% recovered max temp	report	287 8 (370)	
90% recovered max temp	report	243 3 (470)	
Final boiling point max °C (°F)	300 (572)		
Distillation residue max %	1 5	1 5	
Distillation loss max %	1 5	1 5	
Flash point min °C (°F)	(37 8) 100		D56 or D3243[L]
Gravity max °API (min sp gr) at 15 6°C	51 (0 7753)	57 (0 7507)	D1298
Gravity min °API (max sp gr) at 15 6°C	37 (0 8398)	45 (0 8017)	D1298
Vapor pressure max kg (lb)		1 36 (3)	D323
Freezing point, max °C	40 Jet A[E] 47 Jet A 1[E F]	50[E]	D2386
Viscosity (-20°C) -4°F max cSt	8		D445
Net heat of combustion min kJ kg (Btu lb)	42 795 (18 400[H])	42 795 (18 400[H])	D1405 or D2382
Combustion properties one of the following requirements shall be met			
(1) Luminometer number min or	45	45	D1740
(2) Smoke point min or	25	25	D1322
(3) Smoke point min and	20[G]	20[G]	D1322
Naphthalenes vol max %	3	3	D1840
Corrosion copper strip 2 h at 100°C (212°F) max max	No 1	No 1	D130
Thermal stability one of the following requirements shall be met			
(1) Filter pressure drop max mm Hg	76 2	76 2	D1660[I]
Preheater deposit less than	Code 3	Code 3	
(2) Filter pressure drop max mm Hg	25	25	D3241[J]
Tube deposit less than	Code 3	Code 3	
Existent gum, max mg 100 ml	7	7	D381
Water reaction			
Separation rating max	(2)	(2)	D1094
Interface rating max	1b	1b	D1094
Additives	See 4 2		
Electrical conductivity pS/m	K	K	D2624 or D3114

[A] The requirements herein are absolute and are not subject to correction for tolerance of the test methods If multiple determinations are made, average results shall be used

[B] The test methods indicated in this table are referred to in Section 9

[C] Fuels with an aromatics content over 20 volume % but not exceeding 25 volume % are permitted provided the supplier (seller) notifies the purchaser of the volume distribution and aromatic content within 90 days of date of shipment unless other reporting conditions are agreed to by both parties This footnote is subject to reapproval in 1982

[D] The mercaptan sulfur determination may be waived if the fuel is considered sweet by the doctor test described in 4 2 of Specification D484 for Hydrocarbon Drycleaning Solvents[3]

[E] Other freezing points may be agreed upon between supplier and purchaser

[F] The −47°C maximum freezing point limit for Jet A 1 is subject to reapproval in 1983 If not reapproved, the value will revert to −50°C maximum

[G] Fuels having a smoke point less than 20 but not less than 18 and a maximum of 3 volume % of naphthalenes are permitted provided the supplier (seller) notifies the purchaser of the volume, distribution and smoke point and naphthalenes content within 90 days of date of shipment unless other reporting conditions are agread to by both parties This footnote is subject to reapproval in 1982

[H] Use for Jets A and A 1 the value calculated from Table 3 or Eqs 5 and 9 in Method D1405 Use for Jet B the value calculated from Table 6 or Eqs 5 and 7 in Method D1405 Method D2382 may be used as an alternative In case of dispute Method D2382 must be used

[I] Thermal stability test shall be conducted for 5 h at 148 8°C (300°F) preheater temperature 204 4°C (400°F) filter temperature, and at a flow rate of 2 7 kg/h (6 lb/h)

[J] Thermal stability test (JFTOT) shall be conducted for 2 5 h at a control temperature of 260°C but if the requirements of Table I are not met, the test may be conducted for 2 5 h at a control temperature of 245°C Results at both test temperatures shall be reported in this case Tube deposits shall always be reported by the Visual Method, a rating by the Tube Deposit Rating (TDR) optical density method is desirable but not mandatory

[K] A limit of 50 to 450 conductivity units (pS/m) applies only when an electrical conductivity additive is used and under the condition at point of use

$$1\ pS/m = 1 \times 10^{-12}\ \Omega^{-1} m^{-1}$$

[L] Results obtained by Method D3243 may be up to 1 7°C (3°F) below those obtained by Method D56 which is the preferred method In case of dispute, Method D56 will apply of aviation turbine fuels (see references 4, 5, and 6). It soon became evident from those early studies that:

1. It was possible, in principle, to manufacture jet fuels from syncrude which can meet current specifications.

2. The additional refinery processing that was required to bring the synthetic fuels within the specifications would undoubtedly result in higher costs to the consumer and higher process energy consumption at the refinery.

3. Jet fuels from shale oil would be similar to petroleum derived jet fuels, while jet fuels from coal syncrude would be either heavy in aromatics or, after hydroprocessing, highly dominated by napthenic based components.

4. Considerations related to capital investment, risk minimization and other competing sectors from industry indicate that, in the short term, jet fuels will still be produced from petroleum sources, although with increasing participation of the heavier distillate fractions.

5. Before the year 2000, some jet fuels will probably be partially manufactured from shale oil, although the percentage penetration in the jet fuel market is uncertain and will undoubtedly be dependent on the energy supply/demand scenarios that result from growth or no-growth economic trends.

6. Before the turn of the century, it is expected that some fuels will be derived from coal. It is believed that given the higher tolerance to napthenic based fuels by the gasoline, diesel, and heating oil consumers, the contribution of coal will be more noticeable in those sectors, leaving the petroleum based crudes freed for the manufacture of aviation turbine fuels (see reference 7).

In summary, in the short term, current trends indicate that future jet fuels will be manufactured with medium distillates obtained from heavier petroleum fractions by means of cracking and hydroprocessing. In the intermediate future, before the turn of the century, shale oil derived jet fuels could become an important fraction of the total aviation turbine consumption; and in the long term (50 years from now), if jet aircraft are still powered with hydrocarbon fuels (there are strong indications that liquid hydrogen and liquid methane could become major fuels for large aircraft after the turn of the century), most of them will be derived from coal and consequently will be of a napthenic and aromatic base.

The National Aeronautics and Space Administration, and the Department of Defense, have active programs in jet aircraft hydrocarbon fuel technology aimed at investigating the required technological developments which may be necessary in the future (references 7 and 8). These programs comprise studies and experimental testing. In order to provide a common reference base for the program activities, an Experimental Referee Broad-Specification (ERBS) fuel was recommended (see reference 9). Quoting from this document, the ERBS fuel was suggested to be used as a base case fuel in the programs aimed at developing new engines and fuel systems.

The proposed specifications are given in table 3. In this study, the ERBS fuel was intentionally selected to provide a feedback and ascertain as to whether the tentative limits originally recommended for the ERBS were or were not adequately established (at that time, very few results were available from the airframe industry to provide a sufficiently sound basis on which to recommend the working ERBS limits).

4.2.1 Freeze point. - The freeze point is presently specified in ASTM D 1655-81 at -40°C (-40°F) for Jet A fuel, and is defined as the temperature at which suspended waxes in the fuel disappear while on a warm-up cycle after a previous chilling cycle. Visual detection of these waxes is subjective and the reproducibility has been established to be within 2.6°C (4.7°F) (see reference 10). It is important to notice that the volume percentage of these waxes could be extremely small. The materials present in the precipitate belong usually to the highest boiling point paraffinic chains that make their way into the fuel during the distillation or blending process. It should be expected that under equal circumstances a jet fuel with a napthenic base will have a freezing point somewhat lower than one with a paraffinic base. Coal derived fuels still have a substantial portion of heavy paraffins and the dependence of the freezing point on the end boiling point and the crude base may be even weaker than the dependence on the nature of the refinery process employed to manufacture the fuel.

Reference 11 reports on the examination of the distribution of published inspection data of Jet A fuel in an eleven year period, from 1969 to 1979. It was found that a distribution of freezing points in the 676 samples presented a component centered very near the present specification at -40°C (-40°F). In fact, it was found that the freezing point was one of the controlling near-specification properties. A relaxation to a higher limiting value would obviously have a positive effect on the availability of Jet A fuel. Furthermore, laboratory low temperature experiments conducted by Lockheed for NASA (reference 12) in an L-1011 simulated tank have shown that there is a one percent probability that present aircraft may encounter extreme temperature conditions which could result in a freeze-out of 1.2 percent of unusable fuel. Although this quantity is lower than the reserve fuel and would melt during descent, it is evident that the freezing point specification imposes a constraint on the fuel system. The freezing point is then an important property to be considered when designing advanced fuel systems for broadened property fuels. A limiting value of -20°C (-4°F) was selected for this study, which is 3.3°C (6°F) higher than the ERBS recommended limit to reflect the type of freezing point that a diesel fuel could exhibit if manufactured from a blend of napthenic and paraffinic crudes.

4.2.2 Thermal stability. - In the most general sense, fuel stability refers to the ability of a fuel to resist chemical changes due to a shift in environmental variables (temperature or pressure), exposure to foreign matter (materials compatibility), or long term effects (storage stability). Thermal instability makes its presence noticed by the appearance of carbonaceous deposits, film lacquers and varnishes on those components of the fuel system whose surfaces have come in contact with the fuel while being exposed to high temperatures. Storage instability is normally associated with the fuel aging processes and manifests itself by the formation of sediment gums in the fuel tanks. In subsonic flight the components of the fuel system which are vulnerable to thermal instabilities are those where the fuel is subjected to moderately high temperatures for long periods of time, or has come in contact with hot spots for short dwelling times. However, the chemical mechanisms responsible for thermal and storage instabilities are not too different, i.e., they may be different manifestations of the same phenomena at different temperatures and residence times.

TABLE 3 - PROPOSED SPECIFICATIONS FOR EXPERIMENTAL REFEREE BROAD-SPECIFICATION (ERBS) AVIATION TURBINE FUEL

Specifications	ERBS Jet Fuel Value	Proposed Test Method
Composition.		
Hydrogen, wt %	12.8±0 2	NMR
Aromatics, vol %	Report	ASTM D1319
Sulfur, mercaptan, wt %	0.003, max.	ASTM D1219
Sulfur, total, wt %	0.3, max.	ASTM D1266
Nitrogen, total, wt %	Report	Kjeldahl
Naphthalenes, vol %	Report	ASTM D1840
Hydrocarbon compositional analysis	Report	GCMS
Volatility:		
Distillation temperature, °C (°F)		ASTM D2892
Initial boiling point	Report	ASTM D2892
10 Percent	204.4 (400), max.	ASTM D2892
50 Percent	Report	ASTM D2892
90 Percent	260 (500), min.	ASTM D2892
Final boiling point	Report	ASTM D2892
Residue, percent	Report	ASTM D2892
Loss, percent	Report	ASTM D2892
Flashpoint, °C (°F)	37.8 (100), min.	ASTM D56
Gravity, API (15 6°C)	Report	ASTM D287
Gravity, specific (15.6/15.6°C)	Report	ASTM D1298
Fluidity:		
Freezing point, °C (°F)	-23.3 (-10), max.	ASTM D2386
Viscosity, at -23.3°C (-10°F), cS	12, max.	ASTM D445
Combustion:		
Net heat of combustion, kJ/kg (Btu/lb)	Report	ASTM D2382
Thermal stability:		
JFTOT, breakpoint temperature, °C (°F) (TDR, 13; and Δ P, 25mm)	237.8 (460) min.	ASTM D3241

The ASTM D 1655-81 specification for thermal stability offers two alternatives to comply with the requirements: a) the ASTM CRC Fuel Coker described in the specification D 1660, or b) the Jet Fuel Thermal Oxidation Tester (JFTOT) described in specification D 3241. The JFTOT test consists of pumping the fuel sample through a polished aluminum heater tube, and then through a 17 μm (17-micron) filter for 2.5 hours. The temperature at which the filter plugs (a pressure drop of 3.3 kPa (25 mm Hg)), or at which the deposits on the heater tube become darker than a light tan, is known as the breakpoint temperature. The present specifications require a minimum temperature of 260°C (500°F) when operating at 3.45 MPa gauge (500 psig).

The CRC Coker and JFTOT test times are of the order of hours, at temperatures well above those to which a fuel system component is normally exposed, except for perhaps the injection nozzles themselves. These temperature and time ranges have been selected for practical reasons, since otherwise the sample of fuel under test would be too large should the test proceed for periods of several days. The test temperatures are then selected sufficiently high to obtain detectable results.

The formation of deposits, varnishes and lacquers at ambient or intermediate temperatures are perhaps better represented by the specification for allowable gums, which ASTM D 1655-81 sets at a maximum of 0.07 kg/m^3 (7 mg/100 mL), described in specification D 381. This specification was developed to characterize storage stability of a fuel, rather than the thermal stability. Present refinery practices must comply with other Jet A fuel specifications, such as odor improvement, sulfur reduction, organic acidity, color, gums and aromatics. On achieving these goals, experience has shown (see ref. 13) that the fuel stability specifications are automatically satisfied; and if for some reason an unusual jet fuel happens to be beyond the thermal stability limit, it is sold instead as heating fuel. Thus, in spite of the importance that thermal stability has to aircraft engine and fuel system designers, refineries presently have to worry little, if at all, about meeting the specification.

From the extensive effort conducted during the past twenty years in investigating the cause and nature of thermal and storage instability, it is realized that the phenomena, although very complex, are all related to fuel auto-oxidation (ref. 14). The oxygen is present in the fuel in two forms: a) chemically bound oxygen, and b) dissolved oxygen. The chemically bound oxygen occurs usually in the form of hydroperoxides. These compounds are unstable, but at ambient termperatures they may remain in the fuel for long periods of time unless they are exposed to light and/or to traces of some metals (copper or vanadium have been shown to accelerate auto-oxidation). At the higher temperatures encountered in aircraft fuel/oil heat exchangers, fuel control components, manifolds and injection nozzles, auto-oxidation can proceed at a much faster pace. Whether one is concerned with storage stability or thermal stability, the formation of deposits has always been correlated with the onset of fuel auto-oxidation. These deposits do not necessarily result from the products of auto-oxidation themselves, but rather it is believed that the presence of the free radicals triggered by auto-oxidation are directly responsible for the chain of reactions leading to the formation of sediments and deposits. An examination of these deposits has shown that their hydrogen/carbon ratio is lower than that of the fuel, and that the content of heteroatom compounds, i.e., oxygen, nitrogen, and sulphur, is much higher than the average fuel content of those elements. This suggests that the resonance stabilized free radicals derived from auto-oxidation are more reactive with heteroatom compounds and unsaturated hydrocarbons. Quantitative analyses have also shown that the dissolved oxygen participates very heavily in the advanced stages of the oxidation process (see ref. 15). In brief, the formation of sediments and deposits may proceed along the following lines. After an induction period (nucleation) auto-oxidation commences on a series of active centers distributed on the fuel system surface, as well as in the bulk of the fuel. In the case of thermal instability, and taking into account that most of the heat transfer into the fluid occurs by heat conduction through the surfaces, it is improbable that the active centers in the bulk of the fuel will significantly contribute to the deposits; the phenomenon is clearly of a boundary layer type. The traces produced by this first stage of auto-oxidation are still soluble in the fuel, and may diffuse throughout if the residence time is sufficiently long; they may remain in a thin diffusion boundary layer near the wall if the flow velocity is sufficiently high. Further oxidation, in which even the dissolved oxygen may participate, results in products which are no longer soluble and appear in a solid phase as a sediment.

Current engine trends show that in the future, higher efficiency, higher power density and lower emission requirements are going to come in conflict with fuel thermal stability, even if present fuel specifications are not relaxed. This is primarily due to three factors: a) the trend towards higher engine pressure ratios and consequently higher air temperatures flowing through or about the injection nozzles; b) tighter tolerances in the spray pattern which make the nozzles more vulnerable to incipient fouling; and c) the engine manufacturer's desire to take advantage of the fuel flow to cool the engine oil, and thus redirect the oil heat back into the engine in an attempt to improve the specific fuel consumption. This latter approach results in fuel temperatures, at the high pressure shutoff valve, manifold, and distribution valves, which are sufficiently high to experience (after continuous operation in tropical climates) unusually high varnish and lacquer formation rates on those component surfaces. If in addition to engine design trends, the jet fuel specifications are broadened, it is certain that the thermal stability could become a most limiting property.

One of the major drawbacks of the CRC Coker and JFTOT is that the behavior of these jet fuels in the fuel system and engine is not represented by laboratory tests. There is underway a considerable amount of work in this area, supported by NASA, the Air Force and the Navy, with the general objective of closing the gap between the laboratory results and the fuel system design criteria. These efforts (ref. 16) consist of modified laboratory tests under boundary conditions which approximate those used in the fuel system. Other efforts (ref. 17) include fuel systems simulators provided with sophisticated instrumentation rigs able to capture deposits for further laboratory analysis.

For the purposes of this study, the fuel stability will be represented by the JFTOT. A review of the literature, as well as some of the correlations attempted for a variety of petroleum, shale oil and coal-derived fuels, has not shown a definite range where the JFTOT breakpoint temperature would fall. This is mostly due to the lack of current experience in present fuels. Assuming that future jet fuel will have a strong naphthenic base (such as the Alaskan North Slope crude), and that an extreme situation could be incurred in which a diesel type fuel could be considered for utilization as a jet fuel, a JFTOT breakpoint of 204°C (400°F) was jointly agreed upon by NASA and Lockheed for consideration in this study. It is understood, however, that this is a severe relaxation from the present 260°C (500°F)JFTOT specification and it is much below the 237°C (458.6°F) recommended for the ERBS specification.

4.2.3 Aromatics. - ASTM D 1655-81 sets a maximum limit of 20 percent by volume for aromatics. Up to 25 percent is permitted, provided that the supplier notifies the purchaser of the volume, distribution and aromatic content within 90 days of shipment. An increase in aromatics in a fuel increases the radiant heat transfer from the flames in the combustor to the liner and reduces combustor life. A study of the impact on the combustor when relaxing the specification limits is beyond the scope of this effort, and the primary interest here in connection with aromatics is their deleterious effect on certain elastomers and other nonmetallic materials which are used in seals, membranes and filters throughout the fuel system.

It was found in ref. 11 that the aromatics content is a dominant controlling property, and its relaxation is expected to affect refinery output. The ERBS does not recommend any specific value of aromatics but requires that it be determined by test and reported by the refinery. There are reasons to believe that in the near future, and more so towards the turn of the century, the aromatics content in fuels will increase considerably. This projection is based on the fact that the present trend in refineries towards the processing of heavier distillate fractions makes use of catalytic, thermal, and hydrocracking, and results in the formation of a high content of aromatics, which are then brought down to specification limits by hydro-processing. This is the reason why some refineries are already having difficulty in supplying jet fuels with aromatics content below 20 percent. In the future, however, if shale oil and coal become important sources for jet fuels, hydrocracking will be practiced in a much larger scale, resulting in substantially higher aromatics contents. It is obvious therefore, that a modification in aircraft fuel systems, allowing for the utilization of high aromatic fuels, will have a most significant impact on fuel cost, energy savings and fuel availability. For this study a 35 percent maximum content in aromatics was selected for the purpose of studying the fuel system materials compatibility. This figure appears to be representative of refinery outputs that may be experienced before the turn of the century.

4.2.4 Viscosity. - Kinematic viscosity is related to the ability of the fuel system to pump the fuel and deliver it to the engine. The kinematic viscosity for Jet A fuels is defined by the D 1655 specification to be not more than 8 mm^2/s (8 cSt) at -20°C (-4°F). The ERBS recommends its new limit at 12 mm^2/s (12 cSt) at -23.3°C (-10°F). The present specification freezing point of -40°C (-40°F), translates into a 20°C (36°F) margin above the freeze point for measuring the viscosity. In the ERBS the viscosity measurement temperature equals the freezing point. Fuels may still be chilled below the freezing point before their flowability is severely impeded (pour point). The difference between the freezing point and pour point varies from less than 1°C (1.8°F) up to 10°C (18°F) depending on the type of fuel. It is felt here that better reproducibility in measuring the viscosity can be accomplished if the freezing point is well below the measurement temperature for viscosity. For example, the viscosity could be specified as 9 mm^2/s (9 cSt) at -8°C (17.6°F).

For the present study, however, it was proposed that the viscosity be relaxed even further in order to explore the ability of the fuel system to pump fuels that are heavy in aromatics and napthenics with distillation fractions as heavy as diesel fuel No. 2. A limiting value of 15 mm^2/s (15 cSt) at -17.8°C (0°F) was therefore selected, which ensures that the fuel is in a liquid state when its viscosity is measured.

4.2.5 Lubricity. - This property is the ability of a fuel to minimize friction and wear between moving adjacent surfaces in fuel system components. When two solid surfaces slide over one another, the friction coefficient between them can be diminished by the introduction of a liquid film. As the two surfaces are brought closer by applying a normal load, the liquid film is squeezed out at a rate which is a function of its viscosity and surface tension. As long as the thickness of this film layer is above 0.5 or 0.6 μm, the surfaces are said to be hydrodynamically lubricated and the friction coefficient is independent of the film lubricity. When the film becomes thin to the point that it reaches less than 0.1 μm (1000 angstroms), the molecular structure of the film (as well as the solid surface texture and atomic structure) become most dominant in determining the friction coefficient. In the extreme event that both surfaces become locally in contact, they heat locally and melt, resulting in progressive wear. This regime is known as solid friction or partial fluid friction. The thickness of the film layer depends on the normal load, the relative speed between the surfaces, the degree of surface finishing, and the concentration of polar molecules in the liquid.

Lubricity as a jet fuel property is not covered by present specifications, but in those cases where the jet fuels are produced by intense hydrotreating, or whose contaminants have been separated using clay filters, the lubricity has deteriorated to the point of causing isolated but important cases of accelerated wear in the fuel pumps, particularly those using sliding piston-cylinders. Hydrotreating appears to reduce the number of functional groups in the polar molecules or their length, thus reducing the lubricity of the fuel.

In those cases where poor fuel pump lubricity has been detected, it was found that a corrosion inhibitor additive would resolve the situation. The problem would also disappear when the steel cylinder sleeve was replaced by a highly carbon-treated steel. Since unusual amounts of corrosion inhibitors are incompatible with free water removal methods, it is more desirable to change the materials of the fuel pump than to employ additives.

It is suggested in ref. 18 to utilize the Wear Scar Diameter (WSD) as obtained in the Ball-On-Cylinder Machine (BOCM) as a method and specification for ranking jet fuels according to their lubricity. The ability of this method to correlate fuel lubricity with fuel pump life will have to be determined by testing, but the method seems to be able to distinguish between fuels provided with different lubricity additives.

It is consistent to assume that jet fuels with a higher end boiling point, from a broad spectrum of crudes, will have a tendency to exhibit better lubricity. It is also conceivable, however, that those occurrences encountered in Europe, where lubricity was affected adversely because of intense hydrotreating (see ref. 19), could become a trend in the near future, in which case a lubricity standard may be necessary. As suggested in ref. 18, a WSD of 0.45 mm could be representative of a marginal fuel lubricity, requiring special component design considerations.

4.2.6 Water separation. - Before a fuel is delivered by the supplier, the water is separated from the fuel by decantation and solvent extraction. In theory, water is not soluble in hydrocarbons, but the presence of the functional groups and peroxides retain water molecules dissolved in the fuel. The higher the molecular weight of the fuel, as well as the higher the number of branches and naphthenic and aromatic compounds, the larger will be the traces of polar molecules in the fuel, and consequently, the higher water solubility. The capacity of a jet fuel to dissolve water by means of such surfactants is detrimental to the fuel system, as an increase in the number of ice particles is expected in the fuel at low temperatures, which could in this manner affect filter matrices and produce wear in the pumps. The requirements on the concentration of surfactants for water separation and lubricity are in conflict and need to be compromised.

4.2.7 Electrical conductivity. - From the point of view of aircraft safety, a fuel with adequate electrical conductivity will prevent electrostatic discharges in the fuel tank (localized sparking). The triboelectric charge separation, occurring at the interface of a fuel and a solid surface, becomes evident when the fuel is in relative movement with respect to the dielectric solid surface. Electrostatic discharges in the fuel can cause a fire hazard. Aircraft jet fuels sometimes use antistatic additives to improve the fuel conductivity and eliminate the fire hazard. A limit of 50-450 picosiemans per meter (pS/m) applies when an electrical conductivity additive is used. Relaxing the end boiling point of a jet fuel will result in higher water solubility and therefore in a higher initial electic conductivity. Relaxing the front end of the distillation curve has, however, the opposite effect. The amount of additive required in different cases is variable and impacts the cost of fuel delivered to the aircraft.

4.2.8 Flash point and vapor pressure. - The flash point is the temperature at which a fuel exposed to air will form an ignitable mixture. This temperature does not guarantee sustained combustion of the fuel, but only an initial flash after ignition. This property is important because it is related to aircraft safety and to engine re-light at altitude. Because of its relation to the volatile fraction of the fuel (initial boiling point), it is anticipated that the flash point will be no lower than the present D 1655 specification limiting value of 37.8°C (100°F). The flash point is determined primarily by the front end of the distillation curve. This fraction is presently at a premium in the fuel market for the manufacture of gasoline. According to the refinery industry in the future there will be a decrease in gasoline demand, and this could release large amounts of the volatile fractions. As to whether such an assumption is reasonable or not the concern is reflected in the present study by extending the range of availability of jet fuel on both ends of the distillation curve. To this end, a flashpoint of 27°C (80.6°F), which is well above the flash point of Jet B fuel, was agreed upon with NASA for this study.

The vapor pressure is closely related to the flash point in the sense that at a given temperature the partial pressure of vapor fuel, in coexistence with the liquid phase, is determined by the distillation curve (primarily by the front end). In jet fuels this property is given in terms of the Reid vapor pressure as measured by specification D 323. The present specification for Jet A fuel has substituted the flash point for the Reid vapor pressure limit. Jet B and JP-4 fuels use instead the vapor pressure to specify the volatiles. This property is related to the tendency of the fuel to form combustible vapors and lose volatiles through fuel air vents at high altitude. ASTM D 1655 specifies a limit of 13.8 to 20.7 kPa absolute (2 to 3 psia) for Jet B. In accordance with the flash point value of 27°C (80.6°F) selected above, the vapor pressure considered in this study will be assumed to be less than 13.8 kPa absolute (2 psia).

4.3 Summary of Fuel Properties Selected for Fuel System Analysis

The property limiting values discussed in the previous paragraphs and selected for use in this study are summarized in table 4. These properties do not reflect a real fuel, but bracket the range of variation which would be spanned when considering relaxation of jet fuel properties in an effort to increase its availability, lower costs and provide energy savings.

TABLE 4. FUEL PROPERTIES SELECTED FOR FUEL SYSTEMS ANALYSIS

Freezing Point, °C (°F)	- 20 (-4)
Thermal Stability, JFTOT, °C (°F)	204 (400)
Aromatic Contents, % vol.	35
Viscosity, mm^2/s (cSt) at -17 8°C (0°F)	15 (15)
Reid Vapor Pressure, kPa absolute (psia)	13.8 (2)
Flash Point, °C (°F)	27 (80.6)
Lubricity, WSD, mm	0.45

5. ANALYSIS OF IMPACT OF PROPERTY CHANGES ON BASELINE AIRCRAFT

The following sections discuss the analysis conducted to determine the impact each of the previously defined fuel property changes has on the operation of the baseline aircraft.

5.1 Freeze Point

Fuel temperature surveys to determine the maximum freeze point to be allowed for specification fuels have generally been applied to bulk fuels. Below the freeze point temperature, there is the possibility that the free flow of the fuel will be impaired causing loss of fuel availability from the fuel tanks, excessive pressure drop in fuel lines, and possible malfunction of fuel metering controls where tight clearances between moving surfaces are encountered. However, bulk fuel temperature does not adequately represent the significance of freeze point in the aircraft and engine fuel systems. Of far greater significance is the temperature of the fuel immediately adjacent to surfaces which are in contact with the cold slipstream air and which will be significantly below the bulk fuel temperature. In this section, the characteristics of a thermal model which was developed to generate temperature profiles in the L-1011 fuel tanks, the method of model verification, and the fuel temperature-profiles developed through use of the model are discussed.

5.1.1 Fuel tank thermal model. - To estimate fuel temperatures in the baseline aircraft fuel tanks, a computerized fuel tank thermal model was developed. Separate versions of the model were created for Tank 1/Tank 3, Tank 2-inboard, and Tank 2-outboard. The primary inputs to the model are initial fuel temperature and curves of ambient temperature, ambient pressure, Mach number and fuel quantity, all as functions of time. The model gives as outputs the bulk fuel temperature and the temperature of the fuel tank's bottom surface as a function of time. The temperature of the bottom surface of the fuel tank is of primary importance because this is the first location at which fuel freeze-out is likely to occur.

The fuel tank thermal model developed for the study is essentially one-dimensional, primarily treating heat transfer in the vertical direction. The assumption is made that there is no thermal gradient within the upper and lower tank surfaces nor within the fuel in the horizontal direction. Using the one-dimensional approach, the model gives average temperatures for bulk fuel and for tank structures. the thermal analyses include the effects on cooling rate of various tank internal structural members, variable wetted surface area, radiation from unwetted surfaces, and fuel transfer from other fuel tanks.

In developing the fuel tank thermal model, the most difficult task was the determination of convective heat transfer coefficients both within the fuel and on the external tank/wing surfaces. Empirical relations were used to estimate these coefficients and then in some cases, practical adjustments were made to achieve correlations with flight test measured temperature-time histories. Two important conclusions were reached during this process: (1) the predicted temperature of the fuel in the wing tanks of the L-1011 tends to reach the boundary layer air temperature during a long flight regardless of the internal and external convection coefficients used, and (2), the heat input to the fuel tank required to prevent fuel freezing depends primarily on the external convection coefficients.

The fuel tank thermal model was developed using the Lockheed thermal analyzer program as the basic tool. Using this program, the solution to transient heat transfer problems is effected by converting the three dimensional physical system into an analogous electrical network of lumped thermal capacities (small volumes with essentially uniform temperature) connected by thermal resistors. The resistors may represent heat transfer by convection, radiation or conduction. Transient temperature histories are computed using the lumped-parameter, or finite-difference approach by applying Kirchhoff's law at each lump (node) of the R-C (resistor-capacitor) electrical analog network. By specifying any quantity as a function of any other, it is possible to include the effect of various nonlinear parameters, e.g., variable thermal properties and arbitrary boundary conditions as a function of time and/or temperature.

The development of the fuel tank thermal model can be divided into three parts, modeling of tank structures, modeling of external heat transfer, and modeling of internal heat transfer.

5.1.1.1 Modeling of tank structures: In the integral wing tanks of the L-1011, the structural components which contribute most significantly to heat transfer are the upper and lower wing surfaces, the stringers, the ribs, and the spars. Of these, the most dominant heat transfer contribution is made by the upper and lower wing surfaces. These surfaces have the largest convection areas, the shortest conduction lengths, and provide the most direct heat transfer paths from the fuel to the freestream air. High convection rates exist in the vicinity of these surfaces both on the fuel side and on the freestream air side. The stringers and ribs have a significant effect on heat transfer because they act as fins adding convective area to the wing surfaces inside the tank. For example, in one location of the L-1011 fuel tanks, the stringers have the effect of adding 41 percent to the surface area. The wing spars which form the fore and aft boundaries of the fuel tanks have a less pronounced effect on heat transfer than the structural elements previously discussed. This is due to the low air velocities in the wing cavities adjacent to the spars forward and aft of the tank boundaries, and to the spar's relatively small convective areas. The heat transfer through the spars which does occur is primarily the result of convection from the fuel at the vertical spar surfaces and conduction through the spar structure to the upper and lower wing surfaces.

Figure 19 shows a general thermal analyzer R-C network representing a section of the lower boundary of the fuel tank. In the model, this network is simplified to the equivalent network shown in figure 20. The heat paths through the various structures are computed using average values for the structural dimensions so that this network represents a typical section of the lower boundary rather than a specific location. An identically structured network is used to represent the upper tank boundary with conduction resistors computed using the appropriate average structural dimensions. These boundary networks allow the separate determination of average temperatures in the surface, the stringers, and the ribs. The effect of heat transfer through the spars is not treated directly but is included with that through the upper and lower tank boundaries.

For use in a later section of this report, figure 21 shows the R-C network for the lower boundary, modified to account for the presence of insulation. The R-C networks shown in both figure 20 and 21 can be used to model the addition of electric foil heating. This is accomplished by adding a heat input into the heat balance calculations performed at nodes a, b, and c.

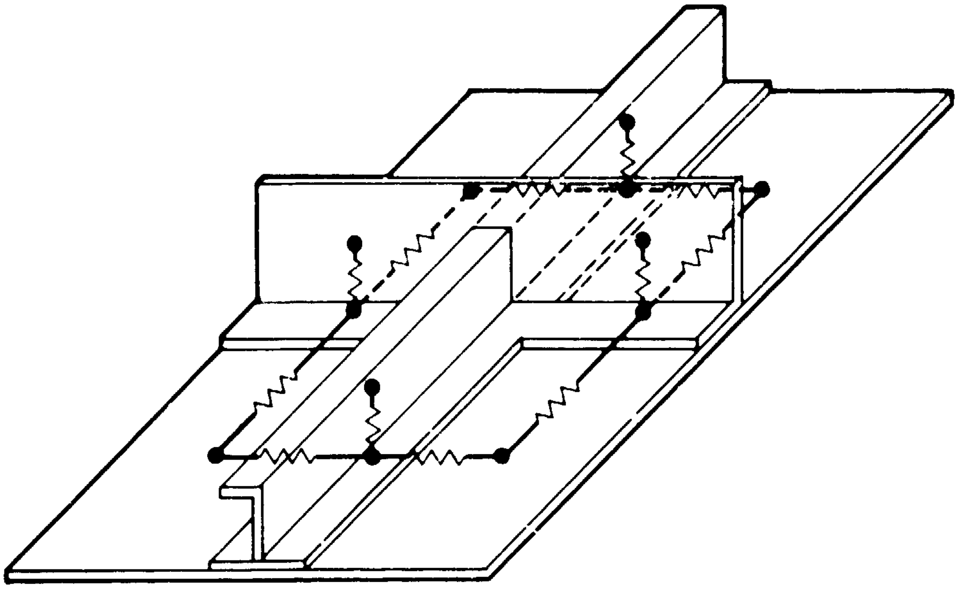


Figure 19 - General thermal analyzer R-C network
for fuel tank lower boundary.

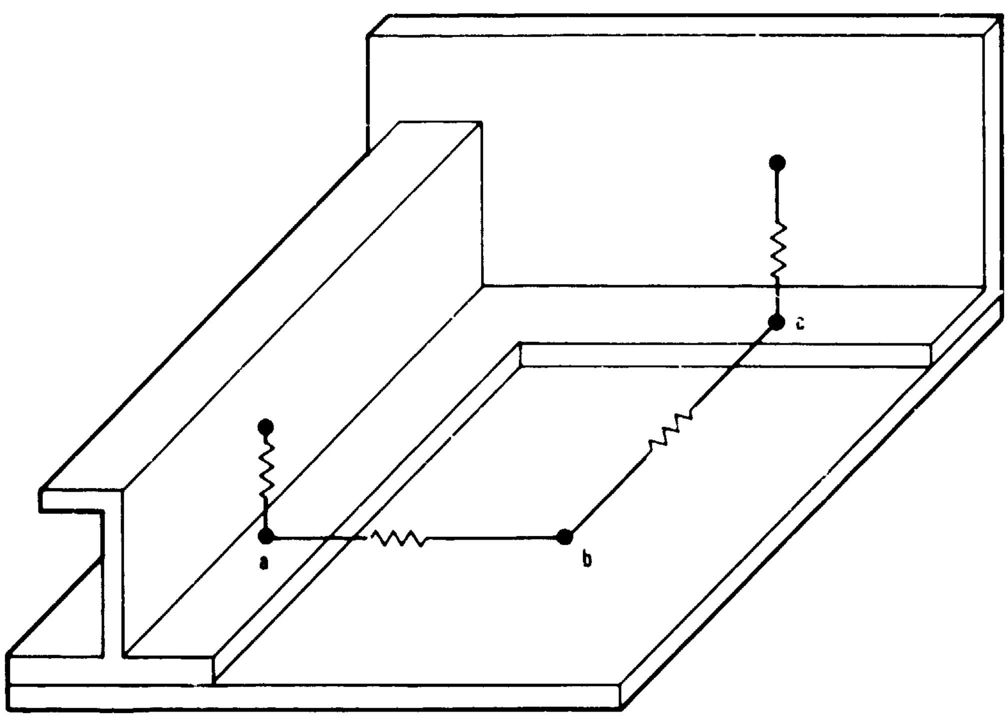


Figure 20 - Simplified schematic of wing fuel tank lower surface model.

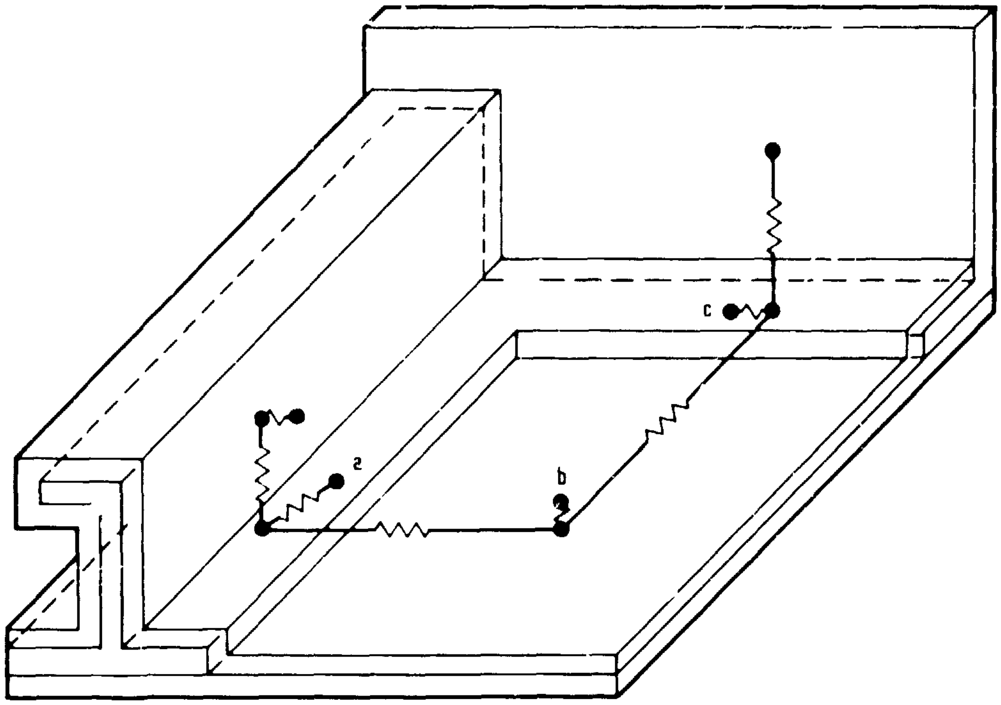


Figure 21 - Simplified schematic of wing fuel tank lower surface with insulation.

5.1.1.2 Modeling of external heat transfer: For the integral wing tanks of the baseline aircraft, the upper and lower surfaces of the wing form the respective boundaries of the fuel tanks. The external heat transfer from the fuel tanks therefore consists of convection from the wing surfaces to the boundary layer air stream and radiation from these surfaces to the surrounding environment.

The convective heat transfer falls into the well documented regime of forced convection. In the model, average convection coefficients were computed separately for the upper and lower surfaces of the wings. These coefficients were computed continuously throughout a simulated flight using flat plate relationships. Because the leading edges of the fuel tanks lie beyond the point of transistion to turbulent flow, turbulent flow relationships were used. Average values of pressure coefficients were estimated for the upper and lower wing surfaces near the center of each fuel tank using flight test data measured under cruise conditions. These pressure coefficients were used in the model with free stream pressure and velocity to compute average values of local pressure and velocity. The local pressure and velocity were used with the air film temperature and the tank's position in the wings to compute the average convection coefficients.

Radiation heat transfer between the wing surfaces and the sky and ground are accounted for in the model as is solar radiation to the upper surface. Compared to the convective heat transfer, the contribution of radiation is relatively minor. The variables affecting radiation are the emissivity and absorptivity of the surface, the ground and sky temperatures, and the upper and lower surface temperatures. An emissivity of 0.18 was used in the model for the wing surfaces. This value corresponds to unpolished and unpainted aluminum. A corresponding absorptivity of 0.50 was used for the upper wing surface in computations of solar heating. A ground temperature of 10°C was used for the validation flight cases in which the actual ground temperature was not recorded. For the contract missions, the ground temperature used is that given by the appropriate mission temperature profile shown in Section 3.2. The sky heat sink temperature is assumed to be -273°C for all cases. Solar heating of the upper wing surface was considered in the validation cases. For the contract missions, however, solar heating is not considered so as to produce the lowest fuel temperatures which would occur at night.

The upper and lower surface temperatures were computed by the model using heat balance calculations which include the previously discussed modes of heat transfer plus heat transfer within the tank.

5.1.1.3 Modeling of internal heat transfer: The model of the interior of the fuel tank considers the variable heat capacity of the fuel and heat transfer from the fuel to the inner tank structure by convection and radiation. In addition, the model considers the effects on fuel temperature of fuel transfer from other fuel tanks.

The entire heat capacity of the fuel is treated as a single thermal unit. This method of analysis requires the assumption that all of the fuel in the tank exists at a single 'bulk fuel temperature' at any given point in time. This assumption is supported by the limited data obtained in L-1011 flight tests (reference 20). The fuel temperature profiles in a vertical section were recorded as functions of time. The results show that the temperature of most of the fuel within the vertical section does fall within a narrow range, although the percentage of the total fuel within that range varies with cooling rate and fuel height. Layers of cooler fuel at the top and bottom of the fuel remain relatively thin most of the time.

An important consideration in predicting the rate of heat transfer to the fuel is the area of the tank surfaces in direct contact with the fuel. Since this wetted surface area varies with the quantity of fuel in the tank, it will decrease as fuel is consumed by the engines during the flight as will the quantity of bulk fuel.

Heat transfer from the fuel to the colder tank structure consists of convection within the fuel and radiation and convection across the ullage space above the fuel. While the model considers all three modes of heat transfer, convection within the fuel is by far the most dominant mode.

The analysis of convection within the fuel is broken down into separat computations for each of the main structural components, the upper and lower surface, the stringers, and the ribs. Only the portions of each of these structures that are actually in contact with fuel at a given time are considered in the computations.

Separate convection coefficients are computed for the upper and lower tank structures. The available relations for horizontal flat plates were used to estimate these coefficients. Empirical adjustments obtained in correlations of temperature predictions with flight test data were then made to the estimated coefficients. Further discussion of these convection coefficients is given in the following section.

5.1.1.4 Correlation of model temperature predictions with flight test data: The validity of the fuel tank thermal model was verified using flight test data obtained under NASA contract (reference 20). Flight tests were conducted in which time dependent vertical temperature profiles were obtained for two L-1011 fuel tanks, Tank 1 and Tank 2R-inboard. For Tank 1, temperature profile data were obtained in cases where fuel was withdrawn from the tank during the flight and in cases where the tank remained full of fuel. For Tank 2R-inboard, applicable data were obtained only for cases in which the tank was maintained approximately 90 percent full of fuel.

The first step in the validation of the fuel tank model was to verify that the temperature predictions showed the proper trends and roughly the proper temperatures. Figures 22 and 23 show that the proper trends were given by the model in correlations of predicted temperature with flight test temperature-time histories. In the figures, the predicted and actual bulk fuel and lower surface temperatures in Tank 1 are shown. Figure 22 gives the results for a flight in which Tank 1 remained full of fuel, figure 23 for a flight in which the tank was emptied.

The second step in the validation of the model was to make adjustments to the estimated convection internal and external coefficients to improve the correlations. The required adjustments were different for Tank 1 and Tank 2R-inboard. Most of the required adjustments were not more than 25 percent of the original values given by the empirical relations, which is generally within the accuracy of these relations. However, two cases emerged in which significant changes were required.

The first case in which a significant modification to the convective heat transfer computation method was required occurred inside Tank 1. While satisfactory correlations of bulk fuel and lower surface temperatures were given by the model for cases in which the tank remained full of fuel, in all cases in which the tank was emptied, the recorded temperature of the lower surface was found to be higher than that predicted by the model. Since the thermocouple rake in Tank 1 was in the vicinity of the fuel pumps, it is hypothesized that warmer fuel from the bulk was mixed with that near the surface resulting in higher recorded surface temperatures in this area. Further investigation showed that a good correlation for lower surface temperature was obtained, for cases in which fuel was withdrawn, by the addition of a local internal convection coefficient used only in the computation of the lower surface temperature. This local internal convection coefficient is higher than the overall internal convection coefficient in order to account for the high degree of motion in the fuel in the vicinity of the fuel pumps. Because the higher surface temperature recorded in the vicinity of the fuel pumps is believed to be a localized effect, the lower surface temperature predictions for the study missions, which are to be representative of the entire lower surface of the tank, were computed without the higher local convection coefficient.

The second case where a significant modification to the convective heat transfer computation was required occurred inside Tank 2R-inboard. It was found that a convection coefficient significantly higher than that given by the empirical relations was required to obtain a good correlation of both lower surface and bulk fuel temperatures with flight test recorded temperatures. This was true for all of the test flights that were analyzed. The higher than estimated convective heat transfer to this surface is incurred because fuel sloshing within the tank is more prevalent in Tank 2-inboard due to its shape and the continual flexing it experiences because of its position further from the aircraft's center of gravity. Sloshing within the tank would tend to increase the rate of heat transfer from the fuel to the lower surface by mixing warmer bulk fuel with the colder layer of fuel near the lower surface. Further investigation showed that a constant value of internal convection coefficient applied to the lower tank surface of Tank 2-inboard consistently gave satisfactory time-temperature correlations for all flight tests. For the study missions, it was premised that the same value of convection coefficient was applicable.

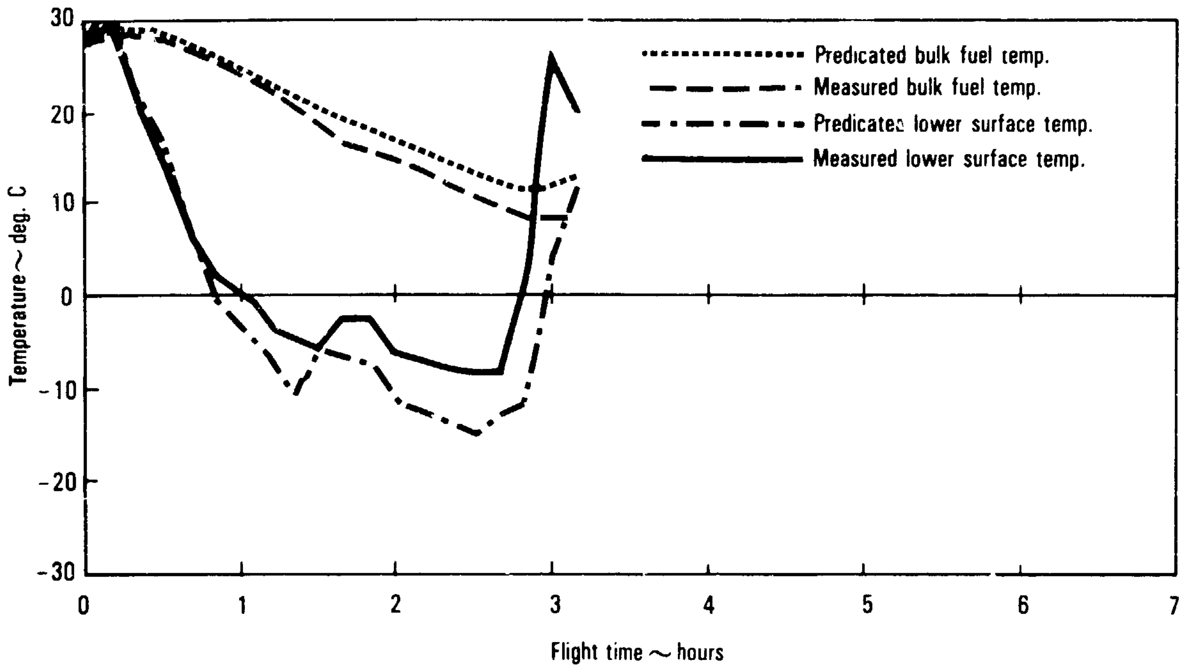


Figure 22 - Bulk fuel and lower surface unadjusted temperature correlations - full tank.

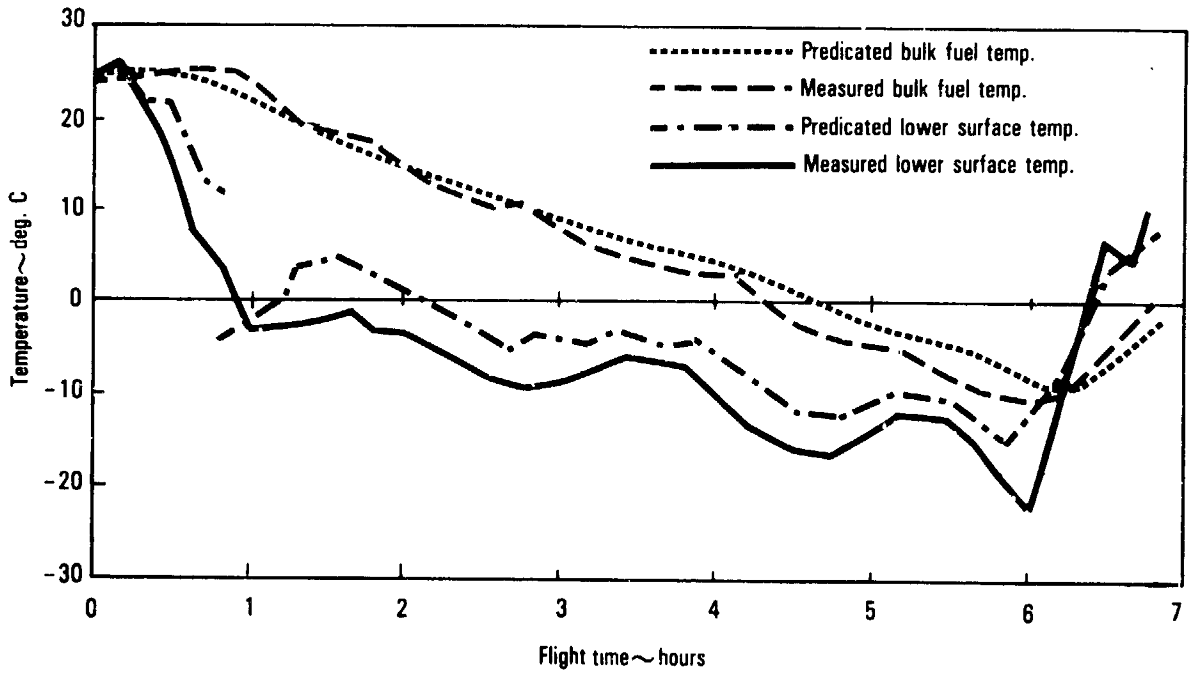


Figure 23 - Bulk fuel and lower surface unadjusted temperature correlations - emptying tank.

Since flight test data are not available for the determination of internal and external convection coefficients for Tank 2-outboard, it must be assumed that those determined for Tank 2-inboard are also applicable to Tank 2-outboard. This premise is justified because Tanks 2-inboard and 2-outboard have similar geometries and because both are located in the outboard portion of the wings.

Figures 24 through 28 show corrected correlations of predicted bulk fuel and lower tank skin temperature-time histories with flight test measured results. Figure 24 shows the correlation obtained for Tank 1 in a flight in which the tank remained full of fuel. The prediction of the lower skin temperature tended to lead the recorded temperature during the initial climb and final descent phases of the flight. This effect is a result of the modeling technique in which all of the thermal mass of the fuel is treated as a single concentrated homogenous unit. During these periods of rapid temperature change, a thickening of the temperature stratification layer is observed in the measured vertical temperature profile accounting for some deviation from predicted results obtained by the bulk fuel concept used in the model. A greater thermal lag actually exists in the vicinity of the lower skin than is considered by the model. However, during the cruise portions of the flights, a more pervasive bulk fuel temperature is established in the fuel tank increasing the accuracy of the bulk fuel concept. The temperature transient observed in the lower skin temperature between the second and third hours of the flight were caused by a 0.16 drop in Mach number.

Figures 25 and 26 show the correlations obtained in Tank 1 for two flights in which fuel was withdrawn from the tank. Again, the tendency of the predicted lower skin temperature to lead the actual temperature can be observed. The good correlations achieved in these two cases demonstrate the validity of the model's treatment of the changing wetted surface areas inside the tank as fuel is withdrawn.

The correlation shown in figure 24 for a case in which the tank remained full of fuel supports the validity of the overall heat transfer analysis but does not indicate whether the separate heat transfer rates computed for the upper and lower tank surfaces are properly proportioned. Since the lower surface is the only significant heat path when the tank is partially filled, the later portions of the correlations shown in figures 25 and 26 for cases in which fuel was withdrawn sh the validity of the lower surface heat transfer analysis.

The periodic temperature transients observed in figure 25 were caused by the aircraft's repeated traversing through a weather front. It is not known what caused the temperature transients observed in figure 26. It appears, however, that the recorded temperature profiles tend to approach the predicted profiles toward the end of the flight where the transients were not present.

Figures 27 and 28 show corrected correlations obtained for Tank 2-inboard. In both cases, the tank was maintained approximately 90 percent full of fuel throughout the cruise portion of the flight. The temperature transients observed in the recorded lower surface temperature during the first two hours of the flights were apparently caused by aircraft maneuvers which resulted in mixing of the warm bulk fuel with the colder fuel layer near the lower surface. In figure 27, the temperature transient observed in the predicted lower surface temperature at approximately 4.5 hours into the flight was caused by a change in altitude. As before, the predicted lower surface temperature tended to lead the measured lower surface temperature during the rapid change in ambient temperature.

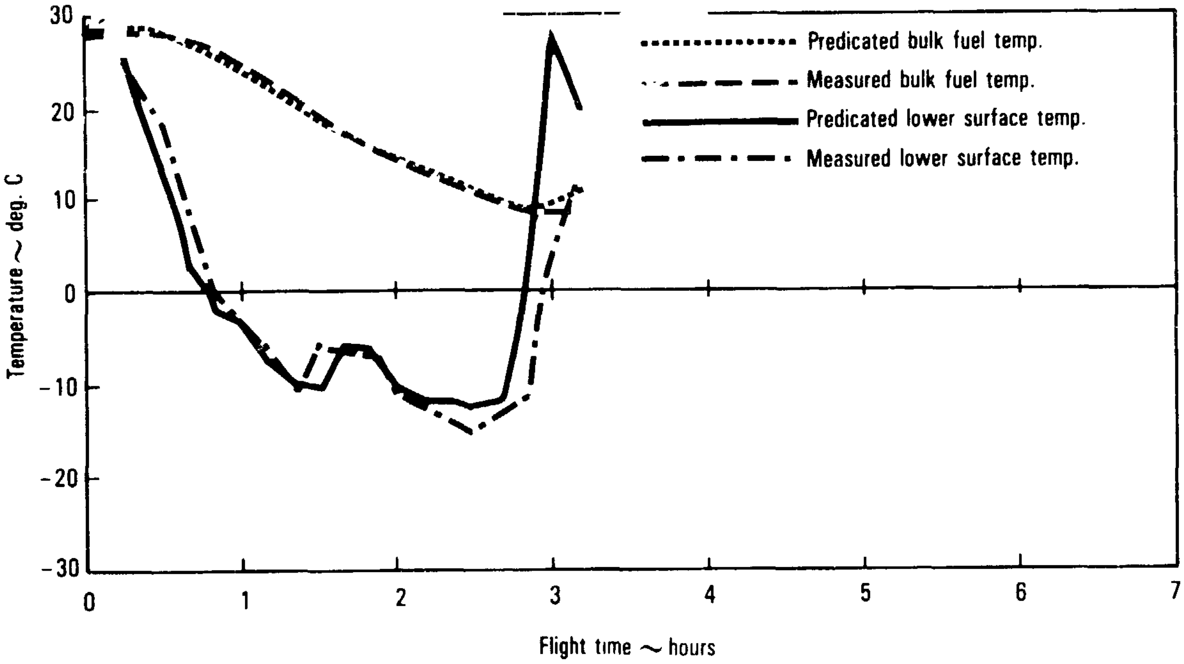


Figure 24 - Tank-1 bulk fuel and lower surface corrected temperature correlations - full tank, flight T-1686.

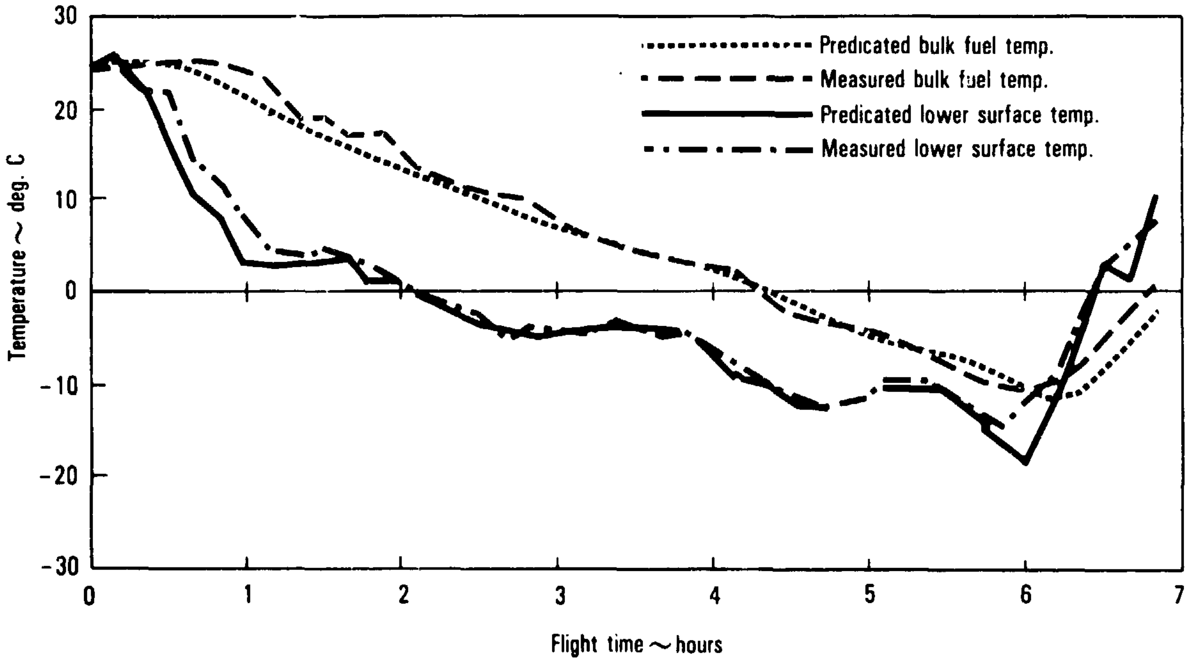


Figure 25 - Tank-1 bulk fuel and lower surface corrected temperature correlations - emptying tank, flight T-1640.

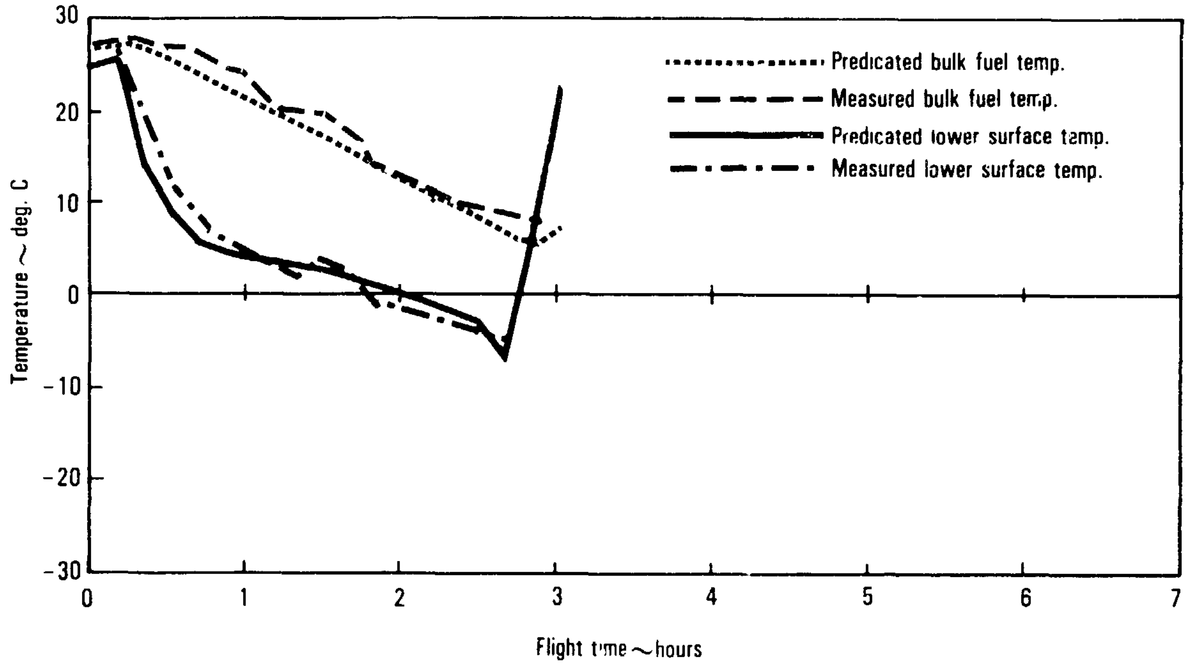


Figure 26 - Tank-1 bulk fuel and lower surface corrected temperature correlations - emptying tank, flight T-1676.

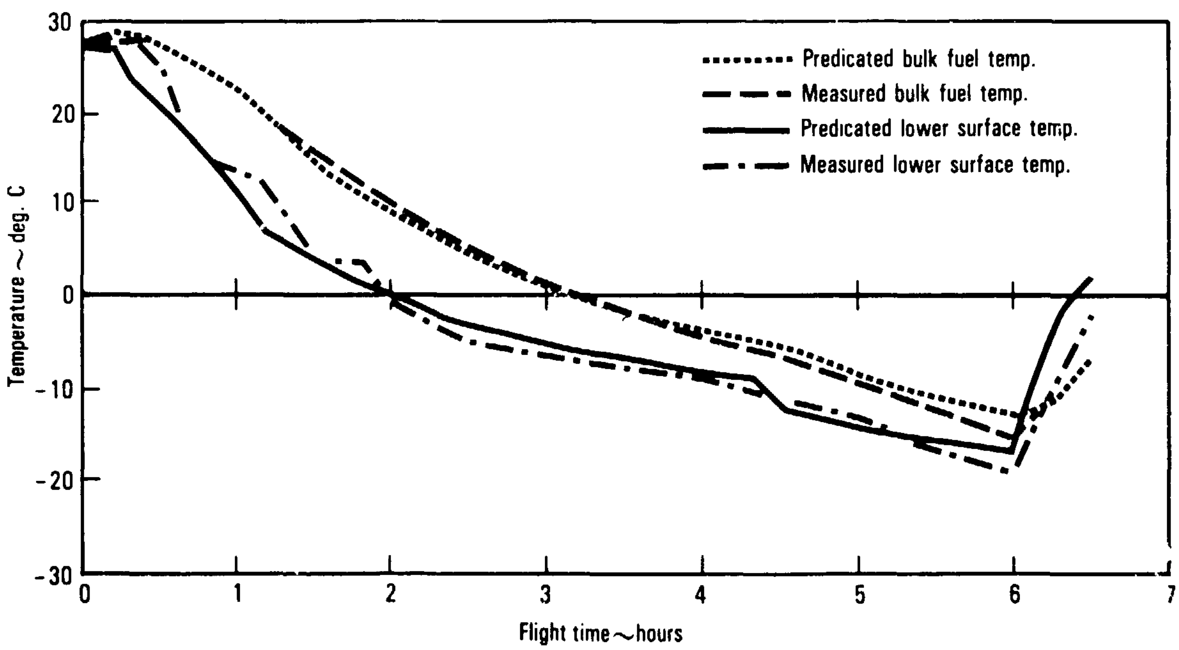


Figure 27 - Tank-2 inboard bulk fuel and lower surface corrected temperature correlations - 90% full, flight T-1653.

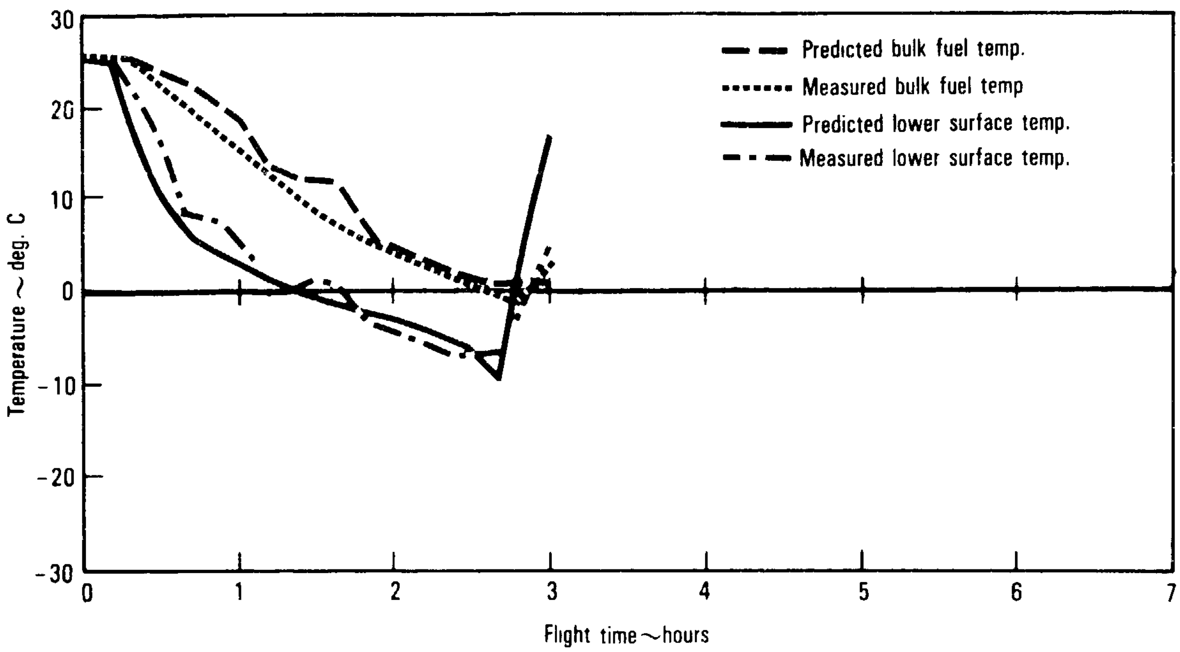


Figure 28 - Tank-2 inboard bulk fuel and lower surface corrected temperature correlations - 90% full, flight T-1676.

5.1.2 Fuel tank predicted temperatures. - To determine the impact of the use of the high freeze point fuel on the operation of the baseline aircraft, predictions of fuel temperature-time histories were required for each of the three cold day missions. The fuel tank thermal model discussed in the preceding section was used to generate these temperature-time histories. Using the mission profile data presented in Section 3.2, including altitude, Mach number, ambient temperature, and fuel quantity as functions of time, the model gave predicted bulk fuel and tank lower skin temperature for each of the fuel tanks.

Since it is within the cold fuel layer on the lower tank wall that the first accumulation of fuel freeze-out occurs, emphasis is placed on the predicted temperatures of this layer of fuel. Although the fuel tank thermal model actually gives the temperature of the aluminum structure of the lower tank boundary, this is very nearly equal to the temperature of the adjacent fuel. In the discussions that follow, the lower surface temperature given by the model is used to express the coldest and most critical fuel temperature.

The predicted bulk fuel and lower surface temperatures for the 9260 km (5000 n.mi.) mission are shown for Tank 1, Tank 2-inboard, and Tank 2-outboard in figures 29, 30, and 31, repectively. An initial fuel temperature of -17°C (1.4°F) which is 3°C above the freezing point of the fuel, was selected for the cases shown in the figures. The selection of this initial temperature and its effect on the overall fuel temperature profile is discussed later in this section. The figures show that the bottom surface temperature is lower than the bulk fuel temperature throughout most of the flight. The exceptions are 1), between the first and second hour of the mission, a period during which the aircraft climbs through a thermal

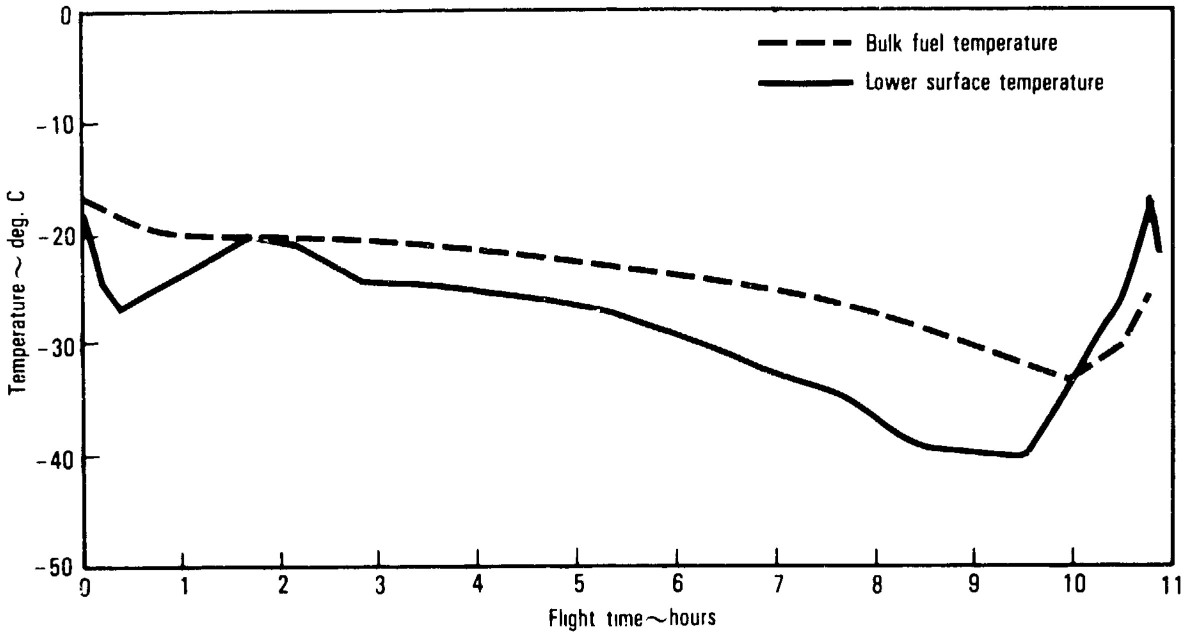


Figure 29 - Tank-1 bulk fuel and lower surface predicted temperature - 9260 km (5000 n.mi.) cold day mission.

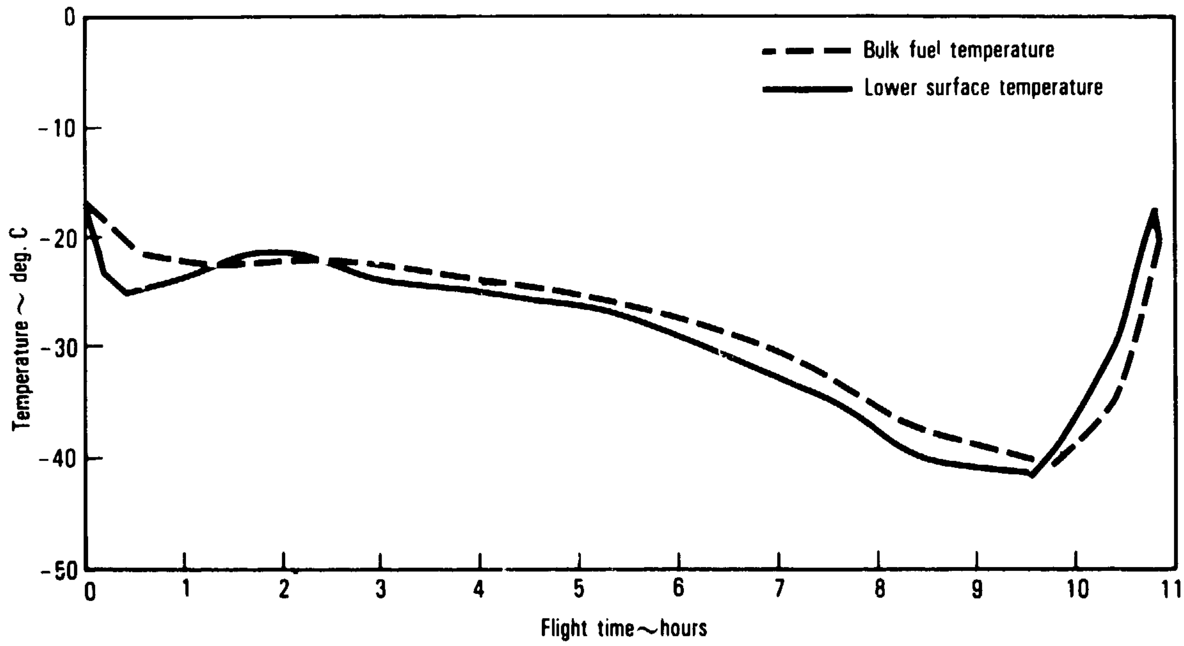


Figure 30 - Tank-2 inboard bulk fuel and lower surface predicted temperature - 9260 km (5000 n.mi.) cold day mission.

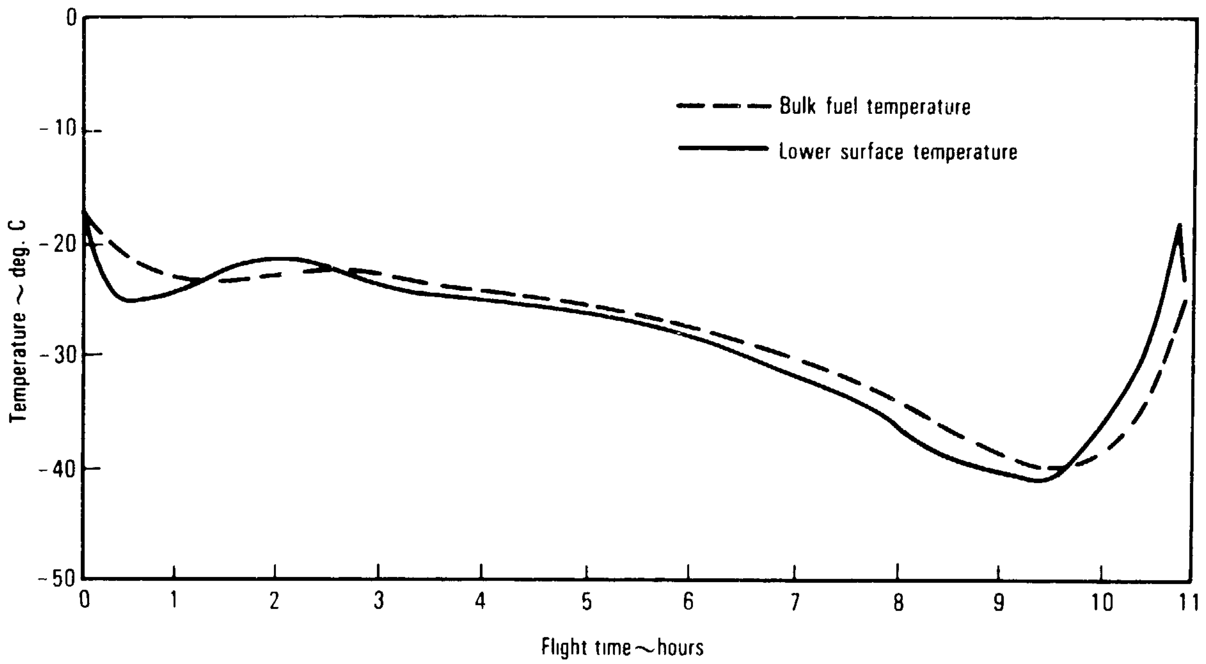


Figure 31 - Tank-2 outboard bulk fuel and lower surface predicted temperature - 9260 km (5000 n.mi.) cold day mission.

inversion layer built into the ambient temperature profile, and 2), during the descent at the end of the mission. In all tanks, the lower surface temperature tends to approach the recovery temperature.

The figures show that the spread between the bulk fuel temperature and the lower surface temperature is quite different in the different fuel tanks. In Tank 1 which has a relatively low surface to volume ratio, the temperature spread is relatively large. In Tanks 2-inboard and 2-outboard both of which have much higher surface to volume ratios, the temperature spread is much less. For all missions, the bulk fuel and lower surface temperatures in Tank 2-outboard are consistently lower than the corresponding temperatures in the other two tanks. Therefore, Tanks 2-outboard, left and right are considered to be the most critical fuel tank in regard to fuel freezing.

The lower surface temperature-time histories in Tank 2-outboard for the 926, 3704, and 9260 km (500, 2000, and 5000 n.mi.) cold-day missions are shown in figure 32. As expected, progressively lower minimum fuel temperatures are attained as the mission length is increased. However, the difference between the minimum temperatures attained in the various missions is not as large as may have been anticipated. This is due to the lower fuel quantities carried on board for the shorter missions. The reduced fuel quantity provides less thermal mass in the fuel tank and results in an increased rate of change in temperature.

The effects of initial fuel temperature on the lower surface temperature-time histories in Tank 2-outboard are shown in figure 33. For each of the three cold-day missions, temperature profiles are shown for initial fuel temperatures of -17°C and

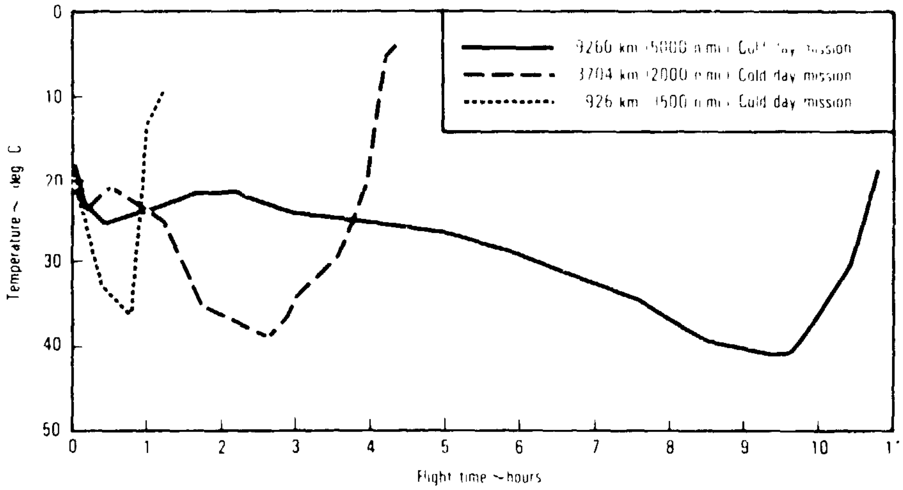


Figure 32 - Tank-2 outboard lower surface predicted temperature - 926, 3704, and 9260 km (500, 2000, and 5000 n.mi.) cold day missions.

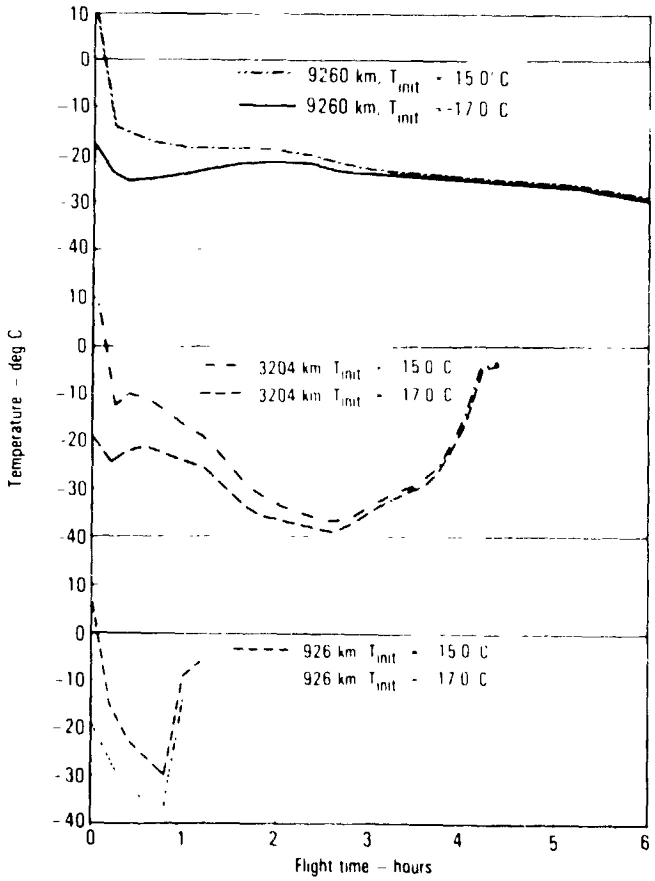


Figure 33 - Tank-2 outboard lower surface predicted temperature - +15.0 and -17.0°C initial fuel temperature.

+ 15°C. The impact of the initial fuel temperature on the minimum lower surface temperature varies with mission length from a 6.7°C temperature difference in the 926 km (500 n.mi.) mission to an insignificant temperature difference after the fourth hour in the 9260 km (5000 n.mi.) mission. The results of this study show that with a 15°C initial fuel temperature, a fuel with a freeze point of approximately -29°C would be required for the 926 km (500 n.mi.) mission and a freeze point of approximately -37°C would be satisfactory for the 3704 km (2000 n.mi.) mission. However, the study results suggest that for the 9260 km (5000 n.mi.) mission, no freeze point advantage could be obtained by maintaining any reasonable initial fuel temperature.

5.1.3 Effects on baseline aircraft performance. - The low fuel temperature of -41°C predicted in the previous section for an extreme cold day (figure 32) is representative of the minimum temperatures expected with a probability of occurring one day per year based upon statistical summaries of data recorded by IATA member airlines (reference 21). The airlines have experienced little difficulty with these temperatures while using ASTM D 1655 specification fuels. In considering an increase in fuel freeze point to -20°C, it is recognized that fuel freeze-out will occur. Consequently, the changes in the physical properties of the fuel must be thoroughly understood in order to evaluate the impact of these changes on the baseline airplane performance.

Although the initial freeze-out occurs at the bottom of the tanks coating the skin with a thin layer of wax, most of the minute wax particles which constitute freeze-out remain dispersed in the liquid fuel and rarely affect the aircraft performance. However, as the freezing process continues, fuel in the form of wax particles agglomerate and fall to the bottom of the tank forming a matrix which traps additional liquid fuel. The total mass of the wax and trapped liquid fuel can be many times that of the wax alone. This combination of wax and trapped fuel, referred to as hold-up, remains unavailable to the engines. In extreme cases, the wax could block the fuel tank exits rendering large quantities of fuel unavailable to the engines.

Agglomerated wax particles in the fuel stream can create blockage in scavenge ejector motive flow filter screens and can slow down the response rate of close tolerance valves. For components which are normally shut down in flight, the problem is more acute. For example: in normal operation of the aircraft, the APU is shut down shortly before takeoff. The APU remains shut down throughout the flight unless it is required as an emergency power source. Since the APU compartment and the fuel lines from the wing fuel tanks are not heated, the possibility of these components becoming blocked by frozen fuel exists. This occurrence would cause the APU to be inoperable at the end of the flight and unreliable as an emergency power source in flight.

On rare occasions, situations arise when it may be desirable to shut down an engine while in flight but retain the option to restart the engine prior to landing. An example of such a situation would be the occurrence of a warning indication on an engine monitoring instrument. In this case, it is likely that the shut down engine would be restarted prior to landing and held in reserve in case the landing had to be aborted. If an engine were shut down in flight, the temperature of the fuel in some of the engine fuel system components would drop very quickly. Thus, the possibility exists that frozen fuel could block the engine fuel system and prevent the engine from being restarted.

In the fuel tanks, small amounts of fuel hold-up could be tolerated as part of normal operating procedures. This would be true if extra fuel were carried on board to maintain the required quantity of usable fuel. However, a practical procedure for predicting hold-up or for measuring hold-up in flight must be developed for this practice to be feasible. For the present, changes in the normal operating procedures of the baseline aircraft will be considered to eliminate the possibility of hold-up in the fuel tanks. For safety reasons, frozen fuel must not affect the operation of the other vital fuel system components.

The options available consist primarily of placing restrictions on the minimum initial fuel temperature and adiabatic wall temperature established by Mach number and ambient air temperature. Figure 33 indicates that an initial fuel temperature of 15°C is not sufficiently high to prevent freezing of the -20°C freeze point fuel even for the short range mission. Initial fuel temperatures higher than 15°C are not considered practical as a standard requirement for winter operation. It is apparent that for the ambient temperatures considered in this study, a restriction on the minimum initial fuel temperature would delay but not prevent fuel freeze-out in the fuel tanks. Of course, restrictions on initial fuel temperature would do nothing to prevent fuel freezing in inoperative engine and APU fuel systems.

The restrictions on the allowable combinations of Mach number and ambient temperature to prevent fuel freezing are dependent upon the atmospheric conditions encountered in flight. These restrictions can be evaluated in terms of the adiabatic wall temperature. The adiabatic wall temperature is the temperature of the boundary layer air stream adjacent to external surfaces of the aircraft. Since the boundary layer is the primary heat sink to which aircraft heat is rejected, the adiabatic wall temperature is very nearly the lowest possible temperature obtainable by any aircraft component. Radiation to the environment, the other normally considered mode of heat rejection is nearly insignificant under the conditions of subsonic commercial aircraft flight. Therefore, if the aircraft is operated under flight conditions such that the adiabatic wall temperature exceeds the freeze point of the fuel, then the possibility of fuel freezing is eliminated.

For any given ambient temperature/altitude profile, there is a locus of altitude, Mach number combinations which give an adiabatic wall temperature equal to the freeze point of a specified fuel. In figure 34, lines of constant adiabatic wall temperature equal to the -20°C freeze point for the study fuel are shown for three ambient temperature profiles, ISA Day, and the 50 percent and 0.3 percent probability days discussed in Section 3. Aircraft operation at altitude/Mach combinations on or to the right of these lines would prevent fuel freezing during flight in the respective environments.

However, the maximum Mach number allowable at various altitudes is restricted by both legal and aircraft operating limitations. The maximum Mach number as a function of altitude for the baseline aircraft is indicated in figure 34 by the dotted line. Below 3048 m (10,000 ft), the line indicates the maximum legal calibrated airspeed of 129 m/s (250 kt). Above 3048 m (10,000 ft), the line indicates the structural limitations of the aircraft. This Mach limit is valid for all three ambient temperature profiles since the legal and structural limitations are independent of atmospheric conditions.

To preclude hold-up in the fuel tanks, the aircraft must always be flown at speeds greater than the minimum Mach number for the atmospheric conditions prevailing. This capability could be made available to the pilot through use of a

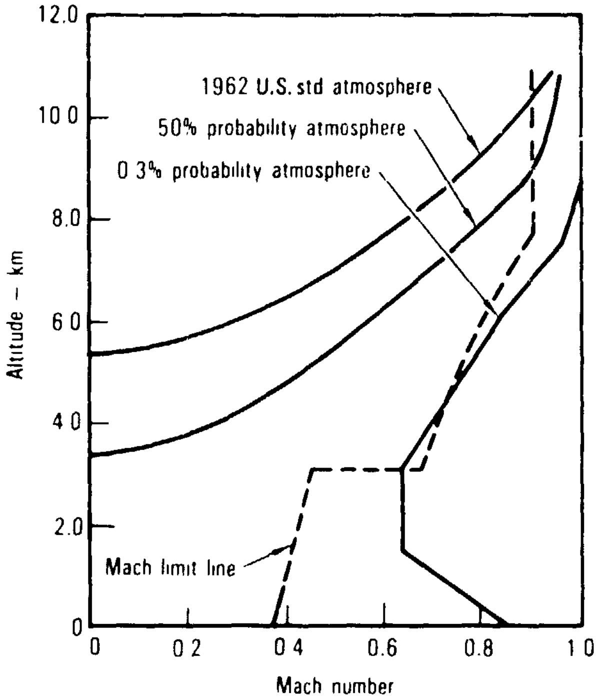


Figure 34 - -20°C adiabatic wall temperature profiles.

chart relating Mach number (M) to stagnation temperature (T_o) to maintain an adiabatic wall temperature equal to or greater than the fuel freeze point. For -20°C freeze point, the flight Mach number must be greater than

$$M = 0.827 \frac{(T_o + 20.)}{(1.0 + 0.123\ T_o)}$$

which can be presented in chart form to the pilot. However. this airspeed must never exceed the aircraft Mach limit line.

An alternate means of insuring safety of flight under low ambient temperature conditions is to measure the fuel temperature immediately adjacent to the bottom tank skin at a point which represents the lowest fuel temperature encountered in any tank. Although this would appear to be a direct method of indicating to the pilot the limiting conditions for safe flight, the selection of a representative position which would apply for all flight conditions and fuel distributions in the tanks is difficult.

It is apparent from figure 34 that the use of the -20°C freeze point fuel in the baseline aircraft is not practical under very cold conditions. For the 0.3 percent probability profile, the figure indicates a very narrow envelope between 3048 and 3962 m (10,000 and 13,000 ft) in which flight would be possible. Of course operation of the baseline aircraft at the altitude and Mach number indicated for this ambient points would have to be used. If, while in flight, an aircraft encountered such low ambient temperatures that it was impractical to maintain the -20°C adiabatic wall temperature, the aircraft would have to take some precautionary action such as landing at the nearest available airport.

The use of the high freeze point fuel in the baseline aircraft is practical under more temperate conditions. Table 5 shows the percent decrease in fuel economy (in terms of specific air range) caused by deviating from the optimum flight altitude and Mach number to maintain the -20°C adiabatic wall temperature. This impact on fuel economy is given for three temperature profiles. The -20°C adiabatic wall temperature profiles were selected to maximize fuel economy by using the highest possible even thousand foot altitude while not exceeding the normal cruise Mach number. The specific air range values are for level cruise flight at the specified Mach number, altitude and temperature for a weight of 175 958 kg (388,000 lb). This simplification does not consider that additional fuel would have to be carried to complete a given mission using the -20°C adiabatic wall temperature profile. Note that the fuel economy percent difference represents instantaneous difference which is valid only for the cruise point Mach numbers, altitudes and weight specified. The fuel economies and percent differences are included only as an example of relative values for the conditions discussed here. The fuel economy percent differences have no relation to mission block fuels because climbs, descents, and amount of time in cruise have not been accounted for.

TABLE 5 - IMPACT OF HIGH FREEZE POINT FUEL ON PERFORMANCE OF BASELINE AIRCRAFT

Atmosphere	Profile	Mach Number	Altitude Meters	Altitude (feet)	Ambient Temperature °C	Ambient Temperature (°F)	Fuel Economy km/kg	Fuel Economy (n. mi./lbm)	Fuel Economy Percent
1962 U.S Standard Atmosphere	a	0.82	10 668	(35,000)	-54	(-66)	0.132	(0.0322)	-6
	b	0.82	9 449	(31,000)	-46	(-51)	0.124	(0.0304)	
50% Probability Profile	a	0.82	10 668	(35,000)	-56	(-68)	0.132	(0.0322)	-13
	b	0.82	8 230	(27,000)	-46	(-51)	0.115	(0.0281)	
0.3% Probability Profile	a	0.82	10 668	(35,000)	-68	(-90)	0.132	(0.0322)	-35
	b	0.68	3 658	(12,000)	-40	(-40)	0.086	(0.0211)	

a—Optimum altitude for weight - 175 958 kg (388,000 lb), Mach - 0.82
b—Altitude and Mach number necessary to maintain a -20°C adiabatic wall temperature

5.2 Thermal Stability

Present jet fuels which comply with ASTM specifications for thermal stability, produce coking rates in current fuel systems that are acceptable from the aircraft operator's point of view. The fuel system can be cleaned, if necessary, by flushing it with approved detergents during the scheduled maintenance program. The criterion for designing a fuel system able to handle lower thermal stability fuels is based on the fundamental assumption that these presently tolerable deposition rates are not to be increased when relaxing the JFTOT rating to 204°C.

The coking rates in a fuel system component depend primarily on the fuel temperature, wall temperature, fuel flow rates, and pressure, as well as other variables unrelated to the fuel system such as fuel composition, contaminants, exposure to air, etc. Unfortunately, there does not exist as yet a dynamic model for thermal stability which can account for the effects of the environmental parameters mentioned above. Recent efforts by Vranos and Marteney (ref. 22) under the sponsorhip of NASA have resulted in the collection of film deposition rates for a variety of fuels, under geometric conditions approaching a fuel system situation. These data were correlated by those authors using Arrhenius plots of deposition-rates vs. wall-temperatures as shown in figure 35. When operating with Jet A fuel, the wall temperatures may on a hot day reach values as high as 135°C (275°F). From the figure, the coking rates at this temperature appear to be of the order of 1 $\mu g/(cm^2\ h)$. For a home heating fuel, such as the one selected in the figure to represent a reduced thermal stability jet fuel, extrapolation of the experimental data to temperatures as low as 84°C, would indicate coking rates as high as 10 $\mu g/(cm^2\ h)$. This temperature is commonly reached in present diesel engine fuel systems, but there have not been any indications of such a fast deposit build-up. It appears then that extrapolation of the high temperature measurements of ref. 22 to lower temperatures should be exercised with great caution. For the purpose of this study, a reduction temperature interval will be selected based on the decrease in JFTOT rather than on the coking rates criterion described above. This interval will be taken as the difference between the Jet A JFTOT (260°C) and the low thermal stability fuel (204°C), that is, ΔT = 56°C. It is recognized that this interval could be somewhat higher if properly estimated from coking rate considerations.

In order to assess the impact of utilizing the low thermal stability fuel, the following method will be applied here:

a) Establish a tolerable maximum fuel bulk temperature in the fuel system when operating with Jet A. This temperature limit is determined from experience.

b) Decrease this present limit by an amount equal to the temperature interval separating the tolerable coking rates of 1 $\mu g/(cm^2\ h)$.

c) Obtain fuel system component temperature profiles for the thermal stability limiting flight (hot day, short flight).

d) Compare the profiles obtained in (c) with the new fuel temperature limit selected in (b). Assess the impact on fuel systems and aircraft operations if the temperature profiles of (c) are reduced to remain below the new fuel temperature limit.

In the following, the tank outlet fuel temperature will be presented for the limiting short flight on a hot day. The fuel system components affected by low thermal stability will be described, and the temperature distribution throughout the system estimated. The impact of the low thermal stability fuel on components as well as aircraft operations is discussed at the end of this section.

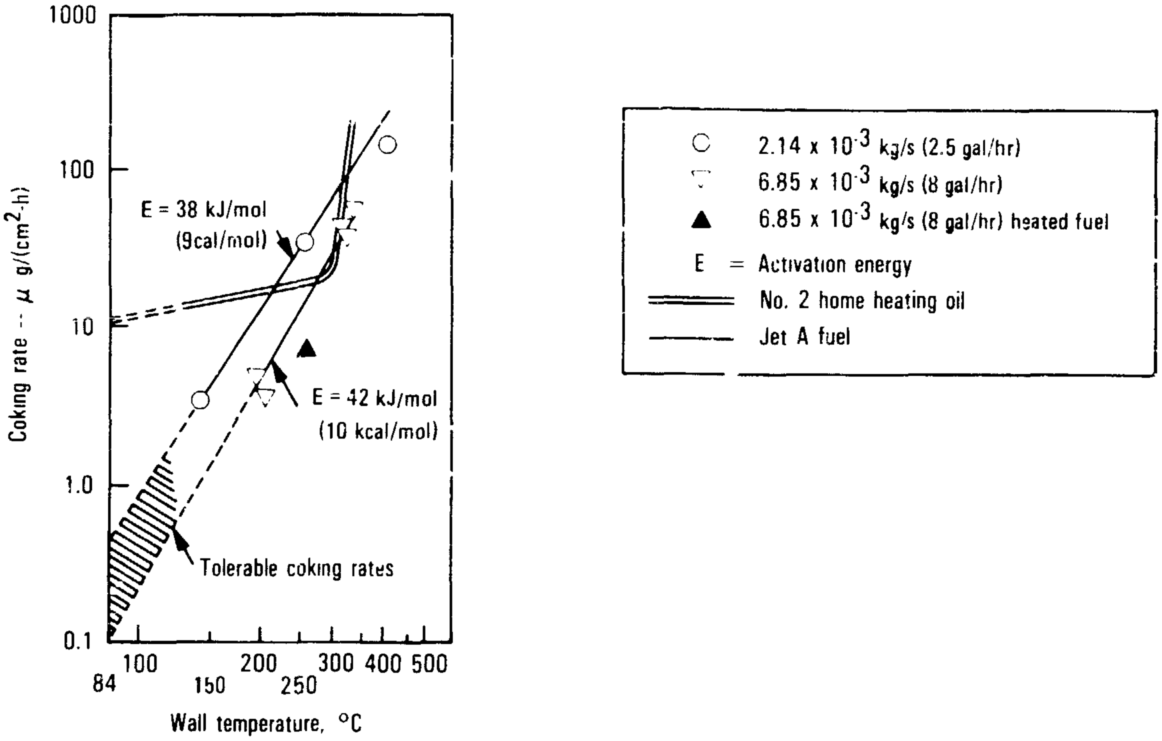


Figure 35 - Coking rates vs. wall temperatures for Jet A and home heating fuels.

5.2.1 Tank outlet fuel temperature. - A short flight of 926 km (500 n.mi.) on the hot day profile established for this study, 54°C (130°F) sea level temperature, is the limiting flight profile for low thermal stability fuel. The tank outlet temperature history was estimated utilizing the wing tank thermal model described in 5.1.1. The flight parameters were described in 3.2.1 and additional information relevant to the present analysis was obtained from table 6. The fuel tank bulk temperatures after refueling were assumed to be 38°C (100°F) which is the reported maximum fuel temperature delivered to aircraft in hot climate airports (ref. 23). In estimating the tank outlet temperature at takeoff, it was assumed that the aircraft cleared the ramp two hours after being fueled. Table 7 shows the estimated fuel tank outlet temperature histories throughout the flight. In Tanks 1 and 3, which supply Engines 1 and 3, the fuel cools at a slower rate than the fuel in Tank 2 outboard which has a higher surface-to-volume ratio. (Fuel from Tank 2 outboard is transferred to the inboard compartment, and from there to Engine 2, the tail section engine). Although the difference between the tank outlet fuel temperature histories is minimal, the Tank 1 outlet temperature was selected for the purpose of this analysis as providing the most limiting condition.

5.2.2 Engine fuel environment. - Thermal stressing of the fuel is most severe on the engine side of the baseline aircraft fuel system. The engine fuel system was described in Section 3.1.2 and illustrated schematically in figure 4. In order to

TABLE 6 - AIRCRAFT AND ENGINE PARAMETERS PROFILES FOR THE SHORT RANGE 926 km (500 n.mi.) FLIGHT, HOT DAY (54°C GROUND TEMPERATURE)

Segment	Mode	Time From Take-Off min	Altitude m	(ft)	Mach No.
1	acc	0		0	0
2	climb	0.45		0	0.226
3	acc	0.98	457	(1,500)	0.232
4	climb	1.47	457	(1,500)	0.388
5	acc	4.13	3,048	(10,000)	0.420
6	climb	4.84	3,048	(10,000)	0.476
7	climb	15.87	8,808	(28,893)	0.820
8	cruise	24.35	11,887	(39,000)	0.820
9	descent	45.10	11,887	(39,000)	0.820
10	descent	57.46	3,048	(10,000)	0.576
11	decel	58.1	3,048	(10,000)	0.452

Segment	Thrust Per Engine kN	(klb)	Fuel Flow Per Engine kg/h	(lb/hr)	HP Rotor Speed (rpm)	Turbine Gas Temperature °C	(K)
1	164.6	(37.0)	7,521	(16,580)	10,050	1818	(1544)
2	146.8	(33.0)	7,466	(16,460)	10,050	1818	(1544)
3	133.4	(30.0)	7,366	(16,240)	10,050	1818	(1544)
4	89.0	(20.0)	4,908	(10,820)	9,500	1629	(1355)
5	75.6	(17.0)	4,518	(9,960)	9,690	1712	(1438)
6	57.8	(13.0)	3,946	(8,700)	9,460	1623	(1350)
7	40.0	(9.0)	2,867	(6,320)	9,490	1673	(1400)
8	27.1	(6.1)	2,005	(4,420)	9,440	1690	(1417)
9	26.7	(6.0)	680	(1,500)	8,130	1306	(1033)
10	2.2	(0.5)	871	(1,920)	7,580	1223	(950)
11	2.2	(0.5)	980	(2,160)	7,800	1229	(955)

TABLE 7 - FUEL TANK OUTLET TEMPERATURE HISTORIES

t*, minutes	Tank 1 or 3 Temperature, ^{0}C	Tank 2 (Outboard) Temperature, ^{0}C
0	47.2	47.2
12	47.2	46.7
16	46.7	46.7
18	46.7	46.1
20	46.1	45.5
26	45.5	43.9
28	45.0	43.3
30	45.0	42.8
32	44.4	42.2
34	43.9	41.7
36	43.9	40.5
38	43.3	40.0
40	42.8	39.4
42	42.8	38.9
44	42.2	38.3
46	41.7	37.2
Beginning of descent		

*Time from takeoff

have a better understanding of the effects of low thermal stability fuels on the engine performance, the functional relationships and thermal environment of the various engine components are discussed in this section. Although the engine fuel feed lines from Tanks 1 and 3 are significantly different from the Tank 2 feed line supplying the aft fuselage engine, all of the engines have identical fuel systems downstream of the engine interface.

In this analysis engine fuel system accessories located in the fan case are discussed separately from those in the engine core area because of the differing thermal environments. Installation of the accessories on the fan case is desirable, since this region is the coolest and can be accessed while the engine is operating. The fuel system components installed in this region include the LP fuel filter, the dual LP/HP pump, the two fuel-cooled oil coolers, the fuel flow regulators (including the fuel enrichment solenoid and the starting fuel flow regulator) and the HP shutoff valve. The core section operates at higher temperatures since it is exposed to the heat rejected from the HP compressor air and the combustors. The fuel system components installed in this region include the HP filter, manifold, distribution valves, and injectors. Ventilation of this region is provided by air bled from the fan duct.

Once the fuel enters the engine side of the fuel system, its temperature increases progressively as the fuel flows downstream through different components and line segments. The bulk fuel temperature in those components located on the fan case are slightly lower than the surface temperature. In the core region the soak-back heat from the hot section of the engine becomes more intense, while simultaneously the fuel velocity increases, causing a larger difference between the bulk and surface temperatures. In the following, those components of the fuel system, which introduce important thermal loads into the fuel system, are reviewed in greater detail.

5.2.2.1 Fuel-cooled oil cooling system (FCOC): The baseline fuel system is designed to improve the engine specific fuel consumption by preheating the fuel while assuring at the same time that under extreme conditions, the fuel is not overheated to form varnishes and carbon deposits. To this end, the fuel is utilized to cool the engine oil, and this results indirectly in a secondary SFC improvement, since it eliminates the need for an air-cooled oil cooling system, which does have an impact on installed performance.

The total heat to the engine oil is approximately a function of the engine speed. The baseline engine has three independent concentric rotors for the LP (fan), IP, and HP compressors. The speeds of the three rotors are coupled aerodynamically within narrow limits permitting the total heat to the oil to be expressed as a function of the HP rotor speed only (figure 36).

A schematic diagram of the FCOC can be seen in figure 37. A pressure pump supplies oil from the oil tank, through a high pressure filter and two oil coolers, and then to the inlet of a second pressure pump which delivers the oil to the gears and bearings. The oil cooling system is comprised of the control valves, the low pressure fuel-cooled oil cooler (LPFCOC), and the high pressure fuel-cooled oil cooler (HPFCOC). The oil flow path through the coolers is established by the temperature of the LPFCOC outlet, which determines the position of the control valves by means of retracting thermal elements. The cold valve element operates between 12 and 25°C, and the hot valve element in the range from 85 to 95°C. If the LP temperature is below 12°C, all the oil flow is directed through the LPFCOC only, resulting in maximum fuel heating. Between 12 and 25°C, a small fraction of the oil

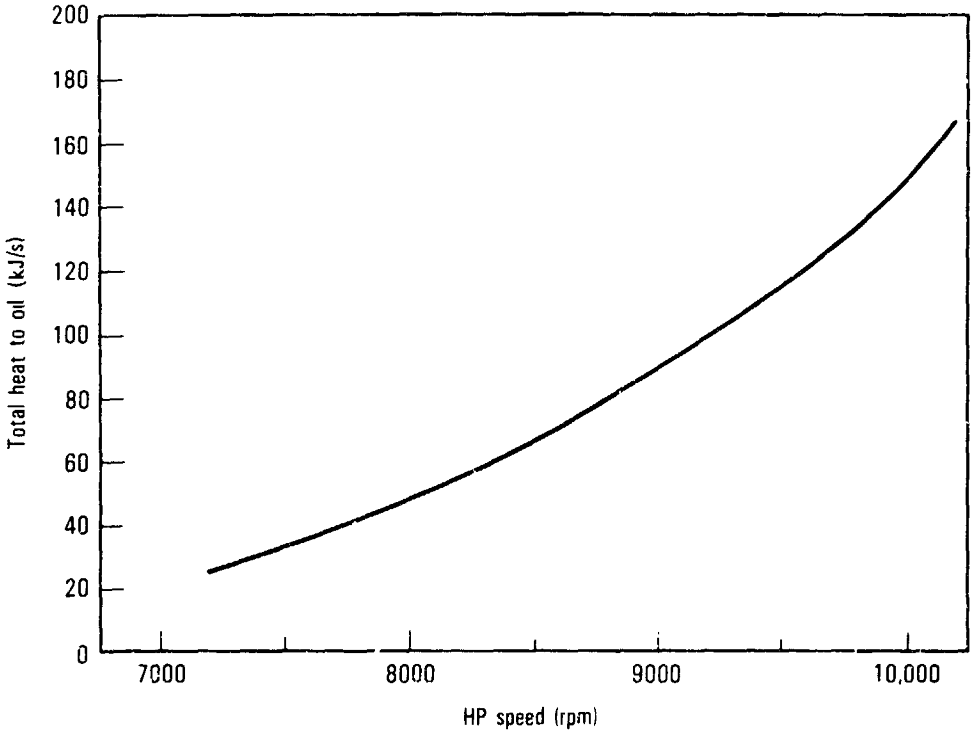
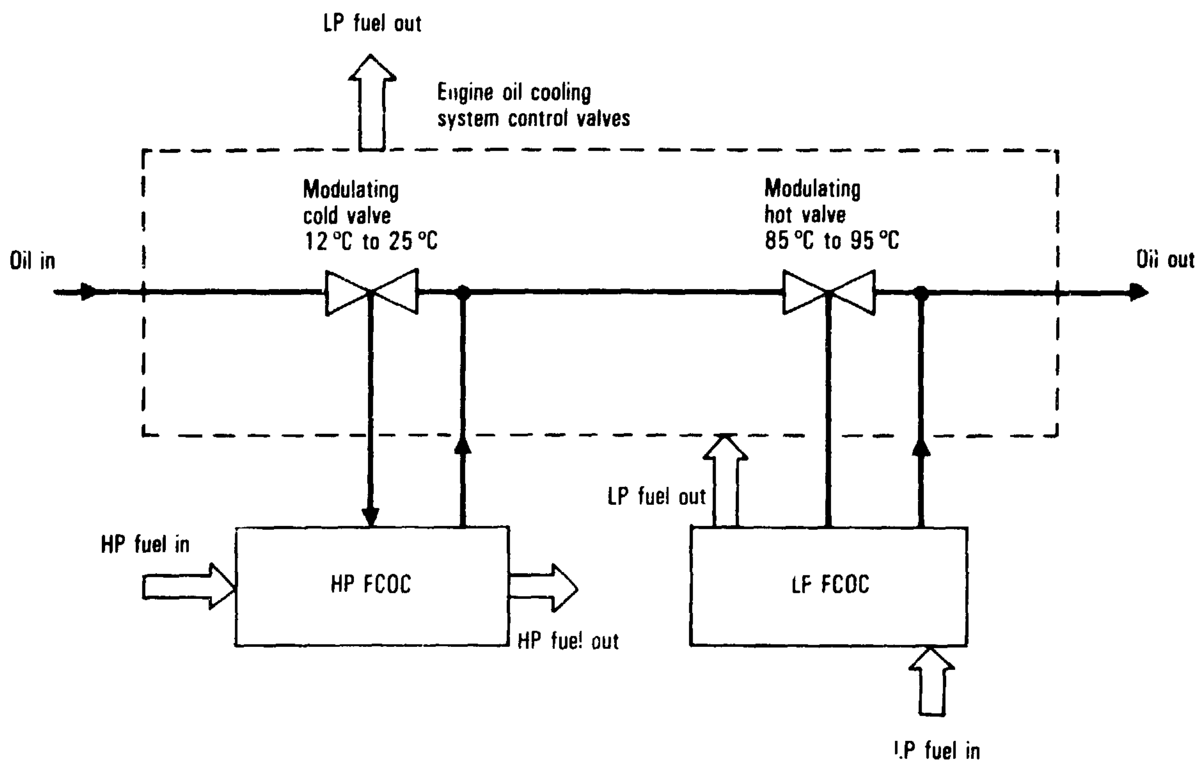


Figure 36 - Total heat input to engine oil as a function of HP rotor speed.



Fuel Temp at LP FCOC Exit	Oil Flow Through	Action
Below +12 deg C oil press above 35 psi	LP FCOC and/or bypass depending on oil pressure	Oil pressure opens LP FCOC relief valve and also hot valve to ensure adequate oil flow to secondary oil pump
Below +12 deg C	LP FCOC	Max LP fuel heating
+12 to +25 deg C	LP FCOC and HP FCOC	Cold valve action progressive with temperature. Increasing oil cooling
+26 to +84 deg C	LP and HP FCOC	Max oil cooling
+85 to +95 deg C	LP and HP FCOC	Reduced LP fuel heating
Above +95 deg C	HP FCOC	Minimum LP fuel heating

Figure 37 - Fuel cooled oil cooling system schematic diagram and operation.

flow has an open path through the HPFCOC for increased oil cooling. From 25 to 85°C, both valves are fully opened and maximum oil cooling is obtained for this temperature range. When the LP fuel temperature is above 85°C the hot valve begins to close in order to reduce the LP fuel heating, and above 95°C it is totally closed and the HPFCOC is the only active cooler. With this control system, the total heat from the oil is directed to the engine. Notice, however, that an extreme condition is possible, although rare, where the LP fuel temperature may be above 95°C, but the HP fuel still receives the heat input from the oil. The HP fuel temperature could then approach (130°C) with an attendant increase in oil temperature.

5.2.2.2 Combined low pressure/high pressure fuel pumping system: The fuel is delivered to the engine nacelle by the wing tank booster pumps. In order to achieve the full flow rates and delivery pressures which are required for engine operation, the baseline fuel system is provided with a compact combined HP/LP pump system. Both pumps are mechanically coupled to the HP rotor through the accessories gear box. For an installation of the unit, see figure 38.

The compactness and structural integration of both pumps result in some thermal "cross-talk" between the HP and LP pumps. For this reason, the increase in temperature of the fuel across each pump may be slightly different for steady state and transient operation. The characteristic temperature increase of the fuel at low power levels is of the order of 5°C for one pass which, depending on the number of passes (as high as 7), could amount to as much as 35°C.

5.2.2.3 Main fuel flow regulator: The main fuel flow regulator has delicate metering air and fuel orifices to accurately schedule the fuel flow rate for every flight condition. The power dissipation in the unit is negligible and requires special considerations when the fuel temperature becomes too high.

5.2.2.4 High pressure shut-off valve: The HP shut-off valve is especially sensitive to materials incompatibility and carbonaceous fuel deposits. The valve functions as a plunger sliding in a cylinder, and sliding surfaces are highly susceptible to becoming sticky when varnishes or lacquers are formed on their surfaces.

5.2.2.5 High pressure filter: A single high pressure filter assures that the small particulates which are suspended in the fuel will not clog the delicate passages of the distribution valves and injection nozzles thus guaranteeing an even delivery of fuel to all of the injectors.

5.2.2.6 Burner system: The baseline engine is provided with a simplex burner system with a bell-mouth feed arm design (figure 39). The main fuel flow from the distribution valve is delivered to the spray nozzle, and goes through a restrictor plug which meters the correct fuel flow for the delivery pressure. The feed arm is directed radially inwards, and ends in the bell-mouth composed of a swirl chamber enclosed by a deflector cone and a shroud. Six tangential drillings form a multipoint injector and produce a spray with an atomization enhancing swirl. The HP compressor discharge air has a dual path through the deflector cone support plates and the radial air feed slots. Figure 40 shows the simplex burner system integrated with the structure of the combustor head, as well as further details on the main fuel manifold, distribution valve, structural supports, and thermal stand-off distances. The feed arm is enclosed in a cast steel body, the conductance of which is the only resistance to heat transfer from the HP air to the fuel flowing within the feed arm. In those cases where a low fuel flow rate is combined with a high HP discharge

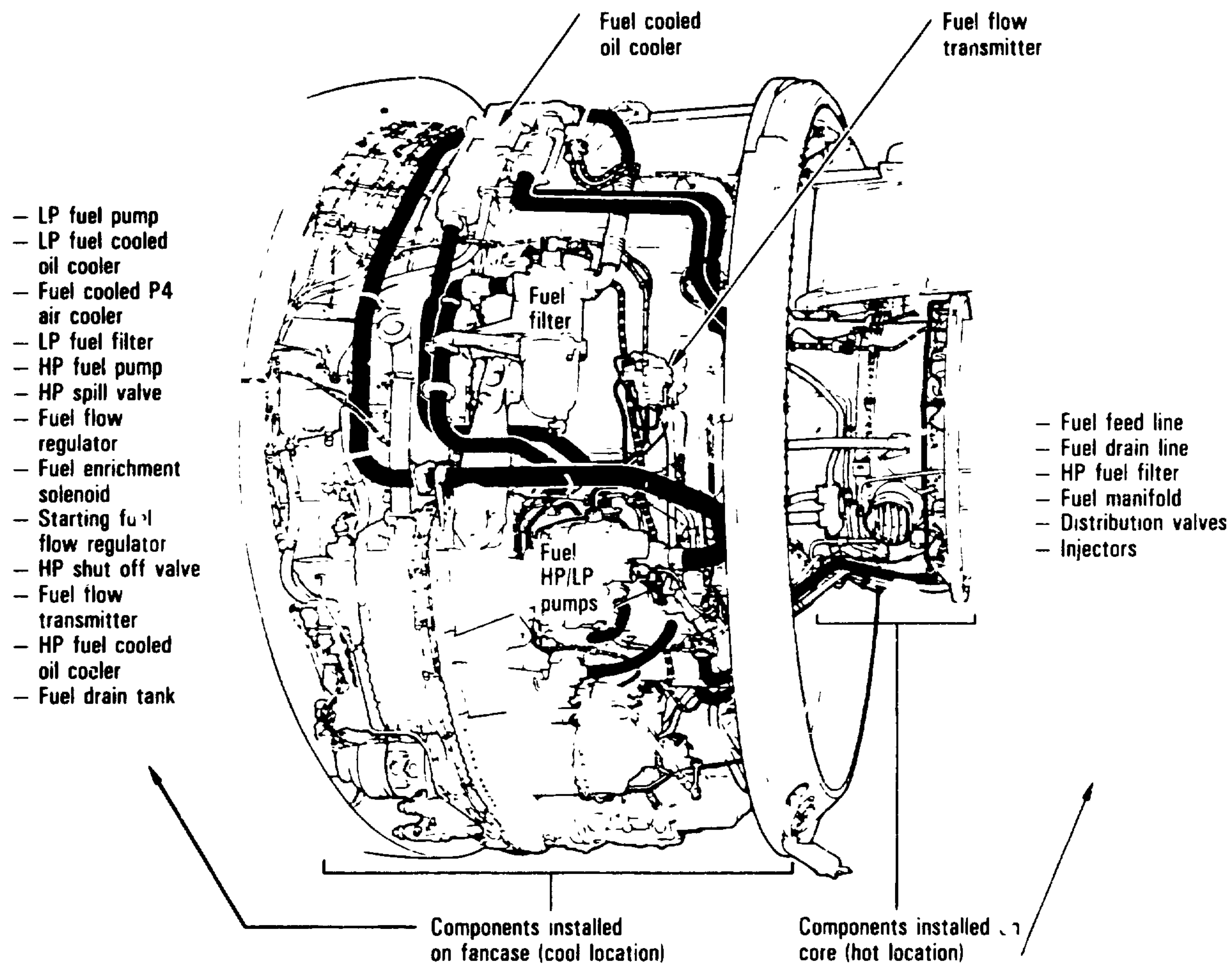


Figure 38 - Location of engine side fuel system components.

Figure provided through the courtesy of
Rolls-Royce, Limited.

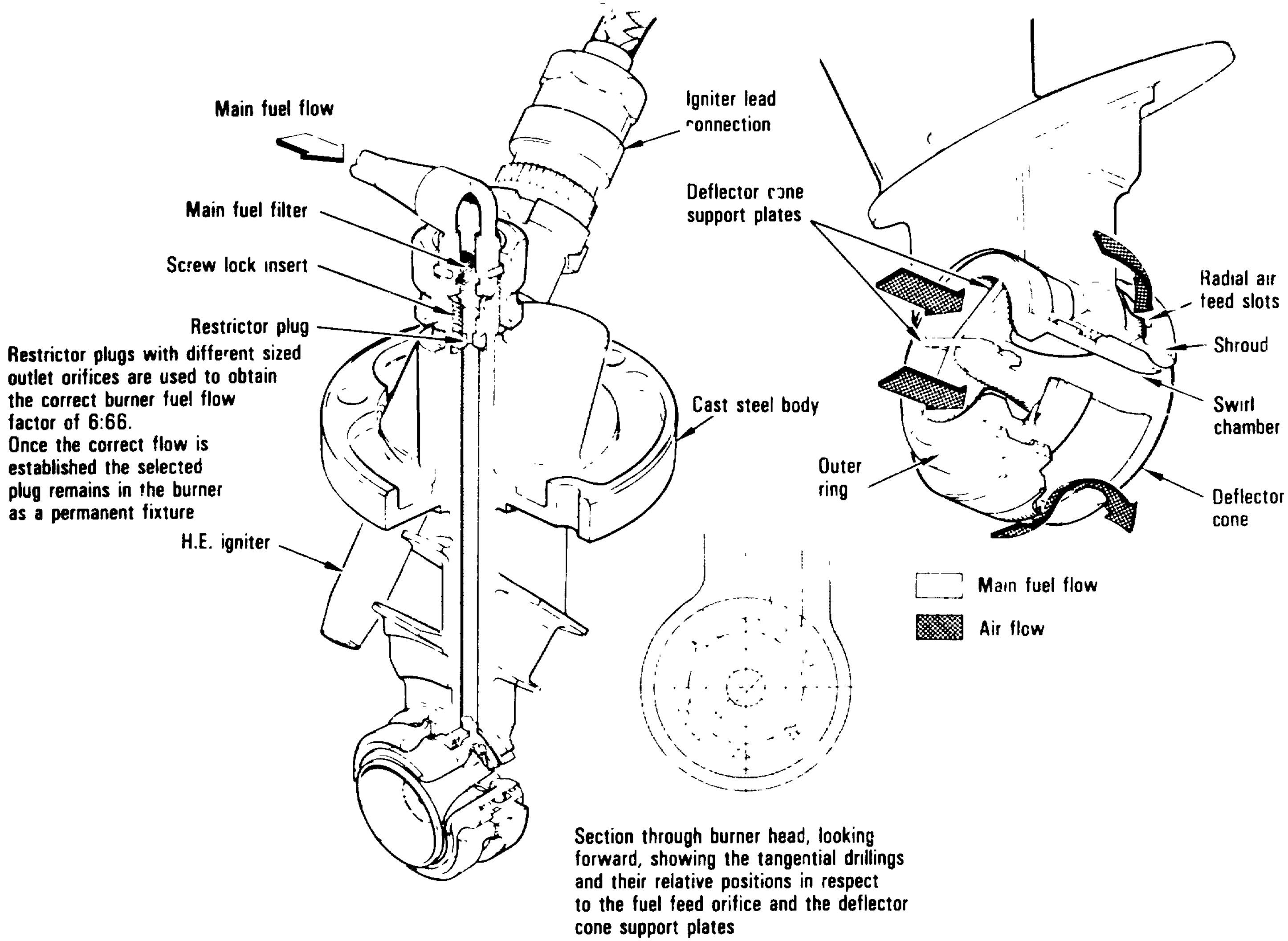


Figure 39 - Simplex burner system - bellmouth burner feed arm with multipoint fuel injection.

Figure provided through the courtesy of Rolls-Royce, Limited.

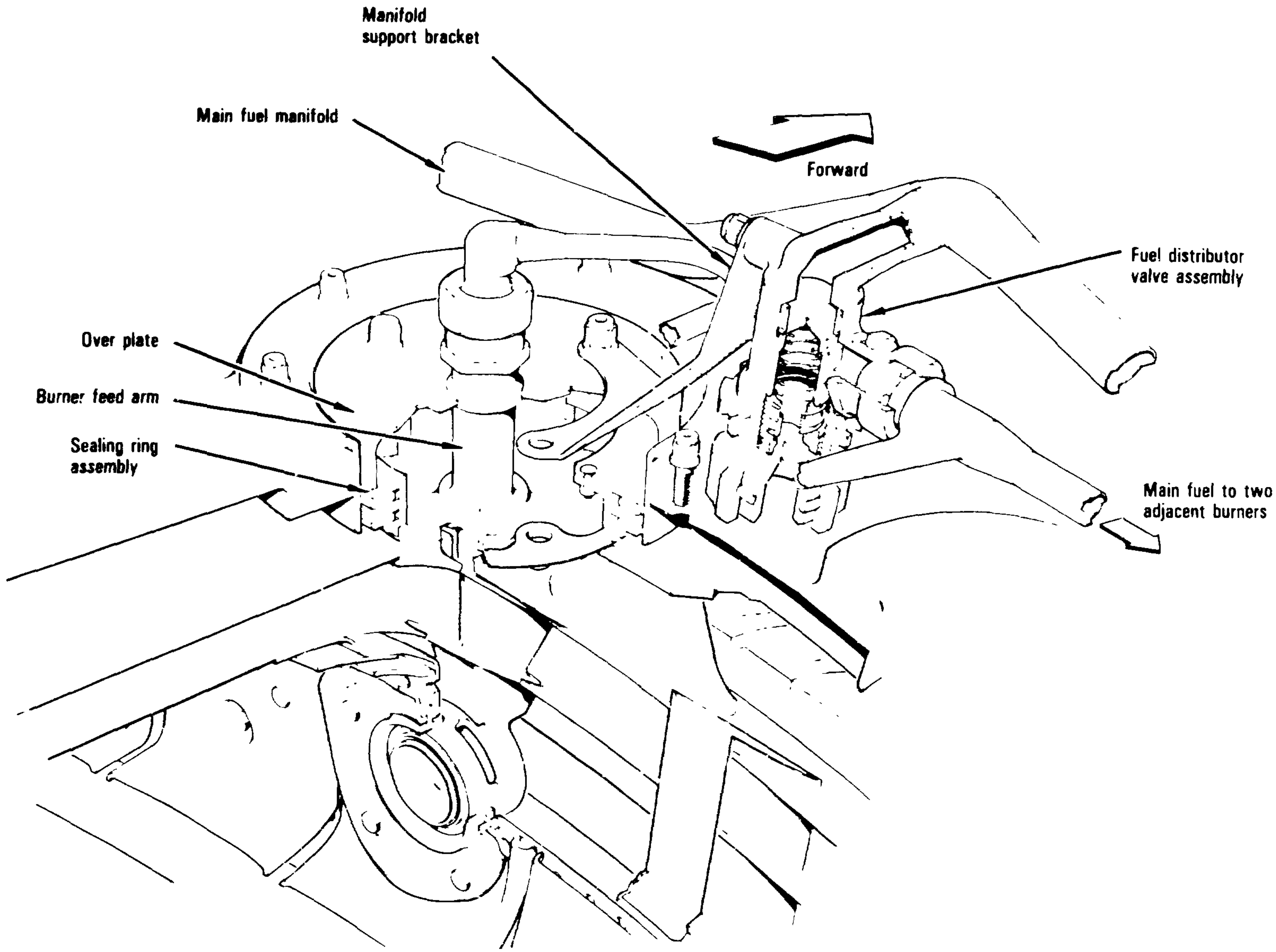


Figure 40 - Combustion chamber head and simplex burner system.

Figure provided through the courtesy of Rolls-Royce, Limited.

temperature, the soakback heat to the fuel in the nozzle could create severe carbonaceous deposits on the tangential drillings. The soakback heat from the combustor head to the nozzle is less severe because of the protective shields installed. The injector restricting plug, as well as the distribution valves, are less sensitive to the soak-back heat because of the much cooler temperatures experienced on the upper plate of the spray nozzles (figure 40).

5.2.2.7 Fuel drain tanks: When the HP shutoff valve is closed, the fuel in the manifold, distribution valves and pigtail conduits is drained down to the level of the lowest position injectors, Nos. 9 and 10. With this procedure, the exposure of fuel for long periods of time to the severe soak-back heat from the core after engine shut-down, is avoided. The fuel is returned by gravity to the fuel drain tank, which is a small spherical container located in the fan case region. The tank is provided with an ejector and a float valve mechanism which redirects the drained fuel back to the low pressure pump when the engine is started again. The float valve blocks any air in the drain tank from entering the fuel system.

5.2.3 Effect on fuel system components. - When a section of the fuel system is experiencing a fuel thermal stability problem, the fuel bulk temperature controls the nucleating rate of chemical reactions responsible for the fuel breakdown, while the component surface temperatures and the fuel velocity control the coking and deposition rates on the component surface. The increase in bulk temperature is caused primarily by the thermal loads in the fan case region, while the highest surface temperatures are experienced in those components which receive the soak-back heat from the HP compressor discharge. The fuel flow in the fan case region is characterized by a low velocity, and moderate surface and bulk temperatures. These conditions are seldom conducive to carbon deposits but may, after many hours of operation, lead to the formation of films on those surfaces which are designed to slide or rest in contact with other surfaces. Since the bulk fuel temperature keeps increasing as it flows through the engine fuel system, the component most likely to suffer from such effects is the HP shutoff valve. Figure 41 shows the temperature profiles at the key sections of the fuel system for the short range, hot day limiting flight. These temperature profiles have been obtained by extrapolating system temperatures, measured during the aircraft certification to the hot day profile of 54°C (130°F) sea level temperature specified for this study. The extrapolation formulae consist of empirical equations developed from previous experience. These temperatures were used to calculate the thermal loads across the components for each engine regime. The temperature increases were then added as appropriate to the fuel tank outlet temperatures reported in 5.2.1. The ground operations from engine start to takeoff were directly adopted from the extrapolated certification measurements and attached to the flight profile to complete the aircraft mission.

An inspection of the figure shows clearly that the HPFCOC outlet fuel temperature (which is nearly the same as the HP shutoff valve inlet fuel temperature) is highest during taxiing and at the beginning of t descent. The other temperatures in the fuel system follow the same trend. This result was unexpected, since it had been reported in previous work (ref. 24) that the beginning of descent was the most limiting condition. The current data show that ground operations are more restrictive (HPFCOC outlet temperature reached between 125 and 128°C during taxiing and ground idle).

Experience has shown that valves using sliding surfaces, such as the HP shutoff valve and the distribution valves, will be free from film deposits, as long as the bulk fuel temperature is kept below 135°C (ref. 25). If the manifold inlet fuel

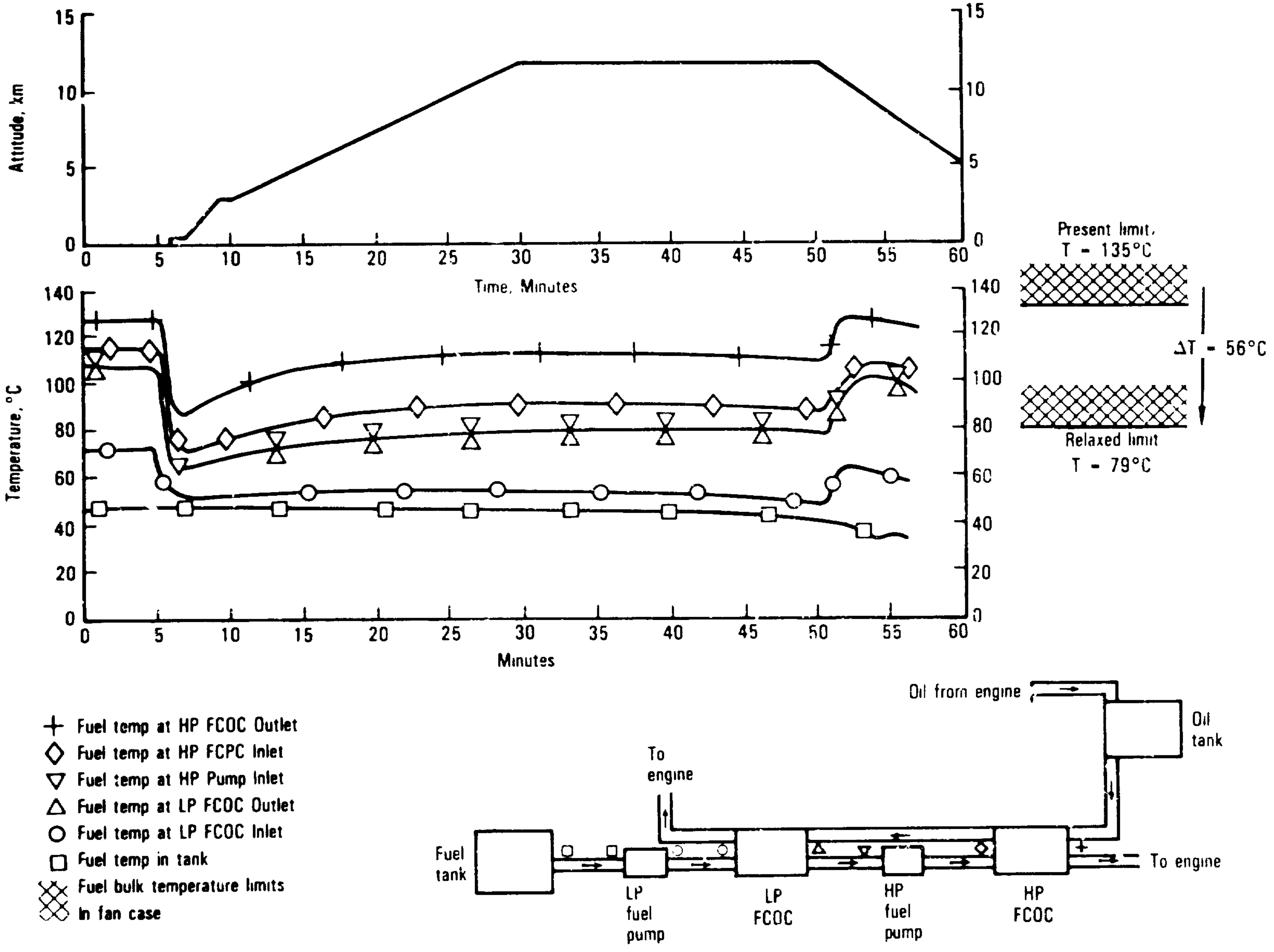


Figure 41 - Fuel system temperature profiles in key sections of fan case fuel system components for the hot day (ambient temperature = 54°C) limiting flight.

temperature is allowed to increase up to 150°C, seizure of the fuel metering valves could occur within 200 hours of engine operation. If one selects 135°C as the upper limit, (tolerable in present Jet A fuels) a relaxation of 56°C in this limiting temperature as derived in Section 5.2 would result in an upper limit of 79°C. On inspecting figure 41 it is seen that this new limit would be surpassed by the HPFCOC outlet temperature for the whole duration of the flight. The HP fuel pump outlet temperature would also exceed this limit for the whole flight except for takeoff and climb up to 3048 m (10,000 ft). The LPFCOC outlet temperature would also be over the limit during ground and descent, and just below the limit during cruise. The LP fuel pump inlet and outlet temperatures would be below the limit for the whole flight.

Further examination of the temperature profiles show that the thermal loads from the oil heat and pumping system are indeed the culprits for the increase in bulk fuel temperature. Notice that the thermal loads are very similar for ground operations and flight idle, and occur during those segments of the aircraft mission when the engine SFC is the highest and the fuel flow rates are the lowest. At cruise, the thermal loads from the oil heat are dominant. At low power levels, the LP fuel pump contributes 30 percent to the total thermal load. The HP pump load remains nearly constant during the whole mission and is the lowest (14 percent) of the total thermal load.

The heat transferred from the oil to the fuel through both heat exchangers has been calculated from figure 41 for the beginning of cruise flight. With the engine fuel consumption of 2005 kg/s and assuming a fuel specific heat of 2.27 kJ/kgK at the fuel temperatures indicated in figure 41, the heat absorbed by the fuel is approximately 58 kJ/s. This represents only 53 percent of the total heat input to the oil at the cruise HP rotor speed of 9440 rpm as indicated by figure 36. At takeoff and climb, the oil heat fraction dissipated by the fuel coolers is even smaller.

The hig[illegible]ng rates on the surfaces are expected to occur in those components whic[illegible]aving the highest surface temperatures, are exposed to the fuel with the highest bulk temperatures. These components are installed in the hot environment of the core region. The temperature distribution along the core for a hot day is given in figure 42. It is measured at the core surface and is not representative of the HP filter, fuel manifold, or distribution valve surfaces, which are mounted on the core surface by means of stand-off supporting brackets. This temperature is representative, however, of the cover plate of the spray fan, as well as of the temperatures in the neighborhood of the metering pl[illegible] of the spray nozzles. For the purpose of this study, a temperature of 300°C has been selected as being representative of the core region.

The burner feed arm is in contact with the HP compressor discharge hot air, and in steady state engine operation, it can be assumed that much of its structure is at the compressor discharge air temperature. This air temperature increases with power levels and flow rates as shown in table 8. At high power, the fuel wetted surface of the feed arm is conside ably cooler than the air and is very near the bulk fuel temperature in the distribution valves. If power is reduced suddenly (aborted takeoff), the fuel flow drops drastically from 7510 down to 680 kg/h. Since at takeoff, most of the external feed arm structure is near 600°C, a severe soakback heat could cause the wetted surface temperature to be excessive. This high temperature situation, however, is expected to last only a short time inasmuch as the feed arm structure is also cooled by the low power HP air.

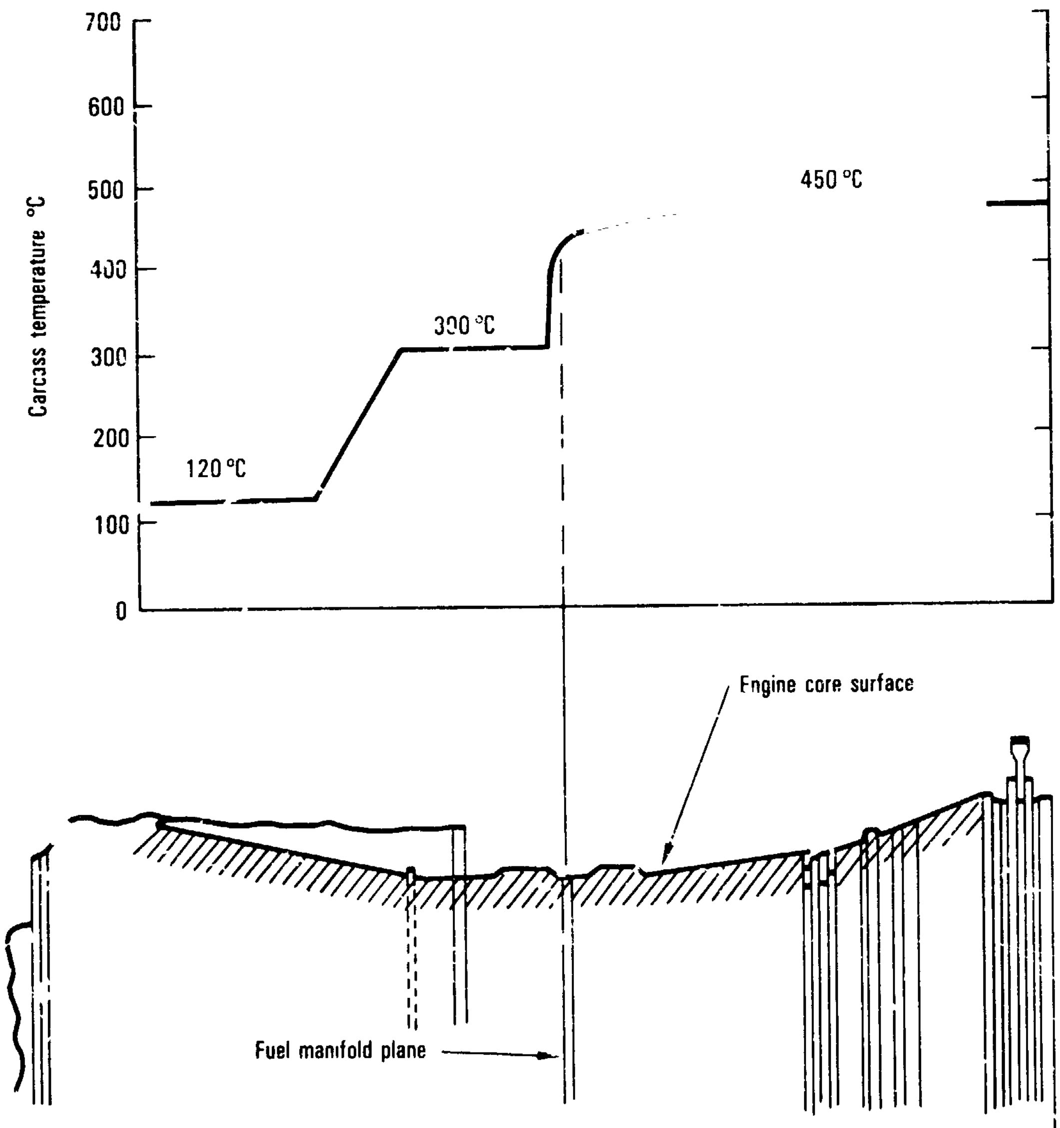


Figure 42 - Measured core temperature distribution for heat rejection rates from an installed engine for an ISA + 39°C hot day at a Mach no. of 0.3, sea level and maximum rpm.

Figure provided through the courtesy of Rolls-Royce, Limited.

TABLE 8 - ENGINE COMPRESSOR DISCHARGE TEMPERATURE AND FUEL FLOW RATES
Hot Day

CONDITION	HP COMPRESSOR DISCHARGE TEMPERATURE, °C	ENGINE FUEL FLOW RATE, kg/h
Max. Cruise 10 668 m (35,000 ft), M = 0.82	507	3,067
Flight Idle 10 668 m (35,000 ft), M = 0.82	300	679
Max. Climb 3048 m (10,000 ft), M = 0.5	561	7.085
Flight Idle 3048 m (10,000 ft), M = 0.5	303	870
Takeoff Sea Level, M = 0	602	8,562

The thermal environment in the feed arms and the spray nozzle cover plates are much hotter than the fuel at the manifold inlet. Cooling of the fuel wetted surfaces is left entirely to the fuel flow itself. Because these components are usually over-designed, the system has little difficulty in keeping the surfaces within 20 or 40°C of the fuel bulk temperature, even at the low flow rates encountered at ground and flight idle. Projections can then be made that, as far as the injector surface temperatures are concerned, the system can absorb a 56°C drop in the JFTOT limit. This, however, is only an estimate, since there is no available temperature data in the close proximity of the feed arm, other than the already reported temperature. Ref. 24 recommends that if a maximum manifold inlet fuel temperature of 102°C is adopted, a modern high bypass ratio turbofan engine can operate free from severe coking rates in the injectors. Referring again to figure 41, it is apparent that present Jet A fuels would already show carbon deposits in the baseline fuel system nozzles when operating on the selected limiting hot day flight. There is, on the other hand, considerable experience in operating the L-1011 aircraft in hot climates at temperatures above 30°C. The temperature in the manifold routinely reaches values above 102°C and no nozzle fouling has been encountered as yet. This experience tends to indicate that the simplex burner system adopted for the baseline aircraft engine is very resistant to nozzle fouling in high temperature operations.

5.2.4 Effect on baseline aircraft performance. - The previous analysis of the effects of operating the aircraft on a hot day and short flights, while using a jet fuel with a JFTOT rating of 204°C, has shown that: (1) the fuel bulk temperature would continuously exceed, during the whole mission, the new engineering limit proposed in Section 5.2.3 of 79°C that must be adopted as a safety margin in order to prevent the formation of varnishes and lacquers. (2) Ground operation at idle and flight idle operation at the beginning of letdown have been identified as the most stringent conditions of the mission. (3) The formation of carbon deposits in the injectors appears to be related to high manifold inlet temperature, and the bulk fuel temperatures are most probably within 20 to 40°C of the wetted metal surfaces.

The results are conclusive. A relaxation of the JFTOT rating down to 204°C would halt operations of the aircraft, not only in hot weather, but even at ground level temperatures as low as 20°C because the distribution valves, HP shutoff valve, and fuel metering devices would almost certainly experience seizure due to the formation of varnishes and lacquers.

A corollary question is - how much could the present JFTOT rating of 260°C be relaxed before operating the aircraft in hot weather and short flights become a problem? The fuel system for the baseline aircraft was designed to take full advantage of any opportunity for improving the SFC. For this particular design the fuel bulk temperature at the manifold inlet is as high as possible without pushing the practical thermal breakdown limit. The formation of varnishes and lacquers on the component surface located in the fan case, as well as on the distribution valves, is a cumulative process, and its presence is felt after many hours of operation. How long it takes to detect them depends on how often the aircraft operates in hot climates, as well as the time that the aircraft spends in ground operations and descent. From this point of view, a statistical approach appears more reasonable for the analysis, rather than the single most-limiting flight approach considered above.

Assume that the aircraft operates 500 hours, servicing routes in a hot climate (for instance, the Southeast United States summer). It is probable that during this period the aircraft could be operating at ground temperatures above 32°C (90°F). Taking 32°C as the ground temperature, down from 54°C used for figure 41, a maximum value of 107°C is projected at the outlet of the HPFCOC. This is also approximately the temperature that the fuel shutoff valves and the distribution valves will be experiencing. If this value is taken as the new practical limit, it translates into a relaxation of the JFTOT down to 232°C.

The estimated lowest JFTOT breakpoint temperature of 232°C at which the aircraft could operate for a sequence of hot days at ground temperatures not greater than 32°C, must be taken with great caution. Much more work is necessary to pinpoint the coking rates at low temperatures which happen to be the prime culprits of the limits imposed to the aircraft.

The limits described above refer to slow, long term deposits on the fuel components. There are situations where the operation of the aircraft (even for short periods of time) in very hot climates could induce injector nozzle fouling in a matter of minutes. For example, after an aborted takeoff, the bulk fuel temperatures are very high and simultanously the structural member of the feed arm is very hot. This could result in carbonaceous deposits on the feed arm at a very high rate if the aircraft is left to idle on the runway shortly after the aborted takeoff.

5.3 Aromatic Content

A significant impact of utilizing fuels with altered chemical composition from conventional petroleum fuels is the effect on the solvent properties of the fuel. Since these fuels must come in contact with certain nonmetallic aircraft components such as seals, hoses, etc, the effect of altered aromatic content of shale and coal derived refined fuels must be addressed. The fuel mixture, treated as an organic solvent, causes swelling of polymers, including plastics and rubber material. Thus, a correlation of fuel solvent action with various polymers provides a useful prediction of detrimental effects of fuel on plastics, elastomers, and other polymeric materials.

The most useful method of predicting polymer solubility is the concept of the solubility parameter δ. The solubility parameter is derived from the chemical and physical properties of a material. In this case, it is a measure of the compatibility in a given solvent of one material relative to another, or of a given material in one solvent relative to another. The solubility parameter values may be used to predict solubility (or insolubility) or swelling of polymers in solvents. In general, solubility or swelling is greatest when the solubility parameter of the solvent is within 0.5 unit of that of the polymer, and diminishes substantially when δ is greater than 1.0 unit from that of the polymer. A few representative values of the solubility parameter for some solvents and elastomer materials are shown in table 9. (ref. 26).

TABLE 9 - SOLUBILITY PARAMETERS OF TYPICAL HYDROCARBONS FROM FUEL

Solvent	δ	Polymeric Elastomers	δ
Iso-octane (aliphatic)	6.85	Natural rubber	8.35
N-decane (aliphatic)	7.75	Polybutadiene	8.45
Benzene (aromatic)	9.15	Buna S	
Toluene (aromatic)	8.9	85%B 15%S	8.55
O-Xylene (aromatic)	9.0	60%B 40%S	8.0
		Neoprene	9.25
		Buna N	9.5 - 9.6
		Polysulfide rubber	9.0 - 9.4

Application of this principle is seen in table 10, which shows the swelling of natural rubber in various solvents. The table shows that the percent swell, Q_{obs}, is largest when the solubility parameter, δ, of the solvent is within $\pm$ 1 unit from that of the rubber, 8.35. As the solubility parameter of the solvent deviates further from that of the rubber, the swelling reduces markedly.

With current technology, compositions of fuels can be determined with considerable accuracy by gas and/or liquid chromatography from which accurate values for solubility parameters may be calculated. A wide variety of polymeric products, such as seals and O-rings, with varying chemical composition, is available to provide desirable seal compatibility with fuels as required. Thus, current commercially available materials will provide adequate compatibility with fuels having aromatic contents to 35 percent. Therefore, polymeric mechanical seal materials may be selected to accommodate the higher aromatic content.

5.4 Viscosity

The viscosity of a fluid is a measure of its internal resistance to motion impressed upon it by external forces. To the fuel system designer this effect is of primary interest in sizing fuel lines and establishing the necessary driving force at the fuel source to insure that fuel is delivered to its destination at specified pressure levels and quantities. A less obvious effect, but one of significant concern, is its effect on heat transfer where fuel becomes the sink for cooling the engine oil. In an existing design, such as the baseline L-1011, the fuel line sizes and components have already been established. Consequently, the increase in viscosity can only be evaluated in terms of its effect on existing component and aircraft performance.

TABLE 10 - SWELLING OF NATURAL RUBBER IN VARIOUS SOLVENTS (ref. 26)

(δ = 8.35)

	δ	% Swell Q_{obs}
	Hydrocarbons	
n-Pentane	7.05	1.12
n-Hexane	7.3	1.18
n-Octane	7.55	2.34
Benzene	9.15	3.95
Toluene	8.9	4.10
m-Xylene	8.8	4.15
Mesitylene	8.8	3.25
Limonene	8.5	4.00
	Ketones	
Acetone	9.9	0.03
Methyl ethyl ketone	9.3	0.71
Diethyl ketone	8.8	1.6
Diisopropyl ketone	7.6	1.9
	Alcohols	
n-Propyl alcohol	11.9	0.02
tert-Butyl alcohol	10.6	0.13
Amyl alcohol	10.9	0.07
n-Hexyl alcohol	10.7	0.15
n-Heptyl alcohol	10.6	0.55
n-Octyl alcohol	10.2	0.85
	Nitriles	
Acetonitrile	11.9	0.04
Propionitrile	10.6	0.06
Capronitrile	9.4	0.70
Benzonitrile	8.35	2.0
	Nitro Compounds	
Nitromethane	12.7	0.03
Nitrobenzene	9.95	1.15

5.4.1 Effect on fuel system components. - The fuel system components affected by increased viscosity include: fuel lines, fuel pumps, heat exchangers in which fuel is one of the heat transfer fluids, and fuel nozzles. The relaxed fuel viscosity is compared to ASTM D 1655 JET A commercial kerosene as a function of temperature in figure 43. The effect of fuel viscosity on component performance is greatest at low fuel temperatures and is almost negligible at high fuel temperatures.

5.4.1.1 Fuel lines: The effect of fuel viscosity on fuel line losses as affected by fuel temperatures is illustrated in figure 44 which compares the pressure drop per meter ($\Delta P/L$) in two sizes of engine feed lines used in the L-1011-500 airplane. During cruise, with a fuel temperature of -40°C, the higher viscosity fuel causes a 250 percent increase in pressure drop in the 3.81 cm (1.5 in.) line supplying fuel to the wing engines and a 350 percent increase in the 5.08 cm (2 in) line supplying the No. 2 engine. However, at temperatures above 15.0°C (60°F), these line loss ratios are reduced to less than 11 percent and 14 percent respectively.

5.4.1.2 Fuel pumps: The primary effect of fuel viscosity on pump performance is a result of the increased discharge pressure required to overcome the increased plumbing line losses. The centrifugal pumps used as boost pumps in the L-1011 fuel tanks and at the engine inlet, will experience some reduction in discharge pressure, fuel flow rate and pump efficiency if a higher viscosity fuel is used. Aircraft pump manufacturers assume these losses to be negligible. However, a method of correcting centrifugal pump performance for viscosity has been developed by the Hydraulic Institute (reference 27). Assuming this method of correction can be applied directly to the L-1011 fuel pumps, pump performance may be degraded by as much as 1.5 percent in output pressure, 4 percent in fuel flow rate, and 17.7 percent in pump efficiency if the higher viscosity fuel is used in cruise in lieu of Jet A kerosene at -40°C.

5.4.1.3 Heat exchangers: Assuming fully developed channel flow in the fuel side of the L-1011 engine oil cooler, the film transfer coefficient (h) may be determined from the following relationship:

$$Nu = 0.0225\ (Rn)^{0.8}\ (Pr)^{1/3}$$

where:

$$N_u = \frac{h\ D_H}{k}$$

$$R_n = \frac{V\ D_H}{\nu}$$

$$P_r = \frac{\rho \nu\ C}{k}$$

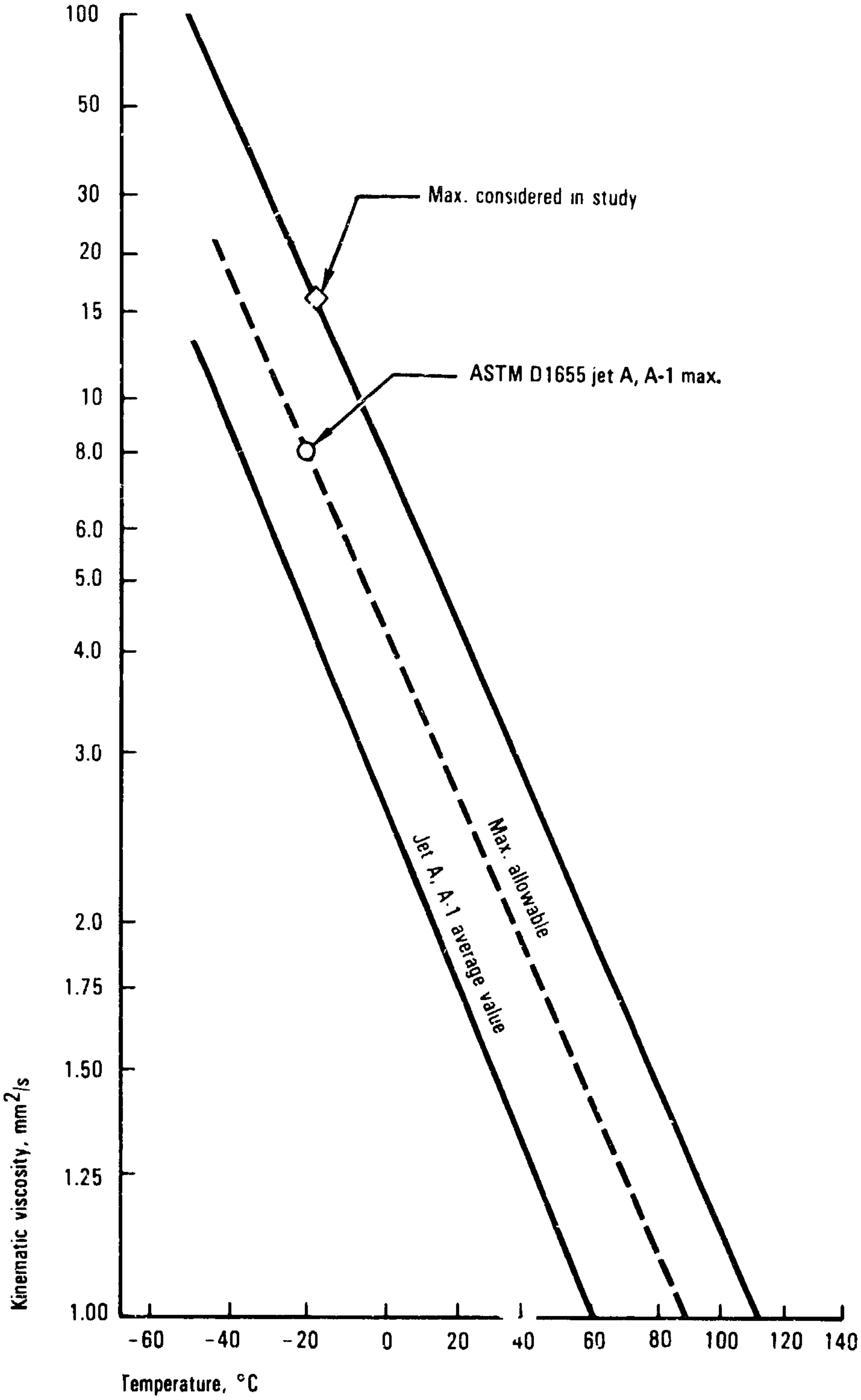


Figure 43 - Commercial aviation jet fuel viscosity.

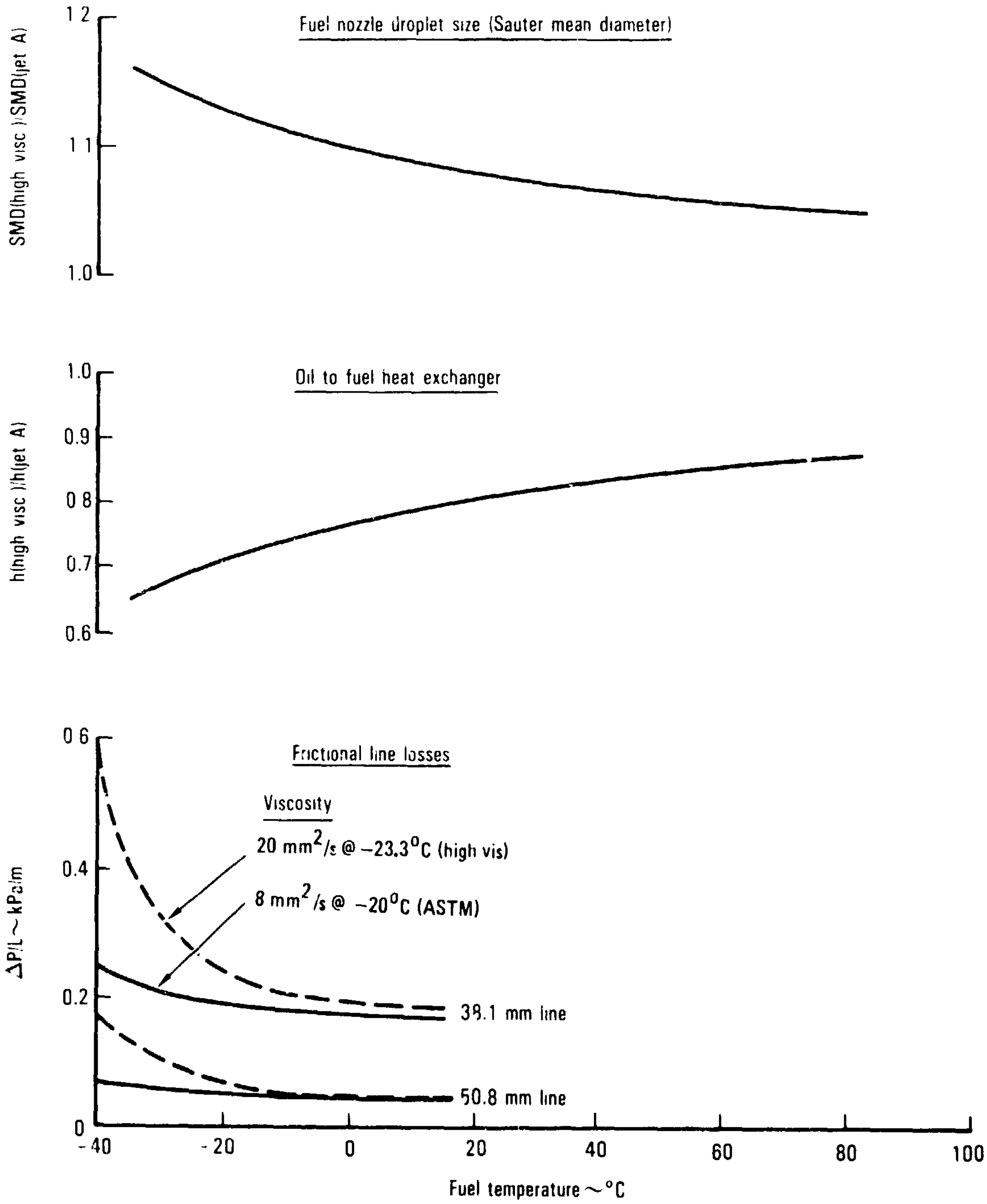


Figure 44 - Effect of viscosity on component performance.

Assuming the flow rate (V), hydraulic diameter (D_H), thermal conductivity (k), density ( ), and specific heat (Cp) are the same for the Jet A kerosene and the higher viscosity fuel, the effect of viscosity (ν) on film transfer on the fuel side will be:

$$h\ (\text{high vis.})/h\ (\text{Jet A}) = [\nu(\text{Jet A})/\nu\ (\text{high vis.})]^{7/15}$$

This relationship is shown on figure 44 as a function of ambient temperature. In the range of critical fuel temperatures above 93.3°C (200°F), the fuel side heat transfer coefficient using the higher viscosity fuel will be down approximately 12 percent relative to Jet A fuel.

5.4.1.4 Fuel nozzles: The impact of viscosity on the Simplex Pressure Swirl Atomizer nozzle used in the L-1011 engines is also shown in figure 44. The comparison is based upon an empirically determined equation for Sauter Mean Diameter (SMD) of fuel droplets reported in reference 28:

$$SMD = 4.4\ \sigma^{0.16}\ \nu^{0.6}\ W_f^{\ 0.22}\ \Delta P_f^{\ -0.43}$$

Assuming the fuel total pressure drop across the nozzle does not change and using compatible values for surface tension (σ) and fuel flow rate (W_f) the droplet size for the high viscosity fuel at combustor inlet fuel temperatures will be approximately 5 percent larger than those using Jet A fuel.

5.4.2 <u>Effect on baseline aircraft performance</u>. - The impact of higher viscosity fuel in the baseline aircraft performance will be most significant at low fuel temperatures. As long as the tank-mounted boost pumps are operating, however, the higher viscosity will not effect engine performance. At sea level takeoff, on a -40°C day, the engine fuel flow is 7597.8 kg/h (16,750 lb/h). Assuming a -40°C fuel, the pressure at the most critical engine, which is mounted in the aft fuselage, may be determined from the following equations:

$$P_{\text{eng. in}} = P_{\text{pump out}} - \Delta P_{\text{fuel line}}$$

The aft engine inlet pressures will vary with fuel viscosity and number of tank-mounted boost pumps in operation as shown in table 11.

From the above, it is apparent that the aft engine can operate at takeoff power with either high viscosity or Jet A fuel as long as the tank boost pumps are operating. With boost pumps inoperative however, the analysis indicates that the aft fuselage engine could not achieve takeoff power with the high viscosity fuel. A similar analysis with boost pumps off for the wing engines shows a fuel inlet pressure of 83.4 kPa absolute (12.1 psia) at the engine inlet indicating no compromise in engine thrust using high viscosity fuel. The higher engine inlet pressure results from the short supply lines and positive fuel head because the engines are mounted below the wing tanks.

5.5 Lubricity

Fuel lubricity has not been a serious problem in the past. However, the trend to moderate and severe hydroprocessing of syncrudes in the future can result in reduced lubricity for aviation jet fuels. The effects on the baseline aircraft components and aircraft performance are discussed in the following paragraphs.

TABLE 11 - AFT ENGINE FUEL INLET PRESSURE

Fuel	Two Pumps		One Pump		No Pumps	
	Ppump, gauge kPa (psig)	Peng, abs kPa (psia)	Ppump, gauge kPa (psig)	Peng, abs kPa (psia)	Ptank, gauge kPa (psig)	Peng, abs kPa (psia)
Jet A	251.7 (36.5)	268.9 (39.0)	225.5 (32.7)	242.7 (35.2)	0 (0)	17.2 (2.5)
High Vis	248.2 (36.0)	237.9 (34.5)	222.1 (32.2)	211.7 (30.7)	0 (0)	-10.3 (-1.5)

5.5.1 Effect on fuel systems components. - Since most fuel system components rely on the lubricating qualities of fuel to minimize friction between moving surfaces in contact with each other, it is to be expected that reduced lubricity will result in increased wear with the attendant reduction in component life. In the baseline aircraft such components include the engine high and low pressure pumps, fuel flow regulator, high pressure fuel shutoff valve, starting fuel regulator, the airframe tank-mounted boost pumps, Tanks 2L/2R flow proportioner, and system shutoff valves. In addition to reduced life for all such components, controls which modulate the fuel flow in the engine may experience sticking or sluggish operation. This latter effect has been observed on a J-79 engine installed in an F-104 airplane which had been refueled by fuel which had its lubricity reduced by passing through clay filters. However, the problem disappeared when the same aircraft was supplied fuel which had bypassed the clay filters. Evidence that the life of some of the baseline L-1011 fuel system components will be reduced by operation with low lubricity fuel has been demonstrated by excessive gear tooth and bearing block erosion in the engine fuel system high pressure pump. Further investigation revealed that the fuel used in these aircraft had been subjected to severe hydroprocessing. Subsequent design modifications to the pump increased its life to an acceptable level while operating with the equivalent low lubricity fuel.

5.5.2 Effect on baseline aircraft performance. - The long term effects of low lubricity fuel on the baseline aircraft is to increase the frequency of component and system maintenance activities with the attendant cost impact. Of more immediate concern, however, is the potential results of sluggish or sticking fuel controls. This could cause safety hazards primarily in takeoff, landing and ground maneuvering where quick response to control movements are sometimes required to avoid accidents.

5.6 Other Fuel Properties

5.6.1 <u>Water separation</u>. - Water in the fuel can become a problem if it exceeds the quantity which will remain in solution. Since its solubility increases with fuel temperature, water saturated fuel taken aboard in a hot humid climate can release significant quantities of free water as the fuel is cooled in flight. As the fuel temperature drops below the freezing point of the free water, small crystals of ice are formed which can remain suspended in the fuel for long periods of time. As they are drawn into the engine fuel system, these fi ice crystals can block fuel filters and cause some system malfunctions. However, a l commercial jet transports must be capable of sustained operation under the most critical conditions for water freeze-out in the fuel. In most cases, this condition is met by using the engine oil heat rejection to ensure that the fuel is well above the freezing point of water before the fuel reaches filters where ice crystals could block the filter.

A more hazardous condition can be encountered if free water is supplied to the aircraft through malfunctioning ground equipment. If the water quantities are large and improperly trained crews neglect to sump the aircraft tanks prior to takeoff, large slugs of water can cause loss of engine thrust. A contributing factor to this condition can be the use of lubricity additives in fuels when severe hydroprocessing has removed much of the lubricating qualities of the fuel. Such additives frequently act as surfactants which reduce the water removing capability of the ground fuel supplier's filter/separator equipment.

5.6.2 <u>Electrical conductivity</u>. - The electrical conductivity of the fuel is an important consideration when evaluating the potential of fuel tank fires while refueling the airplane. As the fuel is transferred from the ground equipment to the aircraft, an electrostatic charge is picked up by the fuel as it passes through the refueling system, especially the ground filter/water separation equipment. This charge is carried to the aircraft fuel tanks where it is gradually dissipated by conduction through the fuel to the oppositely charged airframe. Since charges which are driven to the fuel surface are not neutralized by airframe charges, a surface voltage difference between the fuel surface and upper wing skin develops. If this surface voltage exceeds the breakdown voltage in the ullage space, a spark discharge occurs which can ignite combustible fuel vapors causing an explosive reaction.

If the fuel conductivity is high enough to cause a rapid charge relaxation, the fuel surface charge does not become a potential ignition source. Consequently, a considerable effort is expended by the fuel handler and aircraft operator to insure that the fuel conductivity is at a safe level.

Fuels produced from shale and syncrudes will be subjected to severe processing in the refining operations to remove molecular nitrogen and excessive sulfur. If this processing removes most of the polar compounds, it could reduce the electrical conductivity level to a point where static discharge could become a serious hazard. From a safety point of view, this condition would not be tolerated. Consequently, anti-static additives, which are readily available, would become mandatory.

5.6.3 <u>Flash point and vapor pressure</u>. - Relaxation of the flash point and increasing the vapor pressure of commercial jet fuels can provide a significant increase in jet fuel availibility. Because such changes can increase the possibility of fire, the industry has evidenced a great reluctance to relax these properties. Increasing the vapor pressure can also result in fuel boil-off losses It is probable that contemplated changes in these properties would be well within the present limits of ASTM D 1655 Jet B fuel. Since most commercial jet fueled aircraft, including the L-1011, have been certified to operate with Jet B fuel, none of the fuel system components would be affected by the change.

Jet B fuel vaporizes more readily than Jet A and forms a combustible mixture from approximately 11°C down to -28°C at sea level under stable conditions. However, under dynamic conditions, the lower flammability limit can extend to -65°C. Jet A fuel, which is less volatile than Jet B fuel, has a combustible range from 84°C down to 41°C under stable conditions and down to 5°C under dynamic conditions. It is apparent that the present combustible range of Jet A fuels under stablized coditions has a minimal overlap with normal operating temperatures. Increasing the vapor pressure would increase this overlap with the attendant increase in fire hazard.

6. CONCEPTUAL DESIGNS

This section provides a discussion of the modifications that could be made to the baseline aircraft's fuel system to permit the aircraft to operate with relaxed fuel properties at the critical environmental condition. Of the eight fuel property changes discussed in the preceding section, only two require that major modifications be made to the baseline aircraft design to avoid operational limitations in the extreme environments. These are; 1) the increase in freeze point to -20°C for operation in a cold environment and 2) the decrease in thermal stability for operation on a hot day. Minor modifications may be required to accept fuels with the proposed changes in aromatics, viscosity, lubricity and water separation. Electrical conductivity, flash point and vapor pressure changes are not expected to require any changes to the baseline aircraft design. Accordingly, the major emphasis in this section will be conceptual design modifications which can accommodate the higher fuel freeze point and the lower fuel breakpoint temperature. Several alternative approaches are discussed. The more promising of these are evaluated in terms of their impact on aircraft performance. Each concept was evaluated on the basis of its effect on block fuel weight, increase in takeoff gross weight (TOGW), and percent increase in specific fuel consumption (SFC). Finally, several recommendations are offered.

6.1 Freeze Point

6.1.1 <u>Description of candidate systems</u>. - The use of the proposed -20°C freeze point fuel requires modifications to the baseline aircraft to prevent fuel freezing in the fuel tanks and to control fuel freezing in the fuel distribution system. Systems for the prevention of freeze-out by heat addition in the fuel tanks and distribution system with and without insulation are discussed in the following sections.

6.1.1.1 Fuel tank heating without insulation: One means of preventing fuel freeze-out is to heat the fuel by an amount sufficient to compensate for the aerodynamic cooling. There are several possible sources of heat for the fuel tanks. These include; engine exhaust heat, engine bleed air, engine oil, and electric power. The advantages and disadvantages of most of these heat sources have been considered in previous studies (ref. 29) and therefore will not be discussed in detail in this report. The use of engine exhaust heat has the least impact on the engine specific fuel consumption. Its disadvantages are the large weight penalty associated with having a tailpipe heat exchanger and the necessity of a secondary heat transport fluid to meet safety requirements. Bleed air as a source of heat has the highest impact on fuel consumption. It has the additional disadvantage of requiring a bulky air-to-fuel heat exchanger. Engine oil would be the simplest and most natural selection as a source of heat. Unfortunately, this heat source does not have sufficient capacity to prevent fuel freeze-out. The use of electrical power has the advantages of having: 1) a lower impact on fuel consumption than the use of bleed air and; 2) a lower installation weight and less complexity than the use of exhaust heat. In its present configuration, the electrical system of the L-1011 does not have sufficient generator capacity to provide enough heat to prevent fuel freeze-out in the aircraft's non-insulated fuel tanks. However, the electrical system can be modified to increase its capacity.

Heat can be input into the fuel by either of two methods, 1) by a bulk fuel heater located near the center of the tank and 2), by a heating system which applies heat directly to the bottom of the tank. The first method has the advantage of simplicity. A single, centrally located, heater can use electric power, compressor bleed air, or engine exhaust energy to heat the fuel. The fuel circulation caused by the existing boost pump scavenger system would tend to maintain a roughly uniform temperature within the bulk of the fuel. However, fuel temperature profiles obtained in flight tests show that the fuel near the tank's lower surfaces is generally much colder than the bulk fuel. It is in this cold layer of fuel that fuel freeze-out would first occur.

To prevent freeze-out in the fuel adjacent to the tank lower surface with the first method, it would be necessary to maintain the bulk fuel at a temperature that is considerably above the fuel freeze point. The elevated bulk fuel temperature would result in a higher rate of heat loss from the fuel and correspondingly higher demand for heat from the fuel heating system.

The second method by which heat can be input into the fuel is to use a heating system that covers the bottom of the fuel tanks. The advantage of this method is that it concentrates the heat in the coldest layer of fuel and results in a more uniform temperature throughout the fuel. Natural convection produces sufficient mixing even when the boost pump scavenger system is not operating. The most important advantage of this method, is that it requires much less heat to prevent fuel freeze-out than the first method. If the tanks, bottom surfaces are maintained at a temperature equal to or slightly higher than the fuel freeze point this is sufficient to prevent freeze-out in the fuel tanks. Although the bulk fuel temperature may drop somewhat below the freeze point during long range flights because of heat loss through the upper skin, any freeze-out that occurs in the bulk will eventually drop to the bottom and be melted. Because the bulk fuel temperature is lower with the second method than with the first, the overall heat loss from the fuel tank is much lower and the heat input required from the fuel heating system is considerably less. The large difference in heat requirement between the two methods favors the bottom surface heating method.

The selection of the tank bottom surface heating method leads directly to the choice of electric power as the source of heat. The other sources of heat are incompatible with the bottom surface heating method since they would require a large number of small, independently controlled heaters. Two types of electric heaters were considere ; linear element heaters and foil heaters. Linear element heaters have the advantage of being relatively easy to install. However, because cold spots would occur between the heater elements, they would have to be placed rather close together and would have to be maintained at a higher temperature than foil heaters which provide the most even heat, are lighter in weight, and can be fit into the irregular shapes between the ribs and stringers attached to the tanks, bottom surfaces. Foil heaters were selected for use in the recommended system.

The electric foil heaters incorporating fuel resistant Kapton insulation would be applied directly to the bottom surfaces of the tanks in the areas between the ribs and stringers. Figure 45 shows a typical heater installation. The heater system for each tank is made up of an array of separately controlled panels that cover the bottom of the tank. Each panel consists of a series of heating elements typically made up of the area bounded by a pair of ribs in the spanwise direction and as many as five stringers in the chordwise direction. These heating elements are interconnected electrically and controlled as a unit by means of a centrally located temperature sensing device and a fuel sensing device. The temperature of each panel is independently maintained at a preset temperature. For this study, a temperature of +3°C above the fuel freeze point was selected in order to provide a margin for

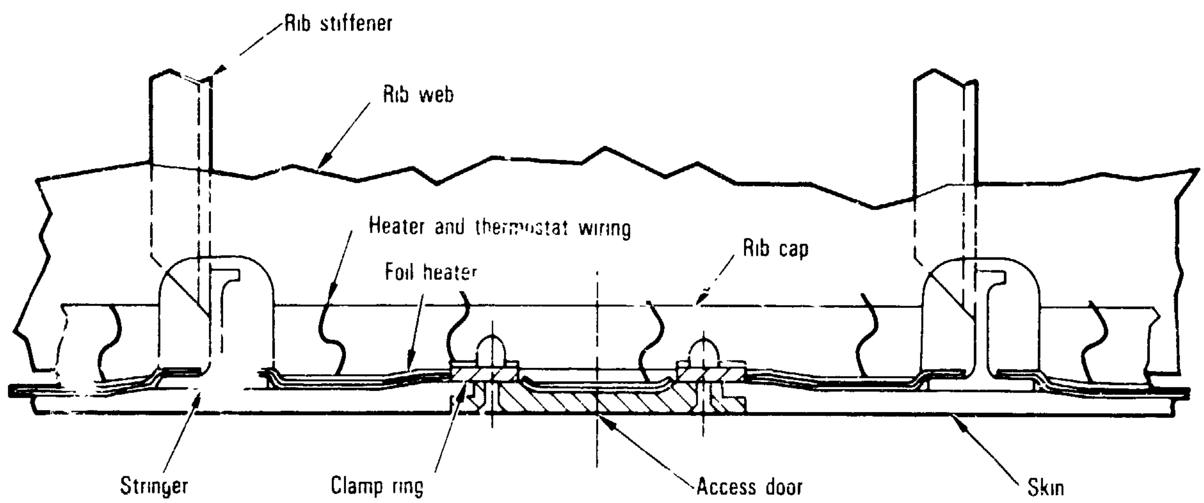


Figure 45 - Fuel tank electric foil heaters w/o insulation (for all tanks).

tolerances in the temperature control system. When, as a result of fuel usage, panels become uncovered by the receding fuel, the uncovered panels are automatically cut out of the system to minimize the heat loss and preclude excessive surface temperatures in the heating panels.

The present electrical system in the baseline aircraft has approximately 101.5 kVA excess capacity to supply the fuel tank heating system in addition to the normal aircraft electrical loads under the conditions of the cold day missions. Electrical heating power requirements predicted by the fuel tank thermal model indicate a peak power requirement for the heating system of 270 kVA (figure 46) for the 9260 km (5000 n.mi.) cold day mission. This excessive power requirement is a valid reason for questioning the selection of a -20°C freeze point. However, the peak power requirement after 8.5 hours exceeds the available excess capacity for a freeze point of -30°C (figure 47) and appears to be marginal if extrapolated to -35°C. Therefore, the -20°C freeze point is retained in this study since the weight and cost systems are essentially identical for any of these freeze point levels.

Three alternative modifications to the electrical system were considered. The first modification involves replacing the three present 75/90 kVA engine driven generators with three 175/220 kVA generators. This constitutes a major change in the L-1011's primary electrical system and, as such, produces several significant problems. First, the entire electrical system would have to be extensively redesigned. Second, the aircraft's new primary electrical system would have to be requalified by the FAA, a process which is costly and time consuming. Finally, limitations on the torque capacity of the engine's generator drive system, the structural load capacity of the generator mounting pads, and the available space inside the engine nacelle may make this modification practically impossible to implement.

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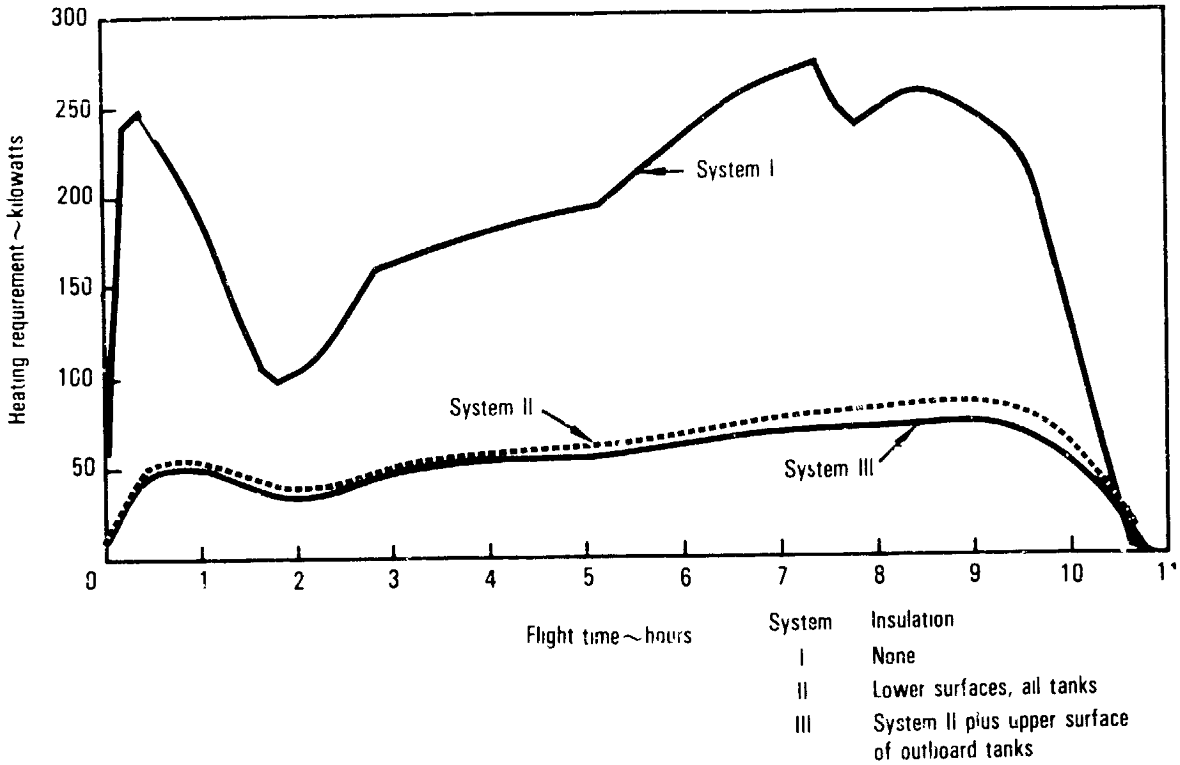

Figure 46 - Predicted fuel tank electrical heating power requirements - 9260 km (5000 n.mi.) cold day mission.

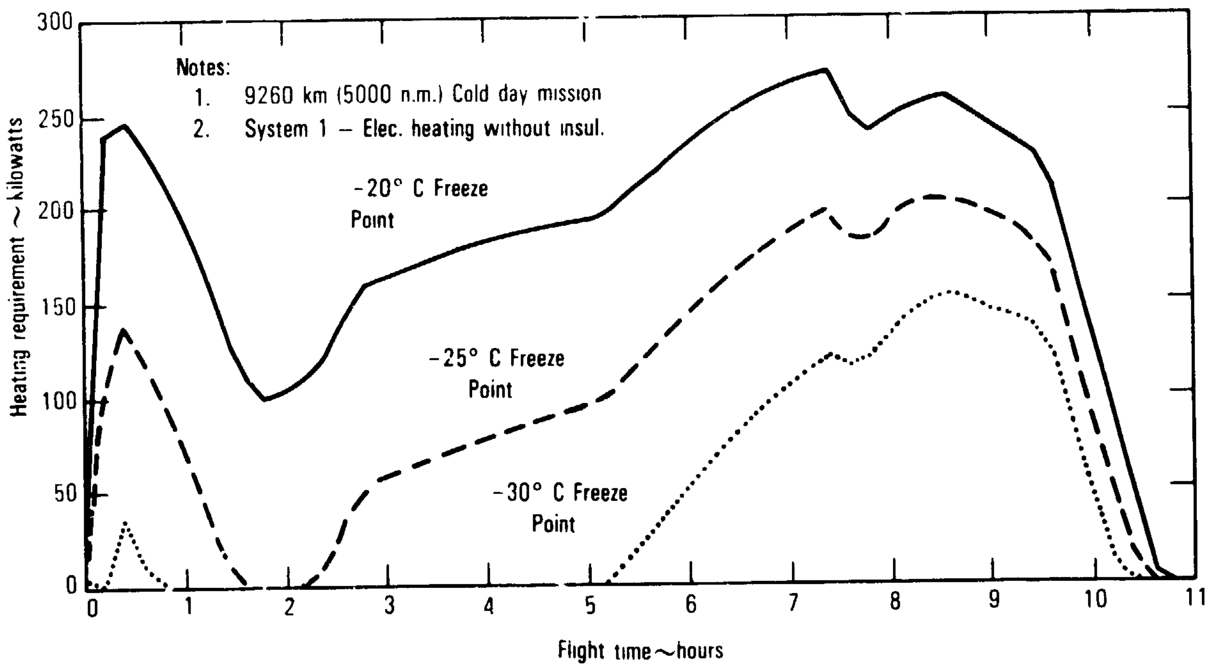

Figure 47 - Predicted fuel tank electric heating requirements for various freeze point fuels.

The second alternative involves adding an additional generator to each engine. The three new generators would be dedicated solely to supplying the electrical requirements of the fuel tank heating system. This less extensive modification requires a less complex FAA requalification but generally involves the same design difficulties previously discussed. In particular, a new generator mounting pad and either a tandem or remote generator drive system pad must be added to the engine.

The third alternative involves a major change in the engine starting system. The present pneumatic starting system would have to be replaced with an electric starter-generator system with the generator portion of the system dedicated to supplying the electrical requirements of the fuel tank heating system. A 150/200 kVA rated SmCo starter-generator, required for engine starting, could be mounted on the existing starter-pad of each engine. In addition an equivalent generator unit would have to be added to the APU to supply electric power for engine starting. Other minor modifications include adding two electrical inverters and the additional wiring and contactors required by the starting system.

The use of starter-generators as the electric power source for the fuel tank heating system has two major advantages. First, the existing starter pad and drive system are able to provide the required torque capacity and accept the physical dimensions of the starter-generator unit. This greatly simplifies the implementation of this modification. Second, because the primary electrical system is not altered by this modification, FAA requalification would be relatively easy. The only major disadvantage of this modification is that it requires a change from pneumatic to electric starting procedures, a requirement that, at the present time, would meet considerable resistance in the airline industry. However, it is anticipated that "all electric aircraft" technologies will gain greater acceptance by the 1990 time period considered in this study. Therefore, of the three sources of electrical energy considered, the electric starter-generator modification was selected to supply the electrical requirements for the uninsulated fuel tank heating system.

6.1.1.2 Fuel tank heating with insulation: The major disadvantage of the above system is that it requires such a large quantity of electric power that a major expansion of the aircraft's generating capacity is necessary. Adding thermal insulation to the fuel tanks significantly reduces the electric power required to prevent freeze-out. The results of analyses conducted using the fuel tank thermal model indicate that properly configured fuel tank insulation can reduce the electric heating loads to a level that is below the excess capacity of the existing generating system. Thus with insulation, the penalties associated with expanding the generating capacity can be avoided. Insulation is only considered in conjunction with fuel tank heating because insulation without heating is insufficient to prevent fuel freeze-out during the cold-day missions.

The results of the thermal analysis led to the definition of two alternative configurations which employ insulation. In the first configuration, figure 48, a layer of insulation is applied to the bottom surfaces of all three fuel tanks between the aluminum lower skin and the electric foil heaters. Other than being applied on top of the insulation, the foil heaters are installed in the same manner as described in the preceding section for the heating system without insulation.

The minimum insulation thickness required on the bottom surface depends upon whether or not the insulation covers the stringers. The thickness required for an arrangement in which the stringers are not covered is approximately twice that required for an arrangement in which the stringers are covered. Because covering the

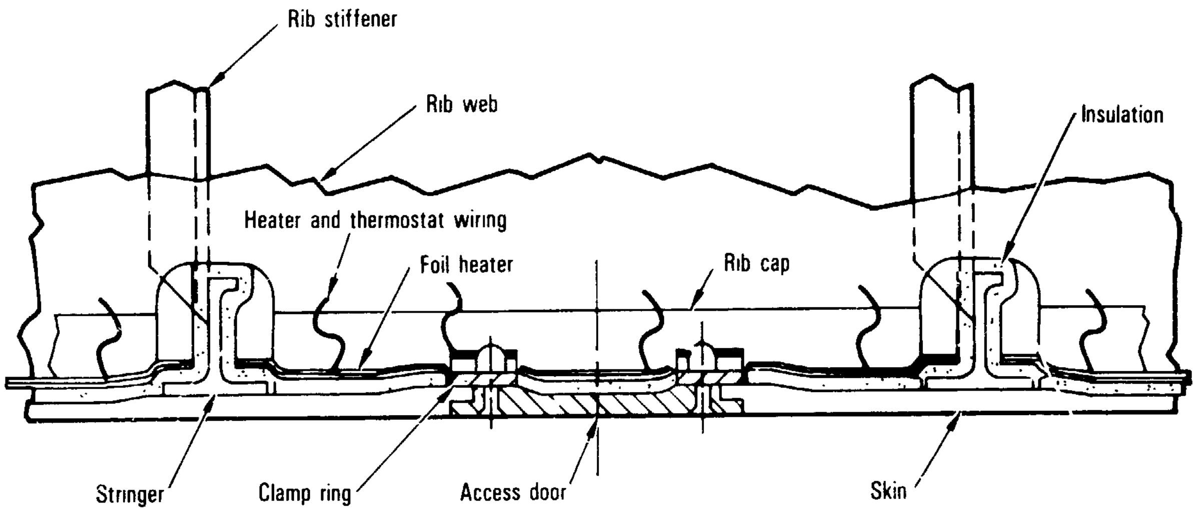

Figure 48 - Fuel tank electric foil heaters with insulation (for all tanks.)

stringers nearly doubles the area of coverage, the volume and weight of insulation required to sufficiently reduce the electric heating requirement is roughly equal for both arrangements. However, the arrangement in which the stringers are covered results in a warmer bulk fuel temperature for the same heat input. The disadvantage of this arrangement is that it is somewhat more difficult to install.

Based upon the thermal analysis, an insulation thickness of 3.175 mm (1/8 in) was selected for the study. This thickness met the objective of maintaining the total aircraft wing heating load within the maximum of 101.5 kW excess capacity of the existing L-1011 generators (figure 46). A further increase in the insulation thickness did not significantly reduce the electrical power requirements.

The second configuration defined is identical to the first configuration in regard to the bottom surfaces of the three fuel tanks. However, in the second configuration, figure 49, a layer of insulation is added to the upper surface of Tank 2-outboard. The installation of insulation on the upper surface of Tank 2-outboard is effective because normal fuel management procedures result in this tank remaining full of fuel for long periods of time. When the tank is full, fuel is in contact with the upper tank surface and a high rate of heat loss occurs. Fuel management procedures are different for the inboard fuel tanks, Tanks 1, 3, and 2-inboard. These procedures result in the fuel in the inboard tanks being consumed first. The inboard tanks are full only for relatively short periods of time and only for long range flights. The air space (ullage space) that forms between the fuel and the upper surface when the tanks are less than full provides sufficient insulation for the upper surfaces of these tanks. Therefore, the installation of the upper surface insulation in the inboard tanks is not advantageous.

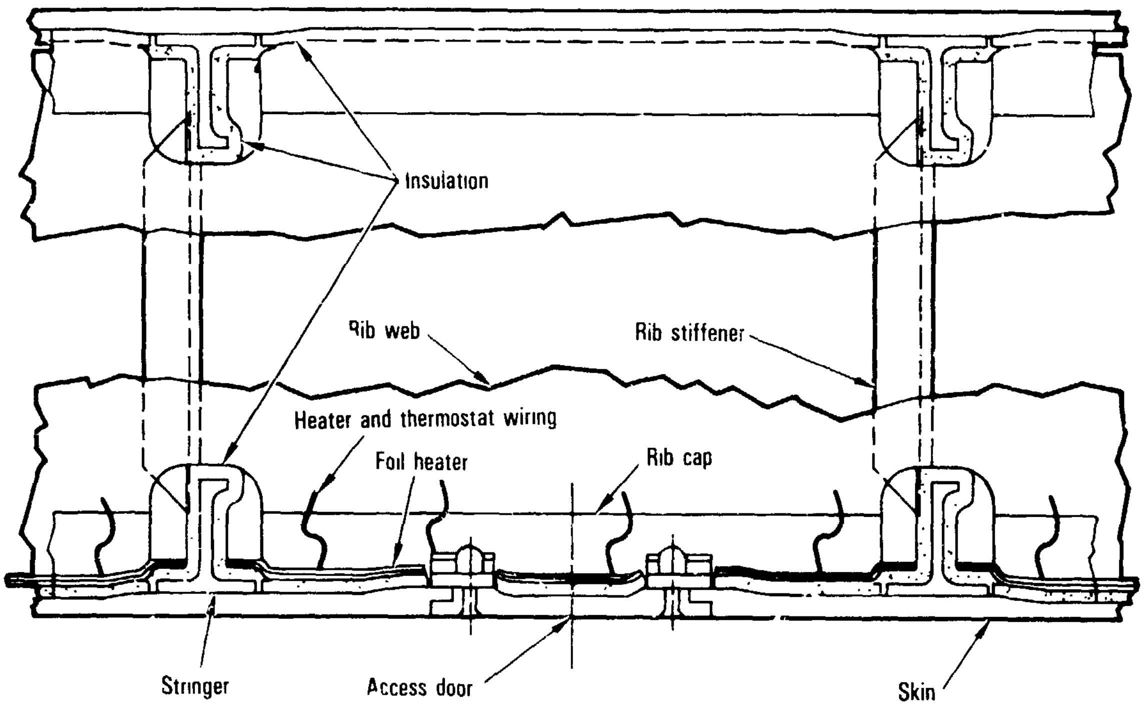

Figure 49 - Fuel tank electric foil heaters with insulation (for tank 2-outboard.)

6.1.1.3 Engine fuel system heating: In addition to the fuel tanks , the engine and APU fuel systems must be protected against fuel freeze-out. Since the engines and the APU produce enough heat locally to prevent freeze-out when operating, the potential for freeze-out exists only when one or more of these units is shut down. In the APU, freeze-out could be a common occurrence because the unit is normally shut down in flight. Because the APU is used in flight as an emergency power source, its fuel system must be made operable quickly. Freeze-out may also occur on the occasions in which an engine is temporarily shut down in flight. Therefore, fuel system heating is provided to insure that freeze-out will not prevent starting a shut down engine or an APU when it is needed.

Two sources of heat were investigated: bleed air from an operating engine and electrical power. Of these, bleed air heating is the recommended method because it is possible to concentrate a large quantity of heat on several adjacent fuel system components by simply directing the air through a discharge manifold onto the components. To achieve the same effectiveness using electrical heating would require that specifically designed heating jackets be fitted around each fuel system component. Maintenance of the fuel system would be made more difficult by the presence of these jackets. Because the heating system would be operated only for short periods of time, the impact of using bleed air heating on the aircraft's overall fuel economy is negligible.

Bleed air is available in the APU compartment and in the engine nacelles through the existing pneumatic system ducting. This system interconnects the three engines and the APU so that bleed can be directed to inactive engines or to the APU from active engines or from the APU when it is operating.

For the APU, air from the pneumatic system is ducted through shrouds which surround the fuel lines within the APU compartment. This air is discharged from the shrouds through manifolds that direct the air onto the fuel pump, oil-to-fuel heater, fuel filter, and fuel control unit. A small portion of the air is used to heat the segment of fuel line that runs from the aft bulkhead to the APU. The entire APU fuel system heater is controlled by a single valve located at the junction between the heating syst.. and the bleed air interconnect line. The installation of the control valve, fuel line shrouds, and discharge manifolds for the APU is shown in figure 50.

For the engines, bleed air from the pneumatic system passes through shrouds surrounding the network of fuel lines within the nacelle. This air is discharged from the shrouds through manifolds that direct the air onto the oil cooler(s), the low pressure filter, and the low pressure pump. A small portion of the air is used to heat the segment of fuel line which, for the wing engines, runs from the wing tank through the pylon to the engine nacelle and, for the center engine, runs from the aft bulkhead to the nacelle. As with the APU fuel system heater, the engine fuel system heaters are controlled by a single valve for each engine. The installation of the control valve, fuel line shrouds, and discharge manifolds for the engines is shown in figure 51.

In addition to the fuel system components previously discussed, the segment of fuel line that runs through the fuselage from the wing tanks to the aft bulkhead must be protected against freeze-out. Without protection, this fuel line would be subject to the slow accumulation of freeze-out in flight. This is due to the constant flow of vent air through the vapor/spillage shroud which surrounds this line. The recommended method for protecting this line is to mix a small quantity of bleed air with the vent air to maintain an air temperature above -17°C.

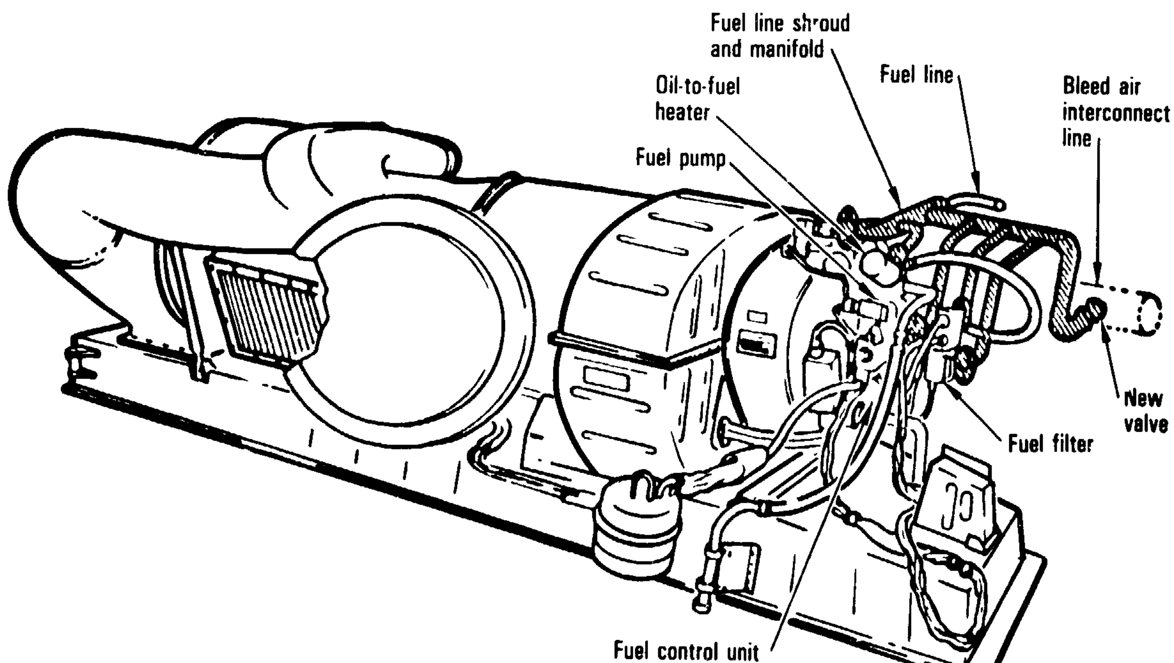

Figure 50 - APU fuel system heating.

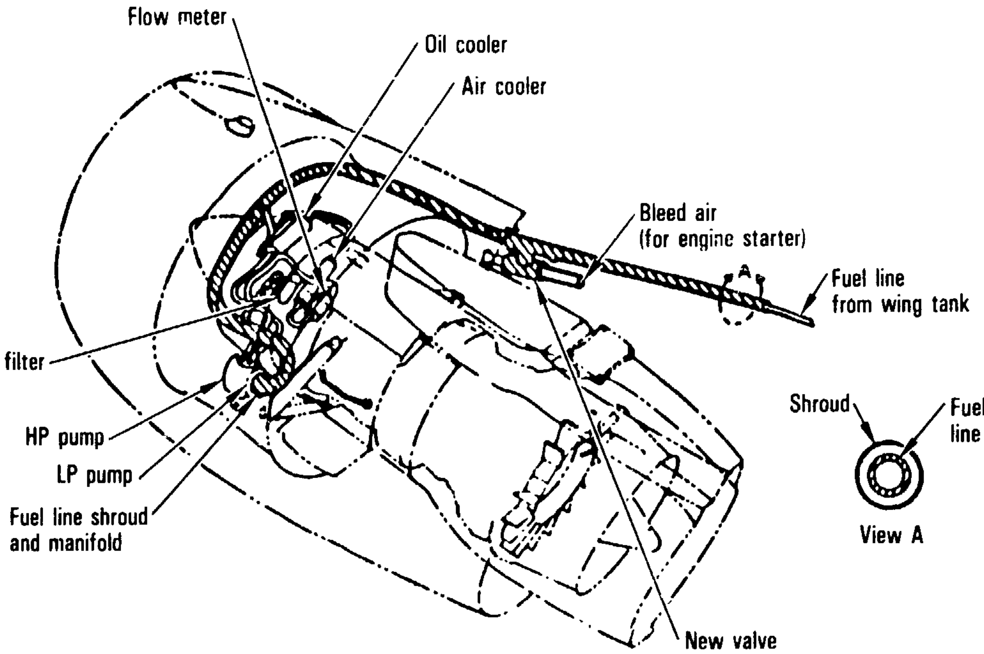

Figure 51 - Engine fuel system heating.

In this and the two preceding sections, various modifications to the baseline aircraft were discussed for dealing with the use of a high freeze point fuel. In subsequent discussions, each of these concepts will be referred to as follows:

SYSTEM	FUEL TANK PROTECTION	ENGINE/APU PROTECTION
I	ELECTRIC FOIL HEATER WITHOUT INSULATION	BLEED AIR HEATING
II	ELECTRIC FOIL HEATERS WITH INSULATION ON THE BOTTOM SURFACE	BLEED AIR HEATING
III	SYSTEM II PLUS INSULATION ON THE UPPER SURFACE OF TANK 2-OUTBOARD	BLEED AIR HEATING

6.1.2 Evaluation of high freeze point concepts. - The critical mission to be examined for the high freeze point fuel is the 9260 km (5000 n.mi.) mission on a cold day assuming that the payload of 18 144 kg (40,000 lb) will be maintained.

The concepts were described in detail in Section 6.1.1. Each concept incorporates foil heaters on the bottom of the tanks but the presence and degree of fuel tank insulation varies.

6.1.2.1 Weight and electrical power assessment: Insulation weights for this assessment are based upon a 50/50 mixture by volume of polysulfide and hollow glass microspheres. This insulation has a weight density of 16.45 kilograms per square meter per centimeter (8.56 pounds per square foot per inch) of insulation thickness. The foil heaters incorporate a fuel resistant Kapton insulation and have a weight of 0.4589 kilograms per square meter (0.094 pounds per square foot).

Electrical power for heating the fuel is supplied by the existing aircraft electrical system modified to produce additional power, if required. The existing L-1011-500 airplane generator capacity is capable of supplying 101.5 kW in excess of ship's requirements when all three engine generators are operating.

The loss of one of these generators can be replaced by actuating the APU generator. However, the probability of this failure occurring is remote and was not considered in this analysis. Its impact on the aircraft performance in the course of a year's operation would not be significant.

The weights and electrical power requirements for each system are summarized in table 12. An inspection of the table shows that System I with no fuel tank insulation has a sizable weight advantage over the other systems considered. However, its power requirement of 270 kW to heat the fuel is 168.5 kW more than is available from the existing electrical system. To meet this requirement, the existing pneumatic starting system would be replaced by a 150 kVA starter/generator mounted on each engine starter pad and an equivalent generator on the APU for self-contained starting of the engines. These starter/generator systems are within the state-of-the-art and would not entail any major development effort (reference 30).

TABLE 12 - WEIGHTS AND ELECTRICAL POWER REQUIREMENTS FOR ENGINE FUEL SYSTEM HEATING

System	Insulation Weight kg	(lb)	Fuel Tank Heater Weight kg	(lb)	Fuel Tank Wiring Weight kg	(lb)	Engine Fuel System Heating kg	(lb)	Starter/Gen Net Weight Added kg	(lb)	Total kg	(lb)	Electrical Power Required By Heaters Kilowatts	Added Generator Capacity Kilowatts
1 Heaters Only Bottom Surfaces	0	0	59	(129)	46	(102)	61	(134)	470	(1036)	635	(1401)	270	270
2 System 1 Plus 3 175 mm (1/8 in) Insulation Bottom Surfaces and Stringers	646	(1425)	59	(129)	46	(102)	61	(134)	0	(0)	812	(1790)	87 3	0
3. System 2. Plus 3 175 mm (1/8 in) Insulation on Top Surfaces and Stringers Tank 2 Outboard	779	(1717)	59	(129)	46	(102)	61	(134)	0	(0)	944	(2082)	74 8	0

From considerations of weight and electrical power requirements, System I, because of its lighter weight, appears to be the most desirable. However, the added complications of the starter/generator system may be a deterrent to its selection. In that case, Systems II and III, which can operate within the existing aircraft electrical power limitation, are more desirable.

6.1.2.2 Impact on aircraft performance: The parameters that best describe the impact of these systems on aircraft performance are the increase in takeoff gross weight and block fuel weight (reference Appendix). In table 13 the total effect on gross weight and block fuel is divided into two parts. The first part is the effect of the increase in the aircraft operating empty weight that is caused by the fuel heating system. The second part is the effect of the increase in the engine specific fuel consumption which is caused by additional power extraction to provide for the increased electrical requirements for the heaters. The total effects show that System I has the least impact on the aircraft takeoff gross weight but requires more fuel to complete the 9260 km (5000 n.mi.) mile range than either of the systems using fuel tank insulation. The increase in TOGW is a built-in penalty which must be evaluated during hot day operation. An inspection of the payload/range curve discussed earlier, figure 7, shows that none of the proposed systems affect either the payload or the range of the aircraft adversely with the study payload of 18 144 kg (40,000 pounds). The primary concern of the operator is that this added fuel consumption increases his operating costs.

TABLE 13 - IMPACT OF FUEL SYSTEM HEATING ON AIRCRAFT PERFORMANCE

All Engines Operating - Cold Day - 9260 kilometers (5000 n.mi.)

Effect of OEW Change

System	Increase In Empty Weight kg	(lb)	ΔTOGW kg	(lb)	ΔBlock Fuel kg	(lb)	ΔTOGW/ΔOEW	ΔBlock Fuel/ΔOEW
I	635	(1400)	1009	(2225)	317	(700)	1.589	0.499
II	812	(1790)	1288	(2840)	408	(900)	1.586	0.502
III	943	(2080)	1497	(3300)	476	(1050)	1.587	0.505

Effect of SFC Change

System	Δ%SFC Due to Additional Fuel Consumption %	ΔTOGW kg	(lb)	ΔBlock Fuel kg	(lb)	ΔTOGW/Δ%SFC		ΔBlock Fuel/Δ%SFC	
I	0.554	494	(1089)	480	(1060)	892	(1966)	868	(1913)
II	0.196	175	(386)	155	(342)	893	(1969)	791	(1745)
III	0.171	152	(335)	132	(292)	889	(1959)	772	(1708)

The table also includes sensitivity coefficients based upon the change in operating empty weight and percent change in SFC incurred by each of the systems for evaluating the impact on payload/range of small changes in payload.

The possibility of modifying the flight profile by increasing flight speed and decreasing altitude to raise the adiabatic wall temperature, which acts as a heat sink for the fuel, was discussed and rejected in Section 5 because of the limited cruise altitudes, 3048 to 3862 m (10,000 to 13,000 ft). Hence, this possibility was not considered in this assessment.

Failure of an engine during takeoff or during cruise must not prevent the aircraft from completing its mission safely. Since the failure of an engine during takeoff is one of the requisites for aircraft certification, the ability of the aircraft to maintain safe flight under such circumstances had to be assessed for each of the systems. Because the aircraft could meet this condition with the maximum increase in TOGW of 1496.9 kg (3300 lb), all of the systems were judged satisfactory for takeoff.

The effects of an engine failure in cruise were also analyzed for each of the systems on the cold day assuming the failure occurred at the midpoint of the 9260 km (5000 n.mi.) flight and the aircraft continued to its destination. Figure 52 shows the adjustment in cruise altitude from 10 688 m (35,000 ft) at Mach 0.82 for three engine operation to 8839 m (29,000 ft) at Mach 0.71 for two engine operation in order to maintain optimum km/kg (nautical miles per pound) of fuel. This flight profile deviation results in an additional block fuel usage of 5142 kg (11,338 lb).

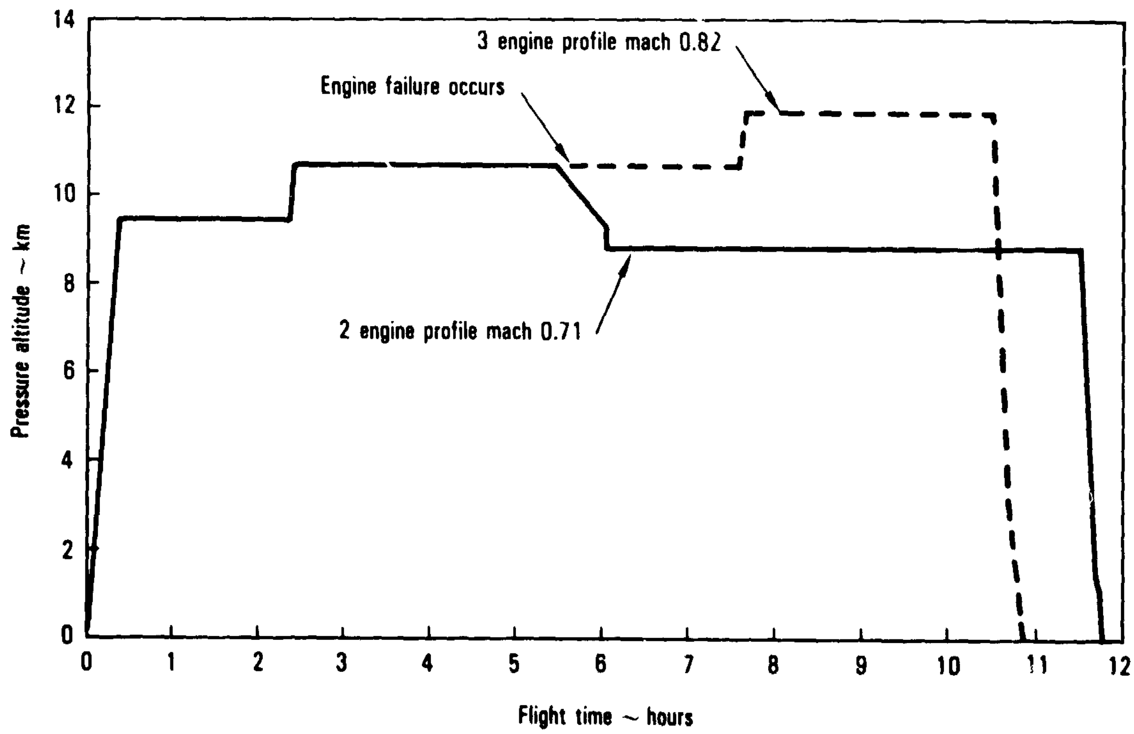

Figure 52 - 9260 km (5000 n.mi.) mission profile with loss of engine in midcruise - cold day.

The engine failure also removes one of the generators which supplies the electrical power for aircraft operation and fuel tank heating. Because the starter/generators used in System I are sized for engine starting, they have sufficient capacity to supply all of the required power from two starter/generators only. On the other hand, Systems II and III require activation of the APU generator to replace the generator from the failed engine. The added fuel consumed during the flight, because of the added power extraction from the two operating engines in System I, and because of APU operation for Systems II and III, is small as shown in table 14.

TABLE 14 - INCREASED FUEL CONSUMPTION DUE TO POWER EXTRACTION WITH ENGINE OUT
Cold Day - 9260 Kilometers (5000 n.mi.)

Systems	ΔBlock Fuel kg	lb
I	181	(400)
II	154	(340)
III	140	(308)

The increase in fuel consumption due to the above factors does not increase TOGW but does result in a decrease in available fuel reserves.

6.1.2.3 Manufacturing, maintenance, and reliability: There is no existing thermal insulation material upon which to base opinions concerning manufacturing methods. Expert opinion based on similar material (polysulfide with phenolic microspheres) is that the material must be troweled on. This will make thickness control difficult. A nylon barrier may be required to prevent absorption of fuel by the polysulfide resulting in swelling which would cause it to pull away from the wing structure and create heat leaks.

There is no prior experience with insulation and heater system inspection requirements. Visual observations during normal fuel tank inspections may be sufficient, but ultrasonic or other inspection methods may be required to detect leaks which would allow fuel-to-skin contact. Maintenance of the physical integrity of the heater elements and associated wiring insulation is critical. During some required structural inspections the heaters and insulation must be removed to perform visual inspections; non-destructive testing (X-ray) may be required.

The life of the insulation will depend on the resistance to fuel absorption and/or the effectiveness of the sprayed-on nylon barrier. Heater life time may exceed aircraft life if they are not physically damaged. Both heater and insulation could be affected by or affect microrganism growth in the fuel tanks.

6.1.2.4 Impact on direct operating costs: A cost analysis of each of the proposed modifications required to allow operation with high freeze point fuels was performed assuming a fleet of 300 aircraft operating over a 16 year period. Parametric cost factors were developed to represent each system in terms of production labor hours and material dollars per kilogram of system weight. These basic data were modified to account for individual design concepts for each applicable major item. Cost factors previously developed for wide body transports for fabrication, assembly, installation, and modification of valves, pumps, and other components of the fuel system were used. The cost of insulation and heater procurement and installation was obtained informally from suppliers.

The premises and assumptions upon which this and subsequent cost analyses were made are shown in table 15. The results of the cost analysis are shown in table 16. Total acquisition costs re reflected largely in depreciation allowances in the DOC computation and to a lesser extent in insurance allowances.

The cost breakdown indicates that System I has the lowest maintenance costs because it requires no insulation in the fuel tanks. To inspect the structural elements of the aircraft, stripping and replacement of the heaters and insulation is required in 8 percent of the fleet. The remainder of the fleet is required to strip and replace the heaters and insulation at one half the aircraft life. These costs more than offset the extra fuel required to operate the SmCo generators to f rnish the 270 kVA needed by the heaters.

System II has the lowest fuel costs but has high maintenance costs because of the insulation. consequently, the DOC of this system at the fuel cost of $1.00/gal is greater than that for System I even though some parameters such as increase in empty weight and block fuel favors selection of System II. As the price of fuel increases, this conclusion will eventually be reversed as discussed in Section 7.2.4.

TABLE 15 - COST PREMISE

Configuration L-1011-500	Based on New Program Production Quantity 300
Operation	
International	
Stage Length	9260 km (5000 n.mi.)
Utilization	4718 Block Hours/Year
Block Time	11.2 Block Hours/Trip
Trips per Year	421
Operational Life	16 years
Cost of Fuel	$1.00/Gallon (International U.S. Trunk - May 1982)
Non Revenue Flying	1.23 percent
Acquisition Costs	Included in Depreciation
Economics	
Year	1982
Labor Rates	Lockheed (1982 Direct, Overhead, G&A, Other)
Profit	10 percent

Maintenance
- Structural Inspection requires stripping and replacing insulation and heaters at 20,000 hours (4 times during life) on 8 percent of the fleet plus 10 percent for miscellaneous checks.
- Life of insulation and heaters is assumed to be one half the aircraft life requiring one stripping and replacement for the remainder of the aircraft.
- Labor Rate $13.93/hour
- Burden Factor 3.13 (International)

TABLE 16 - EFFECT ON DOC OF FUEL HEATING SYSTEMS

(Thousands of 1982 Dollars)

	I	II	III
Acquisition			
Full Scale Engineering Development (FSED)	7350	7350	7350
Procurement	99574	145203	148523
Total Acquisition	106924	152553	155873
Direct Operating Costs			
Fuel	537686	379435	407236
Insurance	4753	6776	6923
Depreciation	96232	127298	140286
Maintenance	100361	260251	270173
Total for Fleet	739032	773760	842618
Cost - $/(Ac Yr)	154	161	172

The added insulation weight in System III increases the cost of fuel, resulting in the highest DOC of the three systems. This also contributes to higher maintenance and depreciation costs, causing System III to be the costliest of all of the systems.

6.1.2.5 Recommended system for high freeze point fuel: At present fuel prices, System I is recommended for use with high freeze point fuels because it has the lowest acquisition and direct operating costs. Although it has the highest fuel consumption due to its high electrical power requirements, this cost is more than offset by its low maintenance costs. System I requires the development of a bonding agent, which must be impervious to hydrocarbon fuel in order to ensure a dependable intimate contact of the fuel heaters with the tank surfaces. In addition to having this requirement, Systems II and III require the development of the 50 percent polysulfide and hollow glass microsphere mixture insulation. A further advantage of System I is that the starter/generator electrical power source is independent of the existing aircraft electrical system which will not have to be recertified, whereas the added power extraction requirements for Systems II and III will require recertification. As fuel prices increase in the future, System II will become the preferred system because of lower fuel usage.

6.2 Thermal Stability

The proposed relaxation of fuel thermal stability from a JFTOT rating of 260°C to 204°C has been shown, in Section 5.2, to require a reduction in peak fuel temperature at the HPFCOC discharge from 135°C to 79°C. In the following paragraphs, alternate engine fuel system design modifications are proposed and evaluated for effectiveness in accomplishing this reduction in bulk fuel temperature.

6.2.1 Description of alternate systems. - The following candidate systems will be divided into two groups: those which decrease the bulk fuel temperature and others which assure low fuel-wet surface temperatures in components located in the hot region of the engine core. In the first group, when considering that the two most important thermal loads in the fuel system originate in the oil heat input and the operation of the fuel pumps, three approaches were investigated: a) rejection of engine oil heat to the atmosphere; b) rejection of engine oil heat to the fuel in the wing tank; and c) reduction of fuel pump heat input. In the second group, the fuel lines and fuel components exposed by their location to the highest fuel bulk heat temperature are proposed to be cooled by air jackets and the feed arm structure by the introduction of heat shields and air jackets. Introduction of advanced ceramic materials with high resistance to structural loads and fatigue at high temperature can be utilized, and thus reduce the thermal conductivity.

6.2.1.1 Rejection of oil heat to fuel tank: This modification permits making full use of the compactness of a single fuel/oil heat exchanger in each engine, while providing an active control of the fuel temperature which is being delivered to the injectors. A schematic diagram of the modification is shown in figure 53. Fuel from the tank passes through the LP fuel pump and FCOC after which a portion of the fuel may be directed back to the fuel tank to maintain a maximum heat sink for oil cooling when the engine fuel consumption is low. The fuel flow rate through the modulating valve is controlled so as to limit the peak fuel temperature at the fuel temperature sensor to 79°C.

This system requires LP fuel and HP fuel pumps of different capacities. The HP fuel pump rating is determined by the engine fuel flow rate requirements at takeoff. The LP fuel pump maximum capacity must accommodate the fuel flow rate at takeoff, in addition to the fuel flow rate which is required to provide adequate oil cooling during the limiting flight. It is estimated that the volumetric capacity of the LP fuel pump is about twice as high as the HP pump. The additional weight requirements for this system are given in table 17.

TABLE 17 - FUEL BYPASS SYSTEM WEIGHT PENALTY

COMPONENT	WEIGHT (Per Engine)	WEIGHT (Per Aircraft)
Increase in LP Pump Weight - kg (lb)	+ 9.1 (20)	+ 27.2 (60)
Fuel Thermal Control - kg (lb)	+ 0.7 (1.5)	+ 2.0 (4.5)
Fuel Return Line - kg (lb) (Engine 2)	+ 30.4 (67)	+ 30.4 (67)
Fuel Return Line (Engines 1 and 3) - kg (lb)	+ 3.2 (7)	+ 6.4 (14)
Total for Aircraft - kg (lb)		+ 66.0 (145.5)

6.2.1.2 Rejection of excess oil heat to atmospheric air: This modification achieves similar results to the scheme described in 6.2.1.1, but avoids the installation of fuel return lines. It requires, however, the addition of an air/oil heat exchanger. A schematic diagram of this modification is shown in figure 54. The

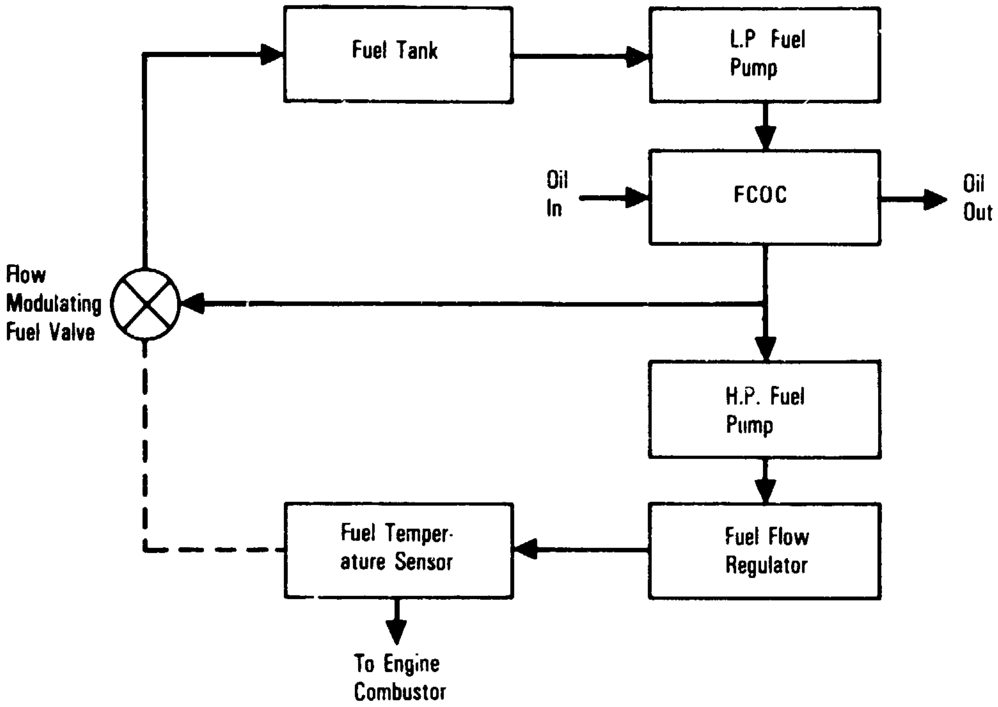

Figure 53 - Rejection of oil heat to fuel tank.

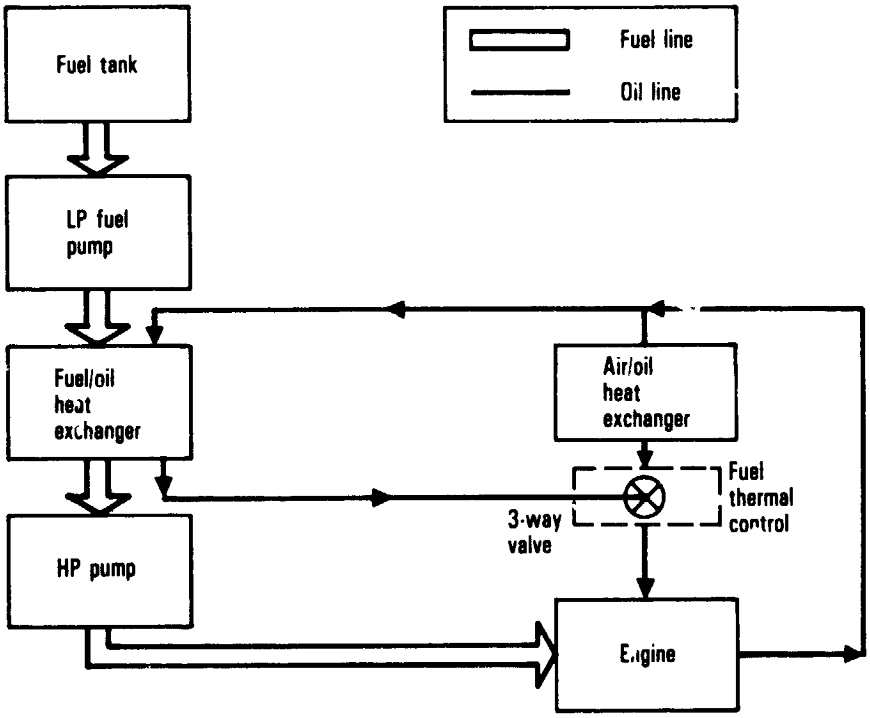

Figure 54 - Rejection of excess oil heat to the atmosphere.

delivered fuel temperature is indirectly controlled by splitting the streams of oil between the air/oil heat exchanger and the fuel/oil heat exchanger by means of one single proportional three-way valve. When the delivered fuel temperature approaches prohibited limits, the three-way valve directs all the oil stream through the air/oil heat exchanger. In intermediate situations, the oil stream is split to achieve a safe fuel temperature and still take advantage of the oil heat to improve the engine SFC. In this scheme, the baseline system suffers a minimal impact. Table 18 shows the weight penalty introduced by this modification.

TABLE 18 - AIR/OIL - FUEL/AIR HEAT EXCHANGER SYSTEM WEIGHT PENALTY

COMPONENT	WEIGHT (Per Engine)		WEIGHT (Per Aircraft)	
Air/oil Heat Exchanger - kg (lb)	+ 6.8	(15)	+ 20.4	(45)
Three-way Valve and Controllers - kg (lb)	+ 0.9	(2)	+ 2.7	(6)
Total for Aircraft - kg (lb)			+ 23.1	(51)

Earlier versions of the L-1011 utilized a similar combination of air/cooled oil coolers and fuel/cooled oil coolers. The difference with the present scheme is that, in those earlier versions, the air/cooled oil cooler was the primary heat exchanger and the fuel cooled oil cooler was receiving only the excess oil heat. In the present modification, the choice of which heat exchanger is the primary cooler does not arise since they are both used in parallel as far as the oil stream is concerned.

When using an air heat exchanger, a decision must be made on whether to use ram air or fan air for the heat exchanger. Since the controlling parameter is the delivered fuel temperature, the worse conditions within a limiting flight have been identified as being those at ground operations and descent. During ground idle and taxiing, ram air is non-existent. Fan air is then the only source of cooling air for the heat exchanger.

6.2.1.3 Reduction of fuel pump heat input into the fuel: During ground operations and descent this thermal load is a major contributor to the heat input into the fuel. The baseline fuel system uses fixed displacement pumps whose speed is mechanically coupled to the engine. The corresponding speed at which the fuel pumping system is driven does not necessarily match, for a given delivery pressure, the volumetric flow rate required by the engine. Accordingly, a bypass circuit is provided for each pump to spill the excess fuel. As a result of this fuel flow adjustment, a significant thermal load is input into the fuel. A flow match to the required fuel delivery pressure would result in a significant reduction in the thermal load. Experience shows that with this approach, the temperature increase due to the pumping system could be reduced from approximately 32°C to 6 or 7°C.

Matching the required fuel flow rates and delivery pressures can be accomplished by either using a variable displacement pumping system or using a variable speed coupler between the pumping system and the engine. Figure 55 shows a schematic diagram of the reduction of fuel pump heat input, using pump speed control. The pump speed control may consist of a variable gear ratio coupler or an electrical variable speed driver motor. The LP and HP pumps are identical to the baseline fuel pumping system, with the exclusion of the spill valve assembly, which has now been substituted by a speed control. The weight penalties of this modification are shown in table 19.

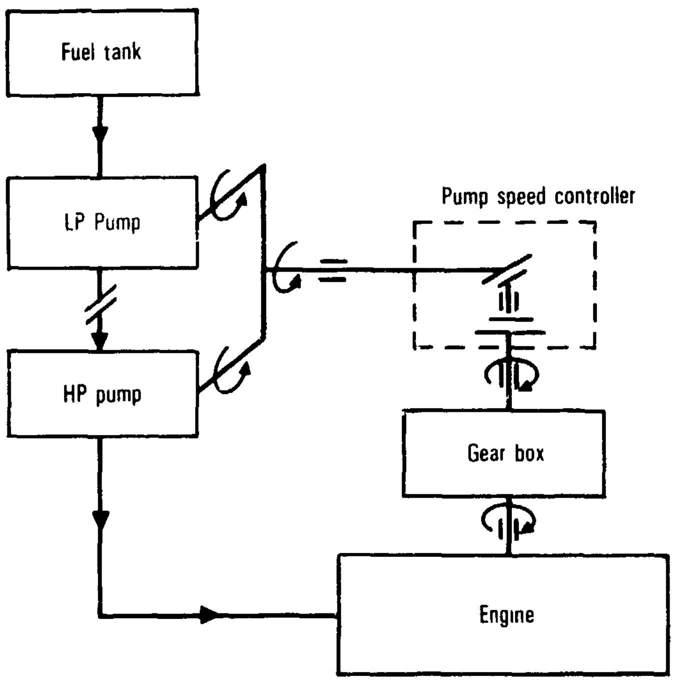

Figure 55 - Reduction of fuel heating using pump speed control.

TABLE 19 - AIRCRAFT WEIGHT PENALTY FOR A VARIABLE SPEED FUEL PUMP SYSTEM

COMPONENT	WEIGHT (Per Engine)	(Per Aircraft)
Pump speed controller (electrically driven DC samarium-cobalt motor, and governor) - kg (lb)	9.1 (20)	27.2 (60)
Total for aircraft - kg (lb)		27.2 (60)
Variable gear ratio (mechanically driven) - kg (lb)	13.6 (30)	40.8 (90)
Spill valve assembly removal - kg (lb)	- 9.2 (-20)	- 27.2 (-60)
Total for aircraft - kg (lb)		13.6 (30)

In this case the electrically driven pump has not been deducted from the total weight since it has been adopted as backup to the pumping system in case of failure of the mechanically driven system.

Figure 56 illustrates how to reduce the fuel pump heat input when using a variable displacement pump. The pump speed coupling to the engine speed is still fixed, but the fuel flow rate and delivery pressures are matched by varying the pump displacement. Flight qualified variable displacement pumps, which are presently available, are mostly of the piston cylinder type and use sliding surfaces. These pumps are more complex than the centrifugal and gear pumps, and have higher lubricity requirements. For the baseline fuel system the maximum volumetric fuel flow rate can be met by a variable displacement pump rated at 55 gal/min and with a maximum displacement of 32.8 cm^3 (2 $in.^3$). A typical weight for one of these pumps is 9.07 kg (20 lb). A dual variable displacement pump of this rating, incorporating an LP/HP capability, could be designed to weigh under 13.6 kg (30 lb). Table 20 gives the weight penalty.

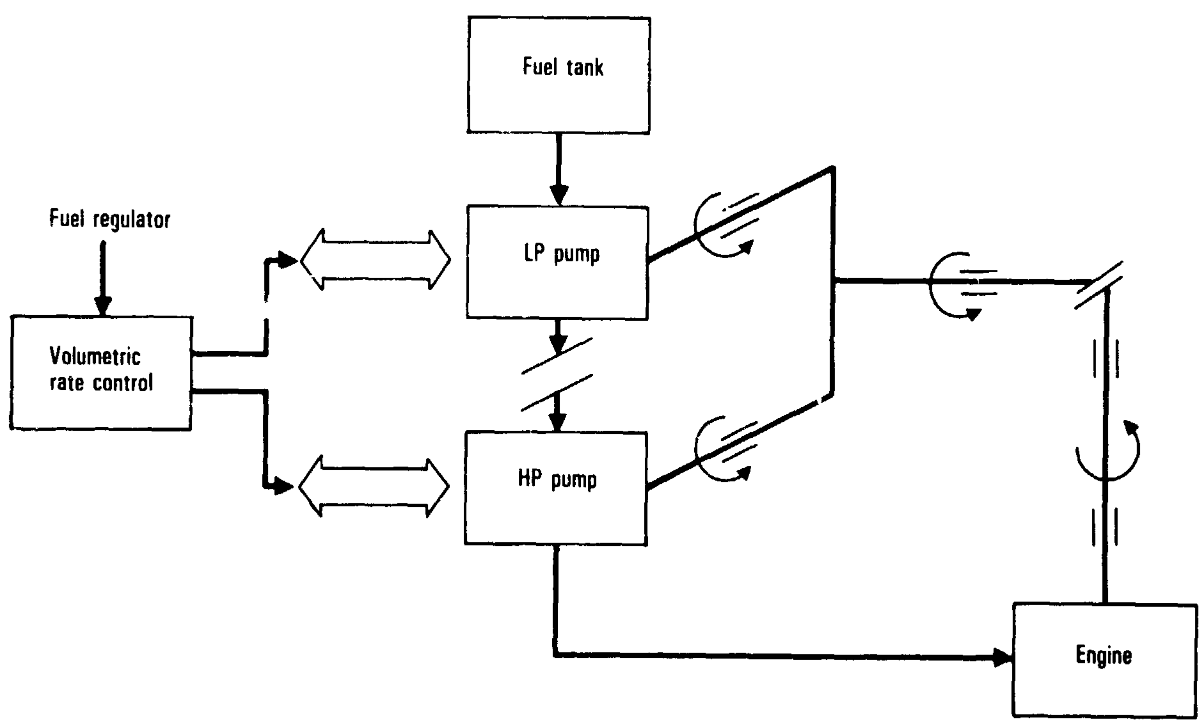

Figure 56 - Reduction of fuel heating using variable displacement LP/HP pumps.

TABLE 20 AIRCRAFT WEIGHT PENALTY FOR A VARIABLE DISPLACEMENT FUEL PUMP SYSTEM

COMPONENT	WEIGHT (Per Engine)	WEIGHT (Per Aircraft)
Removal of baseline LP/HP pumping system - kg (lb)	- 36.3 (-80)	- 108.9 (-240)
Installation of a dual variable displacement pumping system - kg (lb)	+ 13.6 (30)	+ 40.8 (90)
Total for aircraft - kg (lb)		- 68.0 (-150)

Substitution of the baseline pumping system by a variable displacement pumping system results not only in lower thermal loads, but also in a decrease in weight. As discussed later, these advantages have to be weighed against higher complexity, lower reliability and higher maintenance costs, relative to the previously discussed system.

6.2.1.4 Reduction of fuel-wet surface temperatures in the core region: The fuel feed and drain lines, manifold and distribution valves, and injector feed arms are subjected to the hot environment of the core. The injector feed arms are exposed to the most severe temperatures from heat conveyed by the compressor air. The fuel lines, manifold and distribution valves are not so exposed to soak back heat because of the stand-off mounting brackets supporting them to the core surfaces. The modification suggested introduces coaxial stainless steel tubes of about 7.62 cm (3 in.) in outside diameter, which are utilized in place of the existing 3.81 cm (1.5 in.) fuel lines. The outer tube plays the role of a heat shield, and the air gap in between serves as an impedance path to heat conductance. The outer tube can be welded to shrouds which surround the distribution valves, extending in this manner the heat shield throughout all the fuel-wet surfaces located on the core, except in the injectors. During normal engine operation, the cavity between the heat shield and fuel lines and components contains still air, and the fuel flow is sufficient to keep the wet surfaces from reaching prohibitive temperatures. During engine shutdown, air could be blown, at different points, into this gap to reduce the engine soakback heat. Figure 57 illustrates the concept in a section of the coaxial tube.

Two approaches can be adopted to reduce the wet surface temperatures in the feed arms. The first approach, figure 58, uses a heat shield, directly in contact with the compressor air, and protects the feed arm. The feed arm has been reduced to a single tube supported on the heat shield by structural cross members. At the end of the feed arm, the bell-mouth burner penetrates the last sealed structural member. With this disposition, the feed arm itself is kept at low temperatures while only the bell-mouth is exposed to the compressor air temperature. The second approach does not alter the original geometry of the baseline injectors, but it utilizes high temperature ceramic materials instead of cast steel for the structure of the injector. If necessary, the feed arm conduit itself can be lined with stainless steel or other fuel compatible material. In this manner, the high rigidity of the original design is maintained while a high resistance path is presented to heat conduction from the compressor. Table 21 gives the weight penalties for this system.

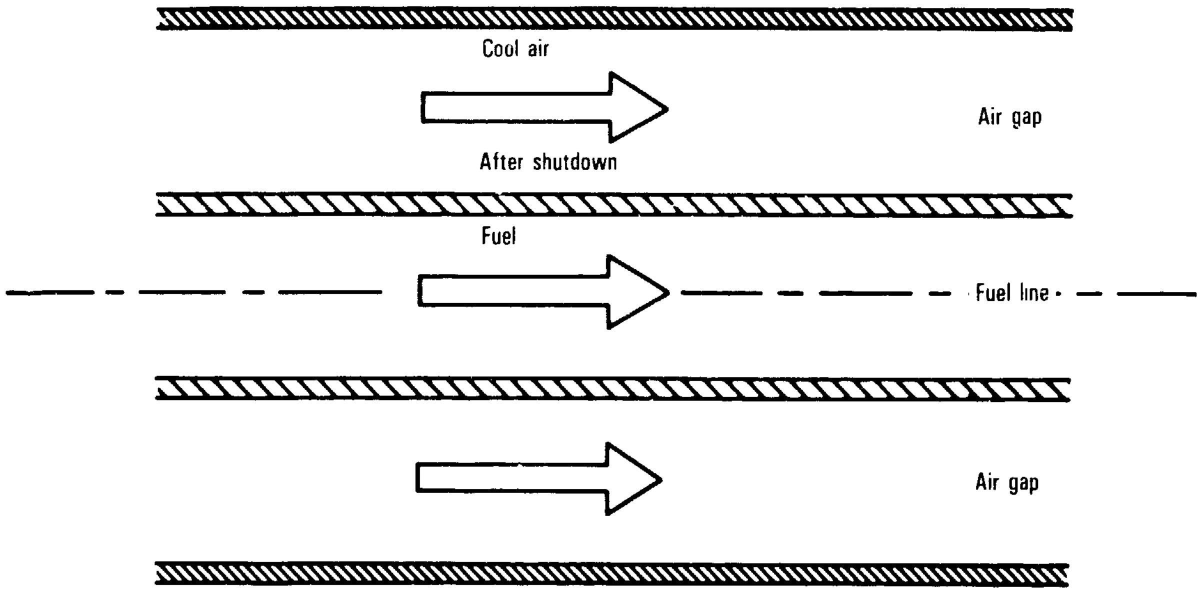

Figure 57 - Heat shield for lines and components located in the hot section region.

TABLE 21 - FUEL SYSTEM HEAT PROTECTION ON WEIGHT PENALTY

COMPONENTS	WEIGHT (Per Engine)	WEIGHT (Per Aircraft)
Coaxial tubes for fuel feed and drain lines, manifold distribution valves and pigtails - kg (lb)	4.54 (10)	13.61 (30)
Heat shield for injector feed arms - kg (lb)	1.36 (3)	4.08 (9)
Total for aircraft - kg (lb)		17.69 (39)

6.2.2 Evaluation of candidate systems. - The modifications to the baseline fuel system that have been proposed in the previous paragraphs, were evaluated here according to certain dominant criteria such as weight increase, electric power requirements and effect on the specific fuel consumption, safety, maintenance, and reliability. Other secondary factors such as technology developments, materials, cost, and suitability for retrofit or adaptability were also considered. In the following paragraphs, the effect on SFC is discussed in terms of: 1) the effect of lowering the delivered fuel temperature to meet the lower thermal limits imposed by the utilization of a low thermal stability fuel; 2) the SFC penalty caused by utilization of fan air, or 3) the higher power requirement to drive the pumping system. The evaluation will be conducted for a representative flight condition, such as cruise at 10 668 m (35,000 ft) and Mach 0.82 during a hot day. Results of the evaluation are shown in figure 59.

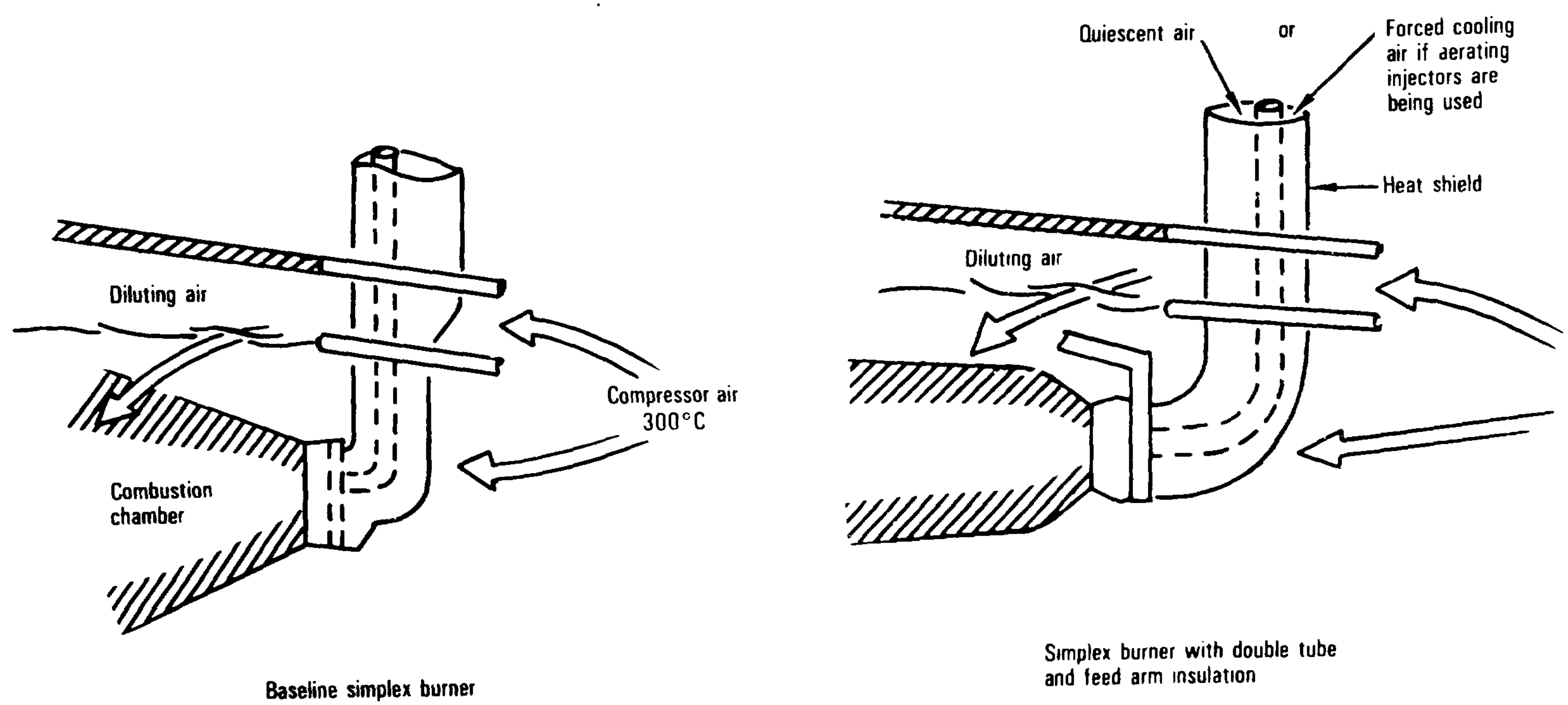

Figure 58 - Improved heat shield thermal design for injectors.

CODE

H - Higher
L - Lower
S - Same
AH - At hand
D - Needs development
G - Good
F - Fair

		Weight, kg	Power, kW	Maintenance	Reliability	SFC increase, %	Safety	Cost	Technology	Materials	Adaptability
1	Oil heat rejection to the fuel tank	+66	+4	S	S	0	S	H	AH	AH	G
2	Oil heat rejection to the atmosphere	+23	0	S	S	+0.2	S	S	AH	AH	G
3	Variable pump speed (electrical)	+27	+12	H	L	≈0	S	H	AH	AH	F
4	Variable pump speed (mechanical)	+14	0	H	L	−0.10	S	H	D	D	F
5	Variable pump displacement	−68	0	H	L	−0.10	S	H	AH	AH	G
6	Heat shield to lines and components	+14	0	H	L	≈0	S	H	AH	AH	F
7	Heat shield for injectors	+4	0	S	S	≈0	S	H	AH	AH	F
8	Ceramics for feedarms	0	0	?	?	≈0	S	H	D	D	G
9	Minor modifications	0	0	S	S	0	S	S	AH	AH	G

Figure 59 - Evaluation of candidate systems.

6.2.2.1 Rejection of oil heat to fuel tank: This scheme requires a return line in addition to a fuel thermal control system. The weight penalty for the aircraft is 65.77 kg (145 lb) and additional power if needed to drive the LP fuel pump. The system should be designed to provide oil cooling during takeoff, and the LP fuel pump must therefore accommodate about 15 876 kg/h (35,000 lb/hr) and be able to boost the pressure by as much as 6894.4 kPa (100 psi). This represents an increase in power requirements from the HP rotor of 1.5 kW (2 HP) at takeoff and climb, and 0.7 kW (1 HP) for cruise. The effect on the SFC of the power required to recirculate the fuel back to the wing tank is negligible.

6.2.2.2 Rejection of excess oil heat to the atmosphere: Rejecting excess heat to the atmosphere has a simpler implementation than the previous system, and the increase in SFC, due to heat loss, is negligible. The air-cooled oil cooler requires fan air in ground operations and in flight. The fan bleed represents an SFC penalty in cruise of approximately 0.5 percent.

6.2.2.3 Reduction of fuel pump heat into the fuel: Besides the weight penalties already discussed, additional electrical power of 12 kW is required to drive the pumping system. In this case, however, the power required for the pumping system, except for efficiency losses, is adjusted exactly to the engine requirements at each flight condition. This matching has a favorable impact on SFC, despite delivering the fuel at a lower temperature. This system requires the installation of a dc motor, and therefore slight increases in maintenance, cost, and a decrease in reliability. When the pump speed is mechanically controlled, the SFC improves by about 0.1 percent; in addition to the weight penalties, there is a decrease in reliability; and an increase in cost, maintenance, complexity, and oil lubrication requirements.

When using variable displacement pumps, some of the criteria appear to move in a favorable direction, including weight and specific fuel consumption. Variable displacement pumps, however, are less reliable and require higher maintenance. From the materials point of view, special liners for the cylinders may have to be built from carbon treated steels to cope with poor lubricity fuels. The cost of these pumps is also higher.

6.2.2.4 Reduction of fuel wet-surface temperatures in the core region: All the schemes described in paragraph 6.2.1.4 can be implemented with minimum impact on the aircraft, save for the increase in weight and the decrease in structural rigidity of the injectors. The suggested introduction of ceramics as a structural member for the injectors would be a new technological development. Because of the lack of detailed temperature data in the area of the core, it is difficult to predict the decrease in surface temperature and the benefits which will be obtained with such schemes, and further work in this area is recommended. Furthermore, it is not known whether these surface temperatures are high enough to constitute a source of carbon deposition even when utilizing fuel with a JFTOT of 204°C.

6.2.3 Recommended system for low thermal stability. - Limiting the engine manifold fuel temperature to 79°C on a 54°C day cannot be accomplished by any one of the proposed systems. However, a combination of heat rejection to the fuel tanks and to the atmosphere coupled with the use of a variable displacement high pressure fuel pump and heat shielding of the fuel injectors can approach the 79°C target. This system is illustrated in figure 60.

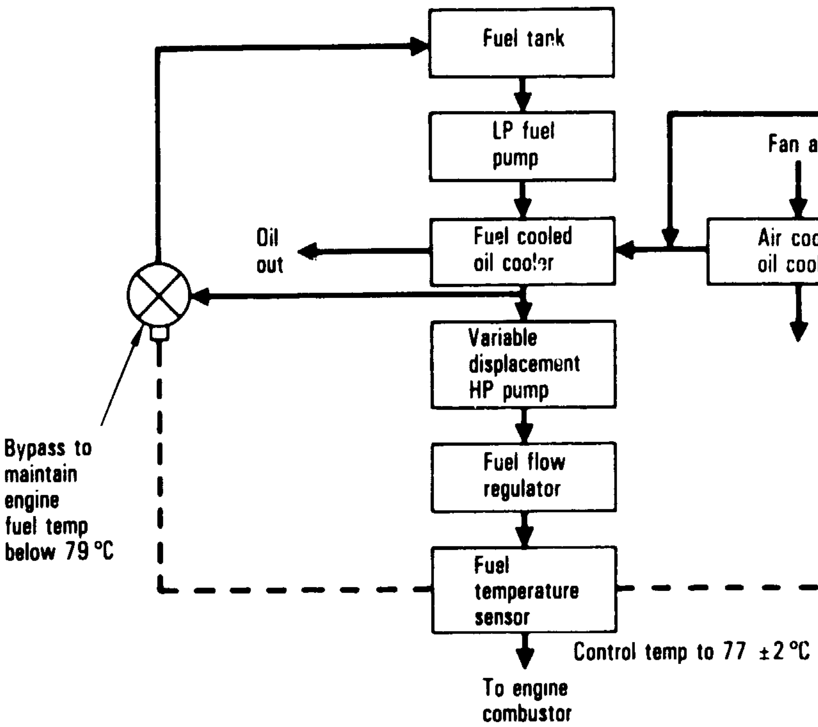

Figure 60. Recommended system for low thermal stability fuel.

The primary heat sink for cooling the engine oil is the engine fan air with any additional heat rejection required to limit the manifold fuel to 79°C going to the fuel tanks. A fuel temperature sensor at the discharge of the fuel flow regulator supplies a signal to two flow modulating bypass valves which control the amount of heat rejected to the fan air and to the fuel tanks. If the fuel temperature at the manifold exceeds 79°C, all of the engine oil is directed through the air-cooled oil cooler and the fuel return bypass valve modulates fuel returned to the fuel tank to supply any additional cooling required. If the fuel temperature drops below 75°C, the fuel return bypass valve remains closed and the air-cooled oil cooler bypass valve bypasses oil flow around the cooler to maintain the required minimum temperature.

A weight summary of these modifications in table 22 shows an OEW increase of 38.56 kg (85 lb).

TABLE 22 - WEIGHT EVALUATION OF RECOMMENDED SYSTEM FOR FUEL COOLING

(Recommended Low Thermal Stability Fuel Only)

(1)	Heat Rejection to Fuel Tanks - kg (lb)	+ 65.8 (145)
	Heat Rejection to Atmosphere - kg (lb)	+ 23.1 (51)
(2)	Replacement of HP pump with Variable Displacement Pump - kg (lb)	- 45.7 (100.8)
(3)	Shields - kg (lb)	+ 4.1 (9)
	Net Increase in Empty Weight - kg (lb)	+ 47.3 (104.2)

Replacement of the LP/HP pump by a direct centrifugal LP pump and a variable displacement HP pump has the effect of removing only 45.7 kg (100.8 lb) as contrasted to the 68.04 kg (150 lb) reduction cited in table 23 for replacing both LP and HP pumps by variable displacement pumps.

The engine fan air bleed will increase fuel consumption in flight and using the tank fuel as a heat sink will require that significant quantities of fuel be retained in the fuel tanks for extended engine operation on the ground in hot weather.

The weight penalties of the recommended system on TOGW and block fuel for the 926 and 9260 km flights are small as shown in table 23 (reference Appendix) as is the increase in direct operating costs of these changes shown in table 24.

TABLE 23 - WEIGHT PENALTIES DUE TO RECOMMENDED SYSTEM FOR FUEL COOLING

Flight Length km (n.mi.)	Day	ΔOEW kg (lb)	ΔTOGW kg (lb)	Δ Block Fuel kg (lb)	$\frac{\Delta TOGW}{\Delta OEW}$	$\frac{\Delta \text{Block Fuel}}{\Delta OEW}$
9260 (5000)	Cold	38.6 (85)	68.0 (150)	22.7 (50)	1.762	0.588
926 (500)	Hot	38.6 (85)	45.4 (100)	4.5 (10)	1 176	0.177

TABLE 24 - DOC INCREASES DUE TO RECOMMENDED SYSTEM FOR LOW THERMAL STABILITY FUEL

(Thousands of 1982 Dollars)

<u>Acquisition</u>	
Full Scale Engineering Development (FSED)	1753
Procurement	16588
Total Acquisition	18341
<u>Direct Operating Costs</u>	
Fuel	15275
Insurance	816
Depreciation	16507
Maintenance	5296
Total DOC	37894
Cost - $/(Ac.Yr)	7.89

6.3 Other Property Changes

The extensive aircraft and engine fuel system modifications such as are required to adapt to high freeze point and low thermal stability fuels are not typical of the changes required to adapt to the other fuel property changes being considered.

Higher aromatics in the fuel increase the solubility parameter and thus may require the selection of polymeric materials with higher values of the solubility parameter. For example, a transition to Buna-N copolymers with higher acrylonitrile content can be made. The solubility parameter of Buna-N copolymers are known to increase with increasing acrylonitrile content in the copolymers. Another possible modification is to use materials with very low values of the solubility parameter such as the fluorocarbon polymers. Both of these approaches are state-of-the-art and will have no impact on the aircraft performance.

The projected increase in fuel viscosity will cause a small increase in power to overcome the associated increase in fuel line pressure drop and a minor decrease in heat exchanger performance. However, the impact on aircraft performance will be insignificant.

A reduction in lubricity characteristics of the fuel is acceptable providing corrosion inhibitor additives are used in the fuel. However, because of the adverse effect that such additives have on the water separation characteristics of the fuel, a more feasible solution is to change the materials to carbon steel where friction between surfaces is a problem.

Of the remaining properties, water separation and electrical conductivity are easily controlled and the changes being considered for flash point and vapor pressure are well within the range of Jet B fuel characteristics, a fuel on which all existing jet aircraft are certificated to operate.

7. CANDIDATE ADVANCED FUEL SYSTEMS CONCEPTS

In Section 6, a number of fuel system design modifications were proposed which would enable the L-1011-500 airplane to perform satisfactorily with the relaxation of a single property of ASTM D 1655-81 Jet A fuel. Although many fuel properties were considered in the study, the only fuel properties which had a significant impact on the aircraft performance were higher freeze point and lower thermal stability. In this section, three alternative fuel system designs, each enabling the aircraft to operate satisfactorily using a fuel incorporating all of the relaxed fuel properties simultaneously, are described and compared in terms of impact on the aircraft direct operating costs.

7.1 Candidate Descriptions

Table 25 describes the three candidate systems, A, B, and C, in terms of the modifications required to permit the use of a fuel having both a high freeze point and a low thermal stability. Each candidate incorporates the identical means of protecting the engines and APU from fuel freeze-out by ducting hot bleed air to non-operating components and from excessive gum, varnish and coke formation by providing a means of limiting peak temperatures in their fuel systems. The candidates differ significantly, however, in the means of preventing freeze-out in the fuel tanks. All of the candidates use electrical foil heaters on the bottoms of the tanks but candidate A requires an additional dedicated starter/generator to replace the existing pneumatic starter on each engine and an equivalent generator on the APU because of the excessive heat loss through the uninsulated skins. Candidate systems B and C do not require the added generator capacity because the tank insulation reduces fuel heat losses to a level that is within the electrical capacity of the existing aircraft generators. Each of these systems will maintain the fuel temperature in the tanks above -17°C which is 3°C above the projected freeze point of the fuel. All other parts of the aircraft, non-operating engine, and APU fuel systems are warmed by engine bleed air only as required to ensure their operation when activated.

Peak temperatures in the engine fuel system are reduced to levels which are compatible with a JFTOT rating of 204°C by using engine fan air, and wing tank fuel, as well as fuel consumed by the engine, as a heat sink for engine oil cooling and by using a variable displacement pump to reduce the high pressure pump heat rejection.

7.2 Performance Evaluation

Each of the candidate systems impacts the overall performance of the airplane by causing an increase in empty weight. Empty weight increases cause an increase in the takeoff weight and subsequently result in an increase in fuel required to fly the mission. Increased power extraction to meet the higher electrical loads required for fuel tank heating also increases fuel consumption. These changes, however, will not prevent the airplane from operating on the desired routes unless a limit, maximum takeoff weight or maximum fuel capacity is reached. Even in this case the mission can be completed by accepting a reduction in payload.

7.2.1 <u>All engines operating</u>. - The three candidates are compared with each other and with the baseline airplane in table 26 (established from information obtained from the Appendix). The total effect of the systems on takeoff gross weight and

TABLE 25 - CANDIDATE FUEL SYSTEMS CONCEPT DESCRIPTIONS

<table>
<tr><th rowspan="2">Candidate</th><th colspan="2">High Freeze Point Fuel</th><th>Low Thermal Stability Fuel</th></tr>
<tr><th>Fuel Tank Modifications</th><th>Engine/APU Modifications</th><th>Engine/APU Modifications</th></tr>
<tr><td>A</td><td>o Electric foil heater on tank bottoms.</td><td>o Replace pneumatic starter with Sm/Co starter/generator.
o Bleed air heating.</td><td rowspan="3">o Oil heat rejection to air, consumed fuel and fuel tanks.
o Variable displacement high pressure fuel pump.
o Heat shielding of fuel injectors.</td></tr>
<tr><td>B</td><td>o Electric foil heater,
o 3.175 mm (1/8in.) insulation on tank bottoms.</td><td rowspan="2">o Bleed air heating.</td></tr>
<tr><td>C</td><td>o Electric foil heater,
o 3.175 mm (1/8in.) insulation on tank bottoms,
o 3.175 mm (1/8 in.) insulation on top of Tank 2 outboard.</td></tr>
<tr><td colspan="4">Modifications Required by Other Fuel Property changes (No Performance Effects)
o Aromatics - materials changes.
o Viscosity - none required.
o Lubricity - material changes.
o Water Separation - none required.
o Electrical Conductivity - antistatic additive may be added to fuel.
o Flash Point/Vapor Pressure - none required.</td></tr>
</table>

TABLE 26 - COMPARISON OF CANDIDATES TO BASELINE AIRPLANE
All Engines Operating Cold Day -9260 kilometers (5000 n.mi.)

Effect of OEW Change

Candidate	Increase in Empty Weight kg	(lb)	ΔTOGW kg	(lb)	ΔBlock Fuel kg	(lb)	ΔTOGW / ΔOEW	ΔBlock Fuel / ΔOEW
A	674	(1485)	1066	(2350)	340	(750)	1.582	0.504
B	850	(1875)	1349	(2975)	431	(950)	1.587	0.507
C	982	(2165)	1553	(3425)	494	(1090)	1.581	0.503

Effect of SFC Change

Candidate	Δ%SFC Due to Additional Engine Fuel Consumption	ΔTOGW kg	(lb)	ΔBlock Fuel kg	(lb)	ΔTOGW / Δ%SFC	ΔBlock Fuel / Δ%SFC
A	0.554	494	(1089)	480	(1060)	892 (1966)	866 (1913)
B	0.196	175	(386)	155	(342)	893 (1969)	791 (1745)
C	0.171	152	(335)	132	(293)	889 (1959)	772 (1713)

block fuel is divided into two parts. The first part is the effect of the increase in the aircraft operating empty weight that is caused by the change in the fuel system. The second part is the effect of the increase in the engine specific fuel consumption which is caused by additional power extraction to provide for the increased electrical requirements for the heaters. Exchange ratios or sensitivity factors are also presented to enable small adjustments to be made in TOGW and block fuel.

Although Candidate C shows the largest increase in TOGW, this effect is not as significant as the increase in block fuel because the airplane is not weight limited at the 18 144 kg (40,000 lb) payload level.

7.2.2 Engine out operation. - The failure of an engine in flight significantly increases fuel system heating requirements because the lower flight speed and longer time in flight lowers the adiabatic wall temperature that acts as a heat sink for the fuel heat. The loss in generator power can be made up in Candidates B and C by activating the APU. However, the excessive generator capacity required for the Sm/Co starter/generator used in Candidate A is more than adequate to meet the added heating loads without activating the APU. The increase in fuel consumption due to power extraction was shown to be small in Section 6.1.2.2 relative to the increase in fuel consumption due to the change in altitude and Mach number. The total fuel consumption increase due to engine out operation, which comes out of reserves, is shown in table 27.

TABLE 27 - INCREASE IN FUEL CONSUMPTION FOR ONE ENGINE CUT OPERATION
Cold Day - 9260 kilometers (5000 n.mi.)

Candidate	ΔBlock Fuel kg	ΔBlock Fuel (lb)
A	5324	(11,738)
B	5297	(11,678)
C	5283	(11,646)

7.2.3 Maximum payload for 9260 km (5000 n.mi.) range. - The operating limits of the airplane become most critical on the hot day mission. Thrust limitations of the engines in climb plus a higher fuel consumption force the airplane nearer to its operating limits. The lower thermal stability and high freeze point fuels further impact the situation. The payload range curve for the hot day was shown previously in figure 7. At a range of 9260 km (5000 n.mi.) the aircraft has the potential to carry 25 402 kg (56,000 lb) of payload, therefore, it is not limited by the chosen payload of 18 144 kg (40,000 lb). Statistical analysis shows, however, that for the present airline average annual load factor, approximately 4 percent of the time the flights will be full, a load factor of 100 percent. The impact of the increase in OEW on maximum payload for each of the candidates is shown in table 28. The increase in OEW reduces the maximum payload directly which means a reduction in cargo as indicated.

TABLE 28 - COMPARISON OF CANDIDATE SYSTEMS AT MAXIMUM PAYLOAD HOT DAY
9260 km (5000 n.mi.) Range

System	ΔOEW kg (lb)	Max. Payload kg (lb)	Cargo kg (lb)
Baseline	0	25402 (56,000)	2899 (6390)
A	674 (1485)	24728 (54,515)	2225 (4905)
B	851 (1875)	24551 (54,125)	2053 (4525)
C	982 (2165)	24420 (53,835)	1916 (4225)

7.2.4 Comparison of direct operating costs. - The acquisition costs including full scale engineering development and procurement costs for a fleet of 300 aircraft operating for 16 years have been factored into the direct operating costs for each candidate aircraft. Direct operating costs for this comparison included fuel consumption, insurance, depreciation, and maintenance only. The data are expressed in thousands of 1982 dollars in table 29. The effect of changing fuel cost is also shown in the table and is illustrated in figure 61. Direct operating costs were calculated for fuel costing $1.00/gal, $1.50/gal, and $2.00/gal. The total fuel costs for Candidate A are always highest. At the baseline fuel cost of $1.00/gal the depreciation and maintenance costs required for the fuel tank insulation in Candidates B and C more than offset the fuel costs. This is reflected by the lower direct operating cost for Candidate A. However, at the higher fuel costs, the reduced fuel usage of Systems B and C overcomes the advantage of the less complex System A. The crossover in DOC occurs at $1.27/gal for System B which, on the basis

TABLE 29 - INCREASE IN DIRECT OPERATING COST
(THOUSANDS OF 1982 DOLLARS)

Candidate	A	B	C
ACQUISITION			
Full Scale Engineering Development (FSED)	9103	9103	9103
Procurement (300 Aircraft)	116162	161791	165111
(Heater Material Cost)*	(22848)	(22848)	(22848)
(Insulation Material Cost)*	(–)	(5242)	(6250)
Total Acquisition (300 Aircraft)	125265	170894	174214
DIRECT OPERATING COSTS			
Fuel(@ $1.00/gal.)	552960	394710	422510
Insurance	5569	7592	7739
Depreciation	112739	153804	156793
Maintenance	105657	265547	275469
Total DOC	776925	821653	862511
COST $/AC/YR			
Fuel Cost – $1.00/gal.	162	171	180
– $1.50/gal.	219	212	224
– $2.00/gal.	277	253	268

*Cost of heaters and insulation material in the aircraft is included in Procurement.

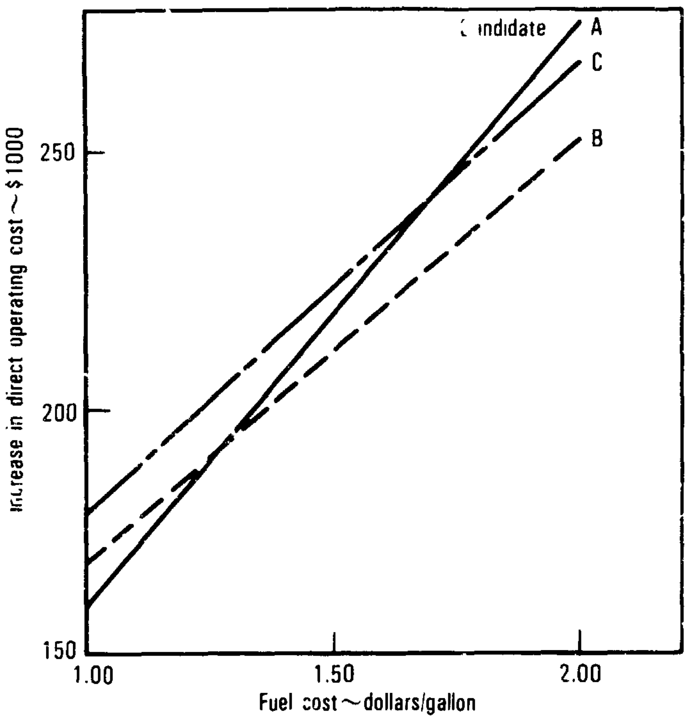

Figure 61 - Impact of fuel cost on DOC.

of cost considerations, is therefore the preferred candidate. These incremental costs were incurred because of baseline aircraft design modifications required to adapt to a fuel with relaxed fuel properties. Assuming the relaxed properties would permit a reduction in fuel production costs, some of which could be passed along to the consumer, the price of fuel based upon 1982 dollars to permit the operator to break even after 16 years of operation was estimated to be as follows:

Candidate	Fuel Price Reduction
Baseline	$1.00/gal.
A	$0.9855/gal.
B	$0.9844/gal.
C	$0.9836/gal.

Conversely, if present fuel properties are maintained regardless of the quality of crudes being delivered to the refinery, the cost of engine and airframe fuel system modifications would not be incurred. As an insurance that present fuel properties be retained, the aircraft operator could afford to pay the following for his fuel depending upon which modification had been contemplated:

Candidate	Fuel Price Increase
Baseline	$1.00/gal.
A	$1.0038/gal.
B	$1.0074/gal.
C	$1.0075/gal.

Although these price estimates would have no effect in fuel prices which are determined primarily by the economic factors of supply and demand, they do represent an allowable fuel price differential to recover costs of installing the various candidate fuel systems or insurance to preclude their need.

7.2.5 Support systems, maintenance, safety. - A Samarium-Cobalt (Sm/Co) starter/generator is incorporated in Candidate A which replaces the pneumatic starter, valve, and ducting. It requires a small gearbox adaptor, electricl converters, contactors, and additional wiring. For the aircraft, this means acceptability problems because of the new starting methodology. This disadvantage in Candidate A is somewhat offset by the necessity of recertifying the electrical system of Candidates B and C because of the higher electrical loads resulting from tank heating.

The control systems for all of the candidates are essentially the same and add a degree of complexity which will entail added maintenance. However, it is not expected to necessitate a change in maintenance intervals. The presence of insulation in Candidates B and C should not require any additional inspection time compared to Candidate A inasmuch as the foil heaters, which all of the candidates use, are in essentially the same locations as the insulation and are subject to the same environment. However, the insulation is expected to require more costly maintenance to ensure its integrity and that of the aircraft structure which it covers.

The insulation material, nylon coating and heaters must be stripped and replaced during periodic structural inspections as defined in the premise outlined in table 15. The stripping and replacing of the insulation material inside the confined area of the wing tanks is a significant maintenance task with the overall replacement requiring approximately twice the original installation time.

The safety aspects of all candidates are identical and are a result of the presence of additional electrical wiring in the fuel tanks. The physical integrity of the heating elements and associated wiring insulation and isolation from fuel vapors must be assured by the design methods employed to insure safe and dependable operation.

7.3 Summary Comparison and Conclusions

Each of the candidates were compared to each other and to the baseline airplane in Section 7.2. All of the candidates are capable of operating over flight ranges of 926, 3704, and 9260 km (500, 2000, and 5000 n.mi.) in the extreme hot and cold environments developed in Section 3.2 using a fuel with the following properties relaxed from ASTM D 1655-81 Jet A fuel:

Freezing Point, °C (°F)	-20	(-4)
Thermal Stability, JFTOT, °C (°F)	204	(400)
Aromatic Contents, % Volume	35	
Viscosity, mm^2/s (cSt) at -17.8°C (0°F)	15	(15)
Reid Vapor Pressure, kPa absolute (psia)	13.8	(2)
Flash Point, °C (°F)	27	(80.6)
Lubricity, WSD, mm	0.45	

For purposes of this study, these fuel property changes are assumed to be the maximum for which system design changes can compensate. Based upon this assumption, the candidate which best meets the goal of operating the aircraft with minimum performance penalty for the specified payload of 18 144 kg (40,000 lb) with low cost fuel is Candidate A. Although this configuration has the highest fuel consumption, its direct operating costs are lowest because it does not entail the high development and maintenance costs necessary to obtain a suitable tank insulation. However, as fuel costs increase in the future, Candidate B provides an increasing cost advantage and is therefore considered the preferred system concept in the long term.

8. RECOMMENDATIONS

Based upon the various analyses conducted and the results obtained in this study, it is recommended that additional research and development efforts be undertaken prior to establishing limits for the allowable freeze point and thermal stability of commercial aviation kerosene. Specifically, these include:

- Experimental determinations of jet fuel properties at and below its freeze point with emphasis on heat of fusion, thermal conductivity, and specific heat.
- Experimental investigation of wax deposition and its effects on heat transfer into the fuel immediately adjacent to the bottom of a fuel tank.
- Details of the freeze-out phenomenon and its influence on fuel hold-up, determined by experimental investigation and analytical modeling.
- Develop a dynamic model for the fuel coking rates, capable of correlating the laboratory characterization of jet fuel in a fuel system operated at high temperature. This model must account for the chemical kinetics of the reactions occurring in the liquid phase as well as the diffusion of the primary reactants and products throughout the fuel itself.
- With the help of the dynamic model, develop a small scale test that will correlate with the maximum temperature that a fuel can be subjected to in a full scale engine fuel system before breakdown occurs. This test should be capable of predicting the impact of time on the breakdown temperature.
- Development of a lightweight insulation which is compatible with jet fuel and which will adhere to fuel tank surfaces when subjected to a wide range of environmental temperatures.
- Analyze requirements for airport facilities to refuel aircraft at environmental temperatures below the fuel freeze point. The cost of operating and maintaining these facilities as well as the equipment required to transport the fuel from the refinery to its destination must be included in evaluating the practicality of raising the fuel freeze point.

Implementation of these recommendations will provide industry with knowledge that can provide a basis for making more accurate assessments of fuel property relaxation on aircraft performance and lead to fuel system designs which are more practical and possibly less expensive than those identified in this report.

APPENDIX

The effects of the increases in operating empty weight (OEW) and engine specific fuel consumption (SFC) for the L-1011-500 are shown as a series of sensitivity curves, figures 62 through 67. These curves can be used to determine the increase in takeoff gross weight (TOGW) and block fuel weight necessary to accommodate changes in OEW and SFC.

The effects of increasing OEW were calculated with reference to the baseline OEW of 111 307 kg (245,390 lb) using the Lockheed Aircraft Mission Analysis Program. The flight profiles for the increased OEW aircraft were flown using the same rules as the baseline OEW aircraft and the profiles are similar to the zero OEW baseline profiles except on the hot day 9260 kilometer (5000 n.mi.) mission where thrust limitations force the profile to lower altitudes for the higher OEW increases. This change in altitudes causes nonlinearity in the Δblock fuel and ΔTOGW lines for this case (figure 64) whereas the lines for all of the other cases are linear. The change in TOGW is greater than the change in block fuel weight for a given increase in OEW because of the fuel necessary to carry the additional fuel and because of reserve increases.

The effects of increases in engine SFC were also calculated using the Lockheed Aircraft Mission Analysis Program. The same OEW of 111 307 kg (245,390 lb) was used for all of the missions and the flight profiles are all similar to the zero percent change in the SFC baseline profiles that they are referenced to. Increases in block fuel and TOGW were calculated by increasing the SFC over the entire length of the mission. Because Domestic Reserves and International Reserves Part II are based on the OEW there are no increases for them for an increase in SFC. International Reserves Part I changes a little however, because of additional fuel flow at the end of the last cruise segment.

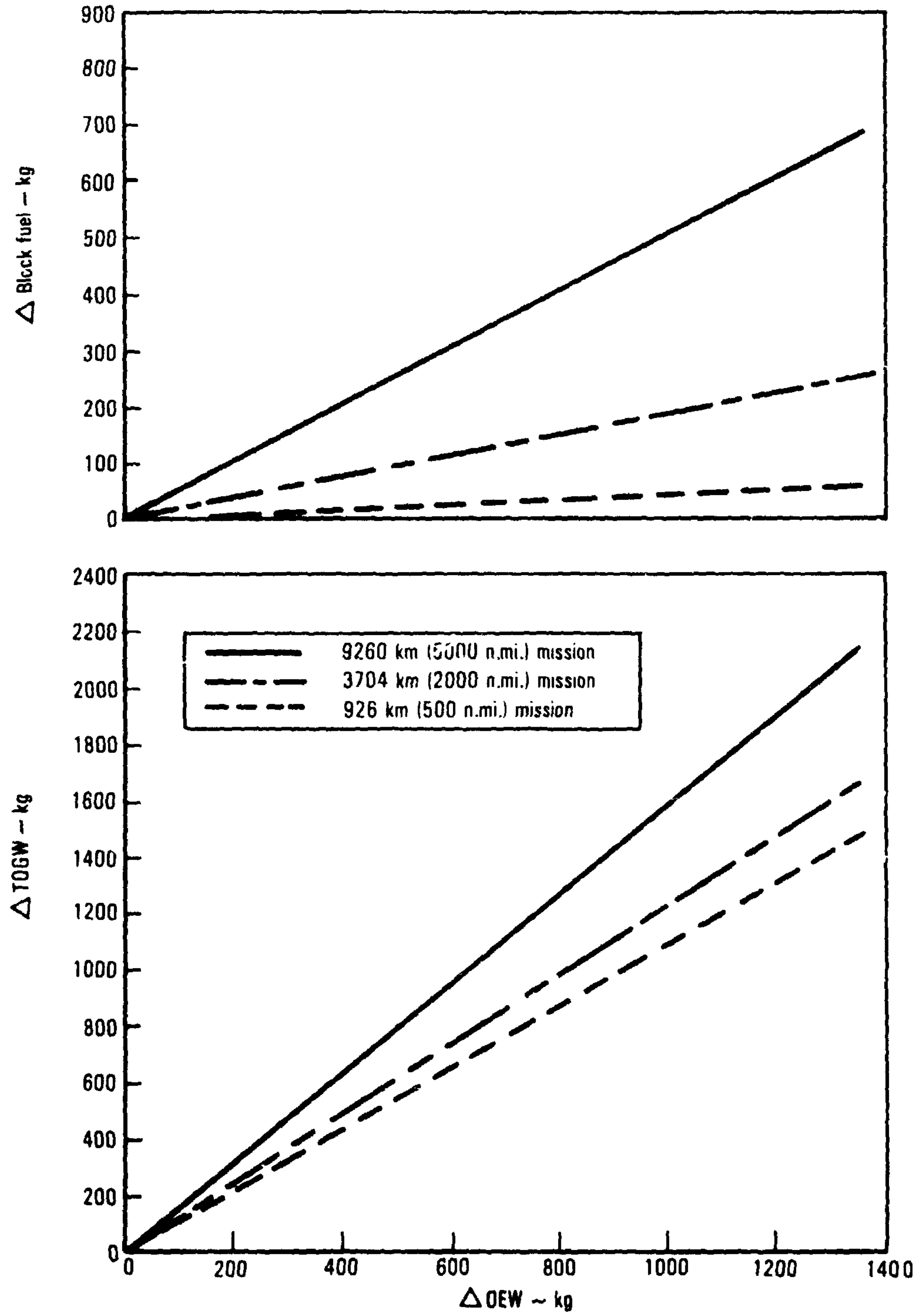

Figure 62 - Effect of OEW on block fuel and TOGW - cold day.

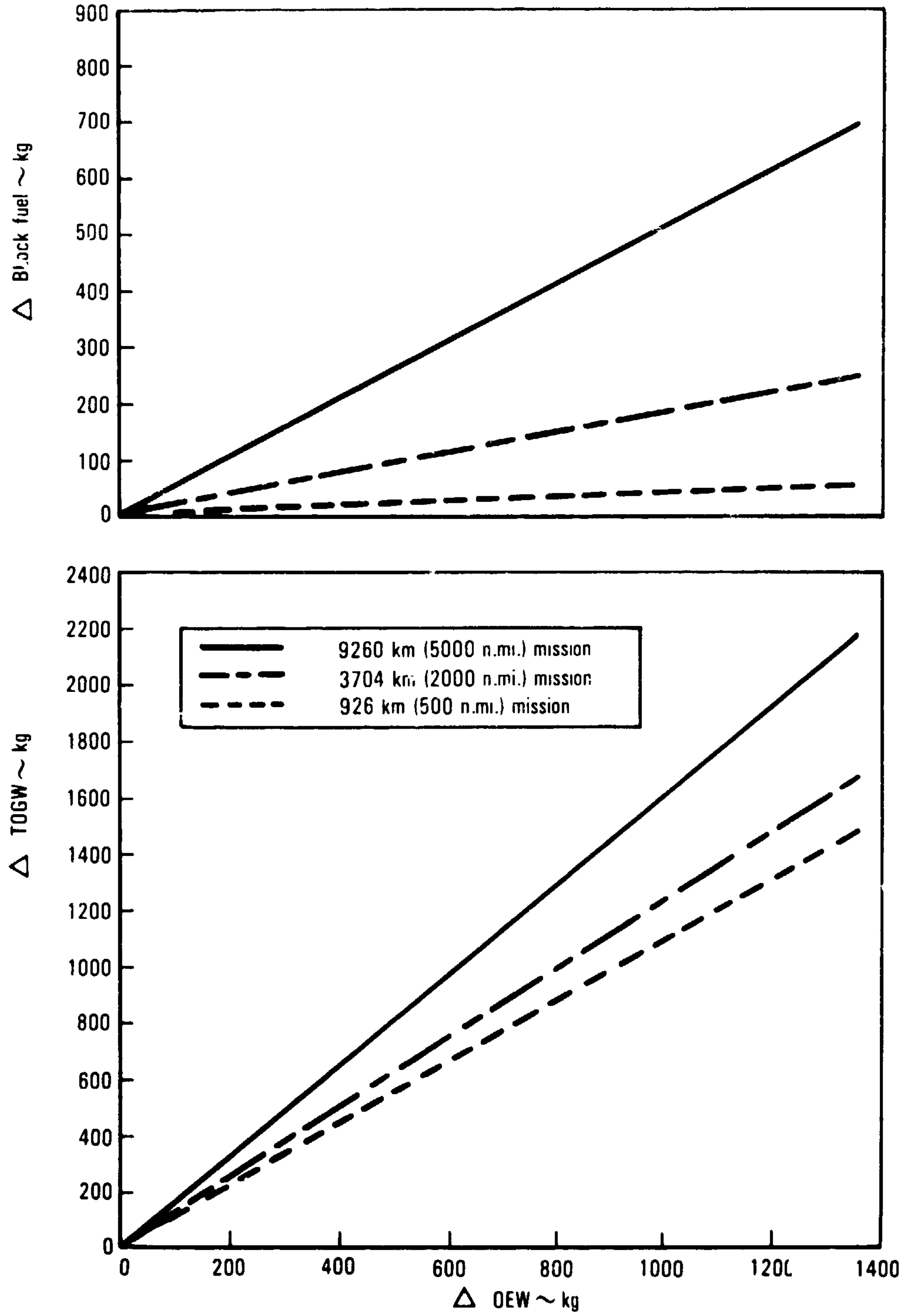

Figure 63 - Effect of OEW on block fuel and TOGW - standard day.

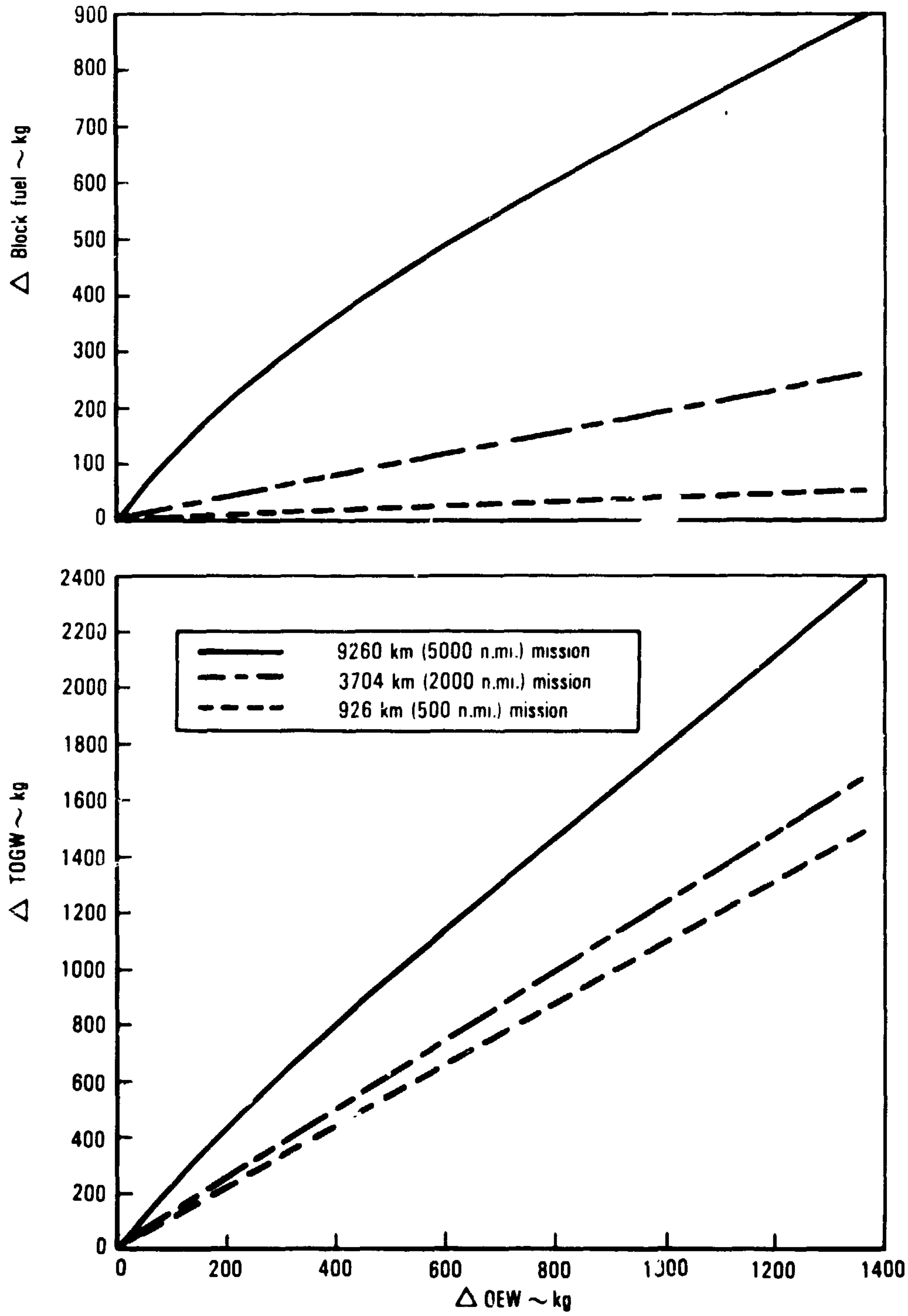

Figure 64 - Effect of OEW on block fuel and TOGW - hot day.

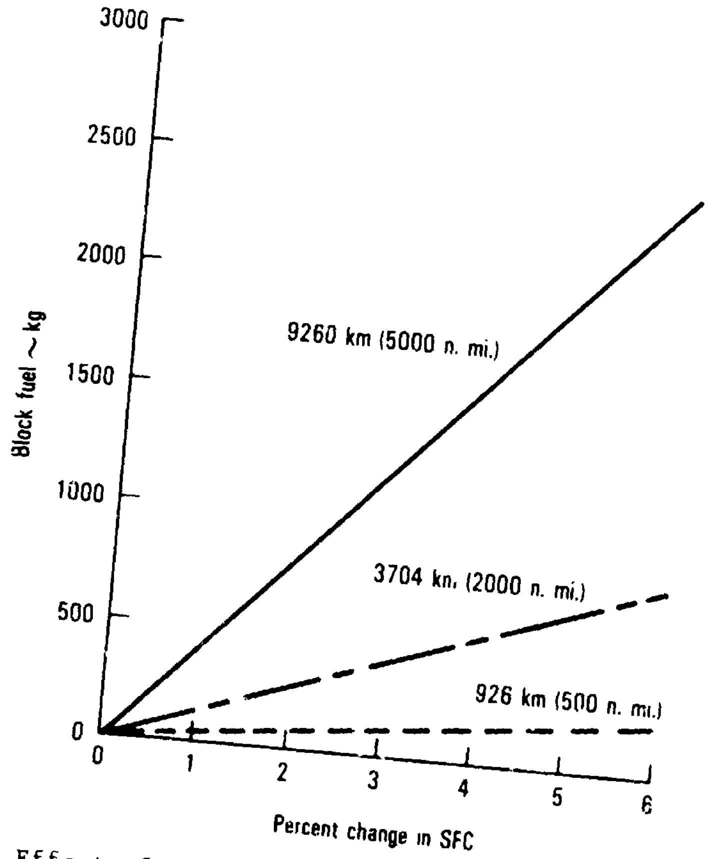

Figure 65 - Effect of percent change in SFC on block fuel - cold day.

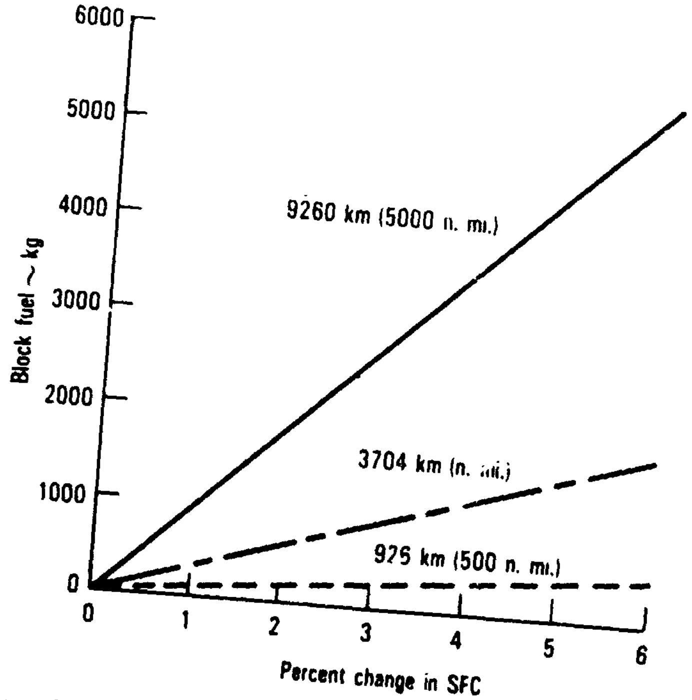

Figure 66 - Effect of percent change in SFC on block fuel - standard day.

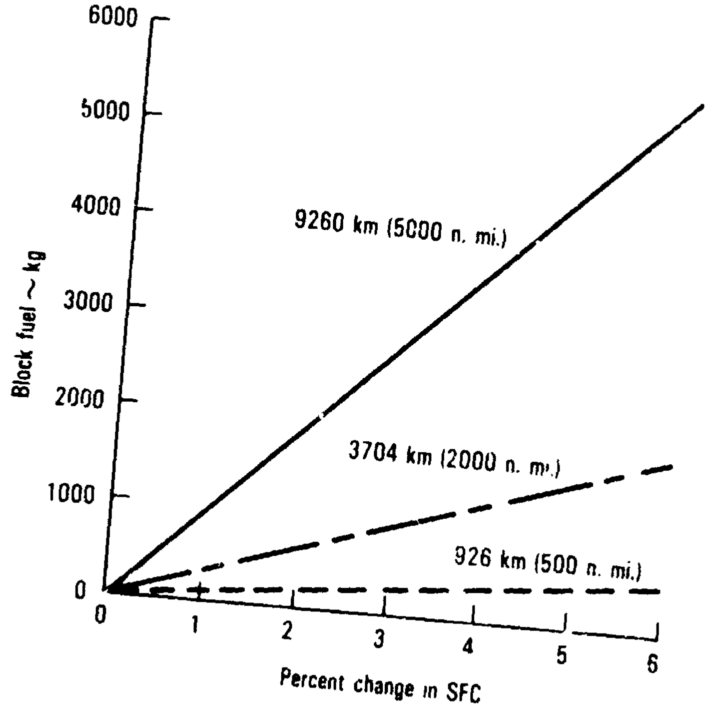

Figure 67 - Effect of change in SFC on block fuel - hot day.

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