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A129 HELICOPTER
LABORATORY AND TESTING METHODS FOR: FLIGHT CONTROL SYSTEM, FUEL SYSTEM, ENGINE COMPARTMENTS

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ABSTRACT

All that is hereunder described regards the basic criteria that were used in the experiment of the fluid systems and engine compartment for the A129 helicopter.

Starting from preliminary tests, whose purposes were to define the flame temperature, it goes on with fire tests and fire resistance tests for a real compartment, reproducing the helicopter operating conditions as air flow, loads and vibrations.

For the fuel system it will be analized the possibility to reproduce in laboratory all the helicopter operating conditions which concur to give vapore lock formation, particularly under high altitude, high temperature and high load factors.

For the hydraulic system and flight controls, the tests will try to reproduce any possible failure examining helicopter controllability and the transient state in the most critical conditions, giving a particular attention to pilot- helicopter integration.

INTRODUCTION

Facing the experiment of the A129 helicopter, (figure 1), regarding power plant installation and hydraulic/fuel systems we have stated the following criteria:

1) Laboratory tests will be realized in order to reproduce the most realistic in-flight conditions.
2) Flight tests, where this is possible, will be a simple verification of the experimental laboratory testing results.

The intents for what we have taken this address are:

1) In flight experiments:
    a- High costs
    b- Great technical risk factors
    c- Extreme difficulties or impossibility to experiment at limit conditions regarding stress and ambient factors.
    d- Very limited helicopter and flight time availability.
2) Laboratory experiments:
    a- Limited costs
    b- Using sufficient equipment it is possible to reproduce any flight condition very closely.
    c- Worthless risk factors or very limited
    d- Extreme possibility to control with an exact and reproducible manner either the ambient or the stress conditions, reaching and exceeding any limit condition without increasing testing risk factor.

Considering the various helicopter systems the laboratory experiment has been divided as follows.

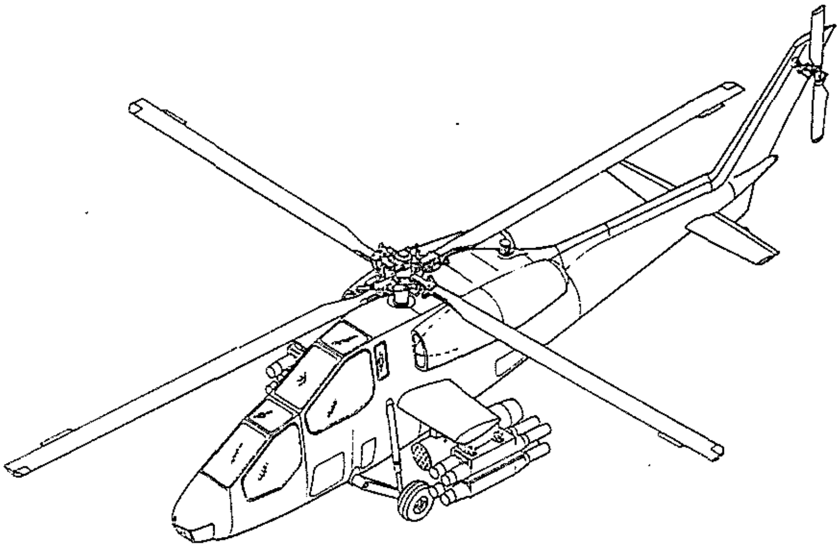


FIG. 1

ENGINE INSTALLATION

We will limit the consideration about the type of experiment deemed necessary to demonstrate the fire resistance of an engine compartment.

Usually the civil and military requirements share the material to be used in the engine compartment in classes.

The most important are: fire proof and fire resistant material.

It was tried further (expecially from civil authority) to determine a test method (see Lennox type burner) by which sample material should be tested without taking into consideration the real operative condition.

Such requirements only give general indication without making distinctions from helicopter installation to an airplane installation, considering besides the same taken light and heavy aeronautical installation.

It becomes therefore extremely important to determine the installation type and the work in order to take into account the real construction conditions.

During design and experiment for the A129 helicopter (figure 2) the following criteria have been taken into account.

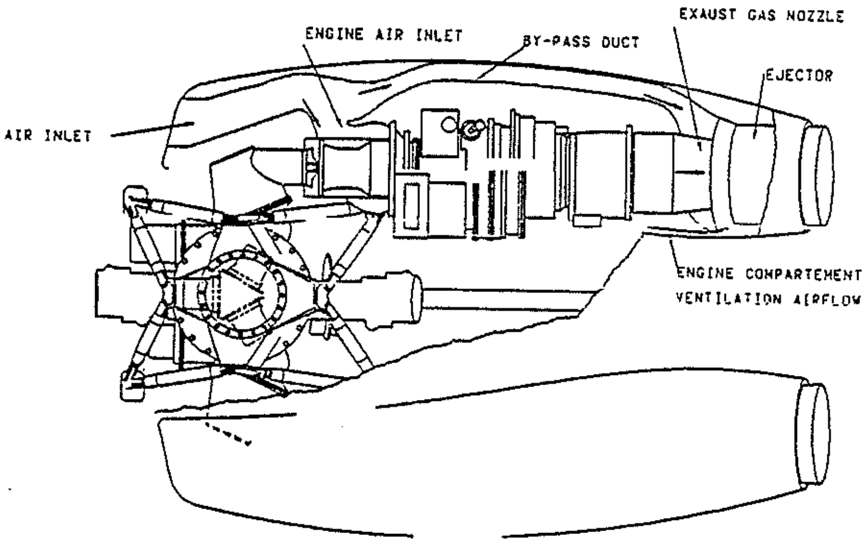


FIG. 2

- Spreaded Flame -

The engine compartment structure in its whole globality will be able to contain a spread flame which may develop in any aerodynamic flight condition, without permitting that the flame propagation may reach the other parts of helicopter.

The typical flame to be considered is equal to the one consequent to a loss or a great quantity of fuel and/or oil inside the compartment, with a subsequent combustion of the escaped fluid.

For these reasons both inside and outside condition, which may influence the combustion, will be taken in account.

Pointing out the most important inside conditions we can find: compartment volume, airflow through compartment (ventilation) the engine installation configuration drain lines and compartment geometry.

It is otherwise important to considerate the configuration and the operation of the fuel and lubrificating systems.

Considering the outside conditions it may be adduced the ventilation caused by external airflow, the geometry of the compartment and the accessories air inlet.

- Local Flame -

This kinf of flame in to be defined as a concentrated flame but at very high temperature which can strike particularly critical structural zone such as engine support, interconnected bulkhead etc.

These tests will be accomplished using a Lennox type burner.

Starting from these concepts the experiment has been planned as follows below.

- Preliminary Tests -

To reply in advance to the answer, how much the real flame temperature inside the engine compartment in case of fire should be a compartment, having the same volume of the A129 helicopter engine compartment (figure 3) was realized.

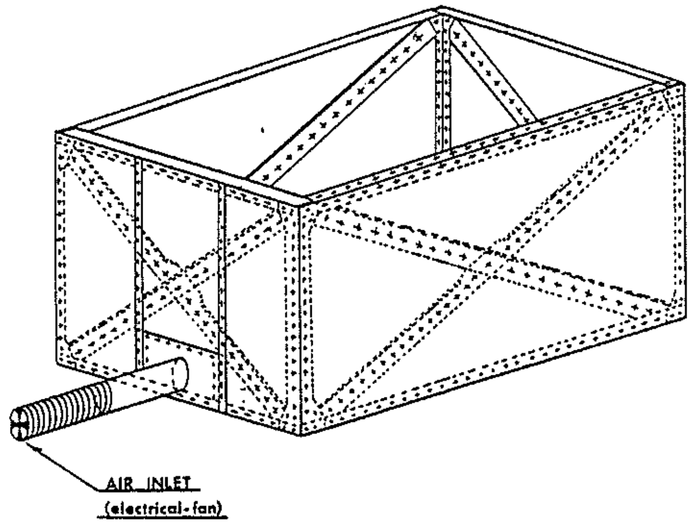


FIG. 3

An electrical fan has been employed to delivere air into the mock-up at a flow as in the real engine compartment, about five cubic meters per minute.

32 CR-AL thermocouples fitted inside and an AGA thermovision system were used in order to measure the inside and surface temperature (figure 4 and 5).

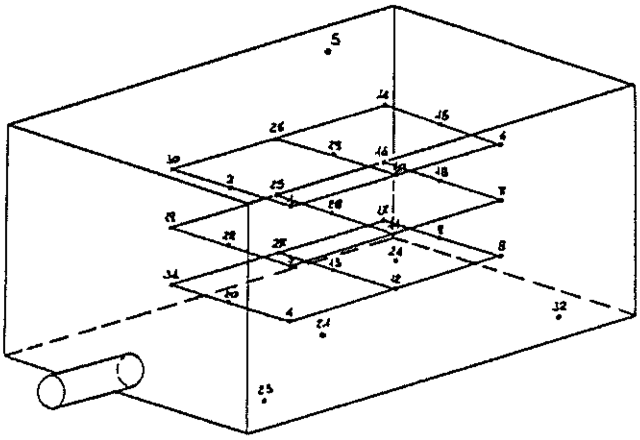


FIG. 4

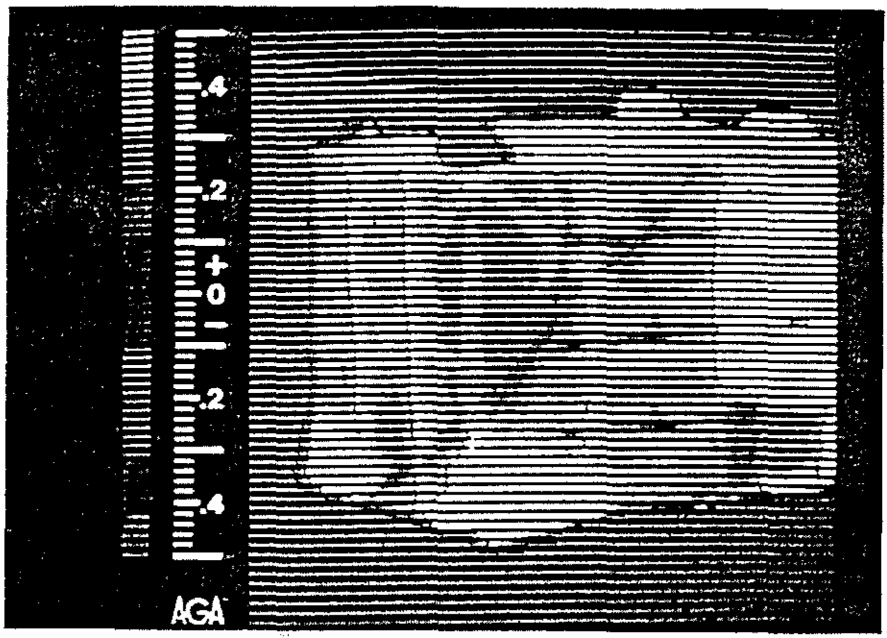


FIG. 5

During this test, as it is shown in figure 6, the temperature has always remained below of 750°C.

TIME	THERMOCOUPLES TEMPERATURE °C															
	1/2	3/4	5/6	7/8	9/10	11/12	13/14	15/16	17/18	19/20	21/22	23/24	25/26	27/28	29/30	31/32
1 Minute	395/363	384/365	340/457	437/409	360/381	402/454	381/457	446/397	304/396	327/311	OFF	113/55	159/159	137/147	150/126	105/108
3 Minute	534/550	492/461	504/592	553/503	525/567	514/466	430/575	580/587	510/660	512/230	OFF	527/135	553/530	506/495	485/467	440/440
5 Minute	617/610	564/495	562/564	570/526	371/592	574/529	331/565	577/554	500/530	566/263	OFF	103/82	539/558	520/561	589/582	581/671
7 Minute	602/606	558/498	566/550	569/521	359/573	568/532	277/567	572/564	505/543	565/230	OFF	69/50	539/565	528/560	527/584	520/678
9 Minute	600/607	558/495	564/551	569/518	364/579	567/529	290/564	579/557	504/548	557/240	OFF	45/39	539/567	521/561	598/593	485/678
11 Minute	592/596	552/495	562/548	565/505	374/579	561/522	297/554	574/559	501/540	557/232	OFF	48/40	530/567	511/556	599/599	515/674
13 Minute	624/640	573/525	582/623	602/557	470/593	588/554	371/629	603/607	555/591	581/294	OFF	48/40	481/OFF	453/553	520/516	471/668
15 Minute	650/661	595/527	583/617	608/560	471/601	588/553	373/621	605/624	554/582	582/281	OFF	48/40	477/OFF	461/554	531/509	469/621

FIG. 6

Considering the results abovementioned tests for the actual engine compartment have been planned.

This mock-up is really equal in geometry and material to the engine compartment and his adjacent zones for the A129 included an engine mock-up.

The typical load and vibration conditions of the engine installation will be reproduced.

The engine ejector flow is generated by an external fan.

A second fan will generate the external air flow up to 40 ÷ 50 Kts (figure 7).

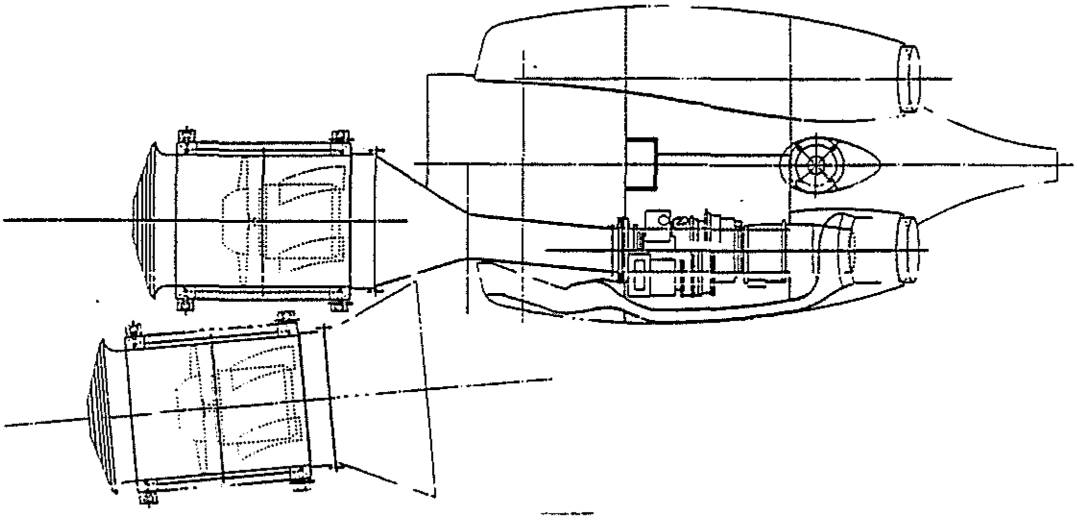


FIG. 7

- Tests with Local Flame -

This sort of tests allows to verify, using Lennox burner with standard flame type (figure 8), the resistance under operating load of some particular parts or surfaces (supports, engine mounting, seals, fire walls, etc.) - figure 9.

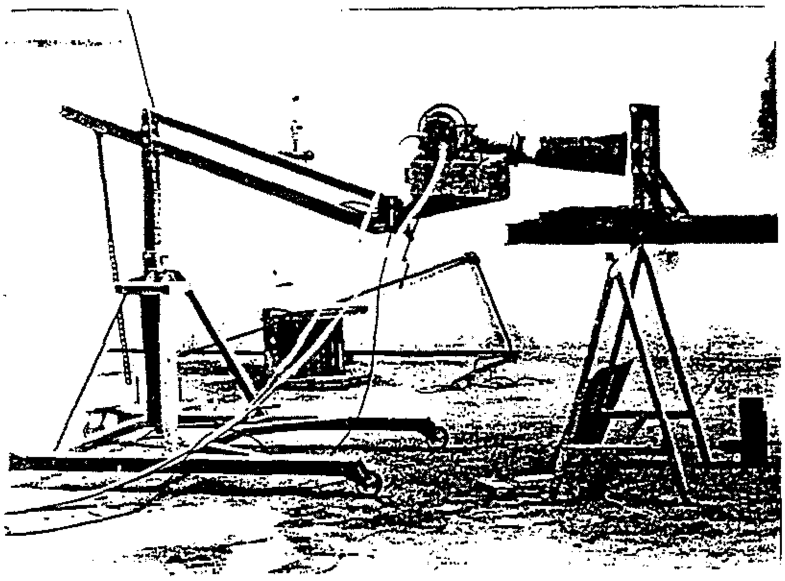


FIG. 8

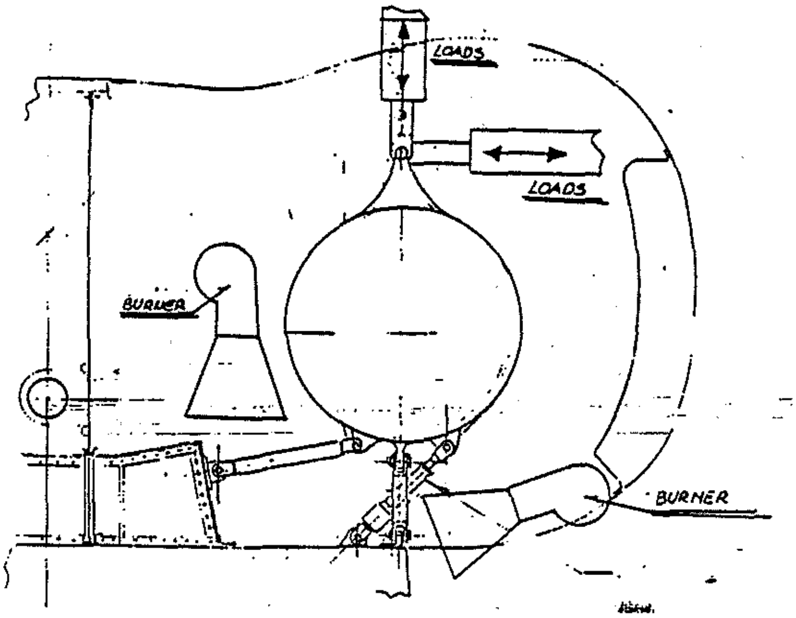


FIG. 9

FUEL SYSTEM

It is now considered a suction fuel system as for the A129 helicopter (figure 10).

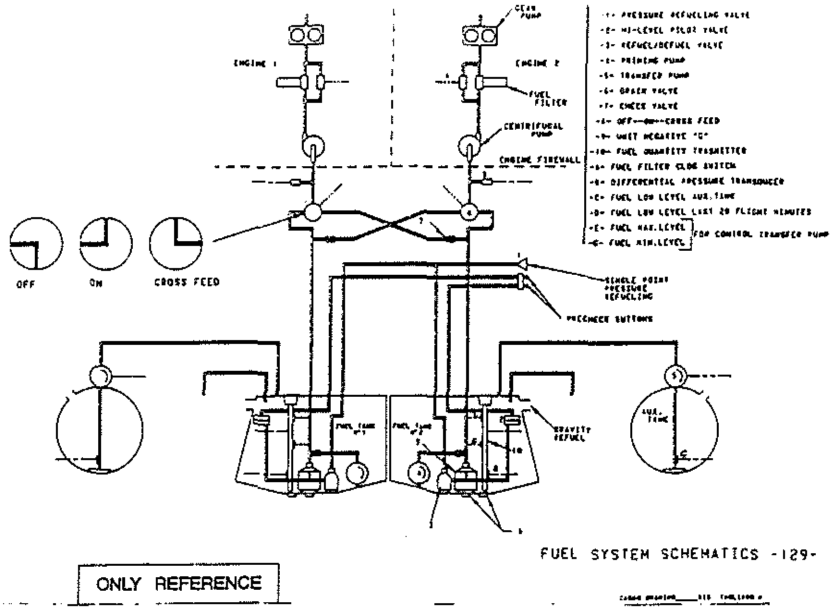


FIG. 10

The most critical aspect to be investigated is its capacity to feed the engine.

This may be accomplished not only supplying fuel at the correct pressure, but also if no cavitation occurs at the engine pump.

To be sure that no cavitation occurs at the engine pump the fuel system shall be investigated under the most critical conditions which concur to cavitation process, such as the following:

- fuel at high temperature
- high altitude
- flight manouvres with load factor > 1 (for the A129 helicopter the maximum permanent load factor during operation will be 3.5 g's).

From this point of view to reproduce in laboratory the operation in the abovementioned conditions for the fuel system, the most critical to be simulated is the one regarding the load factors, and for this reason it is hereunder further exibited.

- Influence of a two-phase fluid (vapor and liquid fuel coexistance) on the engine inlet pressure -

Taking into consideration the figure 11 showing how much the engine inlet pressure may decrease to very low values, so close to the fuel vapor tension, it may be evident that vapor lock formation, is not avoidable.

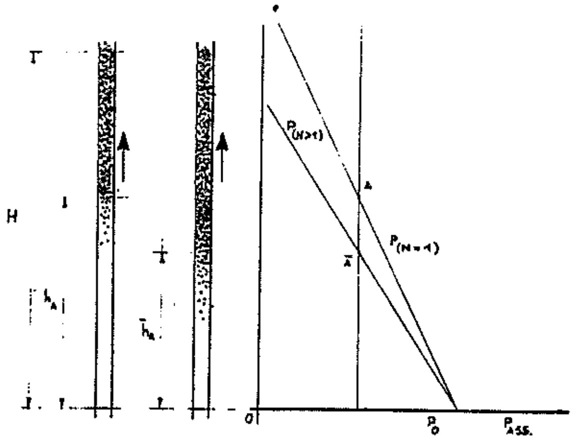


FIG. 11

The maximum vapor rate admitted at the engine inlet, depends on the engine specification.

For the "GEM2" engine installed on the A129 helicopter this value is 31%, to which corrisponds a V/L rate value of 0.45.

The V/L rate value for a fuel used in a specified fuel system is theoretically defined by the following relation, also reported in the ARP-492 report.

$$V/L = 1,54 * K \cdot \left[\frac{P_{amb} - P_{mot}}{P_{mot} - P_{tup}} \right] * \left[\frac{T_c + 460}{95,8 + 0,07\ T_c} \right]$$

However this equation is not considered sufficiently completed, since some important parameters like the system geometry and the time that the fuel remains in the fuel feed line are not taken into consideration.

One more negative point of the abovementioned relation is that the engine inlet pressure is required to be known.

Going ahead and considering simply a straight tube line section as shown in figure 11, it is evident that the system pressure decreases beyond the vapor tension only when the point "A" is overcome.

This point upward vapor locks formation is expected.

It is known that the pressure value in any point of the line is given by the following relation:

$$P = P_o - n\gamma h - \Delta P_c(h) - \Delta P_d(h) \qquad (2.1)$$

for which

P_o = absolute pressure at initial condition
n = load factor
h = geometric fuel height
γ = fuel specific weight
ΔP_c = local pressure losses
Δp_d = distributed pressure losses.

In this equation the factor $n\gamma h$ is purely geometric, while the $\Delta P_c(h)$ and the $\Delta P_d(h)$ factors depend up on the fuel velocity and on which kind of flow is passing through the fuel feed line.

Since here at has been considered the presence of a two-phase fluid; these factors are also in relation with the percentage of the vapor and with the physical characteristics of the vapor locks.

Going on and assuming that in each point of the fuel system, wherein a presence of the two-phase fluid is expected, the pressure should correspond to the vapor pressure; when the pressure decreases, the vapor pressure must decrease and for this reason also the liquid temperature goes down.

But the liquid becomes colder if a part of it evaporates; the volume of this part is given by:

$$QV_{vap} = QV_{liq} \frac{C_{sp} \cdot \Delta T / \gamma_{vap}}{C_{evap} / \gamma_{liq}} \qquad (2.2)$$

for which

QV_{vap} = vapor volume flow
QV_{liq} = liquid volume flow
ΔT = delta temperature
γ_{vap} = vapor specific weight
γ_{liq} = liquid specific weight

C_{evap} = evaporating heat
C_{sp} = fuel liquid specific heat

The vapor percentage is influenced in its turn by the pressure and therefore a vicious circle is encountered, for which a theorical solution is not easily attainable.

For such reason, although interative calcutation methods may be utilized, it is deemed imperative that an answer to this problem can be only given by tests.

- Simulation in Laboratory -

To make the test as close as possible to reality flight load simulation is mandatory.

Starting from the fact that it is not possible to create a load factor grater than 1, artificially in laboratory, a method to simulate it in laboratory must be found out.

On the base of the equation 2.1 the parameters which may be controlled are:

1) Ambient pressure P_0

It is possible to modify the ambient pressure in order to achieve the correct pressure value at the engine inlet without the modification of the fuel system.

However, the ambient pressure is required to reach very low values that should cause the fuel to boil even in the tank.

2) Geometric height h.

It is simply possible to increase the real fuel height for n time, that is:

$$h_{sim} = n \cdot h_{real}$$

making thus it is obtainable the desired pressure value, because an excessive tube lenght is required this will prolong the time for the fuel to run the line, giving rise to an unnatural vapor lock formation.

3) Local pressure drop.

It is possible to increase the value of some local pressure drops by restrictions along the fuel line in order to cause a very high velocity of the fluid and this will correspond to very high changes in pressure.

But this sharp pressure reduction, downstream the restriction, will cause a very high fuel flow speed with consequent abnormal formation of vapor locks.

4) Diffused pressure drop.

It is possible to increase the value of diffused pressure losses.

These kinds of losses, when the fluid flows at high temperature, are very low and may reach very high values only when a long tube line is employed.

However modifying as deemed necessary the diameter size of the tubing (for which losses have a factor relation of the fifth power) it may be obtained significant pressure losses.

What has been presented shows that a modification of a single parameter is non sufficient to create an adequate simulation relative to the vapor lock formation in the fuel system when operates under load factors > 1.

Only operating at the same time on the different abovementioned parameters it is possible to make a valid experiment.

The following criteria will be particularly adopted.

a) To reach the tension vapor in the simulation system, at the same point where it is obtained without any factor load.

b) The fuel subjected to vapor lock formation will remain the same time in the line either with load factor of one or more.
This may be easily obtained increasing the geometric fuel height over the starting separation point and reducing at the same time the diameter size of the tube.

In fact considering (figure 12):

L_h = fuel line length
h_t = geometric height in the real system
h_1 = separation point height
Δ_h = increased head
t = time that fuel remains in tube line
D_o = tube diameter in the real system
D_1 = tube diameter in the simulated system
A = separation point

For the different formulated cases it is obtained:

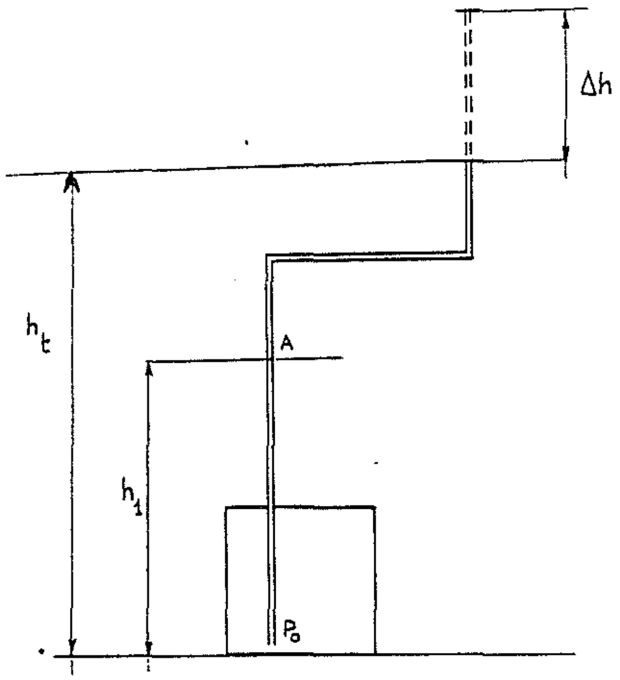


FIG. 12

$$P_a = P_{tvp} \qquad (2.3)$$

$$h_1 = \frac{P_o - P_{tvp}}{n\gamma} \qquad (2.4)$$

$$\Delta_h = (n - 1) \cdot (h_t - h_1) \qquad (2.5)$$

$$\Delta_1 = \Delta_o \sqrt{L_h / (L_h + \Delta_h)} \qquad (2.6)$$

Meanwhile the increment of the distributive pressure losses shall be taken into consideration so that an increment of the fuel height greater than the one determined with the equation 2.5 will be applied.

For which the assumed fuel height will be:

$$\overline{\Delta_h} = \Delta_h + \frac{\Delta}{\gamma} \left(\Delta P_d (\Delta_o) - \Delta P_d (\Delta_1)\right)$$

It is considered that this type of simulation gives sufficient valid results.

Improvements may be obtained acting on the ambient pressure P_o and /or correcting size and length of the horizontal part of the system in which the vapor lock formation occurs.

Making these principles as starting points we are now going ahead with a series of experimental tests on a semplified fuel system model. (figure 13 - 14)

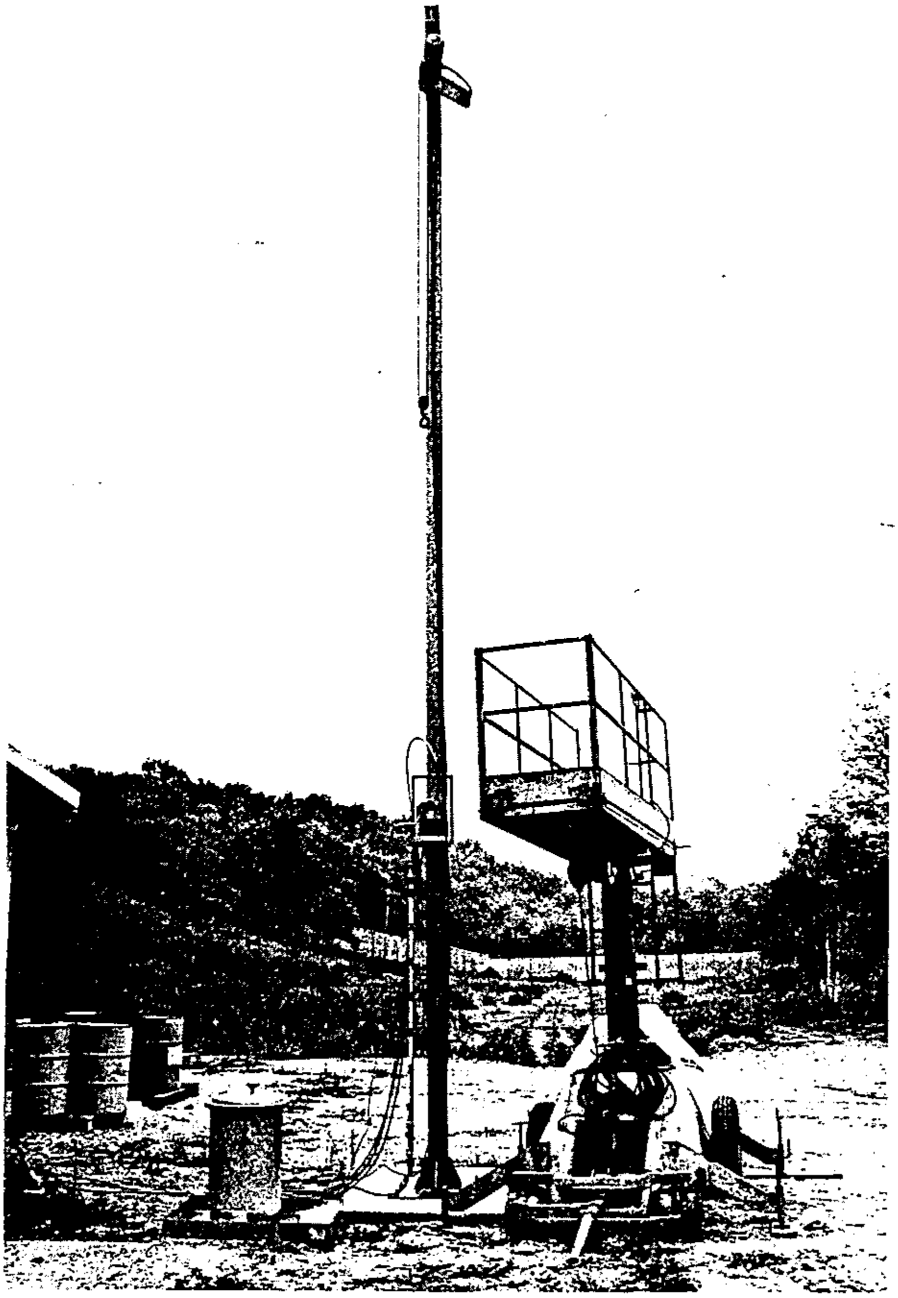
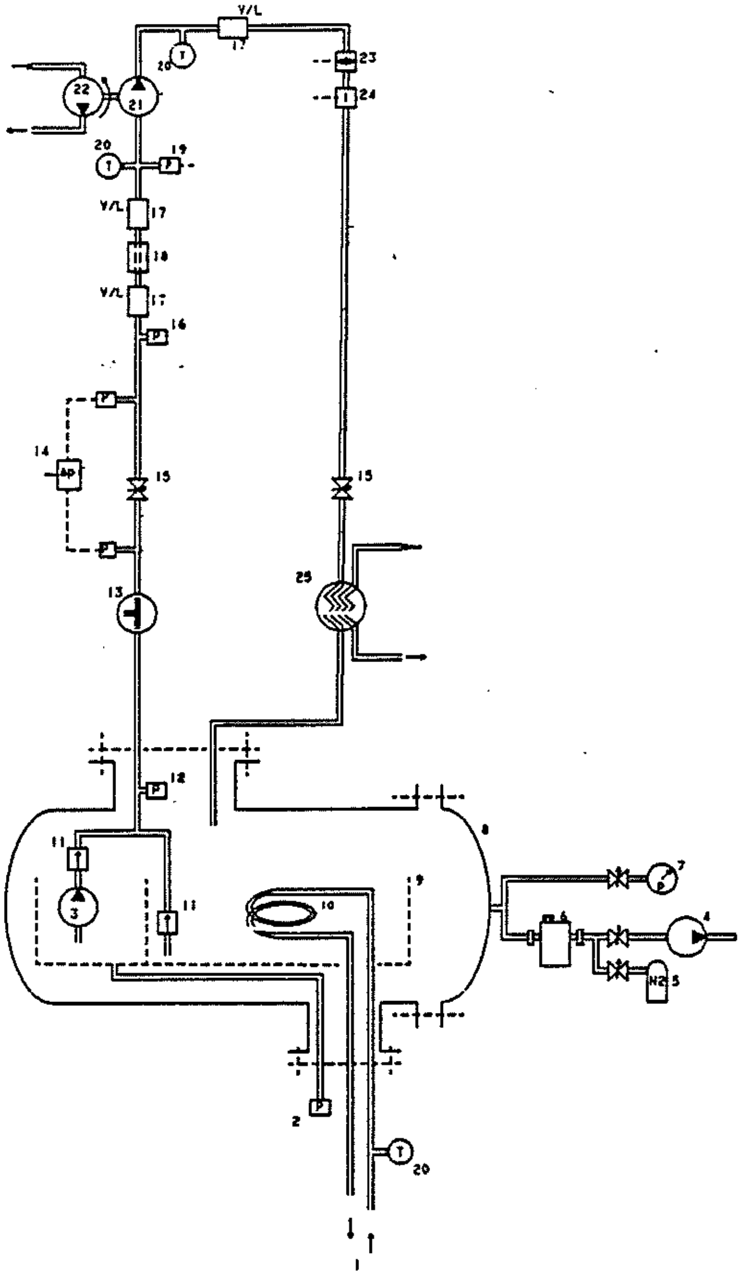


FIG. 13



1 HOT OIL FLOW
2 DIFFERENTIAL PRESSURE TRANSDUCER
3 FUEL PUMP
4 VACUUM PUMP
5 N2 BOTTLE
6 FUEL VAPOR CONDENSER
7 ALTIMETER
8 DEPRESSURIZED CYLINDER
9 FUEL TANK
10 HEAT EXCHANGER
11 CHECK VALVE
12 PRESSURE TRANSDUCER
13 SHUT OFF VALVE
14 FLEXIBLE FUEL LINE
15 SERVO VALVE
16 PRESSURE TRANSDUCER
17 IKOR V/L METER
18 TRANSPARENT FUEL LINE
19 PRESSURE TRANSDUCER
20 TEMPERATURE SENSOR
21 SUCTION PUMP
22 IKOR V/L METER
23 TURBINE FLOW METER
24 ULTRASONIC FLOW METER
25 HEAT EXCHANGER

FIG. 14

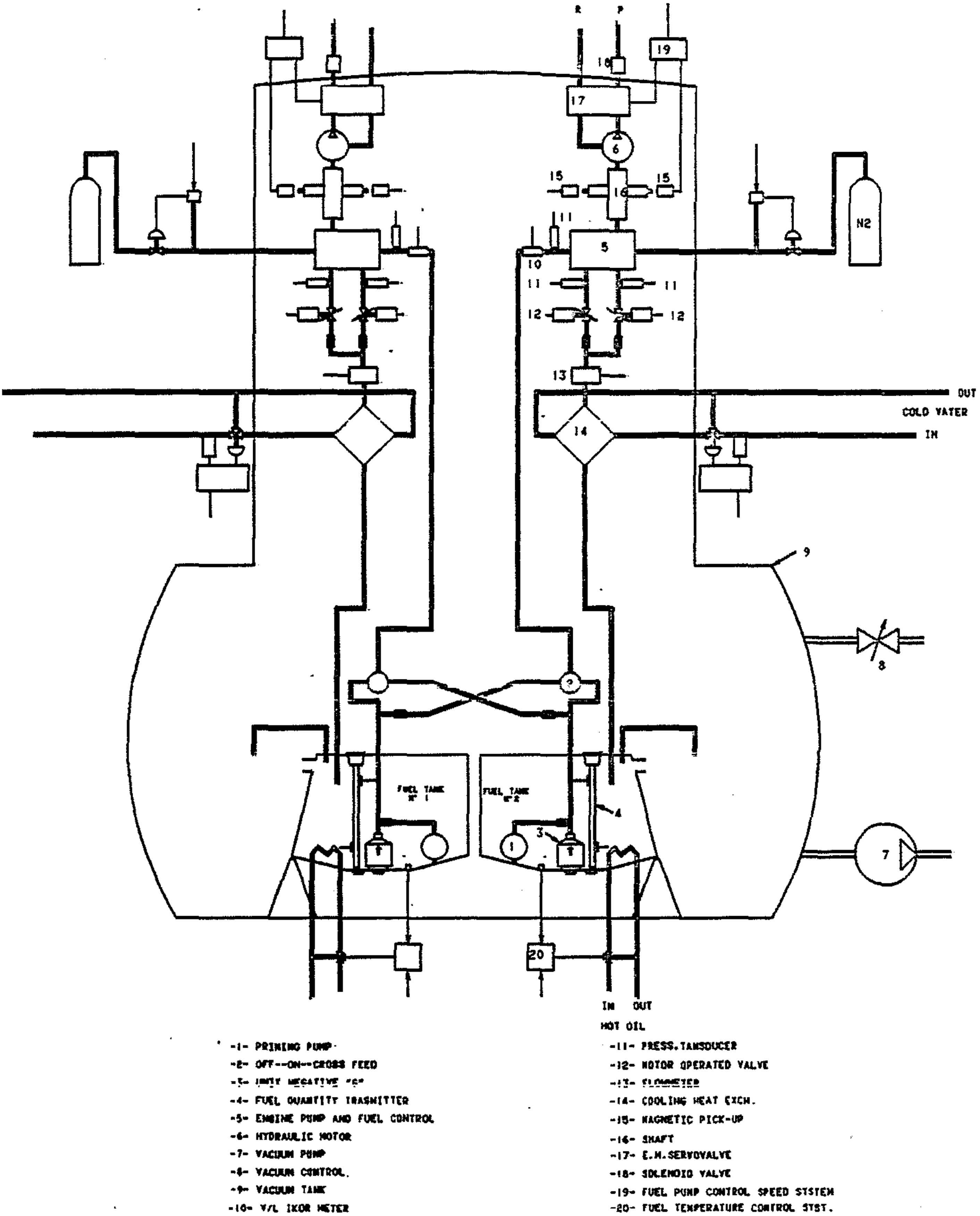

R
P
18
19
17
6
15
16
15
11
5
N2
10
11
11
12
12
13
OUT
COLD VATER
IN
14
9
8
2
FUEL TANK N° 1
FUEL TANK N° 2
4
3
1
7
20
IN OUT
HOT OIL
-1- PRIMING PUMP
-2- OFF--ON--CROSS FEED
-3- UNIT NEGATIVE "G"
-4- FUEL QUANTITY TRASNITTER
-5- ENGINE PUMP AND FUEL CONTROL
-6- HYDRAULIC MOTOR
-7- VACUUM PUMP
-8- VACUUM CONTROL.
-9- VACUUM TANK
-10- V/L IKOR METER
-11- PRESS.TANSDUCER
-12- MOTOR OPERATED VALVE
-13- FLOWMETER
-14- COOLING HEAT EXCH.
-15- MAGNETIC PICK-UP
-16- SHAFT
-17- E.M.SERVOVALVE
-18- SOLENOID VALVE
-19- FUEL PUMP CONTROL SPEED STSTEM
-20- FUEL TEMPERATURE CONTROL STST.

FIG. 15

The fuel is sucted from a tank where it is possible to control ambient pressure and fuel temperature.

The feeding line length may be varied as required and some parts have been manufactured using transparent material in order to check vapor lock formation.

Speaking about the A129 helicopter fuel system, it is now realizing a structure reproducing exactly the helicopter fuel system (see figure 15) which will utilize the same components which are to be installed on the helicopter included the engine pump.

The modifications performed regard only the fuel height and the tube size in that part of the system where vapor lock formation is expected.

To simulate the altitude the whole system will be enclosed integrally in a barometric chamber, in which the ambient pressure is controlled.

The fuel temperature will be altered exchanging heat with a diathermic oil flowing in a separate plumbing.

On the A129 fuel system it will be furthermore studied the sunlight effect for the fuel contained in the tank and in the tube line.

In fact it is known that in an aircraft exposed to sun radiation for some hours, the fuel becomes warmer and warmer far over the ambient temperature.

This could be critical considering the most volatile fluids, as is JP-4.

That is why we will irradiate the fuel system installed in a helicopter structure by a series of solar lamps. (figure 16)

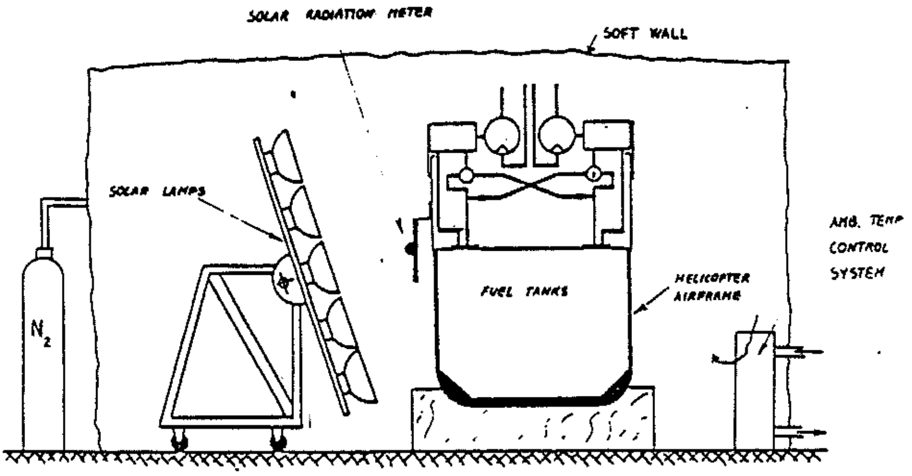


FIG. 16

. All this sort of experiment shall be performed in a neutral atmosphere becouse of the danger caused by fuel at high temperature.

To eliminate the hazard of such an ambient, neutral gas as nitrogen will be used to create a neutral atmosphere.

FLIGHT CONTROL SYSTEMS

The project hypotesis of the A129 helicopter are:

- Mission continuity after a first failure; safe landing after the second failure.
- Chance of the ballistic damage.
- Operating ambient temperature of 50°C.

These points have brought to this helicopter configuration:

- Pilot and copilot tandem seats.
- Non-powered mechanical mode, used by pilot in emergency.
- FBW mode, used by copilot in emergency.
- Tail rotor servo FBW in normal mode and non-powered in emergency mode.

From these assumptions, it is easy to note the flight control system's new aspects; in the same time it appears clear the inherent high technical risks.

Therefore, it is deemed necessary to carry out a deep laboratory testing phase, so that to have a complete analysis and evaluation of the system, reducing the operating "risk factor".

In detail the following aspects will be investigated:

- Analyze the pilot's capability to control the helicopter without using hydraulic power, and to carry out a "safe landing".
- Estimate the copilot's capability to control the helicopter in FBW mode, within a normal flight mission and expecially during a long return flight.
- Estimate the transient during the change from normal control mode into FBW or pure mechanical modes.
- Estimate the transients due to SCAS turning on and off.
- Estimate the behaviour of the hydraulic system at the maximum temperature limits.
- Estimate the consequences of the failures which can be simulated.

It is to underline that the subject of this testing is not only a circuit or a system, but the crew-helicopter integration.

To do that, it is necessary to have at disposal an hardware capable:

- To simulate the helicopter geometrical configuration by using the actual components in respect to:
  a) Flight controls
  b) Servoactuators and hydraulic system

  c) SCAS system
  d) FBW system
- To generate the flight loads on the servos, both in phase and in their static plus oscillatory levels.
- To provide a sufficient and suitable helicopter dynamic simulation, by means of:
  a) Flight attitude
  b) Vertical, lateral and longitudinal speeds
  c) Servos flight loads
  d) An external representation that can be able to simulate a landing.
- To simulate the maximum ambient temperature expected for the servos system.
- To simulate and generate all the possible failures both hydraulic and mechanic.

- Structure -

It has been realized a methallic structure which reproduces the geometry of the helicopter, except for the tail boom.

Neverthless it has been respected the control cable lengths as the real tail rotor command (figure 17).

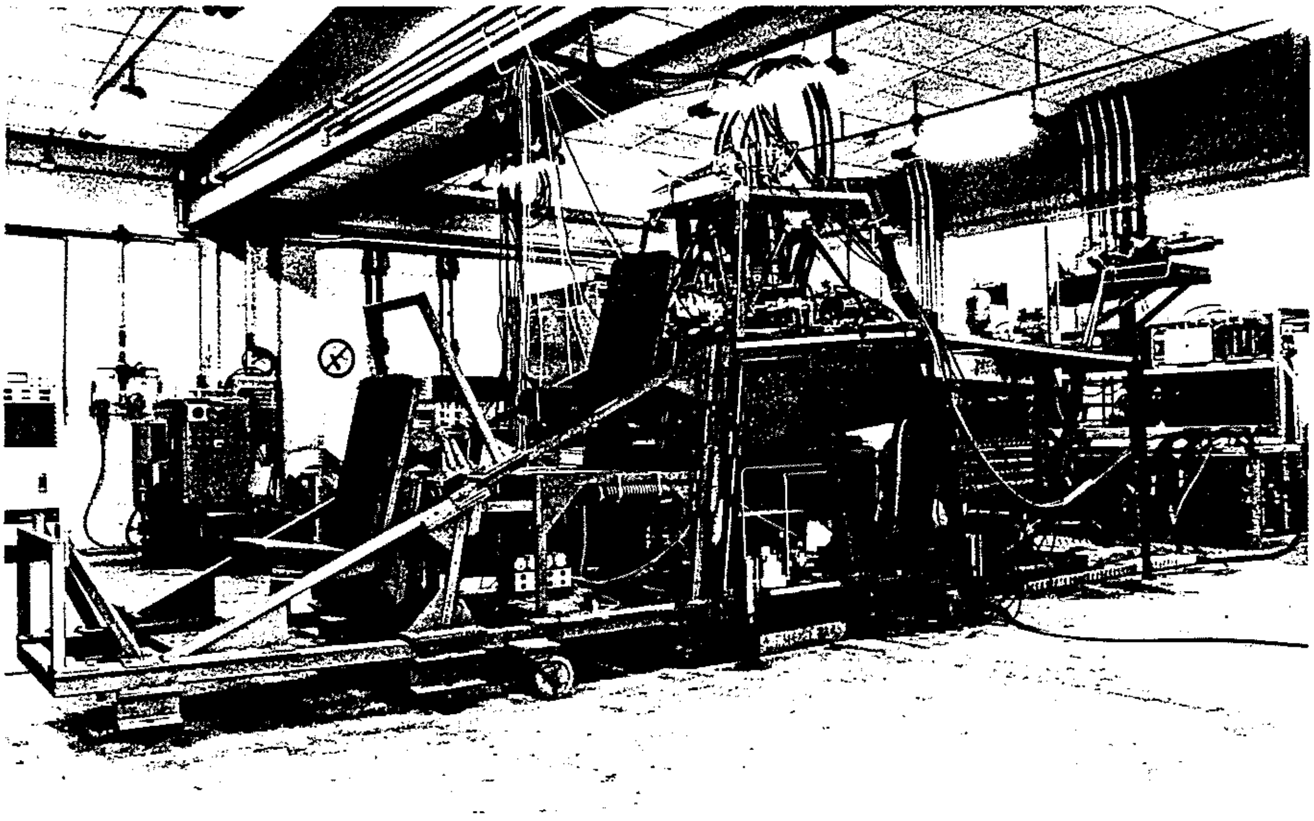


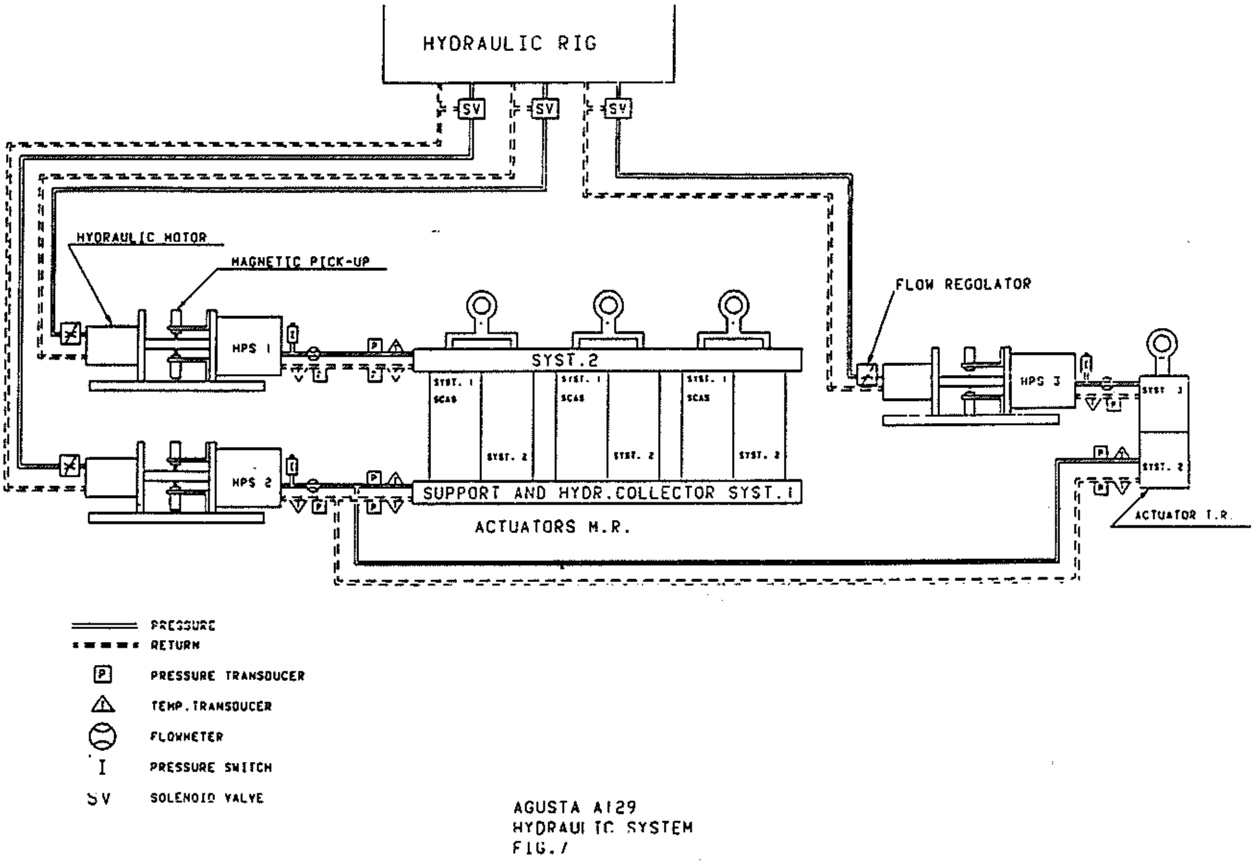
FIG. 17

- Hydraulic System -

The same hydraulic system of the helicopter was installed and the pipelines length was respected.

The system was enclosed in order to control the local temperature.

The system diagram is shown in figure 18.



AGUSTA A129
HYDRAULIC SYSTEM
FIG.7

FIG. 18

- Helicopter Dynamic Simulation -

It has been a simulation hardware that provides sufficient information about the helicopter controllability in the abovementioned conditions, although it has not the accuracy of a flight simulator.

The question was solved in the following way.

The helicopter is represented by means of a set integral-derivative equations.

The mathematical model of the helicpter is the following:

$$\dot{w} = Z_u \cdot u + Z_w \cdot w + Z_g \cdot g + Z_B \cdot B + Z_d \cdot d + g\,(\cos \phi - 1) + U \cdot q$$

$$\dot{u} = X_u \cdot u + X_w \cdot w + X_q \cdot q + X_B \cdot B + X_d \cdot d - g \cdot \sin \theta$$

$$\dot{v} = Y_v \cdot v + Y_p \cdot p + Y_r \cdot r + Y_A \cdot A - Y_b \cdot b + g \cdot \sin \phi - U \cdot r$$

$$\dot{q} = M_u \cdot u + M_w \cdot w + M_q \cdot q + M_B \cdot B + M_d \cdot d + M_p \cdot p + M_r \cdot r + M_A \cdot A + M_b \cdot b$$

$$\dot{p} = L_v \cdot v + L_p \cdot p + L_r \cdot r + L_A \cdot A + L_d \cdot d + L_q \cdot q + L_B \cdot B + L_d \cdot d$$

$$\dot{r} = N_v \cdot v + N_p \cdot p + N_r \cdot r + N_A \cdot A + N_b \cdot b + N_q \cdot q + N_B \cdot B + N_d \cdot d$$

$$U = U_o + u$$

$$\dot{\theta} = q \cdot \cos \phi - r \cdot \sin \phi$$

$$\dot{\psi} = (r \cdot \cos \phi + q \cdot \sin \phi) / \cos \theta$$

$$\dot{\phi} = p + \dot{\psi} \cdot \sin \theta$$

$$\dot{h} = U \cdot \sin \theta - (w \cdot \cos \phi + v \cdot \sin \phi) + W_w$$

$$\dot{y} = U \cdot \sin \psi \cdot \cos \theta + \Big[\cos \psi \cdot (v \cdot \cos \phi - w \cdot \sin \phi) + \sin \theta \cdot \sin \psi \\ (w \cdot \cos \phi + v \cdot \sin \phi)\Big] + V_w$$

$$\dot{X} = U \cdot \cos \psi \cdot \cos \theta + \Big[\sin \theta \cdot \cos \theta \cdot (w \cdot \cos \phi + v \cdot \sin \phi) - \sin \psi \\ (v \cos \phi - w \sin \phi)\Big] + U_w$$

where:

B = Longitudinal cyclic pitch
A = Lateral cyclic pitch
d = Collective pitch
b = Tail rotor control
U_o = Trim speed in the helicopter frame of reference (f.o.r.)
U = Helicopter instantaneous speed in its f.o.r.
$\dot{\theta}, \dot{\Psi}, \dot{\phi}$ = Angular speed components in the Eulerian absolute f.o.r.
$\dot{X}, \dot{Y}, \dot{Z}$ = Helicopter absolute speeds in the ground f.o.r.
u, v, w = Helicopter absolute speed components in its f.o.r.
p, q, r = Helicopter absolute angular speed components in its f.o.r.
U_w, V_w, W_w = Gust speed

The stability derivatives, which represent the coefficients of the equations, are defined in their range of validity by means of the problem C-81, and other similar programs realized by Agusta.

- Servos Loads -

The force operating on each actuator is defined as:

$$F_i = F_{oi} + F_{ci} \cdot \cos \underline{O}t + F_{si} \cdot \sin \underline{O}t$$

where:

$\underline{O}$ is a contant proportional to the rotor speed
F_{oi} is the static component of the load
F_{ci}, F_{si} define the amplitude and the base of the alternative component of the load

Each of this values are defined as a biquadratical function of:
d = Collective pitch
B = Longitudinal cyclic pitch
A = Lateral cyclic pitch
U_x = Longitudinal speed
U_y = Lateral speed
U_z = Vertical speed
n = Load factor

Also in this case, the coefficient of the biquadratical function has been previously computed by means of a dynamic simulation program, as C-81.

The loads so evaluated are applied to the servos by means of electro-hydraulic jacks (figure 19).

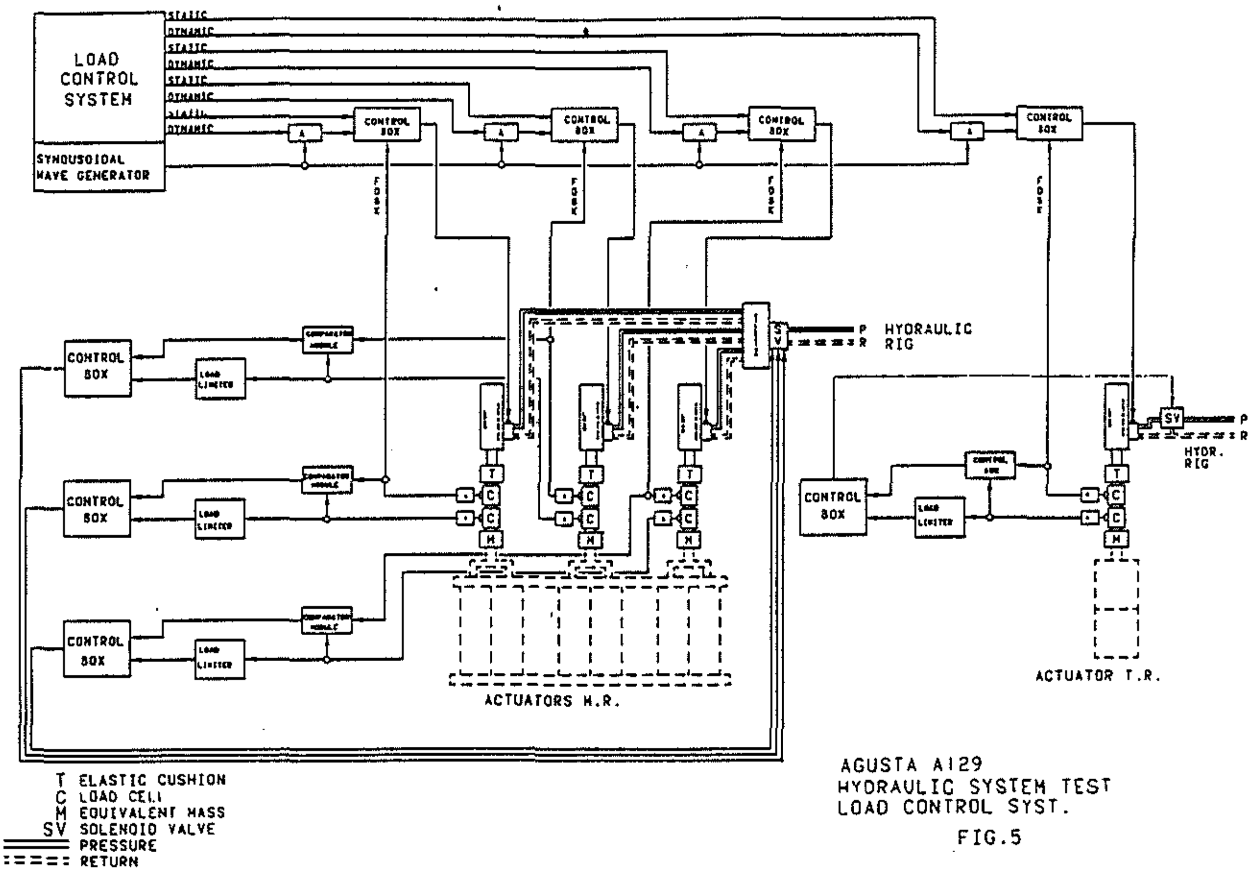


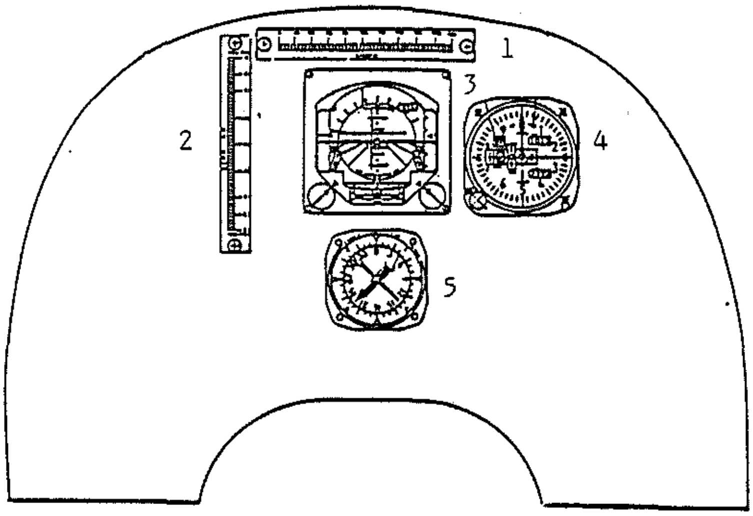
FIG. 19

- Internal Representation -

The purpose of an internal representation is to provide the pilot with the information to be used for the flight control. (figure 20)

The parameters supplied to the pilot are:

- Altitude
- Longitudinal speed
- Vertical speed
- Bearing
- Attitude (pitch, roll and yaw)
- Hovering



Cockpit indicators:
1) Air speed
2) Vertical speed
3) Attitude and hovering
4) Altitude
5) Radio bearing

Figure 20

- External Representation -

The purpose of the external representation is to provide the cues of the external environment to the pilot for a significative reference in a helicopter piloting.

The following parameters are deemed particularly important:
- Horizon
- Ground reference

The method taken in account is to generate an image allowing:

- The horizon line and a perspective representation of runway or equivalent image (figure 21).

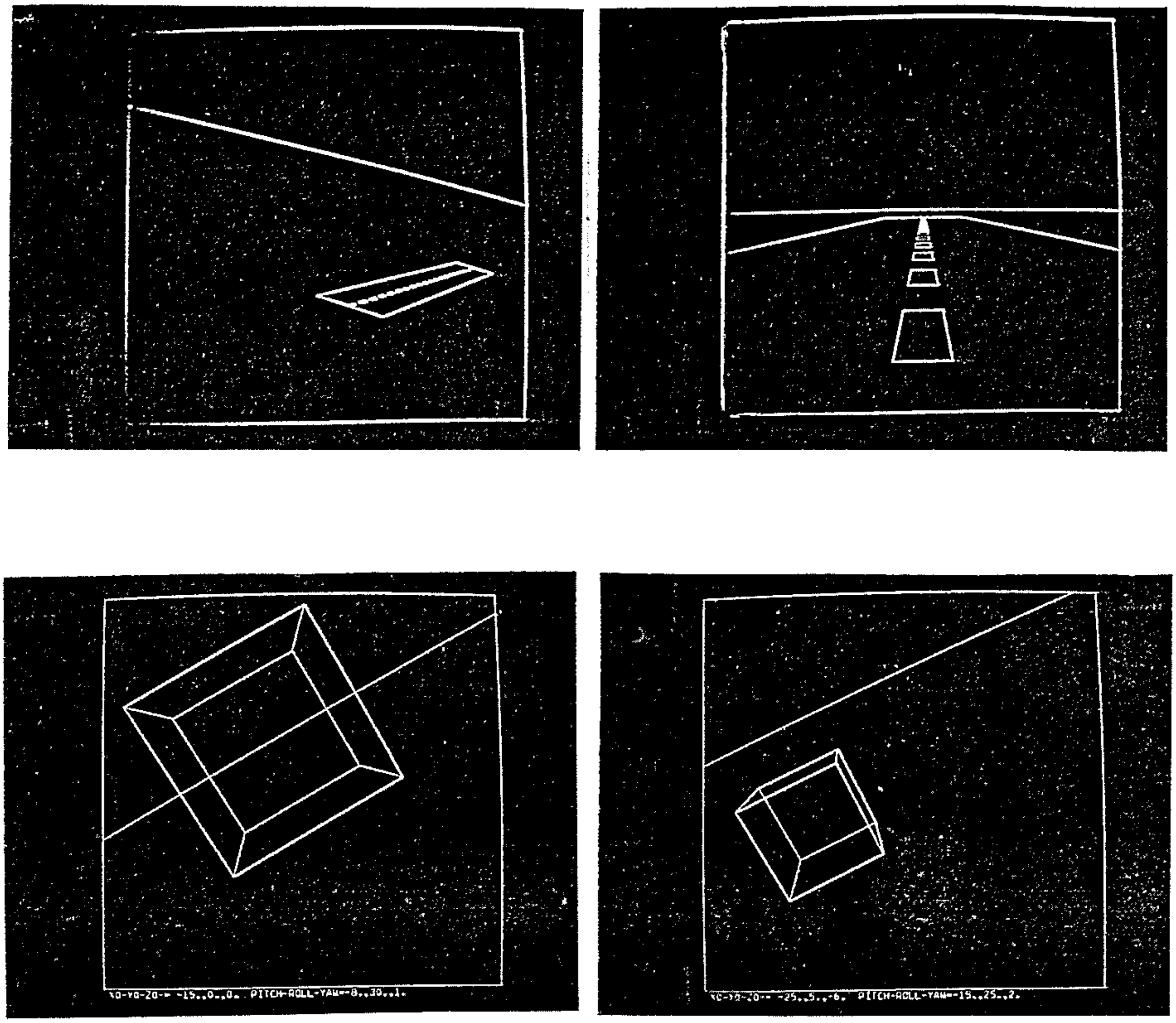


FIG. 21

- Computer -

The operativity of the flight control rig is fully controlled by a computer.

The computer functions are:

- To control the test feasibility
- To acquire the input data from the flight control operated by the pilot
- To compute the helicopter attitude, speed and position, solving a dynamic simulation program, keeping into account:
  a) Actual position of controls
  b) previous flight conditions
  c) Stability derivatives of helicopter stored in memory
- To compute the output parameters in respect of:
  a) Internal representation
  b) External representation
  c) Flight loads on controls

The computer concernes of a multimicro-processors systems, based on the Intel 8085 and 8086 micro, and array processors (figure 22).

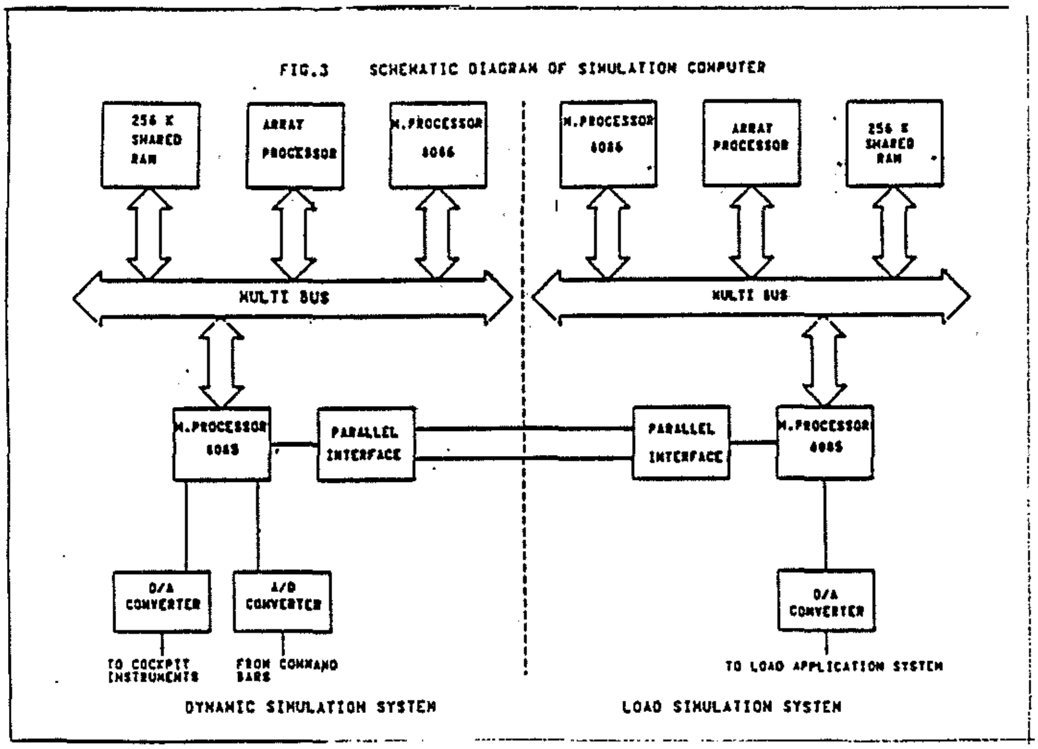


FIG. 22

It is also sized with a 512 Kbytes of RAM.
The computed speed allows to execute a complete processing loop in 50 msec.

- Experimentation -

The experimentation for the A129 helicopter allowed by the system described above, is distincted in the following phases:

1) Preliminar experimentation: this phase has the purpose to give "flight clearance" for the helicopter. It consists of
   - functional examination of the hydraulic system
   - high temperature operation of the system
   - simulation of the principal failures.
2) Dynamics simulation: in this phase we want to evaluate
   - the helicopter controllability in different conditions (manual hydraulically served, manual non hydraulically served, fly-by-wire)
   - the transients that arise passing from one state to another, with particular attention to failure occuring.
3) Duration test: applying electrohydraulic servoactuator to the collective and cyclic control bars and to the pedals (figure 23), it is possible to reproduce the same displacement recorded during typical mission, or an equivalent spectrum.
   The same is for the environmental temperature.
   Utilizing the simulation program, properly modified, the loads are applied on the controls.

These tests provide very important data for the reliability evaluation of the components and for the definition of the relative TBO.

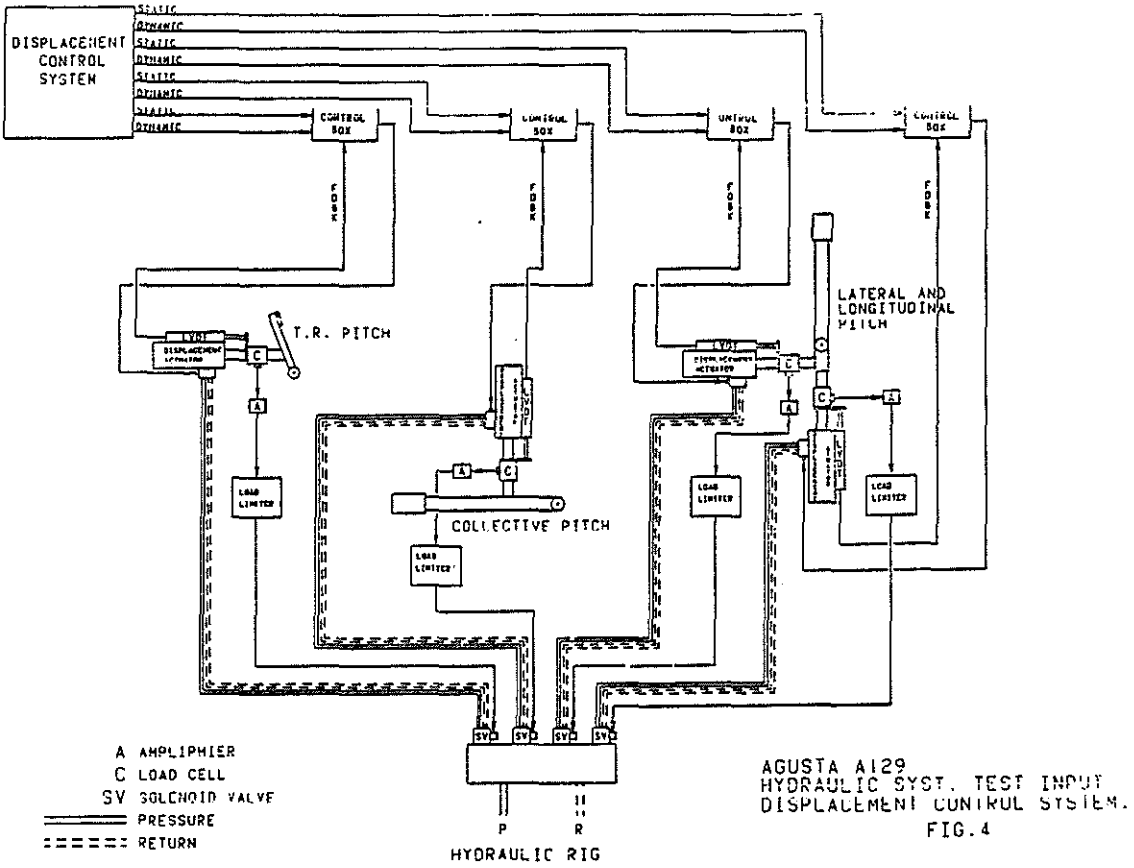
- Data Acquisition -

For each test, the acquisition and the storage of expressive data is managed by a multiprocessor system which is provided with:

- 8 high speed channels (10 Khz/ch)
- 64 low speed channels (300 hz/ch)
- 12 frequency channels (2 Khz/ch)

The storing capability is:

- 20 Mbytes on winchester for the high speed channels
- 10 Mbytes on winchester plus 500 Kbytes on floppy for the other channels.



AGUSTA A129
HYDRAULIC SYST. TEST INPUT
DISPLACEMENT CONTROL SYSTEM.
FIG.4

FIG. 23