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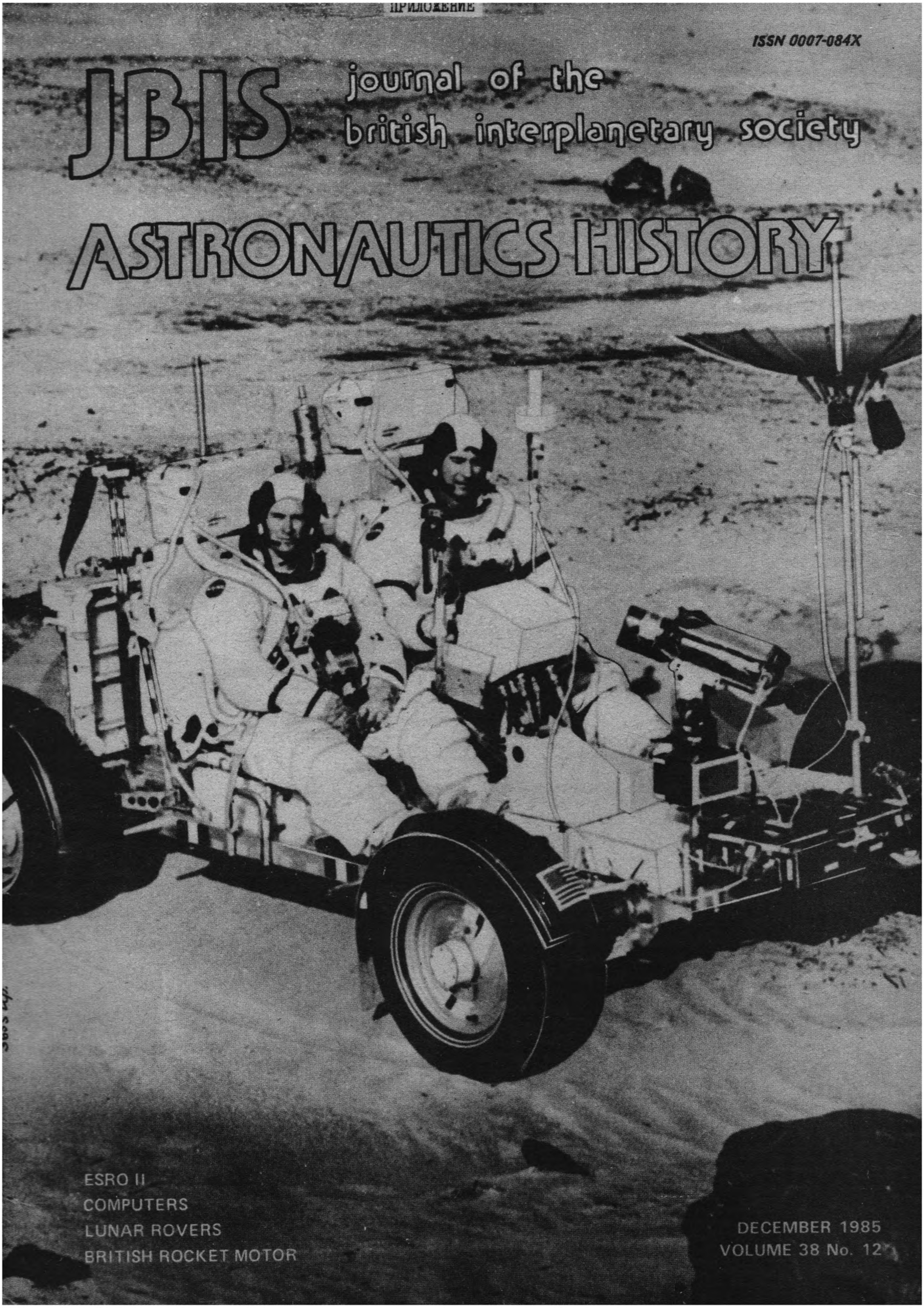
ASTRONAUTICS HISTORY



ESRO II
COMPUTERS
LUNAR ROVERS
BRITISH ROCKET MOTOR

DECEMBER 1985
VOLUME 38 No. 12

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ASTRONAUTICS HISTORY

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VOLUME 38 No. 12 **DECEMBER 1985**

CONTENTS *PAGE*

COVER

Astronauts John Young (right) and Charles Duke train on an Earth-gravity version of the Apollo Lunar Roving Vehicle in preparation for the Apollo 16 mission in 1972. Dr. M. Bekker, in his paper "The Development of a Moon Rover" in this issue, describes the development of such vehicles.
NASA

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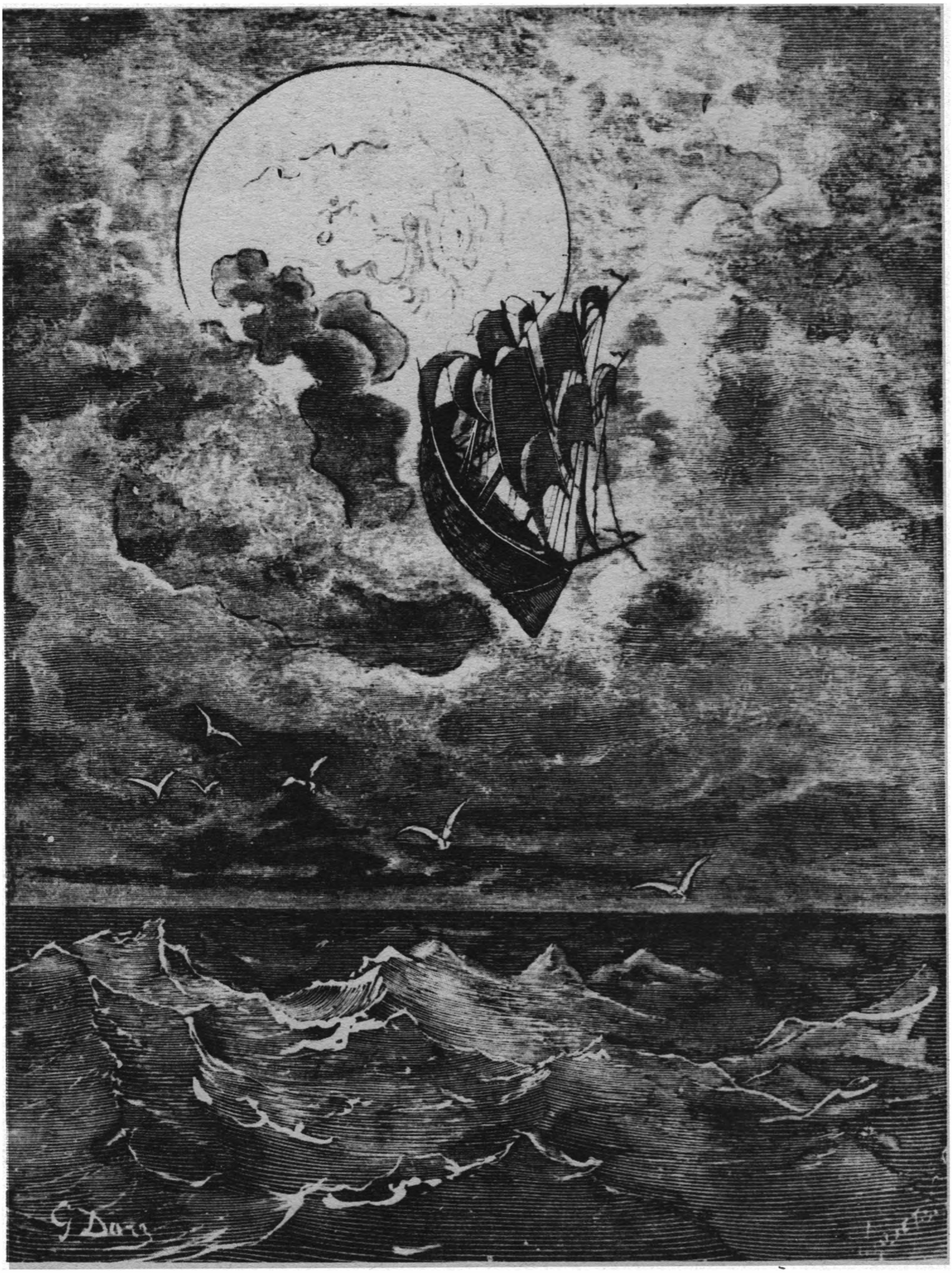


Imaginary concept of a Voyage to the Moon; by an American illustrator, G. Dore, in the late 19th century.

Journal of the British Interplanetary Society, Vol. 38, pp. 531-536, 1985.

LIZZY: THE FIRST BRITISH LIQUID PROPELLANT ROCKET MOTOR

J. GRIFFITHS
Science Museum, London.

This paper traces the development of the first British liquid fuel rocket motor, affectionately called 'Lizzy,' originally designed as an assisted take-off unit for the Wellington bomber. It was built under a Ministry of Supply contract, dated 1941, which was awarded to the Asiatic Petroleum Company (now Shell International Petroleum Limited). The development team consisted of 11 personnel, led by Isaac Lubbock, the total money available from the Ministry being only £10,000 per annum. The original specification was for a rocket motor capable of delivering a thrust of 1,000 lb for a period of 20 seconds. Since little was known at the time of the other liquid rocket work being undertaken in Germany and America, Lubbock and his team had to solve many of the fundamental problems associated with motor development. The paper, for the first time, gives a detailed account of the story of the development of this rocket motor. It discusses the problems Lubbock and his team had to overcome, and looks at the solutions they adopted.

1. PROJECT BACKGROUND

Although much has been written on the rocket development undertaken by the Germans during WWII, it is true to say that there were many significant developments occurring in the United Kingdom at the same time. The solid fuel rocket work of Sir Alwyn Crow [1] has been well documented and is certainly the forerunner of most solid fuel work undertaken today. However, during WWII a small team of scientists and engineers were designing, building and firing a substantial liquid fuel rocket motor on what was, in comparison to the funding engaged by the Germans, a very meagre amount of money. The rocket motor, affectionately called 'Lizzy,' was developed between 1941 and 1943 and there are a few references to its development in the literature [2-6].

The decision to develop the rocket motor was taken in 1941 when the Ministry of Supply decided that, owing to the potential shortage of cordite for rocket motors, there was need to investigate other means of rocket power. The Ministry set out their requirements for an assisted take-off motor for large aircraft which was to supply a thrust of 1,000 lb for 20 seconds duration. At a later date this specification was changed to 1 ton for 30 seconds. The aircraft would be fitted with two motors, one on each wing. The research contract for this work was given, by the Ministry of Supply to Asiatic Petroleum Co. (now Shell International Petroleum Co. Ltd). The contract was titled 'Homogeneous Combustion' and was numbered 294/7629. The arrangement was for the Ministry of Supply to bear expenditure up to a ceiling of £10,000 per year whilst Asiatic Petroleum would pay all salaries, overheads and travelling expenses.

The contract was let in March 1941 and Asiatic Petroleum assembled a team of 11 led by Mr. Isaac Lubbock. Assisting were Mr. G.J. Gollin, Mr. F.J. Battershill (Chief Designer), Mr. L.S. Becker and other engineers and technicians.

A survey of available literature on rocket motors showed that very little had, in fact, been published. In the United States, Robert Goddard's reticence and fear of publicity meant that little of the results of his pioneering work on liquid fuel motors (using liquid oxygen and petrol) was available for study. Also little was known of the work being undertaken in Germany. Therefore Lubbock and his team had to start very nearly from first principles and had to devise solutions to many of the pitfalls associated with the design and development of large liquid fuel rocket motors. The first and most important decision they had to make was in the choice of fuel and oxidiser. Lubbock decided to follow Goddard and use liquid oxygen and gasoline.

As a first measure the team decided to build and test a small scale (approximately quarter-size) test rig. This motor would enable the team to develop techniques to a satisfactory level before applying them to the full size motor.

Of the many problems the team had to solve, some of the most fundamental were:

1. How to design the combustion chamber so that it could withstand the very high heat tansfer during firing.
2. How to initiate the firing.
3. How to ignite the flame.
4. How to introduce the fluids into the combustion chamber.

On top of these there was the problem of how to handle liquid oxygen, something that the team had to solve very quickly.

The very first tests were carried out between February and March 1941 at the Fuel Oil Technical Laboratory in Fulham, London. Liquid oxygen, petrol and water were expelled into the open air from spray nozzles. The next tests were carried out at Cox Lane to prove the feasibility of constructing a large scale rocket motor.

2. THE COX LANE TESTS

Before any 'hot runs' could occur, Lubbock asked for a site where no harm could come to anyone if the motor exploded. The site chosen was at Cox Lane, Chessington, Surrey, England. It consisted of a pit which was originally intended to be the foundations of Shell's new laboratories until the outbreak of war halted construction. This pit was ideal for test firings of the small scale motor. However, it was not very far from factories and houses and it was a constant source of worry to Lubbock and the team that if there should be an accident someone could be hurt. However, the test proceeded and proved to be a great success.

The first test was conducted on 19 May 1941, barely six months after the original contract was let. The aim of these tests was to produce a motor with a thrust of around 60 lb, while at the same time solving the problems outlined earlier. Five 'hot runs' took place on 19 May. The apparatus con-

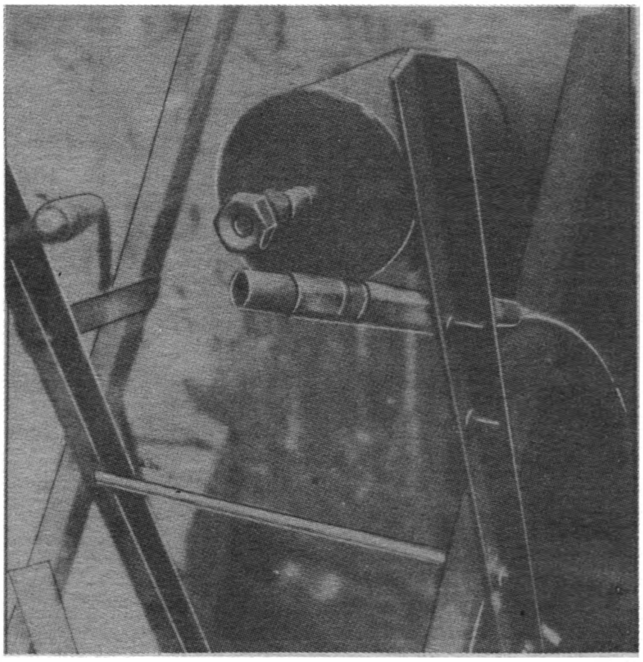


Fig. 1. Close up of the three nozzles set up for the pilot tests at Cox Lane, May 1941. It shows, from left to right, the petrol nozzels, liquid oxygen nozzle and the propane igniting torch.

G.J. Gollin

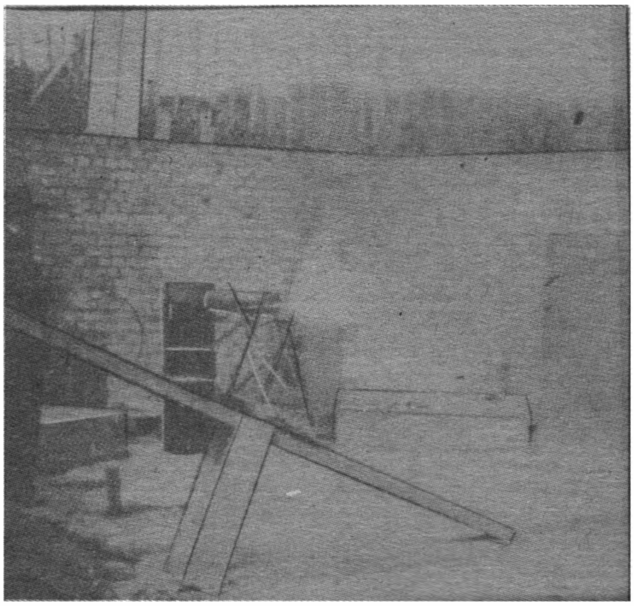


Fig. 2. Test 1, 19 May 1941 (see text for details).

G.J. Gollin

sisted of a liquid oxygen pressure vessel (capacity 2.5 gallons), a petrol pressure vessel (capacity 0.5 gallons) and a propane vessel connected to a propane burner. In order to view the tests in comparative safety, a sandbag wall was erected and a periscope constructed from plane mirrors.

2.1 Test Results

Test 1 consisted of pressurising the petrol vessel and igniting the petrol spray which was emitted from a nozzle. For Test 2 the liquid oxygen nozzle was brought alongside the petrol nozzle. The petrol spray was turned on at 100 lb/in^2 and ignited. The liquid oxygen spray (at 60 lb/in^2) was then turned on and the flame immediately became more intense and much whiter. The propane ignition torch was then extinguished but it was found that the flame, which was about 2 ft long, persisted. Test 3 consisted of the same combination of items as for Tests 1 and 2 except that the firing was arranged to take place into a steel cylinder 8 in in diameter and 18 in long. The end opposite to that holding the nozzles was left open. After ignition, combustion proceeded for about 40 seconds during which time the metal cylinder became very hot and eventually a hole was burned in its side. Test 4 was essentially the same as 3 except that the cylinder was lined with 1.9 in carbon. Combustion proceeded for about one minute during this test. Finally, Test 5 was as before but the cylinder was fitted with an extra cone attachment which reduced the outlet diameter to 3 in. At first the test was not a success due to the propane torch being accidently extinguished. A rerun achieved ignition but it was found that the combustion was not steady. There appeared to be a pulsation which reduced as the run proceeded.

The next series of tests, which took place between 20 May and 6 June 1941 (Nos 6 to 16B) concentrated on calibration of the liquid oxygen output using a variety of nozzle geometries. The tests showed that there had to be a filter placed in the liquid oxygen line in order to prevent any obstruction occurring in the feed line. Also, it was found that adequate time had to be allowed as a purge period to allow the nozzles and pipes to cool down sufficiently for the test to occur.

Another series of burning tests took place on 6 and 7 June (Tests 17A to 21B). The combustion chamber used was the same size as the earlier tests. Liquid oxygen was passed through two pintles placed either side of the petrol nozzle. Ignition was by means of a propane gas torch. Combustion proceeded for 25 seconds, giving a bright flame. However, it took 3 seconds to achieve ignition. Test 18 was similar to 17 except that an annulus-type nozzle was used. Ignition in this case was instantaneous, probably owing to the quicker mixing of the liquid oxygen and the petrol. Test 20 (7 June) resulted in a 17 second run but it was noticed that at the end there was a pulsation occurring in the flame (as was previously noticed in tests on 19 May). However, a change of fuel from the ordinary Pool spirit to Aviation Spirit (77 octane) resulted in a 50 second run (Test 26) which had no pulsation, with a very sharp cut off at the end of the run.

All these early tests had used open-ended combustion chambers. In order to pressurise the chamber it was decided (from Test 50; 1 August 1941) to use a carbon lined (8 in diameter by 12 in long) combustion chamber having a 1.26 in outlet. Use of the propane torch for ignition purposes was now not possible (due to lack of air in the chamber) and it was therefore decided to ignite the flame by means of a firework. Test 53 (19 August 1941) was undertaken with the above configuration and it produced a 30.6 second run. The flame was short, only about 8 in in length with a pale blue and pink hue. Pressures in the combustion chamber were recorded at about 3.5 to 4 lb/in^2 and the whole run produced a very high pitched scream from the motor. Gollin records in his notebook that this was the 'best run ever.' Slight alterations in the configuration followed in the next series of tests. Test 61 (22 August 1941) was undertaken with the firework in a sleeve inserted at 45° to the burner and combustion chamber axis. Previously it had been in a sleeve appendix welded to the combustion chamber parallel to its axis.

From Test 72 (28 August) onwards it was decided to upgrade the tests and increase the petrol flow by a factor of four, to 8 lb per minute. Using the same configuration as the previous tests it was found that runs were failing owing

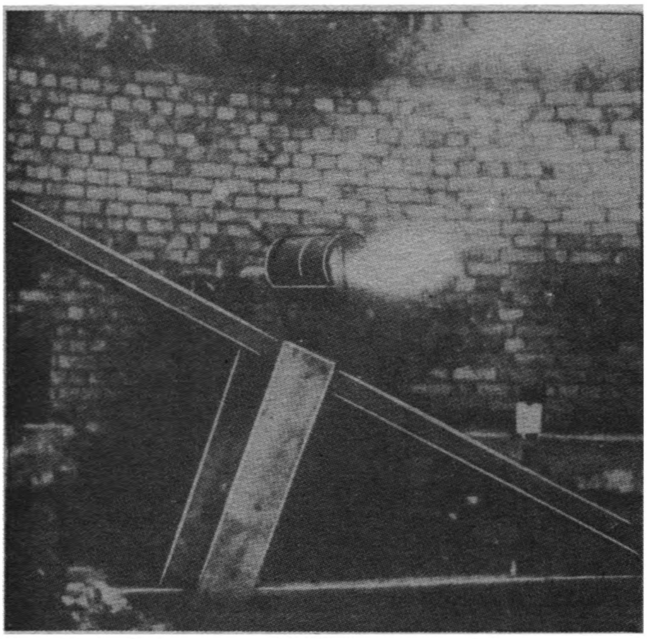


Fig. 3. Test 61 (see text for details).

G.J. Gollin

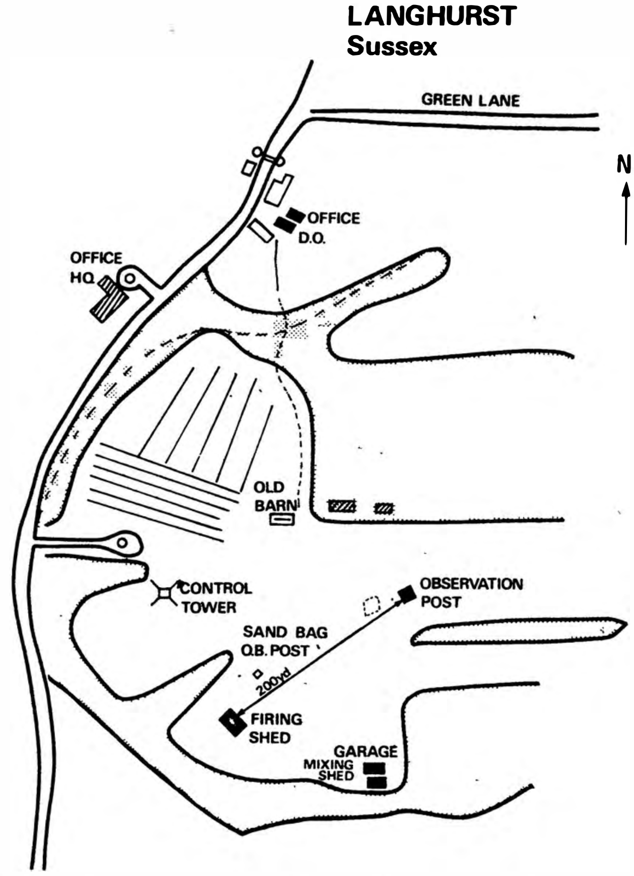


Fig. 4. The layout of the Flame Warfare Establishment at Langhurst during 'Lizzy' trials (1941-1943).

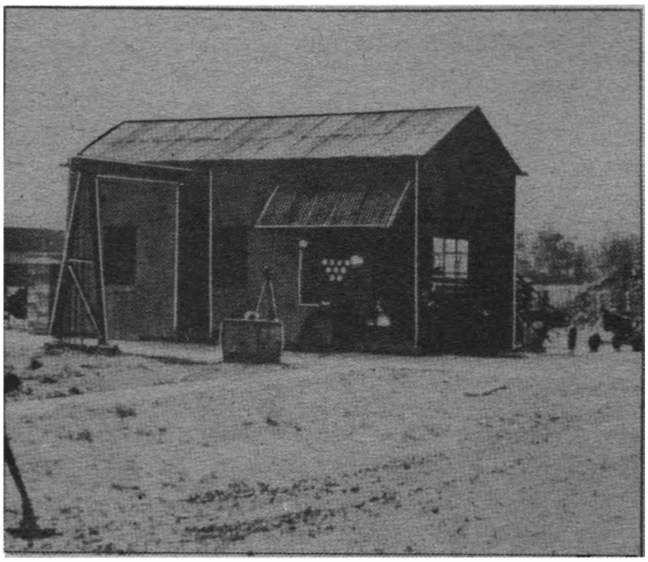


Fig. 5. The Firing Shed, Langhurst, 1942.

G.J. Gollin

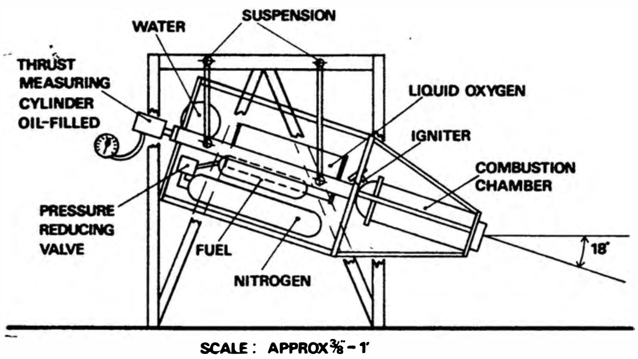


Fig. 6. 'Lizzy' mounted in its trapeze.

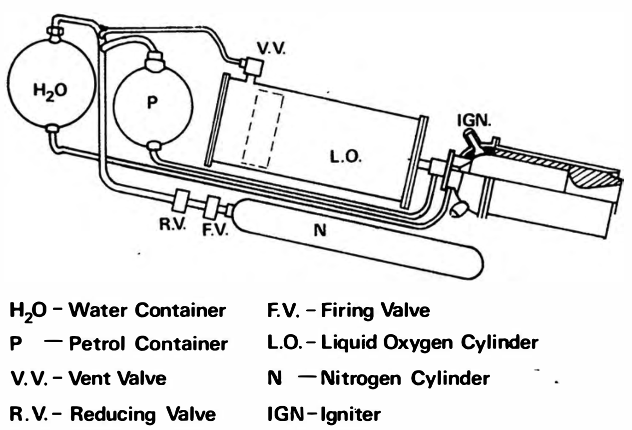


Fig. 7. Schematic layout of 'Lizzy.'

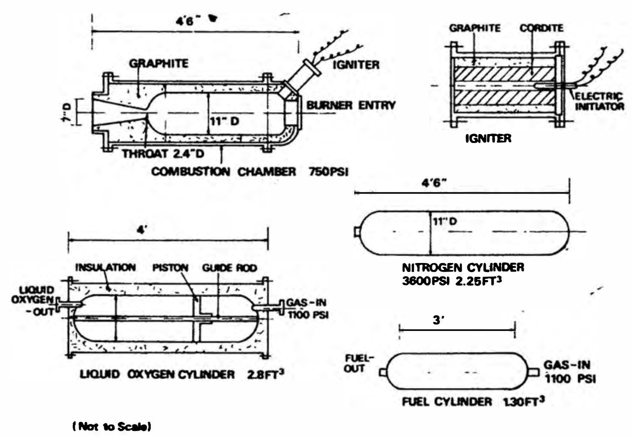


Fig. 8. Cross sections of the main components of 'Lizzy.'

to extreme overheating in the combustion chamber. To reduce the heating it was decided to fit a copper water spray coil, which would cool the nozzle and reduce the temperature in the chamber. It was also decided to introduce scaled down components (quarter scale) into the motor for future tests. Test 81 (2 October) was very successful. It lasted over 37 seconds, until the petrol was exhausted. However, inspection of the equipment after the test showed that cracking had occurred in the combustion chamber and the water spray ring had disappeared. In general later tests preceeded successfully. Alterations to the design were made and by Test 106 (the last pilot test at Cox Lane) the firing was very steady, pressures in the combustion chamber were recorded at 20 lb/in^2. On the way to achieving this consistency there were some alarming moments. Test 97 (27 November) ended up barely 12.5 seconds into the run with a large explosion. On more than one occasion, the local Home Guard at neighbouring factories were seen to dash to the roof and man the anti-aircraft guns, under the impression they were being attacked, simply as a result of the noise from the firing of the motor. Mr. Gollin himself has commented on the noise from this small rocket motor: 'This (noise) must be intolerable when the rocket is one with many tons of thrust but it was fairly alarming with the pilot test rig, which gave only a thrust of 60 lb.'

While these tests were proceeding the design and construction of the full size rocket motor was well underway. The Ministry of Supply put at the disposal of Asiatic Petroleum a part of the Flame Warfare Establishment at Langhurst, three miles north of Horsham, Sussex. It was here that 'Lizzy' was put through her paces.

3. THE LANGHURST TESTS

The layout of the site at Langhurst is shown in Fig. 4. Arrangements were made so that the rocket motor could be automatically fired from any one of three locations; inside the firing shed, from a sandbag emplacement 70 yards away or from a concrete bunker 200 yards away.

To house the rocket motor and conduct the tests a Dutch Barn was especially constructed. This also housed all the control equipment. The motor was suspended in a trapeze structure, the front end of the motor pressed against an hydraulic trust measuring cylinder. The rocket motor and thrust measuring gear were installed in the shed in the autumn of 1941 and in early 1942 tests on the full scale motor began in earnest. The rocket motor itself was assembled at the Fuel Oil Technical Laboratory Experimental Station at 22 Bagley's Lane, Fulham, London.

Figure 6 shows the layout of 'Lizzy' mounted in its trapeze. The pressure reducing valve reduces the pressure from 1,600 lb/in^2 in the nitrogen vessel to 500 lb/in^2 for the expulsion of the rocket fuels. The initial idea of simply expelling the liquid oxygen by direct application of pressurised nitrogen was unsuccessful and a new liquid oxygen cylinder was constructed which had a piston installed between the nitrogen and the liquid oxygen (Fig. 8). This redesign caused a 2½ month delay in the testing of 'Lizzy.'

It was established very early on that one of the most important conditions was that each of the working fluids should arrive at the combustion chamber in the right sequence and the early tests at Langhurst were devoted to taking pictures of the expulsion of water, liquid oxygen and petrol. These initial tests took place between 28 January and 8 August 1942. The combustion chamber was not used. They were hot runs into the open air, with the aim of ensuring the correct delivery sequence for the fluids.

As mentioned earlier, the design of the motor required some inventive steps. With the aid of the results from the pilot tests at Cox Lane, Lubbock and his team solved them and the solutions they adopted are listed below (refer to list in Section 1):

1. It was decided to use a graphite-lined combustion chamber as it was hoped that the firing time would be sufficiently short to permit the heat transfer to be absorbed into the material.
2. Pressurised nitrogen was used to push the water, petrol and liquid oxygen into the combustion chamber.
3. Ignition of the flame would be by means of twin cordite torches.
4. The flow of propellants would be initiated by a special valve which incorporates a burster disc and an explosive charge.

The combusion chamber was fitted during August 1942 and the first hot 'closed' run took place on 15 August. This was a great success for the team but, unfortunately, took place when Lubbock was abroad, in the United States. Gollin cabled Lubbock at Asiatic Petroleum (New York), to say that all went well, the cable, of course being phrased in cryptic terms lest it fall into enemy hands. Back at Langhurst the team celebrated by opening a bottle of champagne.

The run (Test 47) lasted 6.1 seconds (the length of a firing was determined by the volume of fuel used) and achieved a maximum thrust of 1,000 lb. It represents the first successful firing of a large liquid fuel rocket motor in the United Kingdom.

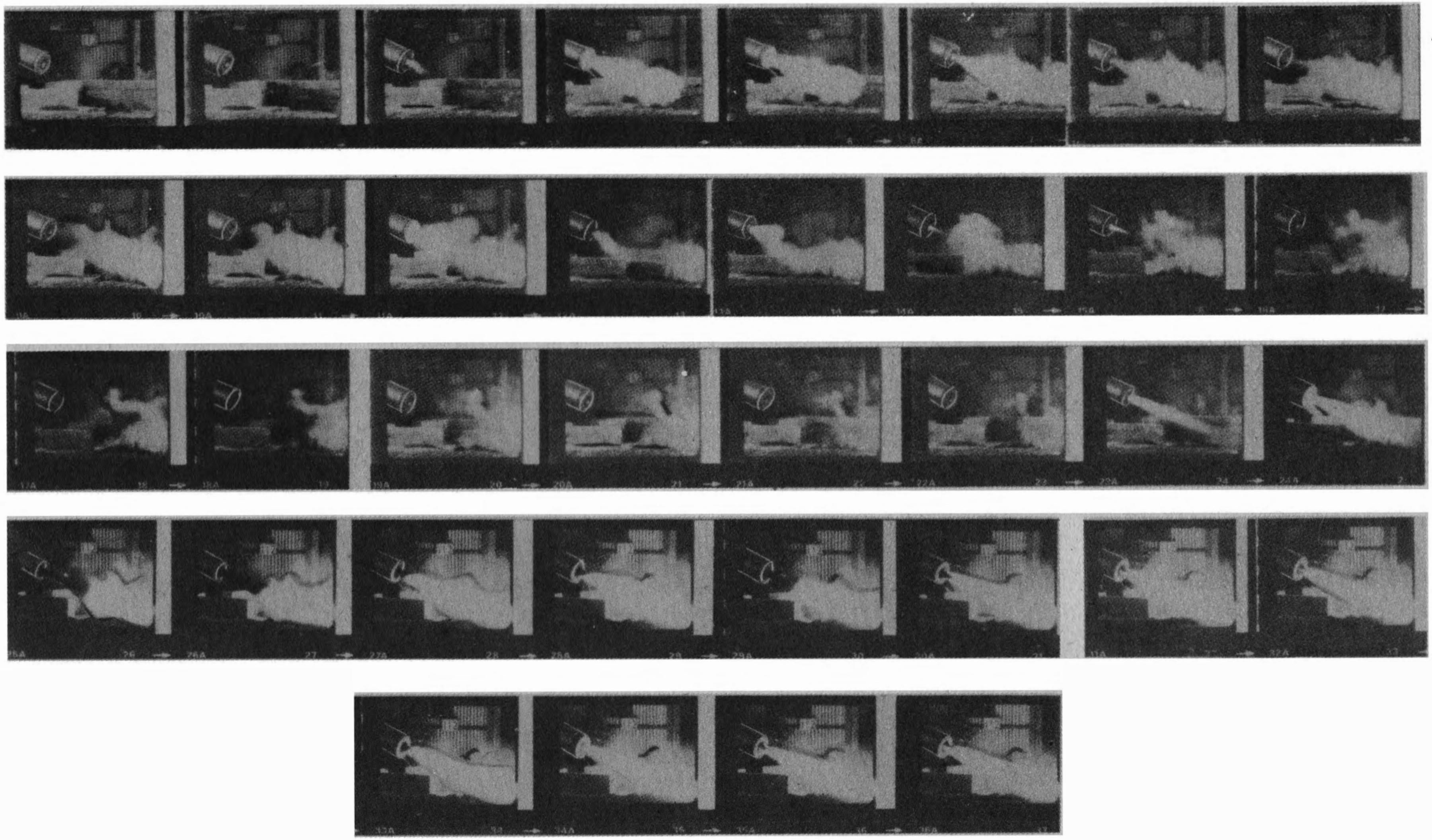
During any one firing run the pressures in seven different areas of the motor were recorded using Dobbie McInnes indicators. They recorded the pressure in the fuel cylinder, water cylinder, downstream of the reducing valve, etc. A high speed camera was also installed to record the important stages of ignition. The firing sequence was as follows:

1. The firing switch was actuated.
2. The cordite in the igniters fired.
3. The diaphragm in the firing valve ruptured and the nitrogen passed through the reducing valve to pressurise the system.
4. Nitrogen pressure on the piston of the liquid oxygen tank caused the vent valve to shut.
5. Diaphragms in the lines to the fuel and oxygen containers burst.
6. Liquids arrive at the nozzle; first liquid oxygen through an annulus, then aviation gasoline through the pressure jet in the centre and then the water through an annular quarl (delivery taking place through perforations in the quarl).
7. Ignition of the fuel and oxidiser mixture took place, pressurising the combustion chamber and causing a thrust.

The thrust of the motor was measured by an oil-filled hydraulic indicator mounted on the trapeze arrangement erected in the Dutch Barn (see Fig. 6).

Figure 9 shows Test 112 (4 June 1943) which gave a 22.3 second run with a maximum thrust of 2,000 lb. The camera was running at 55 frames per second - a clock can be seen just to the right and above the rocket nozzle. This clock makes one revolution per second. A study of the frames can reveal the stages in the firing sequence outlined above.

Soon after the first hot run (15 August 1942) the requirement for the thrust of the motor was raised by the Ministry to 1 ton. A series of runs followed with this increase in



Frame 4 - Firing valve energised, appearance of liquid oxygen vapour.
Frame 12 - Cordite igniters fire.
Frame 12 - 17 Cordite burns.

Frame 24 - Fuel arrives.
Frame 31 - Pressure builds up in combustion chamber.
The camera was operating at 55 frames per second.

Fig. 9. Early frames from the high-speed film of Test 112 of 'Lizzy' (4 June 1943).

G.J. Gollin

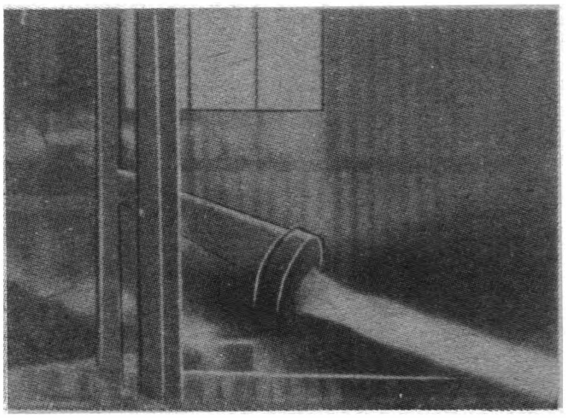


Fig. 10. Close up view of the combustion chamber of 'Lizzy' during firing.

G.J. Gollin

performance. They were also an attempt to prove the reliability, consistency and repeatability of the motor. Test 54 to 59 took place between 30 September and 16 october 1942 and were an astounding success. Thrusts of around 1,700 lb were produced for a period of 22 seconds.

By this time Lubbock and Gollin were so confident in the reliability of the motor that they invited Dr. Crow and his deputy to witness Test 58. The 23 second run went off without incident, except for a 5 minute delay in the firing due to the freezing of the liquid oxygen filling pipe. Present at the firing were Dr. Crow, Mr. Cook and Dr. Church from the Ministry of Supply, as well as representatives from the RAE, the Ordnance Board and the Air Ministry. As Gollin recorded afterwards: 'Dr. Crow and Mr. Cook expressed themselves highly pleased with the result of our demonstration and with the process we have made in the last 20 months. Dr. Crow also expressed his admiration for the ingenuity shown in the design of the unit.'

Further tests proved the reliability of the motor and the development team became used to handling the rocket motor. On one occasion they even ventured into the firing shed during a run. It was felt that, according to Gollin, if the motor did not blow in the first 5 seconds it was comparatively safe. The noise was enormous, and could be heard and felt in Horsham, some 3 or 4 miles away. Figure 10 and 11 show views of the motor inside the Dutch Barn.

A further distinguished group of visitors witnessed a firing on 7 May 1943. This group included Prof. Sir Alfred Egerton, Secretary to the Royal Society (a brother-in-law of Sir Stafford Cripps), Dr. Crow and Mr. Cook. Once again the 23 second run (Test 104) was a great success and Sir Alfred Egerton shook Lubbock's hand and exclaimed: 'It amazes me that you can bring a flame of that size under control using liquid oxygen and petrol. I congratulate you.' Dr. Crow, however, made no such gesture.

Up until 7 May 1943, 103 tests had been carried out using the full scale arrangement of which 34 had been 'hot' runs. Although the tests continued and further firings took place (Test 122 took place on 15 July 1943) the main development of 'Lizzy' slowed down and, as has been documented elsewhere, [2-8] very little use was made of the work Lubbock and his team undertook.

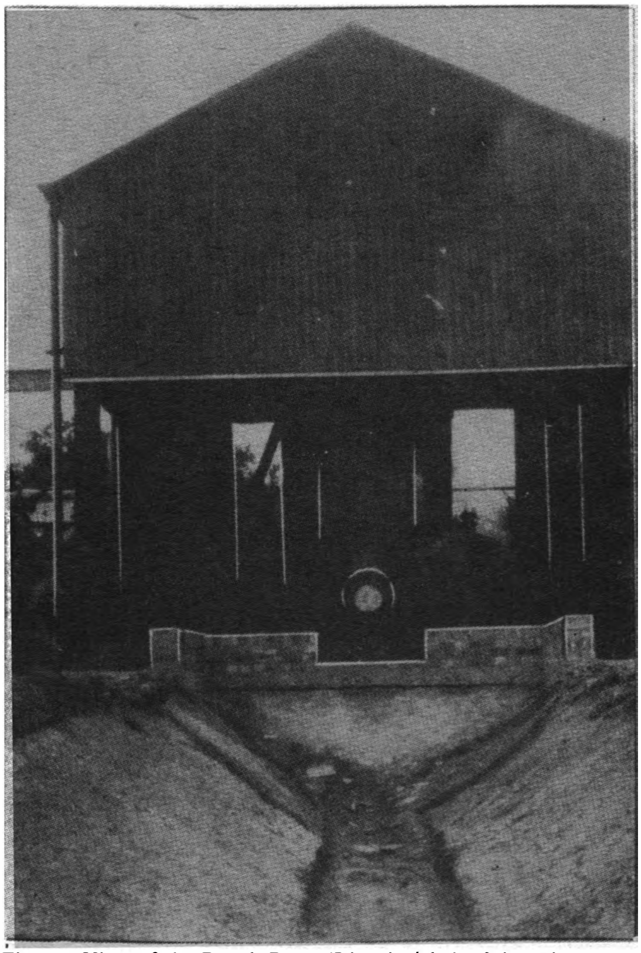


Fig. 11. View of the Dutch Barn, 'Lizzy' and the firing pit.

G.J. Gollin

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THE DEVELOPMENT OF A MOON ROVER

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On 25 May 1961 President Kennedy set the goal of landing American astronauts on the Moon within the decade. Today, this voyage is history. Ground vehicles played a notable role, as demonstrated by the Moon rovers of Apollos 15, 16 and 17 and, to some extent, by the Soviet Lunokhods of Lunas 17 and 21.

From the viewpoint of "hardware," these vehicles might appear as specialised electrical cars engineered within the unconventional state of the art. However, the "software" of American lunar vehicle development represents a unique experience in the field of off-road locomotion. It is that experience, rather than vehicle engineering that is described in this article.

1. EARLY CONCEPTS

The story goes back beyond the date when the Lunar Roving Vehicle (LRV) requirements were first estbalished by the National Aeronautics and Space Administration (NASA) and the contract awarded to Boeing-General Motors in 1969.

Neither were the LRVs left on the Moon by the Apollo astronauts the first rovers developed by the group at General Motors Delco Systems Operations in Santa Barbara, California. In the early and mid-1960's it produced (for NASA-Jet Propulsion Laboratory of the California Institute of Technology and NASA-Boeing Company) working models and prototypes of Surveyor Lunar Roving Vehicles (SLRV) and the Mobile Lunar Laboratory (MOLAB).

The first input to these programmes was related to studies by the author for the Canadian Army during World War II, supported by the National Research Council in Ottawa, in association with R. F. Legget, then of the University of Toronto. After the war these studies were continued at the Stevens Institute of Technology in Hoboken, New Jersey and at the Operations Research Office, Johns Hopkins University in Chevy Chase, Maryland. They resulted in the establishment of the Land Locomotion Laboratory (LLL) by the US Army's Tank Automotive Command in Detroit, Michigan. It was there that a general outline of the engineering mechanics of off-road locomotion was finally realised.

That work became a springboard that helped to launch not only new vehicular hardware but also a specific methodology for its optimisation within systsme analysis.

2. INITIAL RESEARCH FOR LUNAR SOIL ANALOGUES

When organising the Land Mobility Laboratory (LML) at General Motors in Santa Barbara in 1961, the author faced insufficient, conflicting information about lunar soil, from an automotive engineering viewpoint. There were no terrestrial analogues on which to build a rational mechanical model of the soil-vehicle interface. A few examples below illustrate the point.

B. W. Hapke of Cornell University postulated, on the basis of photometric data, that the lunar surface was covered with dust congealed in a porous mass of unknown thickness. However, G. P. Kuiper of the University of Arizona assumed only a few centimetres of loose dust overlaying hard rock. Tests in a high vacuum by J. W. Salisbury of the US Air Force Cambridge Research Laboratory, disclosed possible cold welding of silica particles, producing a hard structure. However, J. D. Halajian, of Grumman Aircraft Engineering Company, showed that silicates melted in a vacuum form a spongy mass. Similar results were reported by others who compared lunar soil to "cotton candy" or "cotton wool."

Subsequent detailed studies by J. K. Mitchell and his associates at the University of California, Berkeley, treated the problem of the lunar surface from a geotechnical engineering viewpoint. They demonstrated the long-standing indeciveness in respect to parametric measures in soil-vehicle systems and recommended changes in the state of the art, as well as further studies.

To resolve the dilemma, analyses of lunar albedo, light polarisation and scattering, thermal conductivity, dielectric constants etc. were carefully followed, including the time-consuming production of "Fairy Castle" soils, in accordance with the specification of Thomas Gold, of Cornell University. Tons of basalt and silica were sifted to obtain various postulated grain size distributions and other properties. The true inner soil remained, however, a mystery as far as locomotion was concerned.

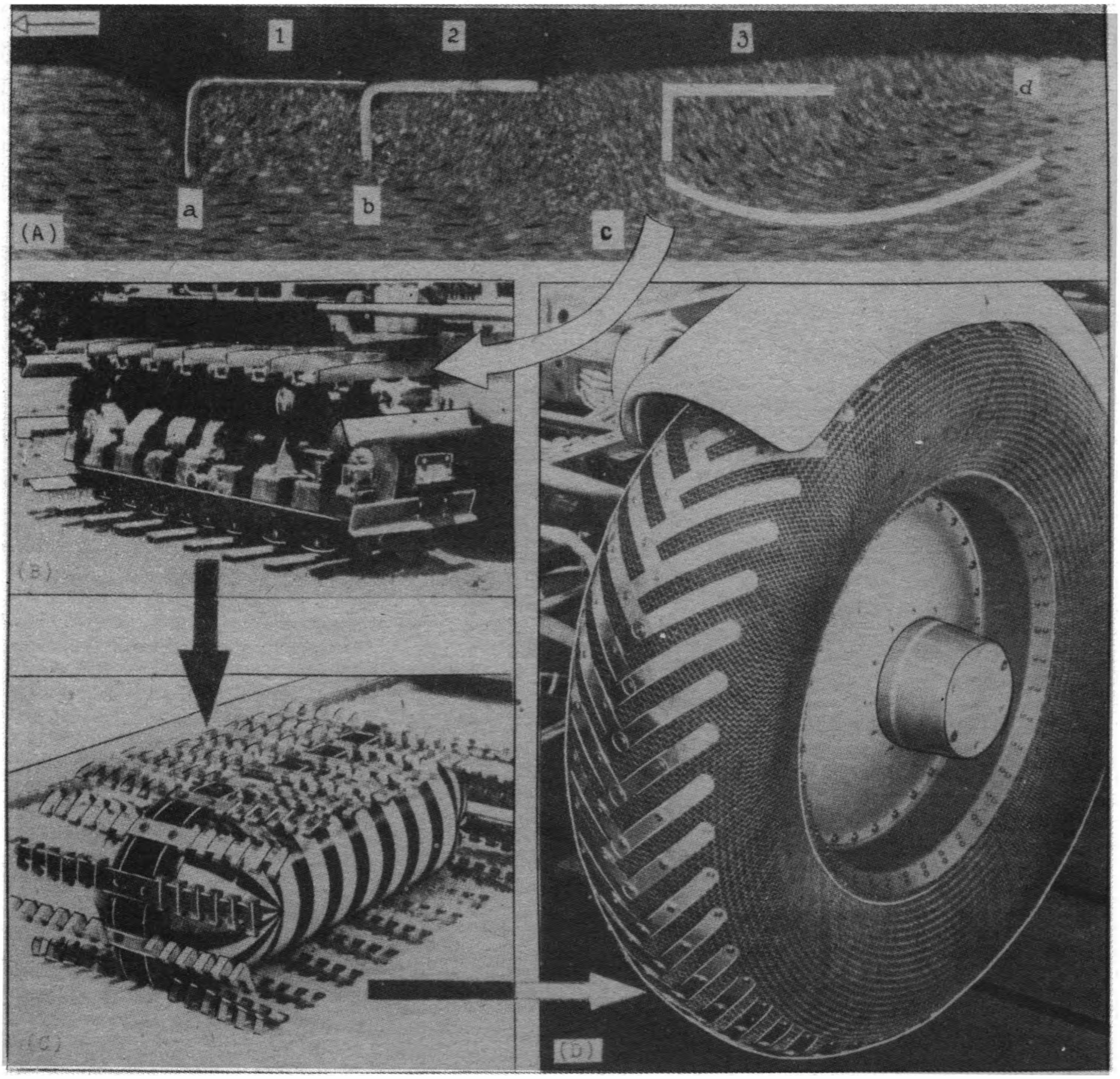
What was missing were certain lunar soil parameters that would indicate its power and effectiveness to support a vehicle in motion and the elimination of qualitative-descriptive soil definitions.

Even as late as the first soft landing on the Moon by Surveyor 1 in 1966, one authoritative opinion claimed that soil properties were similar to those of "medium stiffness snow," which advised caution when walking, while the other considered the "sandy layer" being sufficiently strong to support the future Apollo lunar module.

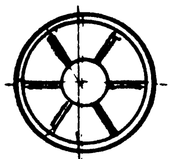
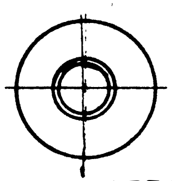
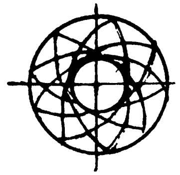
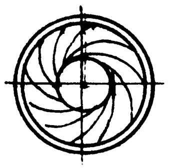
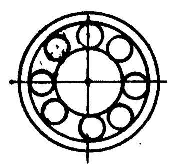
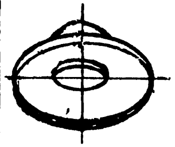
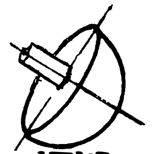
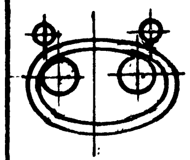
The lack of quantitative parameters compatible with applied mechanics that could be used to calculate optima of vehicles form-size-weight-power aggregates was of great concern to NASA.

Various estimates of the internal friction of lunar soil, cohesion and density could not have served that purpose, except by long-stretched analogues with terrestrial soils. Empirical studies of soil penetration by projectiles originally planned for use on Ranger flights, performed by J. L.

* *In addition to playing a pivitol role in the design and development of the Apollo lunar roving vehicle, Dr. Bekker is recognised as a leader in theory and engineering of land locomotion as well as off-road locomotion. He is a consultant in these areas to indutries and educational institutions in several countries.*



Open ground contact areas of tracks and wheels can support vehicle loads and produce traction as well as, if not better than, the conventional plates and cleats arranged in a continuous surface. The study of these phenomena was performed in the 1940's under the auspices of the Canadian National Research Council, and later at the US Army's LLL. A chronological account of the story is depicted by pictures A, B, C and D. Photograph A shows two cleats 1 and 2 equipped with spuds facing the direction of motion of the vehicle. The cleats were mounted behind a glass wall of a sand bin by means of a special support with a camera. With horizontal and vertical pressures applied, the cleats moved together with the camera and the sheared-off soil mass. Thus the latter remained in focus, sharply showing the sand grains and their surface of failure delineated by blurred soil particles which did not take part in vehicle action upon the ground. The photograph clearly implies that cheat 1, which has no open space between self and cleat 2, shears the soil along surface a-b, much smaller than surface b-c-d sheared by cleat 2 having an open space behind. Thus, the closed spacing does not utilise the bearing-thrust properties of the ground as well as an open spacing. A new location of cleat 2, shown by white lines of cleat 3, would be more effective. This reasoning applies to any number of cleats of any shape, angular, flat, round, etc. Cleat spacing is subject to calculations for maximum efficiency. Photograph B shows an experimental open track mounted on a Caterpillar tractor that pulled much more than the original conventional track. Photograph C shows a track proposed for lunar rovers. Because of low lunar gravity such a track could be made as a very light, elastic structure with open-space bearing plates and spuds. This structure shows commonality with the wire-mesh lunar wheel, photograph D, developed by General Motors. No spuds were provided since they are generally detrimental in "sandy" frictional soils, which were known at that time, to cover the lunar surface. Photographs A, B, C, and D illustrate not only the origin of lunar wheels but also the time span and the line of thought shaping their development.

CRITERIA	RELATIVE VALUE FACTORS	 RIGID WHEEL	 PNEUMATIC TIRE	 WIRE MESH TIRE	 METAL-ELASTIC TIRES		 ELLIPTI-CAL WHEEL	 HEMI-SPHERICAL TIRE	 HUBLESS WHEEL
Mechanical Reliability	15	90.0	67.5	75.0	70.5	70.5	25.5	60.0	28.5
Weight	14	92.0	46.2	121.8	35.0	63.0	14.0	81.2	7.0
Soft Ground Performance*	14	53.0	101.5	101.5	121.1	121.1	114.8	116.4	121.1
Obstacle Performance**	10	68.0	74.0	74.0	64.0	64.0	68.0	74.0	64.0
Steerability	6	43.8	34.8	34.8	12.0	12.0	24.6	39.6	12.0
Ride Comfort	13	ZERO	104.0	117.0	39.0	65.0	78.0	26.0	39.0
Stability	8	64.0	56.0	56.0	22.4	45.6	34.4	56.0	22.4
Wear Resisistance	8	24.0	12.0	42.0	48.0	48.0	48.0	42.0	48.0
Environment Compatibility	6	48.0	ZERO	36.0	42.0	42.0	36.0	36.0	18.0
Development Risk & Cost	6	64.0	8.0	48.0	48.0	48.0	24.0	32.0	16.0
Total	100	Eliminated	Eliminated	706.0	502.0	579.0	467.0	553.0	376.0

* Includes Slopes and Slip

** Includes vertical obstacles and crevasses

Many elastic and rigid-wheel candidates were evaluated using a wide range of criteria shown in the table. Some of the candidates were eliminated at the onset. All were subject to engineering design analysis which yielded such criteria as weights, gradeability, power, traction, etc. These were expressed in terms of their physical and engineering values. Other criteria such as risk, cost, reliability, etc. were defined by the figures-of-merit system. The engineering values and the figures-of-merit were normalised on a 0 to 10 basis to allow a score to be calculated. A straight-line normalising relationship was used following a general practise of operations research. "Relative Value Factor" was included for each score as shown in the second column of the table. The results tabulated in the "Total" score row represent a reasonable basis for determining the relative merit of each candidate. They were based on a large input of views and opinions when making subjective value judgements.

McCarthy *et al.* of the NASA Langley Research Center, and by J. I. Palmore, of the NASA Marshall Space Flight Center, proved totally inadequate.

The remedy was the recognition and introduction by NASA of soil parameters consolidated, in the 1950s, through the pioneering work of the US Army LLL.

Hence, extensive briefings, reports amd lectures concentrated first on inviting NASA's and aerospace industries attention to the LLL soil parameters.

These efforts became very important with the approaching Surveyor flight. B. Milvitzky, of NASA, and H. C. Thorman, of JPL, were the first to introduce the LLL parameter system for defining "locomotive" properties of the lunar terrain. Shortly thereafter, NASA adopted that system in a formal definition of the lunar surface, in the document called "Engineering Lunar Model Surface" (ELMS), and in other documents pertaining to avaluation, development and testing of lunar surface vehicles by all participating aerospace industries.

Opposition was not lacking but the support was overwhelming, particularly in the aerospace industries which, after all, needed numbers in design-performance calculations instead of qualitative, ill-defined analogues and indices. The establishment of a quantitative common language between the user and vehicle developer enabled LML to embark upon a nearly eight-year R&D programme in lunar surface locomotion.

3. DEVELOPMENT OF INSTRUMENTATION

With NASA-JPL support, the first objective became the development for Surveyor flights of a lunar soil measuring device of the LLL type which would telemeter the soil parameters to Earth. Similar devices were built and installed in the laboratory (LML) at Santa Barbara and elsewhere.

The idea of the instrument was based on a simulation of a vehicle's action upon the ground, by the action of round plates, or footings, under proper ground pressures. The ensuing load-sinkage and shear-slip relationships were recorded and then fitted with mathematical equations that yielded the LLL soil parameters.

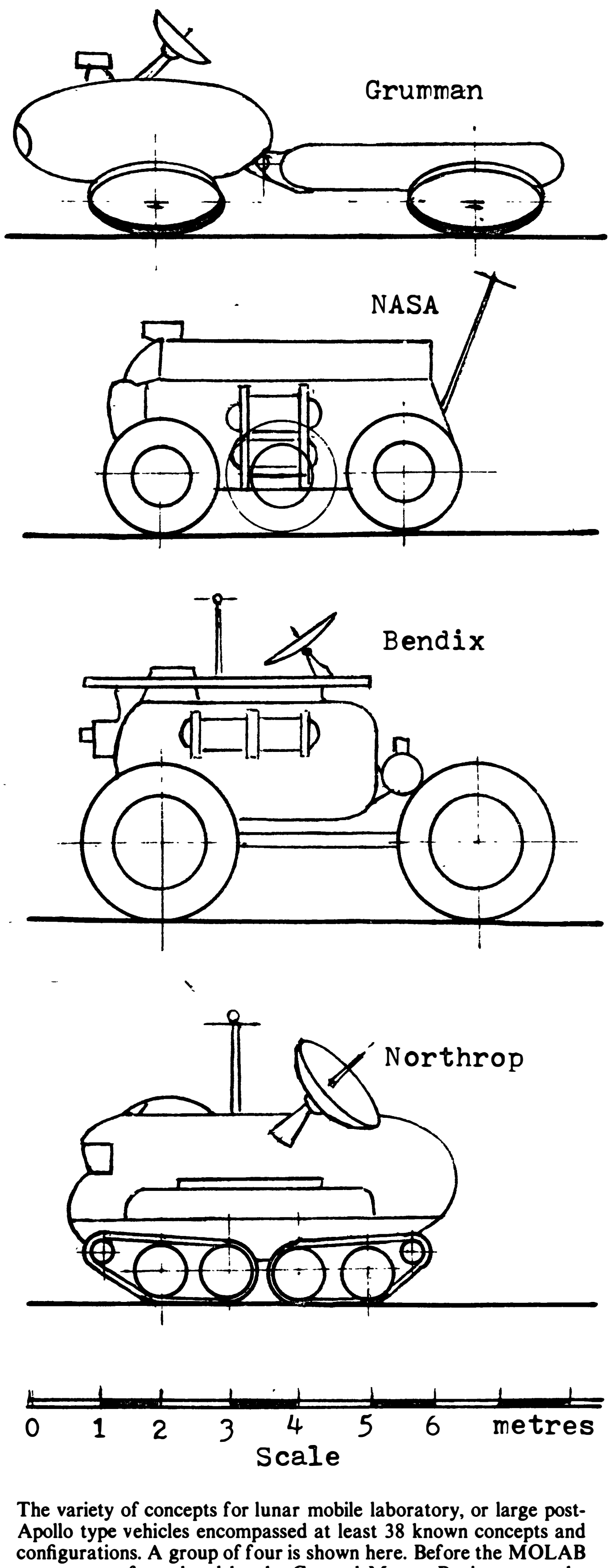
These parameters could be used for prediction of vehicle sinkage and motion resistance, on the one hand, and vehicle slip-thrust on the other. All other derivative predictions of vehicle performance and efficiency as well as of the optimum parameters of design could have been made following the LLL methods which are today the cornerstone of engineering mechanics of off-road locomotion.

Shear-slip relationships are here expressed in a modified Coulomb equation. Load-sinkage relationships are based on work by R. Bernstein who, in 1913, successfully established first the engineering approach to the design and use of gigantic, wheeled steam engines for ploughing.

The Bernstein load-sinkage equation, generalised in the 1920's by Russian researchers of the renowned Automotive Tractor Research Institute (NATI), became the basis of the author's own generalisation in 1955 and was encouraged by the work of D. E. Taylor, of the Massachusetts Institute of Technology (MIT).

He found out much later that V. V. Guskov of the Minsk Institute of Technology, also used similar equations, whereby the exponential function of LLL soil values was replaced by a hyperbolic tangent function – also exponential in nature.

Analysis and verifications of this approach to soil definition were originally performed in a close cooperation with D. C. Drucker, of Brown University, and R. M. Haythornthwaite, of the University of Michigan. E. T. Vincent, also of the University of Michigan, and the author also conducted graduate courses at the university, which included applications of the LLL soil value system.



The variety of concepts for lunar mobile laboratory, or large post-Apollo type vehicles encompassed at least 38 known concepts and configurations. A group of four is shown here. Before the MOLAB concept was formulated by the General Motors-Boeing team, the existing proposals were examined using criteria of compatibility with mission-environment and delivery constraints. To this end, calculations of performance of tracks and wheel assemblies were executed; the results were monitored by numerous tests on small-scale models in soil bins.

More rigorous solutions of soil stress-strain relationships did not yield themselves then, as they do not today, to practical parametric analyses of vehicle mobility, particularly under statistically varying conditions.

NASA use of the LLL soil value system greatly contributed to the popularisation of the approach. Quite recently, J. Y. Wong, of Carleton University, in Ottawa, and T. B. Golob, of the Canadian National Research Council, computerised the LLL parameter processing, using statistical values from a number of projects.

Two versions of the lunar soil measuring device were constructed and extensively tested by the group at Santa Barbara and JPL: one to become a part of Surveyor soil mechanics experiment and the other of the Surveyor Lunar Roving Vehicle (SLRV).

The first instrument used the weight of the Surveyor as a source of pressure applied to the soil-penetrating plates; the second imparted to the plates dynamic loads by means of a falling hammer because of the small weight of the SLRV to which it was attached. It was called the DIBSI (Dynamic Iterative Bearing Strength Instrument). In both cases, regular LLL parameters were determined.

Tests performed at LML in Santa Barbara and in an aircraft flying a trajectory that produced a few seconds of weightlessness showed no effect on LLL values at zero gravity. Similar results were demonstrated by tests in a vacuum.

Application of the LLL parameters to the conjectured lunar terrain models disclosed a wide span of possible ground strengths. The initial lack of indication as to what to expect led to the assumption of a wide spectrum of soil values, which in turn necessitated the consideration of families of different vehicles.

4. PROPOSED VEHICLES

Three classes were considered: screw-driven, bouyant vehicles; tracked vehicles; and wheeled vehicles. By these means locomotion could be produced in very weak, medium strong and very strong soil, if and when required.

In general, the problem was reduced to the mathematical determination of the safe size of the ground contact area, irrespective of what produced that area, since the type of the running gear was immaterial once the footing and the loads were properly selected. Such calculations were routinely performed.

Once the external constraints were considered, however, other probelms were unavoidable: optimisation of the form of the ground contact area (long-narrow or short-wide); the economics (large number of small wheels, or small number of large ones); tracks or wheels, etc.

Envelope-weight-performance optimisation of the vehicles and their components, power required, speed of locomotion, etc., was also a procedure based on LLL soil values. The results were monitored in the laboratory, in a soil bin, on a tread mill and other specialised equipment.

NASA's requirements were stringent. Nothing was taken for granted and everything had to be proven.

At the beginning, the question arose of a walking vehicle. J. D. McKenny of Space General Corporation, proposed among others, such a vehicle for use on the lunar surface.

Luckily, the problem had been studied before, when the author was conducting research at LLL. In a close cooperation with, and the assistance of J. E. Shigley, of the University of Michigan, he arrived at the conclusion that the idea has no practical application. The problem was not and still is not, that of leg movement sequencing and actuation, at least in theory. It is the problem of dynamic loads, non-uniformity of motion, and the strength of materials per unit weight. In a final anaylsis, one has to cope with practically unmanageable accommodation of random gait characteristics and, indeed, of the whole vehicle geometry which must follow the random geometry of terrain surface.

Once one starts designing a specific, viable, mission-environment oriented vehicle, instead of an idealised model, the difficulties become enormous; and a comparison between animal and man-made locomotion becomes a far-fetched premise, if not that based on incommeasurable entities.

H. von Sybel and H. Schwanghart, of Technical University, Munich, though indirectly, have dramatised that point by proposing a half-walking and half-rolling vehicle, with the walk being only an exceptionally casual part of locomotion. It was fundamentally based on wheels.

Additional argument against walking machines may be based on their soft-ground crossing inefficiency compared to that of wheeled vehicles for the same envelopes and loads.

When calculating a vehicle's motion resistance owing to intermittent foot-print making, a prodigious property of walking was once claimed: the resistance decreased proportionally to the increase of the length of the step. Thus, a giant biped would move effortlessly in long strides because of compacting less soil, than a bicycle compressing the ground in a continuous rut, all other conditions being equal.

This theory, however, is fallacious because step increase augments the vertical up-down movements of the centre of gravity of the walker, thus increasing the energy spent. As a result, the gain in lesser compaction resistance is counterbalanced by increased gravity work. Any "balancing" mechanisms complicates the issue.

Considering poor mechanical efficiency of cams, levers and hydraulic actuators, and the extra weight of electric motors and gears, the practical value of a walking machine becomes prohibitive.

In vertical obstacle crossing, the leg was found inferior to the wheel, both having comparable heights. The high degree of animal climbing ability rests with the flexibity of the backbone-leg system rather than legs alone. The backbone pliability, or articulation, enables the legs to follow the ground contour.

This ability may be seen in articulated vehicles as opposed to those with a rigid frame. It adds more obstacle mobility, than other structural features of the vehicle. Consequently, it seemed logical to apply the concept to lunar surface vehicles, following the old "train concept" introduced in the author's earlier studies.

The need for a flexible frame in a Moon rover was also a consequence of the cramped volume of the Apollo lunar lander, which could stow the rover only in a collapsed form.

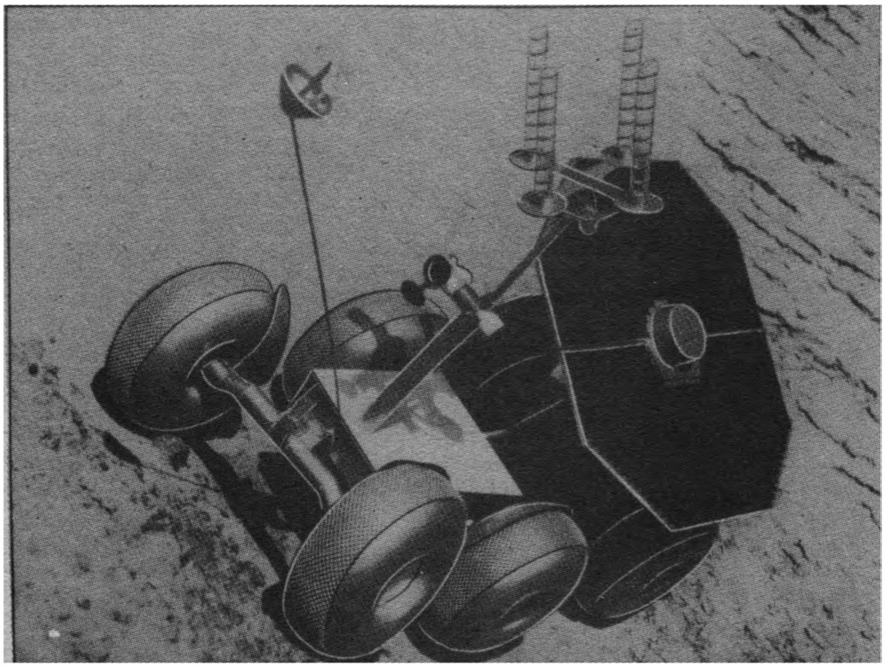
The fear that all kind of lubricated joints of the "backbone," might be contaminated by lunar dust, radiation etc., led to making the vehicle frame of flexible rods connecting particular body segments.

The power of such a vehicle to envelope and overcome terrain obstacles is enormous and unmatched as it could have been seen on a working model which was one of the main attractions of the General Motors display at the 1964 World Fair in New York.

Tests and computations with multi-segmented vehicles, on hard-rough and smooth-soft ground showed that for a compact, small vehicle, two or a maximum of three segmented units provided optimum solution.

Tracks for lunar locomotion were assumed as an alternative to wheels, should Moon soils be of "medium" strength. Considering the low lunar gravity, however, such strength would have to be extremely low by terrestrial standards; and the use of tracks was gradually dismissed.

Nevertheless, for any unforseeable contingency a special lightweight, open "space link" track was contemplated. Analysis of traction effort of such a track, when compared to that of a wheel, showed that tracks might be useful in



Mockup of the final version of Surveyor Lunar Roving Vehicle (SLRV) based on an articulated elastic half-frame and wire mesh tyres. The mission was similar to that of the Soviet Lunokhod rover that was sent to the Moon a few years after the cancellation of the SLRV.

tractor applications, i.e., for towing non-propelled vehicles or implements.

Pneumatic tyres were never seriously considered because of the danger of deflation in the airless lunar environment and radiation. In addition, low lunar gravity would have required the development of a special, highly elastic, thin-walled carcass to accommodate the optimum of inflation pressure for the best ride. Ordinary tyre structure for terrestrial use results in a too-stiff carcass which may carry the load like a rigid wheel, even without the air. Apparently, this happened to the two wheeled "rickshaw" abondoned on the Moon by the Apollo 14 astronauts.

Early tyre models were made of a spongy plastic which possibly could have been further developed for lunar use. However, the lunar vacuum, unknown radiation effects, and great temperature differentials again posed the problem of a costly developmental effort. It was for these reasons that elastic wire mesh was introduced at an early stage with scaled-down vehicle models. The initial technique for mesh fabrication was primitive and resulted in wide openings covered with fabric or plastic pads for test purposes.

The basic concept of an open "space link track," however, always seemed to be the best solution that led directly to a wire-mesh torus without pads or cover. The idea was put in practise by the author and F. Pavlics who devised the method and equipment for the fabrication of the tyre and was later responsible with S. Romano for the engineering of the LRV.

To be sure of the solution, all possible wheel forms and concepts were analysed in a search for the best combination of size, weight, elasticity, performance and reliability. The low lunar gravity was helpful in design. Otherwise, the wire mesh tyre might have been unrealistic.

Calculations and experiment also showed that the tyre elasticity was essential not only for ride comfort but also because of the low motion resistance in soft soil.

This problem was originally controversial as among the numerous competing proposals both rigid and elastic wheels were suggested. Incidentally, the Soviet Lunokhod was equipped with rigid wheels which significantly were of the "open" type: they had rigid rims covered with a wire mesh. They were thus similar to American tyres but did not have their elasticity.

5. THE SURVEYOR LUNAR ROVING VEHICLE (SLRV)

The SLRV was designed as a two-unit elastic-frame six wheel vehicle with wire mesh tyres. Three-unit vehicles were also tried. The vehicle was to be launched on a Surveyor for the purpose of exploring the lunar surface, very much in the manner as the late Soviet Lunokhods.

The Surveyor programme was plagued, however, with delays. When the SLRV launch threatened to overlap with Apollo flights, it was cancelled. The SLRV and the soil-measuring device, DIBSI, never reached the Moon. Had the original schedule been kept, it would have preceeded the Lunokhod by a few years.

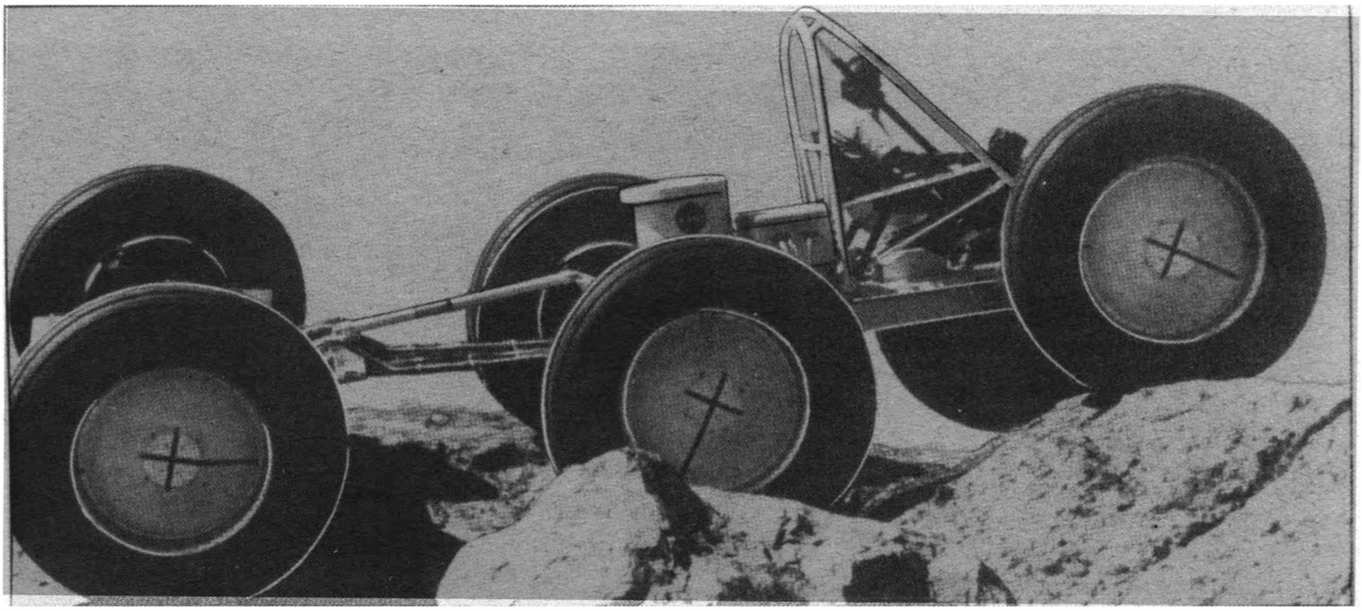
6. THE MOBILE LUNAR LABORATORY (MOLAB)

The MOLAB was developed along the same conceptual baseline as the SLRV: a two-unit, elastic frame six wheeler with wire mesh tyres. It was to provide a two-week habitat and laboratory for two astronauts for post-Apollo explorations. In a stiff competition, some 11 proposals were analysed before GMC/Delco submitted its final version to the Boeing-NASA team. Working models and a full-size chassis of the vehicle were built and tested.

The post-Apollo lunar exploration, like the SLRV never materialised. The vehicles are in storage. But the methodology and experience acquired during their development proved invaluable, when after hesitation the LRV was inaugurated by NASA in 1969.

7. THE APOLLO LRV

Much uncertainty about the roughness and bearing capacity of the lunar surface was resolved, thanks mainly to the instrumented Surveyor landings. Although soil parameters



Full-size MOLAB Chassis was tested in the simulated lunar field and met all requirements. For terrestrial tests, the wire-mesh tyres were filled with pneumatic tubes under pressure to support the vehicle's weight. Their deflection characteristics were kept within the deflection span that would occur on the Moon without the tubes.

of the LLL type were not measured directly on the Moon, their magnitude was inferred from a small number of sinkage and dynamic load data of the landing pads of the probe, and later from astronaut footprints. A meeting with astronauts Neil Armstrong and Edwin Aldrin shortly after their return from the Moon was illuminating in respect to assumptions and conjectures of soil strength, even though it was too late to make changes in the then advanced development of the Apollo LRV.

Analysis of possible vehicle candidates for LRV showed that a four-wheel vehicle with a simple rigid frame could carry two astronauts with the additional payload required by NASA within the restricted shipping conditions of the Apollo lunar module. Even that structure had to travel to the Moon with folded-up wheels; there was no room for a six-wheeler with elastic frame.

The danger of vehicle failure on an obstacle course was, however, minimised. The LRV was not remotely controlled like the SLRV, and the driver-astronaut could avoid catastrophic situations without the three second delay needed for signals to travel to and from Earth.

The continuing lack of *in situ* measurements of lunar soil parameters was disquieting prior to the first LRV landing. Specifications called for a 100 km (60 miles) range, a maximum speed of 10 km/h (6 mph), and a strict adherence to the schedule of vehicle missions, relying on inference and analogues rather than on numerical input. Luckily, what was absent in substance was present in a hard-learned, educated guess. The error in performance prediction was only a few per cent. Had the Moon been less predictable, much disappointment could have resulted.

The matter was also of concern to NASA. Prior to Apollo 14's landing, an urgent teletype from Washington requested for rush procurement a staff-like, simple, lightweight, collapsible, hand-operated instrument for the measurement of LLL-values. The instrument was designed, built and delivered in two weeks. The Apollo 14 astronauts, however, probed the lunar surface with a stick designed to carry the earphones for seismic experiments. The stick was tipped with a cone and used to penetrate the lunar surface when pushed first with one hand, and then with both. This two-step, force-sinkage scale was determined on Earth, in simulated Moon soil, under 1/6 gravity imparted by the trajectory of a KC-135 test aircraft.

8. CONCLUSION

Should any future planetary explorations, notably those of Mars and possibly Venus call for surface rover vehicles, the optimisation of pertinent equipment would not be much different from that described above. Some concepts may even be immediately applicable.

In the meantime, the described methodology has been used for optimisation of locomotion on this planet. As such, it is considered one of the many durable spin-offs of the epoch-making achievements by NASA and the aerospace industry.

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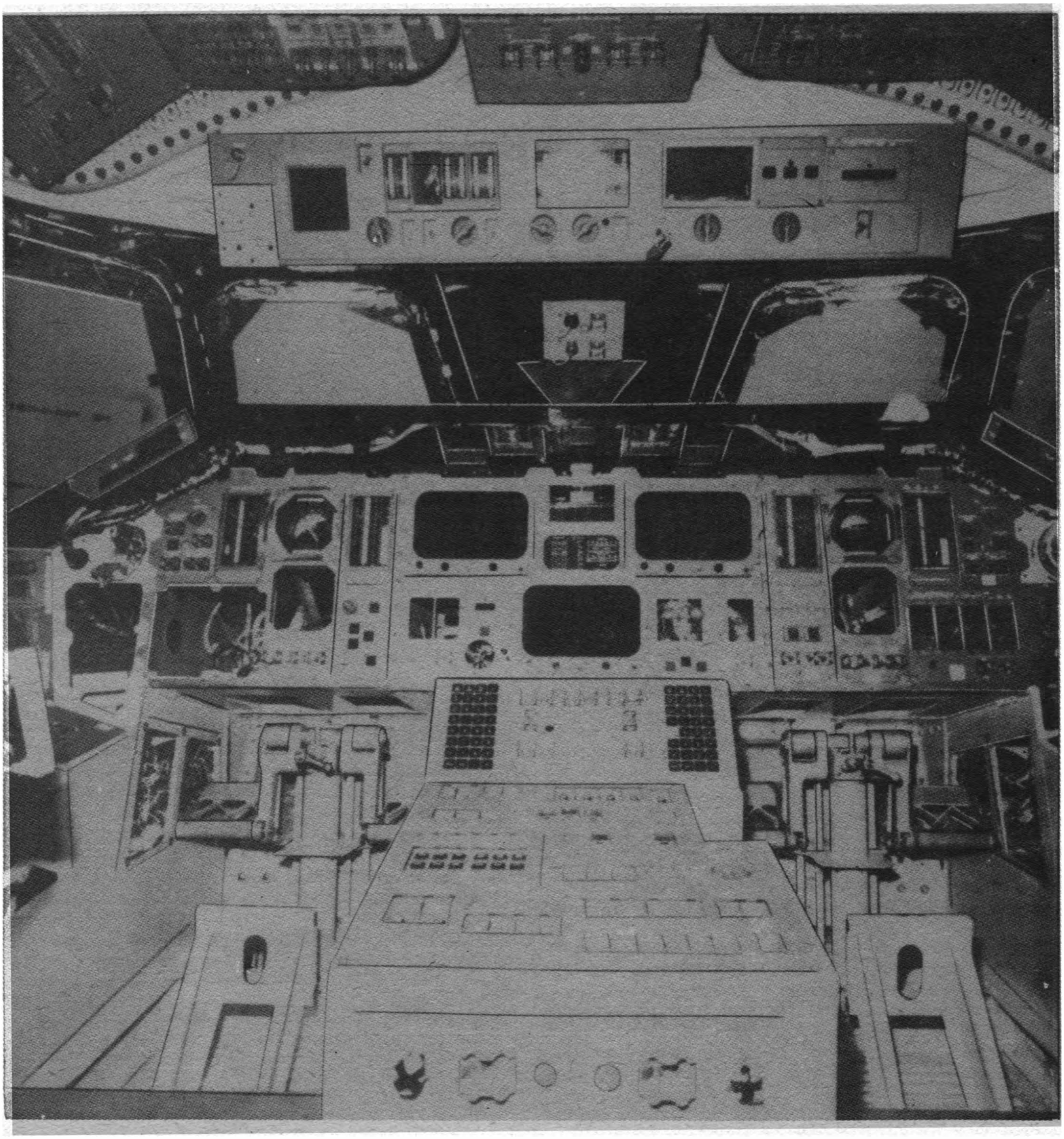

The Space Shuttle carries five General Purpose Computers, as described in the following paper by J. E. Tomayko. The photograph shows the flight deck of *Columbia* during fabrication: the three black rectangles are the holes for the cathode ray display tubes, with the keyboards below. The systems are deployed so that reentry and landing can be made from either seat.

NASA

Journal of the British Interplanetary Society, Vol. 38, pp. 545-552, 1985.

ACHIEVING RELIABILITY: THE EVOLUTION OF REDUNDANCY IN AMERICAN MANNED SPACECRAFT COMPUTERS

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Computers are a key component onboard manned spacecraft. Gemini, Apollo, Skylab and the Space Shuttle all carried computer systems of increasing functionality and complexity. All the computer hardware involved in those systems was rated at 95 per cent reliability or better; yet in no case was a computer system implemented without some alternative method of performing critical functions so that crew safety was assured. How the National Aeronautics and Space Administration (NASA) gained the last five per cent of near total reliability is the story of the evolution of the concept of "backup" to the concept of "redundancy." Success of this evolution is epitomised by the Shuttle, which did what no manned spacecraft had ever done: carry men on its *first* test flight. The main factor in enabling NASA to take such a risk was the redundancy built into the Orbiter [1].

1. INTRODUCTION

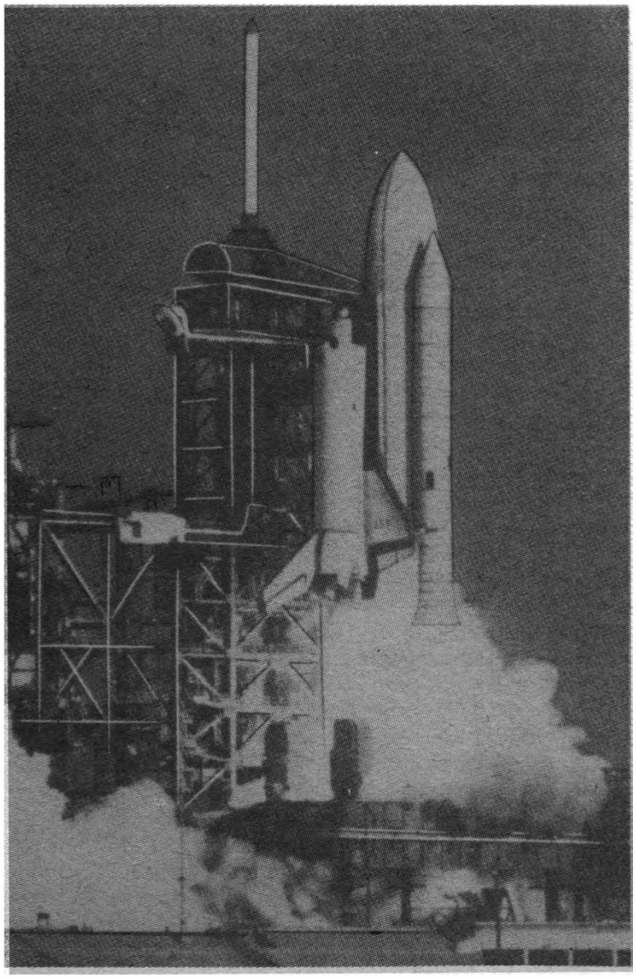
Four terms need to be defined in order to discuss how confidence in the equipment developed. They are: reliability, fault tolerance and redundancy *versus* backup.

Reliability is a measure of success *versus* failure. By studying the life of individual components and testing assembled devices to observe failures, a system's reliability can be established. In the computer and other hardware components, the measure of reliability is the mean time between failure (MTBF), or the average amount of time it takes before the computer fails to operate to specifications. If the MTBF of a computer is 2,000 operational hours and a mission is expected to last 1,000 hours, then the computer's reliability is considered to be sufficient. NASA evaluated both the Gemini and Apollo computers using this measure as the primary consideration. Strict quality control measures were responsible for ensuring a high MTBF.

Fault tolerance is the ability of a computer system to continue operating after a failure. There are two modes of fault tolerance: static and dynamic [2]. Static fault tolerance is characterised by parallel components and usually incorporates automatic switchover in case of a failure. Dynamic fault tolerance means that the system reconfigures itself to continue operation, sometimes at a lower level of performance. The degree of fault tolerance required for the Space Shuttle, for example, was set by the concept of fail operational/fail operational/fail safe. Thus, two failures could occur in a critical component before the fail safe option is reached, causing mission termination.

Redundancy and backups are the means by which fault tolerance is achieved. In redundant systems, components are duplicated (or triplicated, or further). Failed components can be removed (not physically, but electronically) from the redundant set. There is some confusion over redundancy and backup. A redundant component can accomplish all the functions of its partners at the same level of performance. A backup exists to achieve the fail safe objective and most often cannot do everything that the primary system can do. For example, the abort guidance section in the Apollo Lunar Module (LM) could guide the upper stage of the LM to a rendezvous with the Apollo Command Module if a failure occurred in the primary guidance system during the descent to the Moon or after landing. It could not accomplish the landing.

In its early manned space programme, NASA had to settle for backups. Later, to some extent on Skylab and certainly on the Shuttle, redundant systems were used. The transition from backups to redundant systems is the difference between needing extensive ground control and achieving the autonomy desired for the Space Transportation System.



Shuttle *Columbia's* manned first launch on 12 April 1981 would have been impossible without completely reliable computer systems.

NASA

TABLE 1. Performance Characteristics of Computers Used on American Manned Spacecraft.

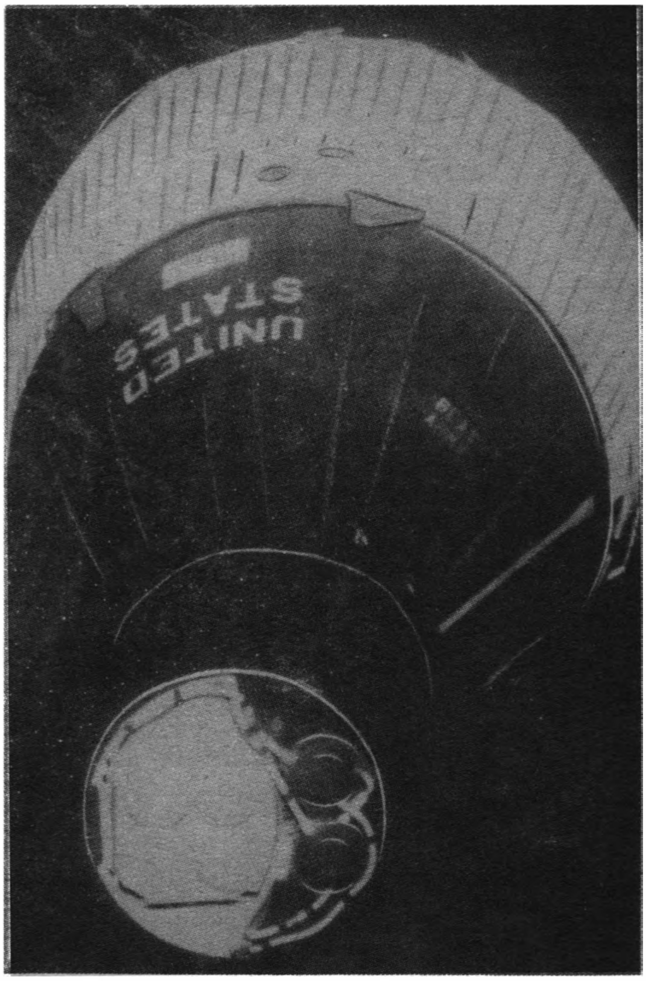
Name	Year	Maker	Word Size (bits)	Adding Speed (microseconds)	Memory Size (bits)	Total Software (bits)
Gemini	1965	IBM	13 (instruction)	140	159,744	390,000
Apollo	1968	MIT/ Raytheon	16	11.7	622,592	1,245,184
Skylab (TC-1)	1973	IBM	16	<1	262,144	262,144
F-8 (AP-101)	1974	IBM	32	0.48	1,048,576	800,000
Shuttle (AP-101)	1981	IBM	32	0.48	3,473,408	16,000,000

2. GEMINI AND APOLLO: SIMPLEX COMPUTER SYSTEMS WITH BACKUPS

Gemini was the first manned spacecraft to carry an onboard computer system. It was a small machine, custom built by IBM using discrete components and a core memory [3]. The computer was needed for onboard calculations of rendezvous manoeuvres and computer-controlled reentries, both techniques critical to the later lunar missions. It also functioned as an active backup for the guidance computer in the Titan II booster. If the Titan computer failed, the crew could engage the Gemini computer to finish guiding the spacecraft into orbit. Therefore, the Gemini computer needed to sample the accelerometers and other inertial guidance devices continuously to keep track of where the booster was flying the spacecraft so that it could instantly take over.

Gemini's computer had no active backup. Manoeuvre calculations computed in the spacecraft were also computed in ground systems in the Real Time Computing Center (RTCC) associated with Mission Control in Houston, where five large IBM 7094 computers were situated. Results of the ground solution, spacecraft computer solution and a hand-calculated solution done by the astronauts, could be compared [4]. If the onboard computer failed, the ground solution could be transmitted to the crew or the manoeuvre simply abandoned. On Gemini 4, the computer did fail just prior to reentry. Its crew flew a manual descent similar to those in the Mercury programme instead of the planned computer-controlled one. It was a prime characteristic of a system with a backup rather than redundancy: the reduction of functionality.

Despite the more ambitious and complex nature of the Apollo programme, NASA remained with the primary/backup configuration in the computer systems. A dual-computer system was considered but rejected in the very early stages of planning due to size, weight, power and cost constraints. The Command Module (CM) and the LM each carried a general purpose computer using some integrated circuits with a combination of erasable memory and permanent memory. MIT's Instrumentation Laboratory (later renamed the Draper Laboratory in honour of its director Dr. C. Stark Draper who was largely responsible for obtaining the Apollo contract) designed the computer, which was built by Raytheon. Apollo's CM computer used the RTCC computers (by then five IBM System 360/75s) as its backup. As with Gemini, both systems calculated manoeuvre burns and other functions, then the solutions were compared. If the ground system failed or the transmitter went out, then the flight had to abort using the onboard computer for return-to-Earth guidance. If the onboard computer failed, the RTCC handled the abort guidance.



Gemini 7's crew took this picture of Gemini 6 shortly after the two spacecraft accomplished the first orbital rendezvous on 15 December 1965. Rendezvous required the use of an onboard computer to calculate timing and duration of engine firings.

NASA

Due to its mission, the LM computer had an onboard backup. TRW Corporation built the Abort Guidance Section (AGS) for the LM with a small computer at its heart. The AGS would take over guidance if the primary computer failed. Its purpose was originally to put the LM into a lunar orbit high enough to clear the terrain features. Then the CM would manoeuvre to rendezvous [5]. This requirement changed to include rendezvous capability and the AGS was

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AUTHOR INDEX

FEATURES

SUBJECT INDEX

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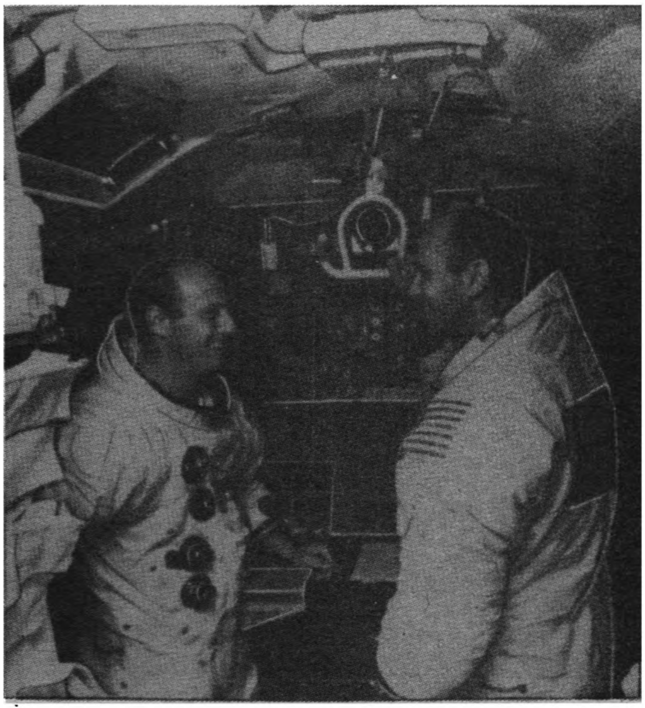
used on the Apollo 11 flight to do just that [6]. One criticism of the AGS came from astronaut John Young, who said that "one mistake in rendezvous, and the whole thing quit" [7]. His comment indicates that some operator errors caused restarts, the most common way for a simple system to get itself out of trouble. Basically, the AGS was like the parachute worn by a jet pilot – absolutely necessary, but hopefully never needed.

Both backup modes for the CM and LM computers involved reduced functions, just as in the Gemini computer. To maintain full capabilities required a movement toward redundancy. That evolution proceeded in stages through several systems.

3. EARLY REDUNDANT SYSTEMS

A number of systems that incorporated redundancy preceeded the fully redundant Shuttle. These systems demonstrated the evolution of the means of redundancy management. The first system was the computer used in the Instrument Unit carried on both the Saturn 1B and Saturn 5 launch vehicles. It contained the vehicle's guidance system. It used triple modular redundant TMR circuits. These circuits consisted of components that were in triplicate. Results of data flow through these circuits were voted by means of hardware at regular intervals. There were some disadvantages to using TMR circuits in larger or more complex computers. They are very expensive and an explosion such as that on Apollo 13 could damage enough of the computer to completely disable it. By spreading the redundancy among several simplex circuit computers and then distributing them in different parts of the spacecraft, the effects of such catastrophic failures were minimised [8].

The first manned space flight of a redundant computer system was that used in Skylab. Whereas the Gemini, Apollo and Saturn computers were used for guidance and navigation, the primary function of the Skylab computer system was attitude control. The large spacecraft had a complex arrangement of control moment gyroscopes that required a computer to assist in pointing the solar telescope. IBM built a dual computer system of its 4Pi series processors with identical hardware and software. Each computer had the capability to fully perform all the functions required. If one failed, the other would automatically take over. However, both computers were not fully functioning at the same time. The computer taking over would have to find out where the other had left off using the contents of a 64-bit transfer register located in a common section with triple-modular

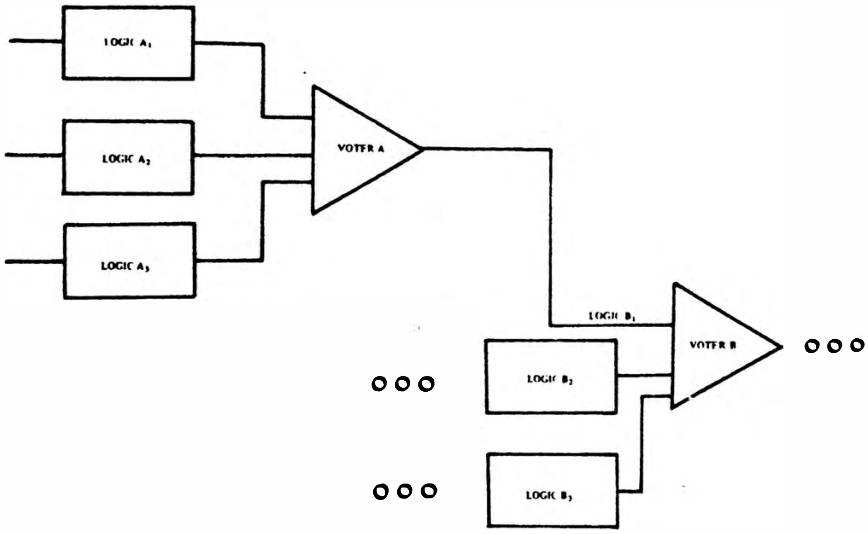


Two computer systems were installed in the Apollo Lunar Module, the first spacecraft to carry an onboard backup processor. Charles Conrad (left) and Alan Bean stand in the Module simulator, with the keyboard and display of the primary guidance computer between them, partly visible to the upper right of Conrad's left hand. About waist height in front of Bean is the keyboard for the Abort Guidance Section.

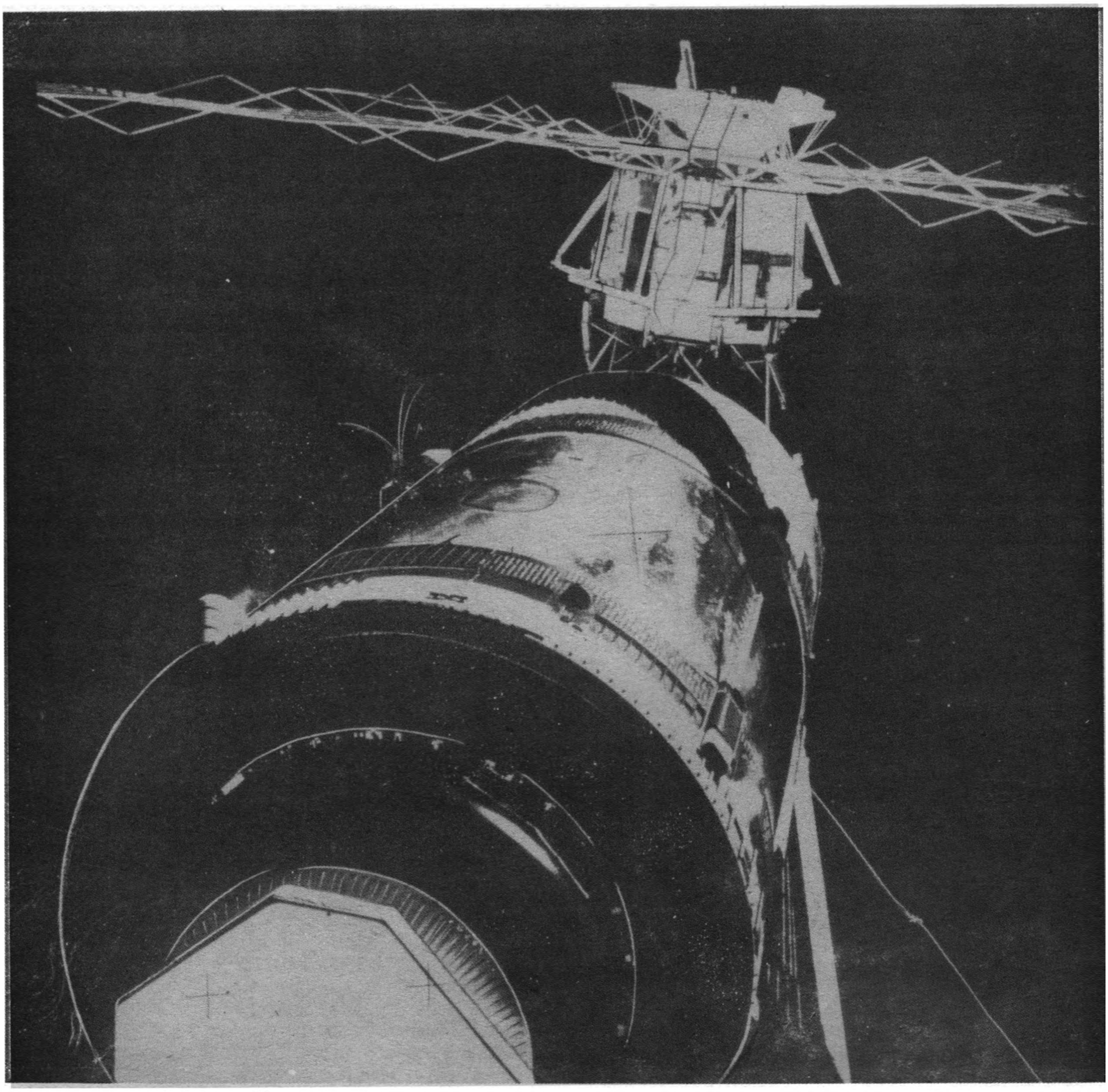
NASA

redundant circuits. It was possible for the Skylab computers to have such a relatively leisurely switchover system because the computers were not responsible for navigation or high-frequency flight control functions. If there were a failure, it was possible for the Skylab to drift in its attitude without serious danger. But the later Space Shuttle would have no such buffer.

The problem with the Space Shuttle is that the computers have many more duties than guidance and navigation. The Orbiter has a fully digital, fly-by-wire avionics system. This means that instead of mechanical linkages connecting pilot



Triple modular redundancy, used in the Saturn Launch Vehicle Digital Computer and the Skylab common section, is based on this concept. Triple logic paths exist in the circuits, with periodic voting.



Skylab's first crew took this photo before repairs were made. The laboratory's micrometeoroid shield has been torn off during ascent to orbit, along with a solar cell array, leaving just the cabling and stump visible on the left side of the cylindrical portion. The right solar panel, visible partially extended, was fully deployed when a jammed cable was cut. During the weeks that the repair plans and equipment were being made, Skylab's redundant computer system kept the spacecraft oriented so that internal temperatures would not rise above a critical point. After repair, the computers pointed the Apollo Telescope Mount, shown in the background. Four years later the computers were reactivated to help control the reentry of the laboratory.

NASA

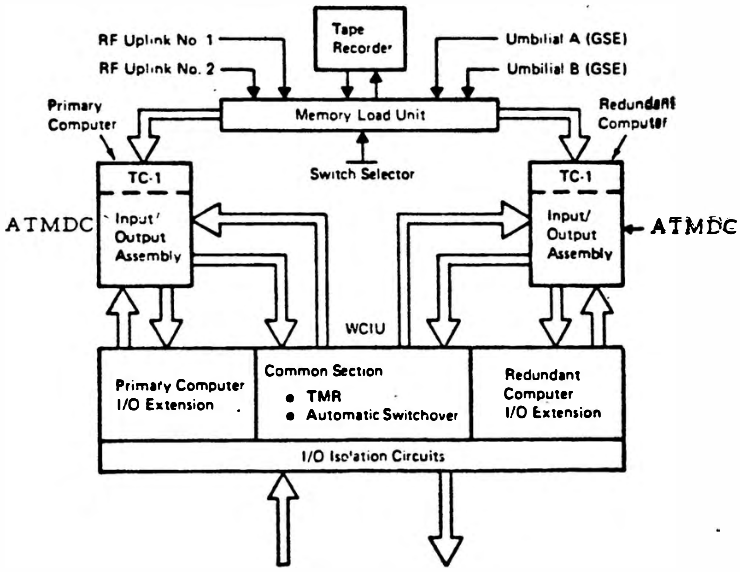
controls such as the stick-and-rudder pedals to aerodynamic control surface actuators, connections are electronic and are routed through the onboard computers. Since the computers are responsible for all flight control laws, their failure means the transformation of the craft from an aerospacecraft to a large inert body. Therefore, it is critical that the Shuttle computers have 100 per cent reliability. That level is achievable only through redundancy.

Redundant computers on the Shuttle need to process information simultaneously and thus stay closely synchronised, presenting a serious problem for the designers of the system. Such a close synchronisation between computers had not been done in previous space programmes. Fortunately, the concept could be proved in another vehicle. In the early 1970's, NASA started a digital fly-by-wire testing programme at the Dryden Flight Research Center in California. NASA procured an F-8 Crusader supersonic fighter from the US Navy and modified it by eliminating mechanical control linkages and replacing them with electrical devices and triplicate actuators at the hydraulics. The Phase I version of this system used a surplus Apollo Guidance Computer in simplex, with an electronic analog backup. For Phase II of the programme the NASA engineers (led by Cal Jarvis, Dwain Deets and Ken Szalai) wanted a duplex

system using a more advanced computer. Johnson Space Center (JSC) avionics personnel noted the similarities between the digital fly-by-wire programme and the Shuttle. Dr. Ken Cox of JSC suggested that Dryden go ahead with a triplex system to move beyond simple one-for-one redundancy [9]. By coordinating procurement, both the F-8 aircraft and the Shuttle were fitted with IBM AP-101 processors. Draper Laboratory produced software for the F-8 and the Phase II flight programme proved the feasibility of computers operating in synchronisation. The F-8 suffered several single point computer failures and successfully flew on without loss of control [10]. This flight programme did much to convince NASA of the viability of the synchronisation and redundancy management schemes developed for the Shuttle. The method of synchronisation and redundancy management are unique in aerospace systems and part of the overall philosophy of fault tolerance followed in the design of the Orbiter.

4. REDUNDANCY ON THE SPACE SHUTTLE: THE MATURITY OF A CONCEPT

The Shuttle accomplishes fault tolerance through a combination of redundancy and backup. There are five general-purpose computers in the Orbiter, as well as small and medium-sized special purpose processors for data bus control and display management. The computers are IBM AP-101 processors with core memory, of the same family as the Skylab computers. Four of the general-purpose computers operate in the "redundant set" during critical mission phases such as ascent and descent. The fifth is used as a backup in that it contains software to accomplish a "no frills" ascent and descent, and nothing else. The computer in which this software resides is identical to the ones in the redundant set. Redundancy is also present at the actuators for the control surfaces and other equipment. Four actuators drive the hydraulics at the aerodynamic surfaces. A pair of redundant computers function as engine controllers on each of the three

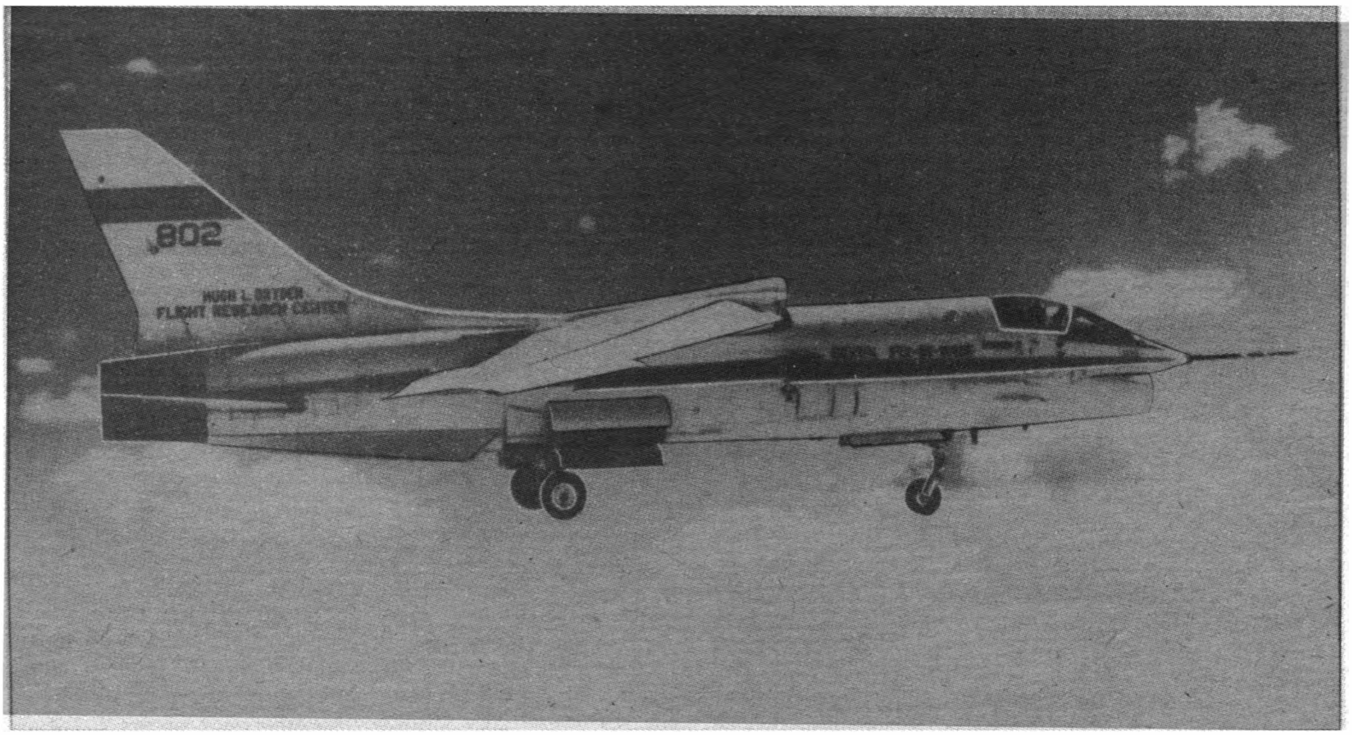


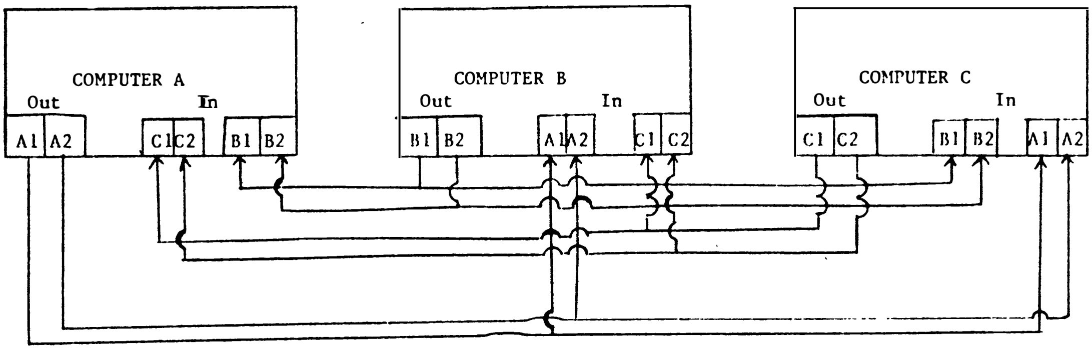
IBM's schematic of the Skylab computer system shows the two processors (ATMDC, or Apollo Telescope Mount Digital Computer), and their interconnection at the common section (bottom) and where the single magnetic tape device could load programs into either computer (top).

main engines.

Managing redundancy raised several difficult questions. How are failures detected and certified? Should the system be static or dynamic? Should the computers run separately without communication and be used to replace the primary computer serially as failures occurred? Could the computers, if running together, stay in step? Should redundancy management of the actuators be at the computer or subsystem level? But first, the number of component failures to be

NASA's F-8 digital fly-by-wire test aircraft in flight over Edwards Air Force Base. Its wing is raised during landing to provide a better angle of visibility for the pilot. Computers are installed on a pallet just aft of the cockpit.





The F-8's three computers pass synchronisation signals to each other using dedicated circuits. (Drawings from Ref. 10, p. 100.)

tolerated must be established.

One key question in redundancy planning is the number of computers required to achieve the level of safety desired. The concept of fail operational/fail operational/fail safe (FO/FO/FS) guided the Shuttle designers. Using that concept, five computers were needed. If one fails, fail operational/fail safe is still maintained. Two failures result in a fail safe situation, as three processors are the minimum required to avoid the feared standoff possible in dual computer systems (one is wrong, but which?). Owing to cost considerations (both equipment and time), NASA decided to lower the requirement to fail operational/fail safe. That meant that the number of computers could be reduced to four. Since five were already procured and designed into the system, the fifth computer evolved into a backup system, providing reduced but adequate functions for both ascent and descent in a single memory load. The other four were kept in the redundant set to provide fail operational/fail safe capability. Using four computers has a basis in reliability projections done for fly-by-wire aircraft. Triplex computer system failures were expected to cause loss of aircraft three times in a million flights. Quad computer system failures would cause loss of aircraft only four times in a *thousand million* flights [11].

The Shuttle's backup flight system computer was not supposed to be necessarily a permanent fixture. When the requirements were lowered, some IBM and NASA people expected the fifth computer to be removed after the Approach and Landing Test (ALT) phase of the Shuttle programme and certainly after the flight test phase of STS-1 to 4 [12]. However, the utility of the backup system as insurance against a generic software error in the primary system outweighed, in many minds, the savings in weight, power and complexity made by eliminating it [13]. In fact, as the first flights approached, NASA's Arnold Aldrich circulated a memo arguing for a sixth computer to be carried along as a spare [14]. He reasoned that since 90 per cent of avionics component failures were expected to be general purpose computer failures and that since a minimum of three computers and the backup should exist for a nominal reentry, then aborts would have to take place after one failure. By carrying a spare, pre-loaded unit with the entry software, the primary system could be brought back to full strength. The sixth computer was indeed carried on the first few flights. In contrast, with this "suspenders and belt" approach, NASA's Jack Garman said that "we probably did more damage to the system as a whole by putting in the backup" [15]. This is because the institution of the backup took the pressure off the developers of the primary system. No longer was their software solely responsible for survival of the crew. Also, integrating the priority-interrupt-driven operating system of the primary computers with the time-slice operating system of the backup caused compromises to be made in the primary, as the backup's operating system is by nature less flexible. Two different operating systems are the result of another NASA safety requirement: the software for the primary system and backup system is done by different companies (IBM and Rockwell respectively) to avoid a possible duplicate error in ascent or descent programs.

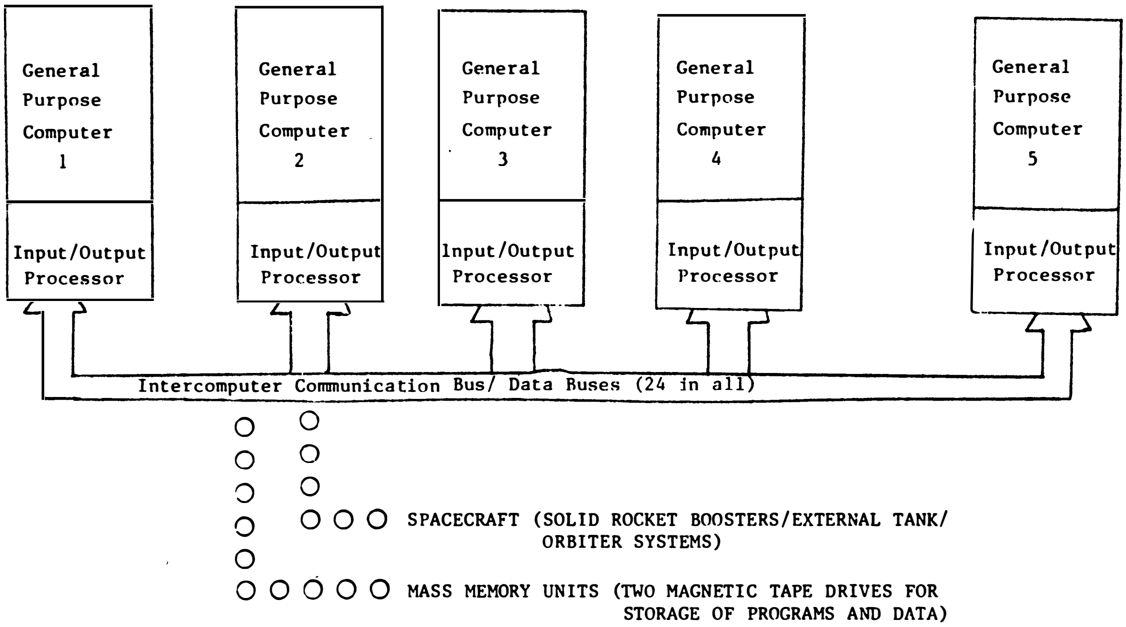
5. SYNCHRONISATION

The most difficult achievement in the production of the Shuttle avionics was the synchronisation of the computers. NASA was breaking new ground in its development of the redundant set architecture. The purpose of synchronising the computers was that it is the best way to decide if a failure has occurred. A computer producing different answers or at the wrong place in the software is instantly detectable.

There are two types of synchronisation used by the Shuttle computers: that used in synchronising the redundant set of computers and that used in synchronising the common set. It took some time to establish how to accomplish these two types.

Shuttle redundancy is, in essence, that all computers in the redundant set could each individually do all the functions necessary at that mission phase. For true redundancy, all computers must listen to all traffic on all data buses, even though they might be commanding just a few. In that way they know about all the data generated in the current phase. They must also be processing that data at the same time the other computers do. If there is a failure, then the failed computer could drop out of the set without any functional degradation. At the start, NASA thought it would be possible to run the redundant computers separately, then just compare answers periodically to make sure that the data and calculations matched [16]. It turned out that small differences in the oscillators that acted as clocks within the computers caused the computers to get out of step fairly quickly. NASA formed a committee made up of representatives from NASA, Rockwell (the prime contractor for the Orbiter), Draper Laboratory and IBM, and headed by Garman, to study the problem caused by oscillator drift [17]. Draper Laboratory made the suggestion that the computers be synchronised at input and output points [18]. This concept was later expanded to place synchronisation points at process changes, when the system makes a transition from one software module to another. The decision to put in the points "settled everyone's mind" on the issue [19].

Synchronisation of the redundant set works as follows. When the software accepts an input, delivers an output or branches to a new process, it sends a three-bit discrete signal on the intercomputer communication (ICC) buses, then waits up to four milliseconds for similar discretes from the

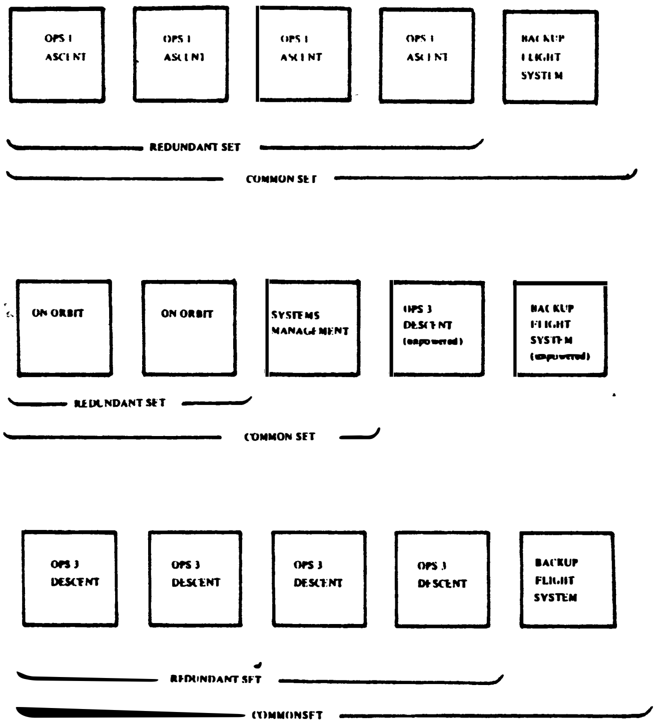


The Shuttle's five computers have an Intercomputer Communications Bus in addition to several data buses going to all parts of the aerospacecraft.

other computers to arrive. Each discrete is coded for certain messages. For example, 010 means an input or output is complete without error, but 011 means that an input or output is complete with error [20]. This allows more information other than just "here I am" to be sent. If another computer sends the wrong synchronisation code, or is late, the computer detecting either of these conditions concludes that the delinquent computer is bad, and refuses from then on to listen to it or acknowledge its presence. Under normal circumstances all three good computers should have detected the single compter's error. The bad computer is announced to the crew using warning lights, an audio signal and messages on the cockpit display screens. The crew must purposely kill the power to the failed computer, as there is no provision for automatic powerdown. This prevents a generic software failure causing all the computers to be automatically shut off by each other.

The form of synchronisation just describes creates a tightly coupled group of computers constantly certifying that they are at the same place in the software. To certify that they are achieving the same solutions, a sumword is used. While computers are in a redundant set, a sumword is exchanged 6.25 times every second on the ICC buses [21]. The sumword typically consists of 64 bits of data, usually the least significant bits of the last outputs to the solid rocket boosters, orbital manoeuvring engines, main engines, body flap, speed brake, rudder, elevons, throttle, the system discretes and the reaction control system [22]. If there are three straight miscomparisons, the GPC's vote the computer involved to be failed [23].

Both the three bit synchronisation code and the sumword comparison are characteristic of the redundant set operations. During non-critical mission phases such as on-orbit, the computers are reconfigured. Two might be left in the redundant set to handle guidance and navigation functions such as maintaining the state vector, which is the Orbiter's true position. A third would run the systems management software that controls life support, power and the payload. The fourth would be loaded with the descent software and powered down, or "freeze dried," to be instantly ready to descend in an emergency and to protect against a failure of the two tape units that store software not currently in use.



During a typical Shuttle mission, these computer configurations are used. The type of software loaded into individual computers is shown with the redundant set/common set alignment.

The fifth contains the backup flight system. This configuration of computers is not tightly coupled, as in the redundant set. All active computers, however, do continue the 6.25 times/second exchange of sumwords. This is referred to as the common set synchronisation [24].

These forms of intercomputer communication are what makes the Shuttle avionics system uniquely advanced over other forms of parallel computing. The synchronisation is accomplished with minor overhead. The software required for redundancy management uses just 3,000 words of a

106,000 word memory and about 5-6 per cent of computing resources, which is a good trade for the results obtained [25].

One reason why the redundancy management software was kept to a minimum is that NASA decided to move voting to the actuators, rather than do it before commands are sent on the buses. Each actuator is quadruple redundant. If a single computer fails, it still continues to send commands to an actuator until the crew takes it out of the redundant set. Since the other three computers are sending apparently correct commands to their actuators, the failed computer's commands are physically out-voted [26]. Theoretically, the only serious possibility is that three computers would fail simultaneously, thus negating the effects of the voting. If that occurs, and the proper warnings are given, the crew can engage the backup system by simply pressing a button located on each of the forward rotational hand controllers used for attitude control by the commander and pilot.

Does the redundant set synchronisation work? The F-8 version, which had identical redundancy management concepts as the Shuttle, survived several inflight computer failures without mishap. On the first Shuttle ALT flight, just as *Enterprise* was released from the 747 carrier, a computer failed. The landing took place successfully. That incident helped to convince the astronaut pilots of the viability of the concept. Gordon Fullerton said, "I personally gained a lot of confidence in that whole idea that I didn't have, never having experienced it before in an aircraft..." [27]. Successful computer operations after a failure also occurred during the STS-9 flight when two computers went down at different times, and in the automatic shutdown of the engines during an attempt to launch *Discovery* when redundant engine controllers could not agree on whether a valve was open. The reliability desired continues to be proven.

Synchronisation and redundancy is the method of ensuring the reliability of the Shuttle computer hardware. It represents the current state of the art in the evolution of reliability from less robust backup systems through to full redundancy. It is why the Shuttle is the most autonomous manned space vehicle ever flown, as it carries onboard the backup capability previously provided from the ground. It can serve as a model for future distributed computer systems in space as well as fault tolerant systems on the ground. This evolution is an important part of the history of astronautics for that reason. Future long duration manned missions to the planets and stars will owe their ability for survival and autonomy to NASA's efforts in the 1960's and 1970's.

ACKNOWLEDGEMENT

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27. Quoted in Poupard, 'Redundant Operation,' p. 24.

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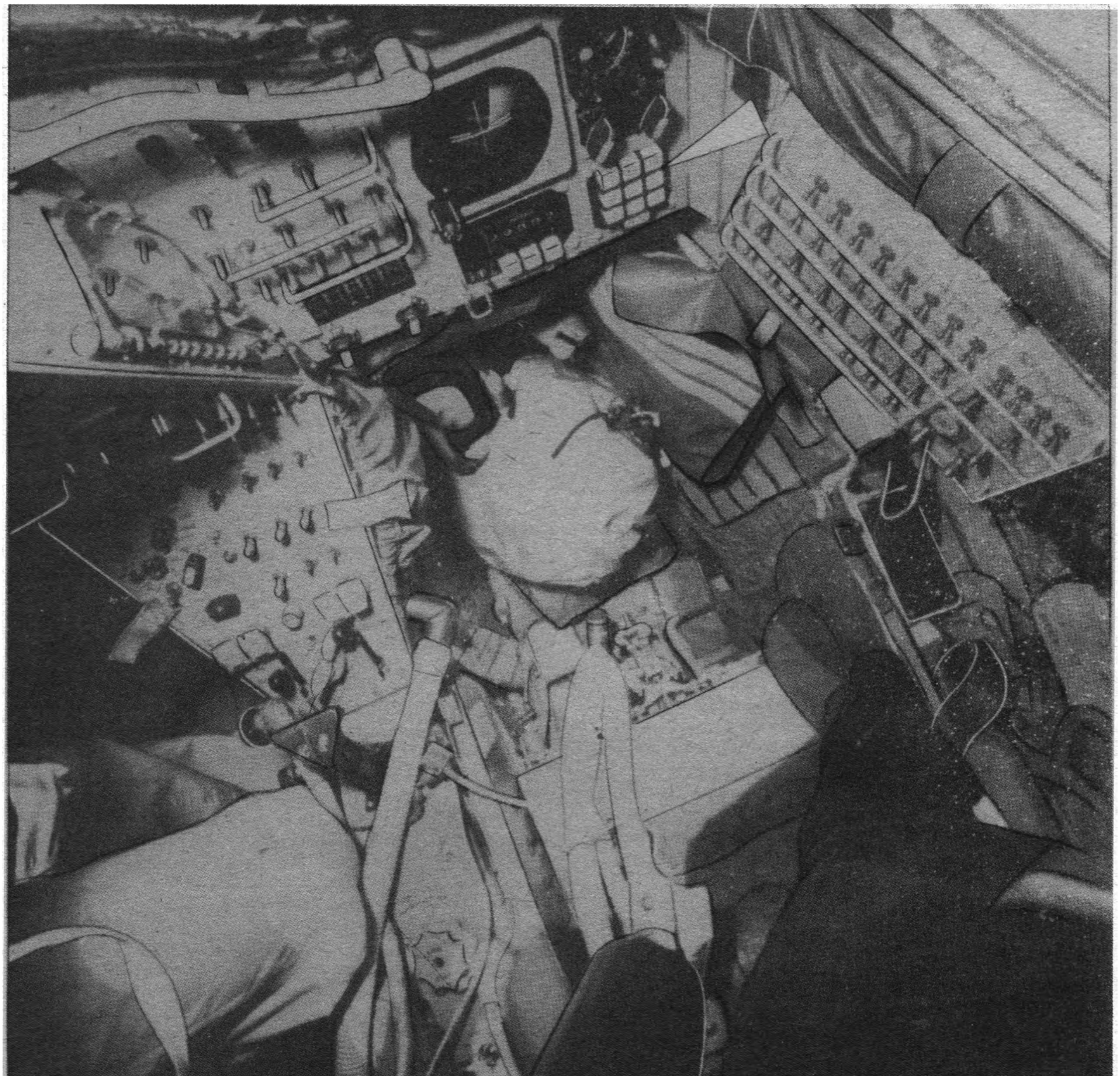

Gemini was the first manned spacecraft to carry a computer system, as described in the previous paper by J. E. Tomayko. The keyboard, in front of the pilot's seat, is arrowed at right.

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ESRO II: 20 YEARS AGO

C.R.J. BRYANT
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1. INTRODUCTION

In the autumn of 1964 Hawker Siddeley Dynamics, Stevenage was awarded the contract for the ESRO II satellite. This was the first satellite contract awarded by ESRO and the first received by HSD. Following nationalisation and denationalisation HSD has, together, with the space interests of the British Aircraft Corporation, become the Space and Communications Division of British Aerospace.

The ESRO II project not only initiated the development of the technical skills relating to space activities within the Company but also led to the setting up of the first Satellite Project Management organisation, under Bob Hume. The co-ordination of the multinational team of Contractors and Experimenters participating in the ESRO 11 project presented a new challenge in management and communication.

ESRO II was a small, scientific satellite designed to study solar X-rays and energetic particles from the Sun and Galaxy. Following injection into orbit, the satellite was given the name IRIS, which stood for International Radiation Investigation Satellite. However, this name was used only for official publications and the satellite continued to be known as ESRO II.

The launch of the first satellite in May 1967 was unsuccessful and it was lost in the Pacific Ocean following a failure of the third stage of the four-stage Scout rocket. The second flight model was launched successfully on 16 May 1968 from the USAF Western Test Range at Vandenberg, California. The satellite operated successfully until re-entry on 8 May 1971. The seven experiments yielded a vast amount of data and made a significant contribution to the study of solar and cosmic radiation.

2. GENERAL DESCRIPTION

The overall configuration of the satellite was determined by a number of considerations, the most important of which were:

1. viewing requirement of experiment sensors;
2. launch vehicle heat shield limitations;
3. gyroscopic stabilisation of the spin axis attitude, requiring the ratio of moments of inertia to be greater than 1.0;
4. body mounted solar arrays;
5. passive thermal control;
6. access requirements to internally mounted equipment;
7. alllowable satellite weight for the chosen orbit limited by launcher capability.

In order to meet these requirements, a configuration was chosen that was basically cylindrical with the axis of rotation along the geometric axis. The viewing requirements of the seven experiments were met by providing three areas of sensor location, one at each end of the cylinder viewing along the spin axis and a central band mid-way along the cylinder length viewing at 90° to the spin axis. This layout was achieved using a thrust tube on the spin axis and two equipment mounting platforms close the centre of the cylinder.

The general configuration and layout is shown in Fig. 1. The outer wall of the cylinder was made up of the solar cell panels positioned symmetrically each side of the central band of experiment sensors. Each panel formed two facets of a 12-sided figure, chosen in order to simplify the mounting of the solar cell modules.

The solar panels were hinged along one edge which, together with the quickly detachable thermal control shields at each end of the cylinder, permitted ready access to the equipment mounted on the two floors.

The thermal balance was achieved by passive control. Rejection of heat from the electronic units was accomplished by radiation from the end cover shields to space.

The floors were bonded honeycomb panels with aluminium skins and 0.005 cm wall thickness core. The remainder of the material in the structure was magnesium alloy.

3. THE ORBIT

At launch the almost polar orbit had an apogee of 1100 km, a perigee of 350 km and a period of 98 minutes. In order to obtain maximum time in sunlight throughout the life of the satellite, the plane of the orbit was placed normal to the solar vector. This meant that the satellite orbit was at first continuously passing over the dawn-dusk line and hence was constantly illuminated by the Sun. However, due to various disturbing torques the orbit plane rotated about the Earth from west to east at approximately 1° per day and thus, after about four months, the satellite started to enter eclipse. The eclipse duration increased to a maximum of 37% of the orbit. The orbit decayed over the three-year lifetime until immediately prior to re-entry the apogee was only 188 km and the perigee 165 km.

4. SYSTEMS DESCRIPTION

4.1 De-Spin

During the final stage of the injection into orbit the complete fourth stage of the launch vehicle and satellite were gyroscopically stabilised by spinning about the longitudinal axis between 140 and 180 rev/min. After burn-out of the fourth stage motor, separation occurred. At this time, the satellite was still rotating at a high speed and this had to be reduced to within the operating range for the experiments, namely 15 to 40 rev/min. A simple de-spin principal was used, known as a 'yo-yo' system. This consisted of two weights

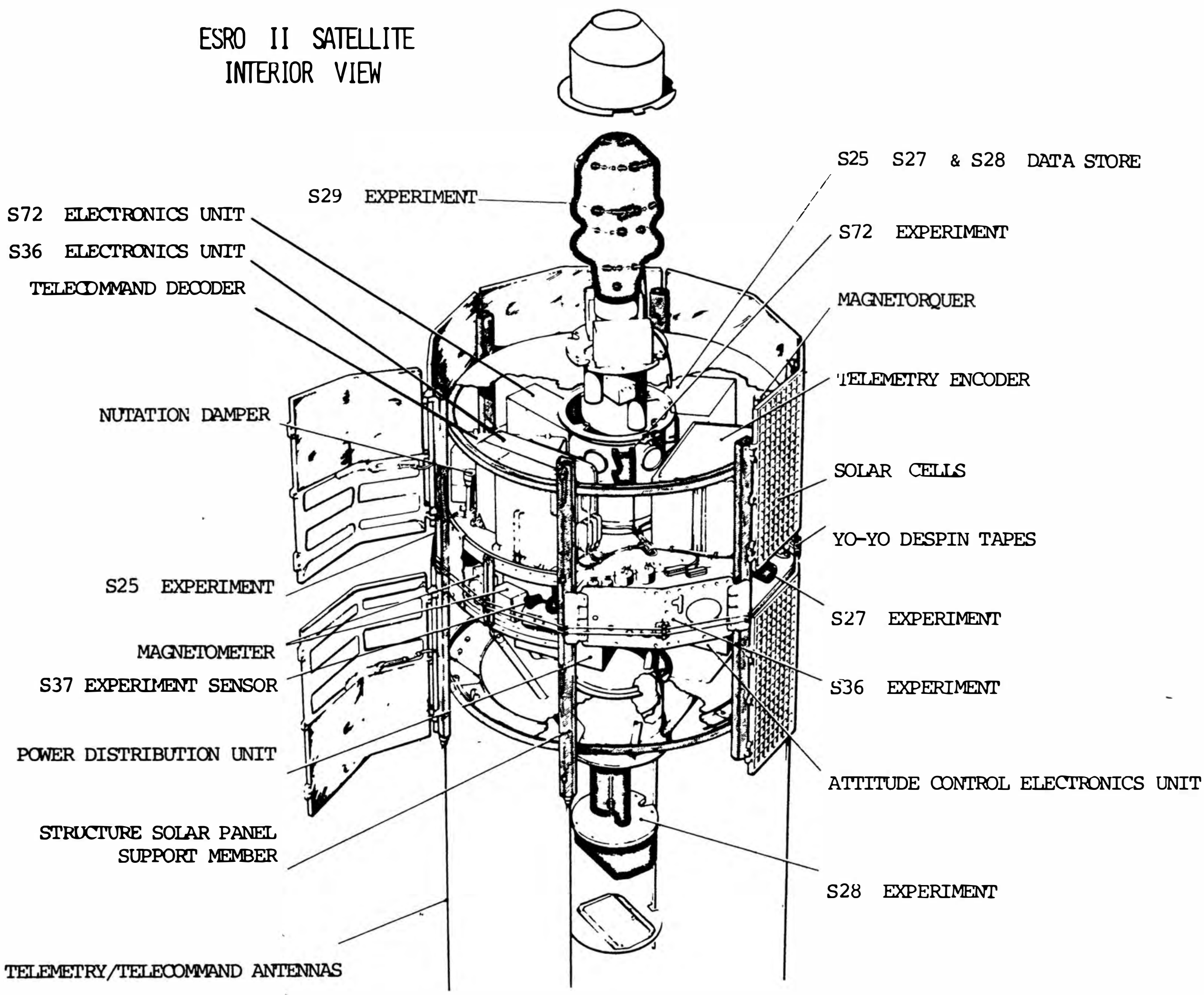

Fig. 1. ESRO II satellite interior view.

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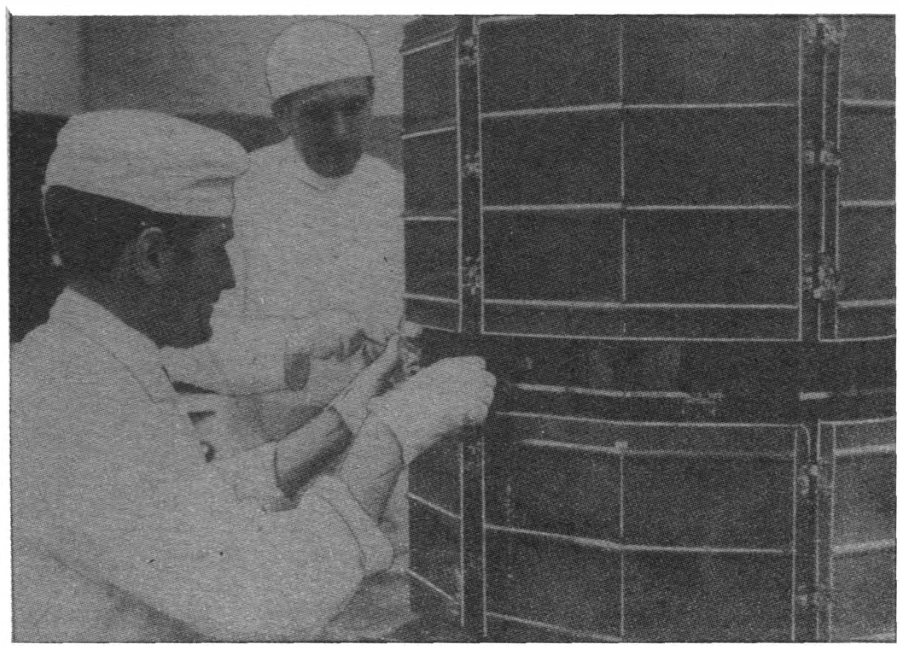

Fig. 2. Fitting of yo-yo despin tapes.

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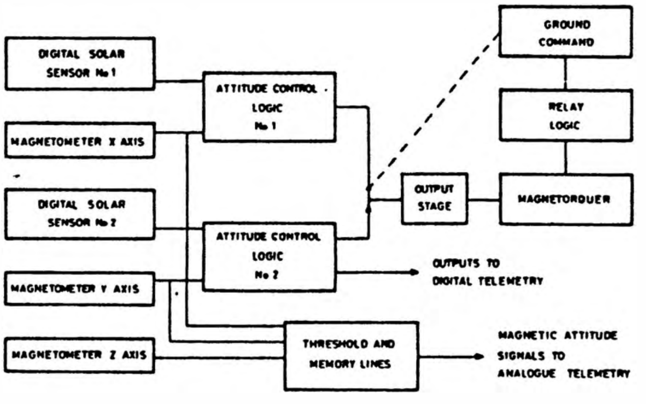

Fig. 3. ESRO II attitude control system functions.

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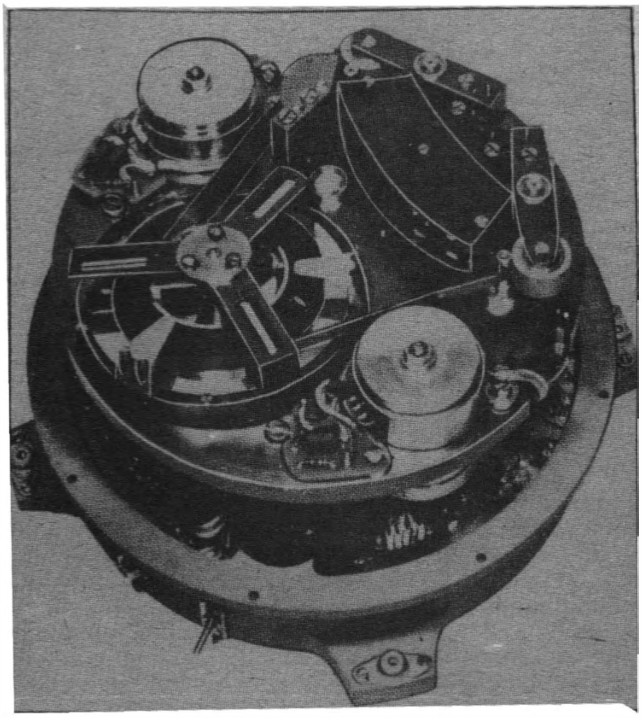

Fig. 4. Interior view of tape recorder.

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attached to the satellite by light, inextensible tapes. The tapes were wound round the centre band of the satellite and terminated at two limited-closure hooks positioned diametrically opposite. The weights were held in position by pyrotechnic latches which were fired simultaneously for release. When the weights were released, the satellite was de-spun due to conservation of the angular momentum and the satellite decelerated as the weights moved outwards. To prevent re-spinning of the satellite, the weights and wires were released following the satellite de-spin. Figure 2 shows the installing of the yo-yo tapes.

4.2 Nutation Damping System

Nutation, or coning, of the spin axis could occur, resulting from asymmetric forces occurring at separation, de-spin, re-spinning or during attitude control manoeuvres. It was essential to minimise any coning angle to ensure that the experiment sensor viewing angles were maintained within the required limits. Additionally, any large coning angles could seriously degrade the performance of the attitude control system. A nutation damping system was therefore provided.

The nutation damper consisted of two tubes mounted diametrically opposite in the satellite at the edge of the floors. Each tube was curved with a radius of curvature of 5 m and contained a ball of 16 g mass. The balls were free in the tubes; each tube was filled with nitrogen gas. The gap between the ball and tube was selected to obtain the correct viscous friction coefficient required to tune the system to the satellite inertia ratio.

4.3 Spin-Up System

It was thought that during the lifetime of the satellite the spin rate would gradually decay due to the effects of retarding torques resulting from:

1. eddy currents generated in the satellite structure and wiring;
2. magnetic hysteresis of ferromagnetic components in the satellite;
3. aerodynamic forces; and
4. impact from micrometeorites and other particles.

The effects of eddy current and magnetic hysteresis torques were thought to be the most significant of these factors.

It was found that the calculations had been over-pessimistic and the spin rate never decayed below 15 rpm. The spin-up system was, however, fired toward the end of satellite lifetime to prove its operation.

The spin-up system was a single gas jet mounted at the circumference of the satellite and fired by the ground command system. Three gas bottles were carried, one for each spin-up, exhausting through the single jet. Each bottle was charged with dry nitrogen at 200 atmospheres. A unique method of actuating the system was used and consisted of an electrically operated burster tube, one for each gas bottle. A tube was wound with a resistance wire, the tension of which was carefully controlled. When spin-up was initiated, an electric current was passed through the wire, causing it to fracture. The hoop tension was thus released from the burster tube and as its strength was inadequate to withstand the internal pressure due to the nitrogen, the tube burst and the nitrogen gas passed to the jet nozzle. It was felt that this system offered more reliability for long periods in space than a pyrotechnic system. The spin rate was measured from a tapping on the solar array and telemetered to the ground.

4.4 Attitude Control System

The satellite spin axis attitude was required to be maintained within ± 10° of the normal to the solar aspect line throughout its lifetime. This requirement was specified in order to satisfy the viewing requirements of the scientific experiments.

The attitude control system was a magnetic torquing system in which control torques were generated by the interaction of the Earth's magnetic field with a field generated within the satellite. The satellite field was obtained by energising an electromagnet positioned parallel to the spin axis. This device was referred to as a 'magnetorquer.'

Sun sensors were used to determine the spin axis attitude relative to the Sun vector and a magnetometer was used to

determine the magnitude of the components of the Earth's magnetic field in the direction of and normal to the spin axis. An automatic on-board logic system was the primary control system with ground command as a back-up. Figure 3 shows a block diagram of the system.

Five command channels were allocated for operation of the attitude control system and could select the "on board" logic or switch current to the magnetorquer directly. Eleven telemetry channels were used for transmission of data for the monitoring of the performance of the attitude control system and could also be employed for attitude reconstitution.

4.5 Power System

The power system was required to supply power to the spacecraft for the whole period of life. It had to operate when as much as 37% of the orbital period was spent in eclipse.

The electrical power was provided by 3500 2 x 2 cm solar cells mounted on the body panels. A nickel cadmium battery supplied power during eclipse periods, thus ensuring that the satellite was fully operational at all times. The only limitations during eclipse were, the loss of the spin rate monitor and Sun sensor outputs.

The main power bus was + 16 volts d.c. derived from the solar cell or battery voltage by means of a PWM regulator. The PWM input voltage was from 17.5 to 24 volts and the normal output power was 15 watts, but it had the capability of delivering 40 watts. The output voltage was 16V ± 2% and output impedance was less than 0.5 ohms up to 10 kc/s; 1 ohm at 10 kc/s to 50 kc/s, and less than 0.2 ohms at d.c. The power conversion efficiency was greater than 85%. A d.c./d.c. converter supplied the experiments and housekeeping system. This converter gave supply voltages of ± 1 volt, + 3 volts, ± 6 volts and -40 volts. Full wave rectification and series regulation were used to achieve the required stability. The power conversion efficiency was greater than 40% at normal full load.

A solar array shunt regulator was provided to limit the maximum working point of the solar array at 23.4 volts. This was necessary to prevent excessive pressure build-up within the battery during overcharge, while allowing for adequate re-charge of the battery. The battery charge logic was temperature controlled.

The power subsystem was designed such that the satellite was still capable of operation in sunlight in the event of battery failure.

4.6 Housekeeping System

A housekeeping unit was provided that enabled ten temperature measurements, three current measurements and a voltage measurement, on the battery line, to be telemetered to ground. It also included the separation timer and firing relays for the de-spin system.

4.7 Aerials

The telemetry and command system included a hybrid and duplexer unit, which formed the interface between the telemetry transmitter, command receivers and the four rod aerials mounted on the base of the satellite. The operating frequencies were:

1. low power transmitter 136.89 Mc/s;
2. high power transmitter 136.05 Mc/s;
3. command receiver 148.25 Mc/s.

Simultaneous operation of all systems was possible.

4.8 Telemetry System

The telemetry system was a pulse code modulation system comprising a low power real time transmitter, a high power transmitter to play back tape recorded data and a PCM encoder.

The PCM encoder generated the following format: bit rate 128 per sec; word length 8 bits; frame length 32 words; master frame length 8 frames; master frame rate 16 sec; and number of words per master frame 256.

The bit rate at 128 per sec was obtained by counting down an internal 2.6 megacycle clock. For synchronisation purposes, the first two words of each frame and master frame were coded and a further two words used to transmit timing information. Ten bits only were employed to count in binary code the number of master frames transmitted, giving a maximum count of 1023 for a period of approximately 274 minutes.

Each experiment transmitted its primary information as serial digital data, using 18 channels between the seven experiments. Some channels consisted of several words. Four eight-bit words were used for on/off data and each bit was fed to the encoder on a separate wire. There were two types of analogue inputs to the encoder, single-ended and balanced inputs, both with a range of zero to 5 volts.

Five commands controlled functioning of the telemetry system and two back-up commands for use in an emergency. The five were record; play back; play back back-up; stop recorder; and tape recorder input to lower power transmitter. A 5-minute timer was built into the system to return from play back to record.

The tape recorder contained an endless 60 m loop of 6.4 mm magnetic tape on a single spool giving a duration of 110 minutes of normal recording at 1 cm/s. Play back took place at 32.5 cm/s. There were two tracks on the tape: one for data and one for basic encoder clock (128 cycles per sec). During play back, the clock signal was used to regenerate the data correctly and the drop out of data was less than 1 in 10^5 bits of information. Figure 4 shows the interior arrangement of the tape recorder.

4.9 Command System

The command system was capable of receiving 36 different commands. The receiver operated at carrier frequency of 148.25 Mc/s. It had two IF frequencies at 10.7 Mc/s and 1.65 Mc/s.

5. THE EXPERIMENTS

The seven experiments carried by ESRO II were supplied by five experimental groups. They were divided into two missions, with S36 and S37 studying solar X-rays in the 1 to 60 Å band and S25, S27, S28, S29 and S72 being particle experiments examining solar, Galactic and Van Allen trapped particles.

The Scientific Mission was directed by Dr. Edgar Price, who was the ESRO II Project Scientist. The experiments were provided by the following groups.

S25, S27 and S28 – from Imperial College, London, were all under the direction of Professor H. Elliot. These three experiments employed a common store for the output of their data.

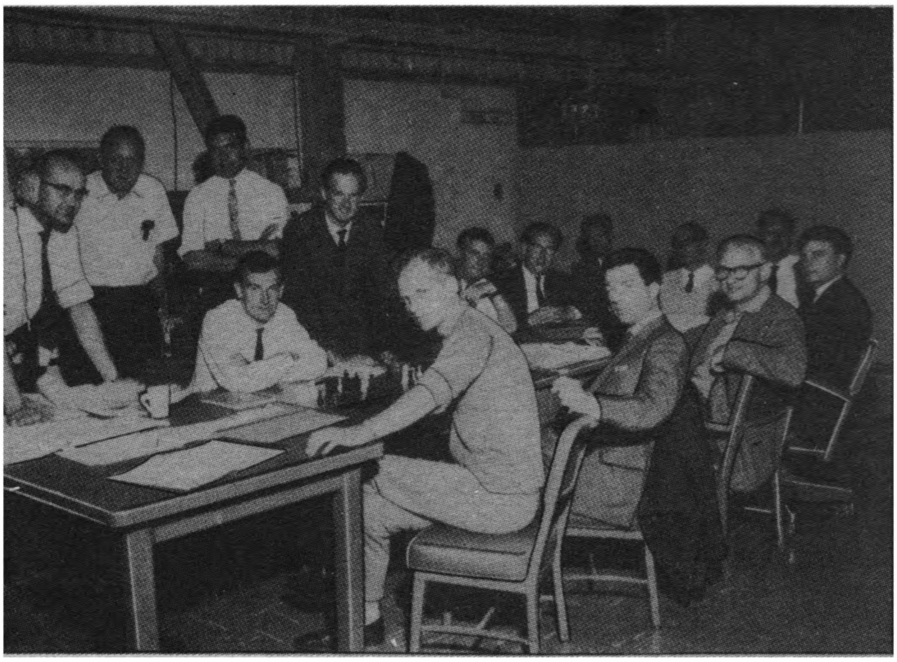

Fig. 5. ESRO II experimenters in the final tense minutes awaiting the launch.

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S25 – The experimenter, Arthur Bewick, was also the Imperial College electronics expert for the mission. S25 contained two Geiger-Muller counters for high and low intensities and measured the flux of Electrons above 1MeV and Protons above 15MeV.

S27 – Bob Hynds was the experimenter and S27 consisted of a Proton telescope comprising four solid state detectors separated by absorbers. Coincidence gating allowed monitoring of Protons in four energy ranges between 1 and 100 MeV and Electrons between 5 and 70 MeV.

S28 – Alastair Durney was the scientist responsible and S28 consisted of a telescope formed by two scintillators, two proportional counters and a Cerenkov counter. Photo-multipliers were used to view the scintillators and Cerenkov counters. S28 measured the time dependence of the flux ratio of protons and alpha particles of magnetic rigidity 0.4 to 0.8 GeV.

S29 – provided by University of Leeds was under the direction of Dr. (now Professor) Phil Marsden. The experimenters were Barry Crowden and Tom Napier. S29 employed a gas filled Cerenkov counter and lead scintillator sandwich with outputs being taken from Photo-multipliers. It measured Primary Cosmic Ray Electrons in the GeV range and Protons of energy greater than 300 MeV.

S-36 – This experiment was supplied jointly by University College London and Leicester University under the direction of Professor R.L.F. Boyd and Dr. A. Willamore. The experimenters were Len Culhane, Peter Sandford and Mike Shaw. S36 consisted of five gas filled proportional counters and measured solar X-rays in the range 1 to 20 Å.

S37 – was supplied by the University of Utrecht under the direction of Professor C. De Jager. The experimenters were Bert Brinkman and Hans Imhof. The experiment used two detectors each consisting of two interconnected proportional counters, one being the detector and the second used to give a calibration output from an internal source. S37 measured the flux of soft solar X-rays in wavelengths between 44 and 70 Å.

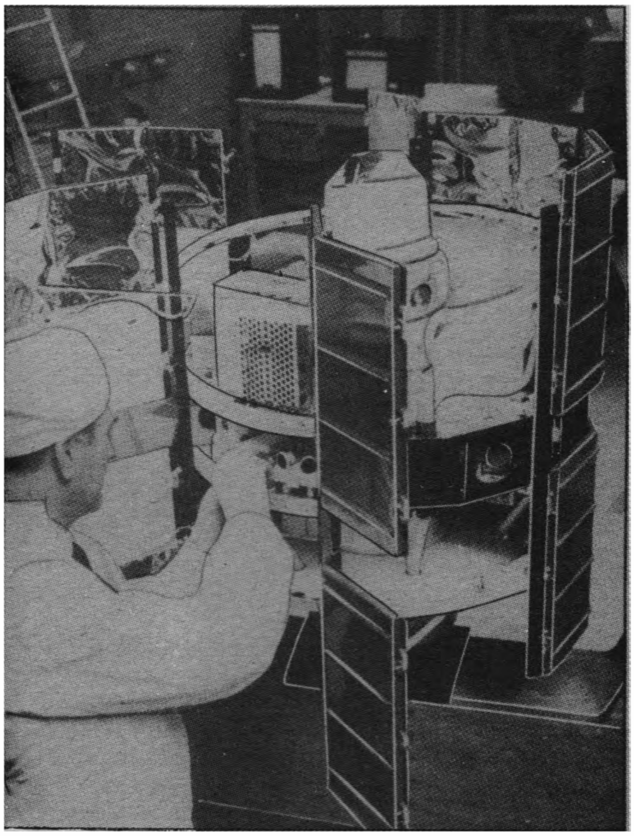

Fig. 6. Installation of equipment into a flight model satellite.

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S72 – This experiment was provided by the Centre d'Etudes Nucleaires de Saclay, France, under the direction of Dr. J. Labeyrie. The experimenters were Yves Amran and Jacques Andrejol. S72 employed a telescope of three solid state detectors and measured the flux and energy distribution of Protons between 25 MeV and 1 GeV.

Figure 5 shows the team of experimenters in the final tense moments awaiting the launch.

6. TESTING

The integration and test team was led by Peter Conchie and contained specialists for each subsystem who, in many cases, had been involved in the design. This gave a small, close-knit team that contributed greatly to the success of the project.

The model philosophy was lavish by today's standard and was possible due to the relatively small size and simple construction.

1. first vibration model to obtain early design information;
2. second and more refined vibration model to confirm structural design. This structure was later to be used for 'Breadboard' integration;
3. thermal model for heat balance testing;
4. P1 design test model to full flight standard to confirm design;
5. P2 Customer design acceptance model;
6. F1 flight model;
7. F2 flight spare.

The initial electrical integration and RF susceptability testing was performed on the bench, using a breadboard harness, prior to performing electrical integration on the ex-vibration model structure.

Figure 6 shows an equipment being installed with the satellite mounted on the wood RF test rig.

Many of the test methods and equipment employed were primitive by today's standards but nonetheless effective. Because of their simplicity it was usually easy to determine whether an anomaly was due to the subsystem or test aid. Figure 7 shows the P2 satellite on vibration test.

The classic illustration of an unsophisticated but most satisfactory test was the "Boy Scout" checkout employed on the attitude control subsystem by Ken Kendall. Apart from the spacecraft on its handling frame and the telemetry and command checkout equipment, all that was required was a powerful electric torch and a pocket compass. The approach was to verify with the compass that the spacecraft was in a reasonably undisturbed part of the Earth's magnetic field; then, by presenting the spacecraft at differing attitudes to the Earth's field and shining the torch into the appropriate Sun sensors, it could be checked by means of the telemetry that the appropriate logic responses were obtained. Since the attitude correction was achieved by the interaction of the on-board magnetorquer with the Earth's field, the overall operation of the system could be verified by the pocket compass alone. A knowledge of the Earth's field level at the test site enabled the magnetic sensors in the on-board attitude measurement subsystem to be verified *via* the telemetry. All the parameters were accurately measured during unit and spacecraft testing, using hard line connections and elaborate fixtures, but the final question, "Will it work properly in orbit?," needed to be verified after the spacecraft had been finally closed. This verification had to be simple for its validity, as far as function and sense were concerned, to be unquestionable. The possibility of calibration shift was very small.

The test documentation would not have satisfied today's Product Assurance managers. However, in the hands of engineers who had "lived" with the satellite from its inception and who each had a personal commitment to its success the test programme was completed with very few serious problems.

Although there were very few significant test problems with the satellite subsystems, a number of difficulties were encountered with the experiments, many of which arose from the high voltage supplies used for the photo multipliers. Under vacuum during the satellite thermal vacuum testing corona discharge was experienced on several experiments. It was also found necessary to perform repairs and modifications on experiments during the launch campaign. Because of the risks involved in disturbing the experiments and in particular the high voltage circuitry it was considered necessary to re-submit the experiments and, in many cases, the satellite to thermal vacuum testing. During the first launch campaign the team was repeatedly 'commuting' the 250 km between Vandenberg AFB and Los Angeles to use the TRW thermal vacuum test facilities at Redondo Beach. On each occasion the satellite and its test equipment had to be packed and shipped in the huge "Air Ride" trucks. Although the test philosophy may have erred on the side of caution, the subsequent success of the project confirmed its wisom.

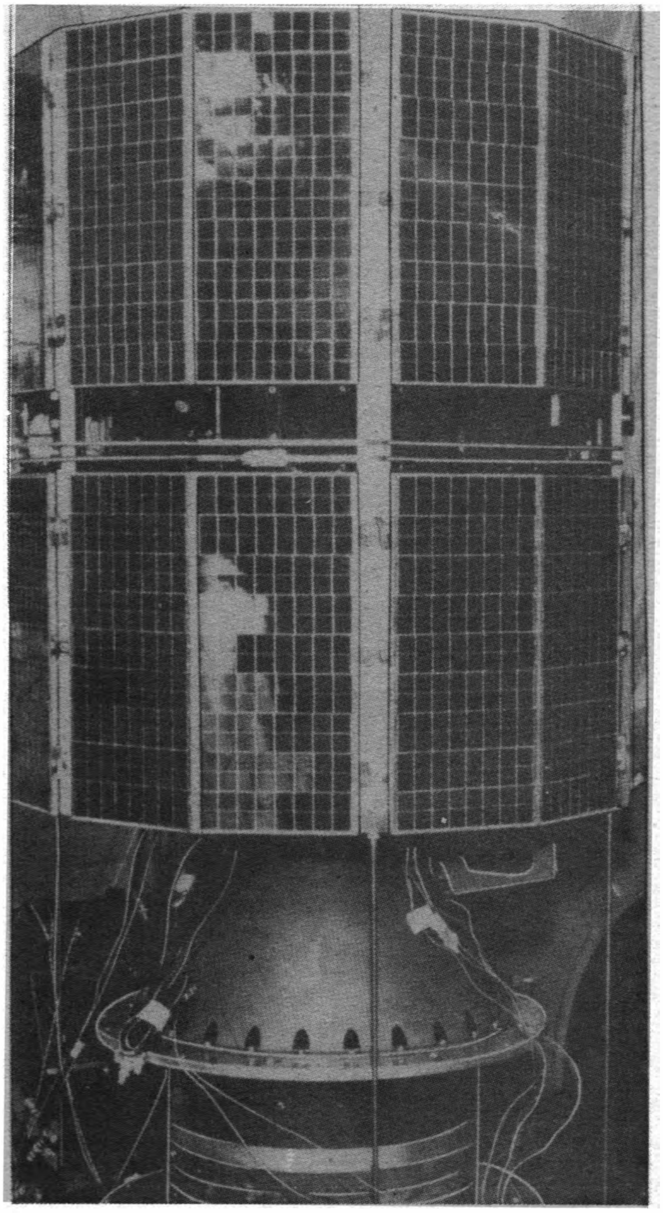

Fig. 7. P2 satellite mounted on vibration slip table.

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Most of the experiments required the use of radioactive sources for their checkout or calibration. This presented a

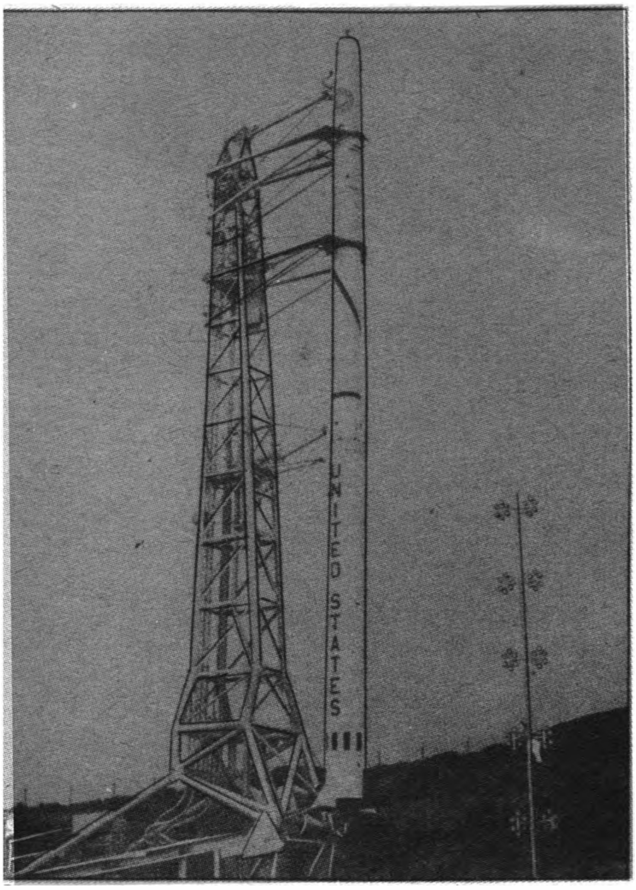

Fig. 8. The Scout rocket in the last minutes of the countdown.

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problem to Hawker Siddeley since the factory authorities had not previously had to store and handle radioactive materials. After much consultation it was agreed that they could be stored in a steel box which had to be locked away in a suitably marked steel cabinet. These precautions were greeted with some amusement by the experimenters who had a more cavalier attitude to the handling of these materials. On one occasion Jacques Andrejol from Saclay arrived at a Customs check with a Hawker Siddeley engineer. The Customs Officer, after reading the import documentation for the S72 experiment, asked to see the radioactive source. Jacques is reported to have produced it from his trouser pocket, in a match box, much to the dismay of the onlookers. Members of the team later noted with interest that Jacques did not appear to have suffered any ill effect.

7. QUALITY CONTROL

Early in the programme a NASA representative, Bob Martin, gave a teach-in on Quality. One statement he made is still remembered and remains true today *viz:* "If you have a dedicated team you get quality for free!" I am certain that this was achieved on ESRO II because we did have a small and dedicated team.

Quality Control was implemented in a thorough and practical manner by a Quality Engineer and Inspectors but lacked the formality and much of the documentation that our customers require today.

High Reliability electronic components were unheard of in those days and ESRO II used components to the American MIL standard. The parts used in experiments were of more dubious origin and experimenters were known to visit the local radio dealers to buy "Radiospares" components. We did not have the requirements for full traceability on all parts and materials that we enjoy today.

8. THE OPERATION OF ESRO II

After the disappointment of the launch failure in 1967, the launch of the F2 satellite in May 1968 was a complete success with performance of both the Scout rocket and the satellite near nominal. Figure 8 shows the Scout rocket on the launch pad shortly before launch. The in-orbit operation of the satellite was excellent and was summed up by the following statement in the ESRO Report for 1968:

> "The behaviour of the spacecraft in orbit has been so near nominal that any description reads very much like earlier explanations of what it was supposed to do."

This most satisfactory state of affairs continued with ESRO II remaining fully operational up to its re-entry on 8 May 1971 on its orbit number 16283.

The only element of the satellite subsystems to fail was the tape recorder. This exceeded the average life of similar recorders and was stopped by a mechanical fault after seven months of operation. During this time the peak flutter value had increased by only 0.4%.

The operation of the experiments did suffer some anomalies but all collected more than enough data to satisfy their mission objectives. Two months after launch one of the Imperial College experiments experienced corona discharge which caused interference on the telemetry system and loss of data. It was necessary to switch all three experiments off since they were all controlled by one command. It was thought that a cavity had been formed in the insulation of one of the EHT supplies which outgassed very slowly only reaching the critical partial vacuum of 10^{-3} Torr, at which corona is likely to occur, after two months. This was confirmed as it was later found possible to switch them on again and they operated without further problems. The two solar X-ray experiments also experienced some distruption of EHT supplies when the satellite was near to perigee. This was believed to be due to build up of plasma causing a temporary increase in pressure which was sufficient to give an increased leakage rate on the EHT supplies, resulting in a desensitising of the sensors.

The experimental mission on ESRO II was declared a success by ESRO and this was confirmed in later ESA scientific papers. As in the majority of scientific investigation no Earth-shattering discoveries were made. However, the ESRO II experiments produced much valuable data which, when correlated with other information, helped towards completing the scientific 'jig saw' and expanded our understanding of the Sun and Galaxy.

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CORRESPONDENCE

Skylab History

Sir, Back in 1965 or so, at the Douglas Company's space facility in Huntington Beach, California, hydrostatic tests of the liquid oxygen and liquid hydrogen tank sumps for the Saturn third stage were conducted. In an over-pressure condition a crack developed in one of the specimens. It was determined that the cause was high discontinuity stresses created by the centroid of the jamb reinforcement ring being eccentric with respect to the tensile membrane of the tank dome hemisphere. The decision was made to change subsequent vehicles.

As long as the hardware was being changed for this problem, however, it was decided that the hydrogen tank opening might as well be enlarged from 71 cm to 109 cm. This was instigated by some friendly collusion between Ted Gordon, head of our preliminary design department, and Wernher von Braun. They were pushing at tnat time for the spent stage programme that later became Skylab. As part of the justification for the change, the authorisation document asserted that the crack formation problem could be alleviated by increasing the hole diameter. Curiously enough, this phenomenon seemed to apply only to the hydrogen tank.

I remember commenting at the time, 'I wonder when someone is actually going to refer to this justification as authority for the theory that crack propagation is prevented by enlarging the opening.'

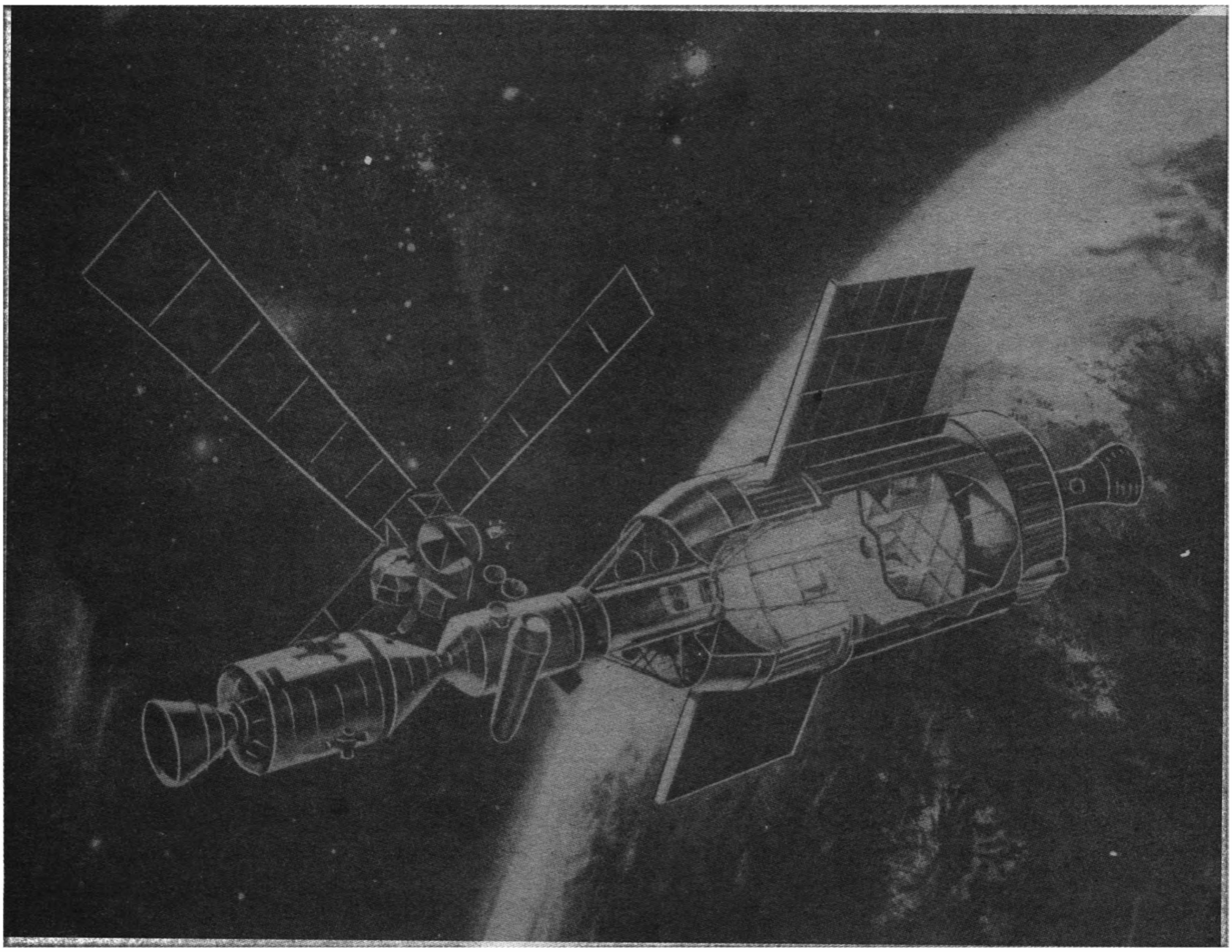
I see from the historical article about spent stages in the April *JBIS* [1] that the action and the excuse for it have now been cited as scientific precedent. I hope that this note will stop the hole-expanding theory of crack stoppage becoming imbedded in the folk lore.

O. P. HARWOOD
Huntingdon Beach,
California. USA.

REFERENCE

1. W. D. Compton, 'The Rocket as Spacecraft: Spent Stages in Manned Space Flight,' *JBIS*, **38**, 147-154 (1985).

An early Skylab concept. The opening into the liquid hydrogen tank is arrowed. *Douglas*



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The Eagle Has Wings (144 pp) traces the development of astronautics in the United States from 1945 to the end of the Apollo era in 1975.

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