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ORION - A SUPERSYNCHRONOUS TRANSFER ORBIT MISSION

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ORION F1 was launched on 29th November 1994 on an Atlas IIA launch vehicle. It was designed, built and delivered in-orbit by Matra Marconi Space Systems Plc and was handed over to ORION Satellite Corporation on 20th January 1995 at its on-station longitude of 37.5°W. The mission differed significantly from that of any other geostationary communications satellite in that the Transfer Orbit apogee altitude of 123,507 km was over three times geosynchronous (GEO) altitude and one third of the way to the moon.

The SuperSynchronous Transfer Orbit (SSTO) mission is significantly different from the standard Geostationary Transfer Orbit (GTO)mission in a number of ways. This paper discusses the essential features of the mission design through its evolution since 1987 and the details of the highly successful mission itself including a detailed account of the attitude determination achieved using the Galileo Earth and Sun Sensor (ESS).

THE ORION SYSTEM

The ORION Launch and Early Orbit Phase (LEOP) was the first use of the Satellite Control Centre at MMS, Stevenage. This was the first LEOP Control Centre built in the UK and was designed and built in-house in preparation for the ORION launch. All of the LEOP operations were performed by MMS staff. The Control Centre made use of the ground stations at Perth, Allan Park and Chilworth for TM/TC and tracking from the TELESAT network, and the newly built ORION station at Mt. Jackson, VA.

QUARTZ, the Flight Dynamics software used during the LEOP was written specifically for, although not limited to, the ORION F1 mission. It consists of a VAX workstation based environment with a central database and a high quality MOTIF user interface. Many of the algorithms had been well tested previously within the mission department of the company and these were brought together into an integrated suite of Flight Dynamics software.

The ORION spacecraft weighed 2361 kg at launch, 1200 kg of which was liquid propellant. Injection into the various transfer orbits was performed using a 490 N Marquardt Liquid Apogee Engine. The spacecraft was passively spin stabilised at 12 rpm during the Transfer Orbit phase. The propulsion system is combined with the on-station thrusters so that any propellant saved during Transfer Orbit was used directly to extend life.

Because of the special nature of the super-synchronous transfer orbit, before describing the results of the LEOP itself, some of the pertinent aspects of the mission design are outlined below.

MISSION DESIGN

The following points provide the reasons and the logic behind the supersynchronous mission design.

The launch vehicle selected for the ORION mission was the Atlas IIA manufactured by Martin Marietta Commercial Launch Services. Due to the latitude of the launch site, the injection orbit would have an inclination between 23 and 27.5 degrees.

If ORION had used a standard transfer orbit (i.e. apogee at GEO altitude) with such an inclination, the propellant costs would have been prohibitive and the lifetime requirements would not have been met. Although the excess launch vehicle propellant could have been used to reduce the inclination further, the lifetime would still have been less than 7 years.

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By selecting the SSTO, the propellant costs to GEO were significantly reduced. In SSTO the majority of the plane change is performed at apogee. It is most efficient at apogee because this is where the spacecraft is moving the slowest and it is close to the ascending node. The higher the apogee the lower the spacecraft velocity and hence, the more efficient the plane change.

Pushing out the apogee radius to 130,000 km, making use of excess performance available from the launch vehicle, made the plane change for ORION much more efficient - see the comparison below:

	<u>ΔV to reach Drift Orbit</u>
Ariane standard GTO	approx. 1505 m/s
ATLAS IIA GTO	greater than 1750 m/s
ORION SSTO	1473 m/s

The SSTO mission therefore significantly decreases the ΔV required by the Liquid Apogee Engine (LAE) and thereby enabled ORION to achieve the lifetime requirement.

Figure 1 shows the SSTO relative to the geostationary orbit.

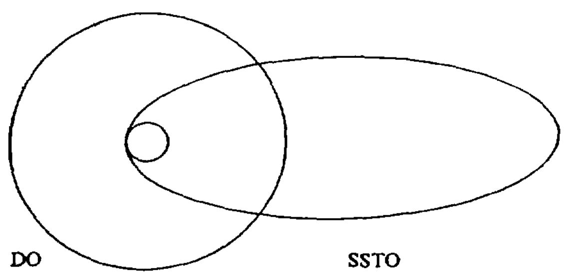

Figure 1 SSTO and GEO

The overall strategy to transfer from the SSTO to GEO is to perform a burn at apogee to raise the perigee to geosynchronous. A retrograde burn at the perigee of this Intermediate Transfer Orbit (ITO) then lowers the apogee down to geosynchronous.

To further minimise the propellant usage during LEOP, two apogee burns were selected rather than one. This improved the mission's robustness to uncertainties in the LAE performance and the attitude determination.

Two perigee burns were also selected at perigees 4 and 6 (see Table 1). The reasons behind this decision are as follows:

Table 1 Nominal LEOP Burn Strategy

Burn	Apse	Nature	ΔV (m/s)	Transfer	$T - T_0$ (hrs)
1	A1	Prograde	643.93	SSTO-ITO1	24.04
2	A2	Prograde	131.89	ITO1-ITO2	86.86
3	P4	Retrograde	329.12	ITO2-ITO3	190.17
4	P6	Retrograde	367.87	ITO3-DO	264.67

The ORION spacecraft has linearly polarised telemetry (TM) and telecommand (TC) bicone antennas with their boresights in the spacecraft XY plane. During the critical Sun and Earth Acquisition manoeuvres in Drift Orbit (DO), which involved the spacecraft rotating about its X-axis, it was necessary to have a ground station which could operate in circularly polarised mode to ensure continuous TM/TC access. There was only one ground station from the tracking network which could provide this service - Allan Park. This ground station also covered the on-station longitude.

Thus it was decided to design the nominal strategy such that in the event of having to adopt a burn back-up scenario, it would always be possible to re-target the longitude of the final perigee burn (entry into DO) to be close to on-station and hence above the horizon at Allan Park.

Various scenarios were investigated, but it was decided that two perigee burns, separated by two orbit revolutions, provided the longitude re-targeting flexibility required. By varying the proportion of the first perigee burn, the final burn longitude could be adjusted. Not only did this provide the flexibility for the burn back-up (BBU) scenarios (see Table 2), it also provided the ability to correct for any errors from the two apogee burns in the nominal scenario.

Table 2 Nominal & Burn Back-up (BBU) Strategies

Strategy	ΔV1 (m/s)	ΔV2 (m/s)	ΔV3 (m/s)	ΔV4 (m/s)	ΔV5 (m/s)	Final Longitude
Nominal	643.9	131.9	329.1	367.9	-	332.5° E
1BBU	644.4	132.0	332.1	364.8	-	332.6° E
2BBU	643.9	131.9	254.9	442.0	-	332.5° E
3BBU	643.9	131.9	303.2	393.8	-	332.6° E
4BBU	643.9	131.9	329.1	53.2	314.6	332.5° E

Note that always having to achieve a certain longitude with the final burn drove the fourth burn back-up (4BBU) strategy to have 5 burns - an extra burn was added to allow longitude re-targeting (see Table 2).

Adoption of a multiple burn super-synchronous strategy was unavoidably going to result in a transfer orbit duration much greater than ever experienced before. The impacts on the spacecraft system design were potentially significant and had to be carefully studied during the spacecraft design phase. In order to minimise the impact of the extended duration, the number of revolutions between burns was kept to a minimum.

The spacecraft Earth Elevation Sensor (EES) cant angle was carefully chosen such that, during any one transfer orbit revolution, two EES data passes (used for attitude determination) were available - one at perigee and another on the ascent to apogee (see Figure 2).

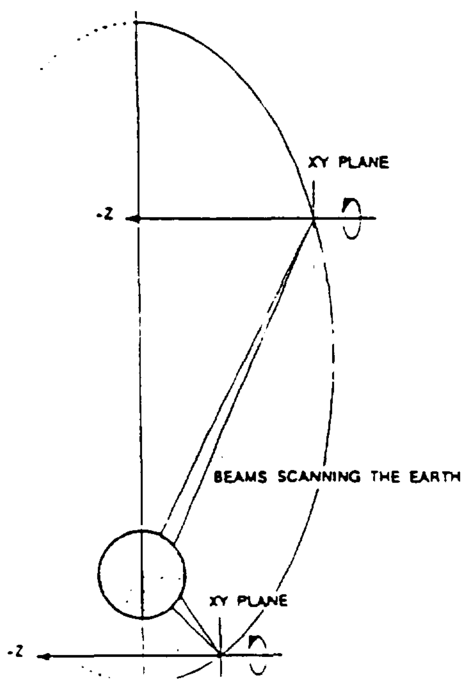

Figure 2 EES Apogee and Perigee Passes

To be able to minimise the number of orbital revolutions between burns, it was decided to use the perigee ESS data passes as well as the apogee passes. However it is not advisable to use the ESS below an altitude of 25,000 km, so the first apogee burn was sized to ensure that the perigee altitude of ITO1 was at least 25,000 km. Therefore, after the first LAE firing all perigee ESS data was deemed usable for attitude determination purposes. This resulted in a fixed 85%:15% apogee burn split.

Major Technical Issues

During the proof of concept phase of the mission a number of major technical issues, themselves a product of the super-synchronous strategy, had to be addressed and resolved satisfactorily. The points below summarise the most significant issues:

- *Attitude Determination:* ORION was to make use of both perigee and apogee ESS data passes (see Figure 2) for attitude determination during transfer orbit. The super-synchronous strategy had raised some interesting questions that had to be answered:

  The apogee ESS passes occur at altitudes in excess of 100,000 km. The sensor had never been used at such extreme altitudes before. The major effect is that the earth appears much smaller in the sensor field of view - hence the earth chords are much smaller. On the other hand, the spacecraft is moving so much slower at the high altitude that each apogee pass lasts for several hours - providing much more data that ever previously obtained.

  Thus, for the apogee passes, the amount of data was not a problem, the questions to be answered were: How would the sensor behave with such small chords and how would these small chords affect the attitude determination solution ?

  Thus a detailed analysis had to be performed, which involved modelling the sensor, to prove that the spacecraft attitude could be determined to sufficient accuracy for manoeuvre planning purposes.

- *TM/TC Link Margins:* The obvious effect of pushing the apogee radius out to near 130,000 km was to place considerable strain on the link margins. It was essential to guarantee adequate link margin since the critical LAE burns were to be performed when the spacecraft was at its greatest distance from the earth. Detailed analysis had to be performed, which involved calculating the link margins at every point in the orbit, to prove that links were available for these critical operations.

- *Long TM/TC Outages:* ORION's TM/TC antenna configuration consists of TM/TC +Z horns plus TM/TC bicone antennas with their boresights in the spacecraft XY plane. The resultant TM/TC nulls about the spacecraft are illustrated in Figure 3.

  Due to the TM/TC null at the rear (-Z) of the spacecraft, TM/TC outages were experienced in every orbit on the descent from apogee. These outages ranged from 4 hours in SSTO up to 19 hours in ITO2.

  Substantial analysis was performed to demonstrate that the spacecraft was robust to failures and had sufficient autonomy to cope with such outages. An on-board applications program was developed to increase the spacecraft's autonomy during outages. This program was configured from the ground prior to outage entry. The analysis concluded that the risk presented by these outages was acceptable. Note, in the event of a spacecraft anomaly which prohibited the spacecraft being out of contact with the ground for any extended period, the back-up solution was to perform a slew manoeuvre to the orbit pole to avoid the outages.

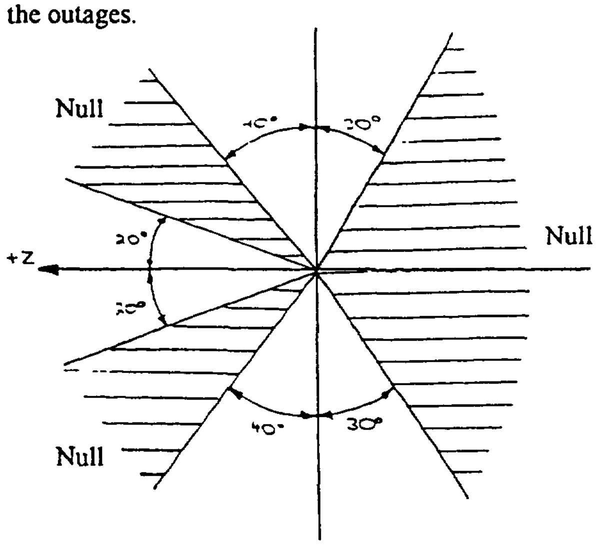

Figure 3 TM/TC Antenna Nulls

- *Orbit Determination:* The orbital geometry associated with the super-synchronous mission was significantly different to any experienced beforehand. Sufficient analysis had to be performed to confirm that the orbit could be determined to sufficient accuracy, within the time available. This was performed using an observations generator to simulate tracking data and using the QUARTZ orbit determination software to establish the attainable accuracy.

- *Launch Window:* Efforts were made to provide launch opportunities on every day of the year. As it turned out, this was not possible.

  The eclipse seasons presented some interesting problems. Midday launches were ruled out, since terrestrial eclipses lasted 6 hours around apogee due to the fact that the spacecraft is moving so slowly. To achieve a midnight launch opportunity during the eclipse season, it was necessary to bias the eclipses away from perigee and hence away from the perigee burns - burning in an eclipse would have meant having no spin rate or nutation data during the burn which was not acceptable. This biasing had major impacts on the allowable Solar Aspect Angle during LEOP.

  Another consideration, unique to the super-synchronous strategy, was the effect of lunar gravity on the injection orbit. Depending on the relative positions of the moon and the spacecraft's orbit, the lunar gravity could have the effect of raising or lowering the injection orbit perigee by hundreds of kilometres. The 1BBU strategy (see Table 3) results in the spacecraft spending 1.5 revolutions in the injection orbit. Potentially, the lunar gravity could lower second perigee to an altitude below 167 km (nominal = 185 km) at which point the resultant heating effect on the spacecraft would be unacceptable. Hence, the launch window would be closed due to these lunar gravity effects.

Table 3 Nominal & BBU Strategies

Strategy	Burn Apses	Total ΔV (m/s)	Transfer Orbit Duration (hrs)
Nominal	A1,A2,P4,P6	1472.8	264.67
1BBU	A2,A3,P5,P7	1473.4	312.53
2BBU	A1,A3,P5,P7	1472.8	336.48
3BBU	A1,A2,P5,P7	1472.8	336.48
4BBU	A1,A2,P4,P7,P8	1472.8	336.47

The LAE Burn Strategy

Figure 4 shows the LAE Burn Strategy and the resultant intermediate orbits between SSTO and Drift Orbit (DO).

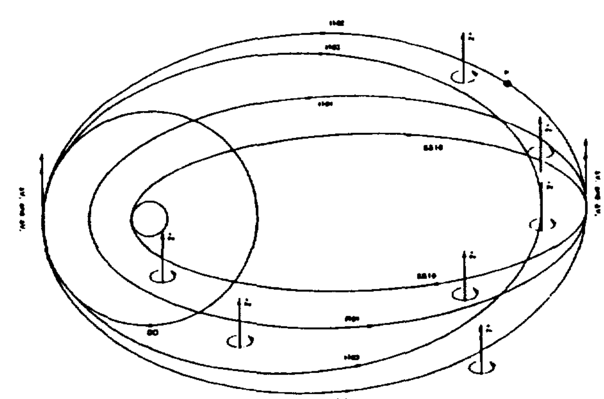

Figure 4 LEOP Nominal Burn Strategy

Table 4 Nominal Strategy - Orbit Details

Orbit	SemiMajor Axis (km)	Eccentricity	Inc. (deg)	Period (hrs)
SSTO	68223.95	0.9038	26.0	49.26
ITO1	80240.39	0.6005	3.12	68.83
ITO2	85296.20	0.5056	1.10	68.86
ITO3	56627.42	0.2553	0.63	37.25

The following points should be noted from the SSTO Strategy tables:

- The possibility of missing any one of the nominal burn opportunities is taken into account in the design.
- The total LAE ΔV requirement is approximately 1473 m/s.
- Going to any back-up strategy results in a negligible propellant penalty.
- The same final sub-satellite longitude is achievable regardless of the burn strategy adopted. (The on-station longitude for ORION F1 is 322.5 East).
- The sizes of the two apogee burns are effectively fixed, while those of the two perigee burns can vary significantly, from one burn strategy to another.
- Almost all the orbital plane change is performed by the two apogee burns.
- The duration of the transfer orbit can range from 265 to 337 hours, i.e. 11 to 14 days.

Once the spacecraft achieves Drift Orbit the operational activities required to achieve three-axis stabilised, Earth pointing, normal mode are relatively standard for this type of spacecraft, i.e. Sun Acquisition, followed by Earth Acquisition, entry into normal 3-axis mode and then station acquisition.

RESULTS OF THE ORION MISSION

The Atlas IIA flight designated AC110 launched at the opening edge of the 84 minute window at 10:21 UT on 29th November 1994. The first attempt on 21st November was delayed due to adverse weather and the second attempt on the 22nd was aborted four seconds before lift-off due to a launch vehicle minor mechanical fault.

The ascent phase of the Atlas and both phases of the Centaur upper stage powered flight were described as perfect by the Atlas launch team. The second Centaur burn was retargeted to reduce the Transfer Orbit inclination from 26.9° to 25.7° as the on-board computer estimated a Propellant Excess of 45 lb. due primarily to favourable winds during the Atlas phase. As the retargeted Transfer Orbit would mean the spacecraft would rise in a different place as seen by the Perth Ground Station, the Atlas launch team relayed the new targeted inclination in real-time to allow Perth to relocate their Antenna position before Acquisition.

Spacecraft separation occurred on time at 10:51. The telemetry beacon was detected by the Perth ground station twelve minutes later as it rose above the horizon, and telemetry lock was achieved at Stevenage at 11:04:08, exactly as predicted. No search pattern was required, hinting at an accurate injection by Centaur. The Atlas launch team estimated a spin rate at separation of 5.053 rpm, confirmed one hour later from the spacecraft sun sensor at 4.937 rpm.

At 11:25, the Atlas launch team provided their estimated Transfer Orbit parameters and attitude prior to separation.

The retargeted and achieved Transfer Orbit parameters were:

	Target	Achieved
ra (km)	129, 885	130, 233
rp (km)	6563.1	6563.5
i (deg.)	25.700	25.686
Ω_G (deg)	173.6	173.5
ω (deg)	179.98	179.98

After 23 hours of tracking we were able to confirm this orbit to within 4 km which was within the error of the estimate. Confirmation of the attitude had to wait until an orbit match was performed through the first apogee firing, and was found to be within 0.5° of the target, well within the 1.5° specified.

The following operations then took place in the next five hours:

- Immediately after initial acquisition by the tracking network, the spacecraft underwent a health check of all subsystems.
- The spacecraft was configured for transfer orbit by switching on the attitude control equipment and pressurising the propulsion subsystem.
- The payload reflectors were then deployed once the injection spin rate was confirmed. This decreased the

spin rate to 4.3 rpm due to the increase in spacecraft inertia.

- A spin-up manoeuvre was then performed to the nominal transfer orbit spin rate of 12 rpm required for stability during the LAE firings.

With the spacecraft now ready for its first LAE firing at Apogee 1, Flight Dynamics activities were to determine the orbit and attitude and to optimise the sequence of firings. The following sections describe the orbit and attitude determination and manoeuvre planning in more detail.

ORBIT DETERMINATION

Range, azimuth and elevation data was received from the TELESAT network via the Ottawa hub. The data was filtered and smoothed by QUARTZ before being passed to a standard Weighted Least Squares algorithm for the orbit fitting.

The software allows for azimuth, elevation and range biases to be solved, considered or fixed for up to three ground stations at once. The software can also solve for the attitude and ΔV of an LAE manoeuvre during the observations, or the Transverse, Normal and Radial components of a station-keeping manoeuvre. Solving for the attitude of the manoeuvre is referred to as orbit-matching. Pre-launch analysis showed this to be potentially very accurate and this was seen during flight. All orbit matches compared well with the attitude determined using the Earth and Sun Sensor, and in general the orbit matched attitudes were used as the attitude on which to optimise the forthcoming manoeuvre.

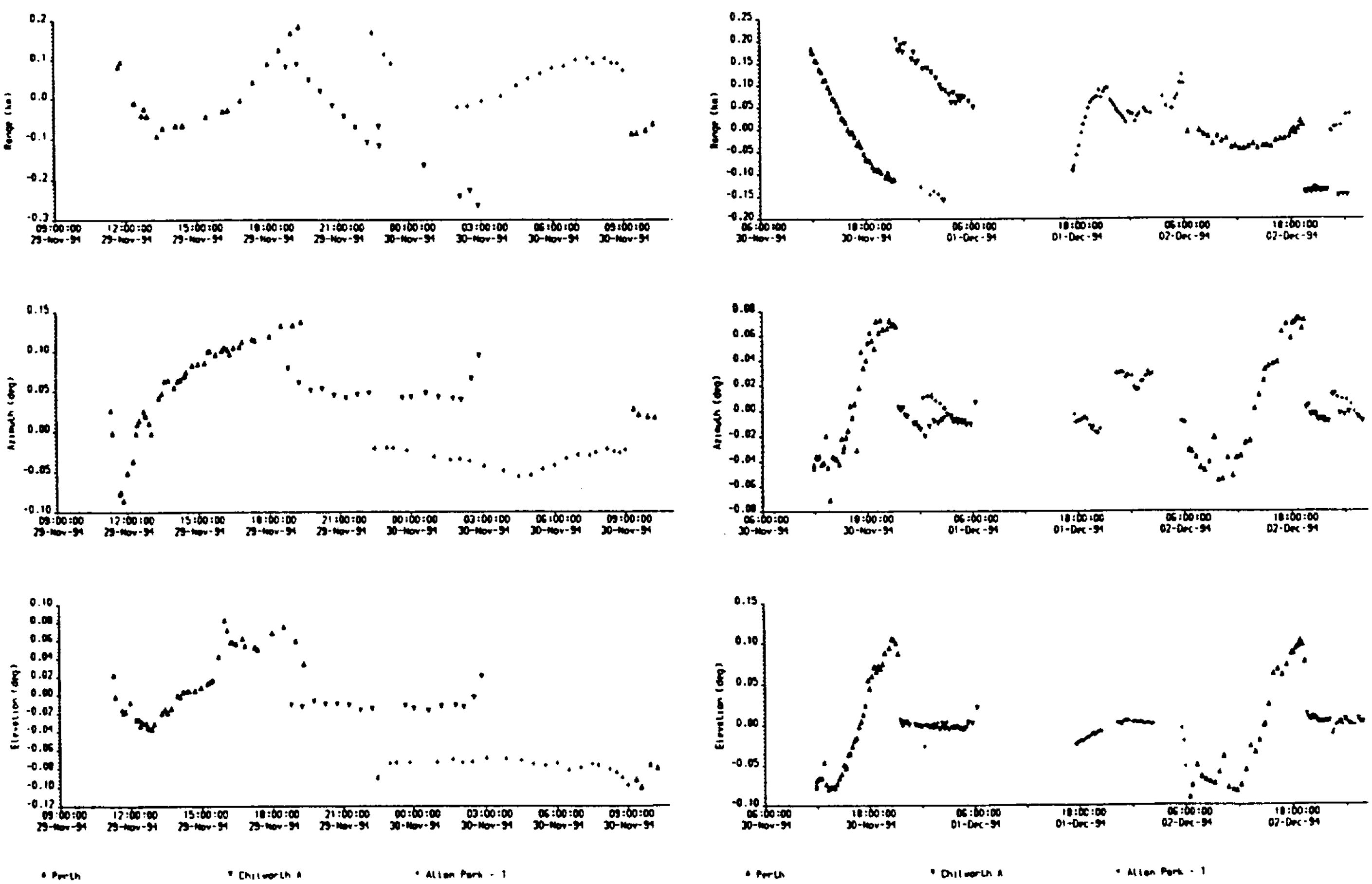

Figure 5 Range, Azimuth and Elevation Residuals prior to LAE 1 and between LAE 1 and LAE 2

The convergence properties during the LEOP were excellent, typically converging in three or four iterations. Various forms of weighting were available, namely:

Covariance scaling: this scales the solution covariance by the sum of the residuals divided by the number of observations, in an attempt to account for the residuals not being truly Gaussian white noise in nature. Note this does not affect the solution, only the covariance matrix.

RMS weighting: this uses the RMS of the measurement residuals from the previous iteration to weight the measurements of the current iteration. This will affect the solution, since it alters the relative weighting of the observations. It was only used once the solution had already converged as it could lead to instability if used initially.

Model weighting: this adjusts the weighting of the measurements according to the size of the correction predicted by the tropospheric model - in other words, measurements with large corrections are weighted less. This ensures that errors in the model do not significantly drive the solution.

Results obtained during the LEOP were excellent. White noise in the measurements was smaller than previously noted in GTO. Time-varying trends in the residuals were rarely greater than 300 m, again smaller than seen in GTO. Figure 5 shows the residuals obtained before and after the first LAE firing.

It was concluded that orbit determination for the SSTO mission was not significantly different from GTO.

ATTITUDE DETERMINATION

ORION F1 uses a Galileo Earth and Sun Sensor to produce sun and earth measurements for attitude determination whilst spinning. The sun sensors are the V-slit type which produce a single pulse when the sun passes through the field of view of the meridian sensor and another pulse when the sun passes through the field of view of the oblique sensor. The earth sensors are 2 pencil beams canted 4° apart and sensitive in the 14 - 16 mm wavelength range. They produce a pulse for space-earth and earth-space horizon crossings based on the derivative of the energy throughput of the detector and the value of the last threshold measured.

The ORION AOCS pre-processes these measurements before telemetering them to the ground where the QUARTZ software converts them into the basic 5 uncorrelated observations. These are: Sun Phase, the angle between the meridian and oblique sun sensor pulses; Chord Widths, the angle between the space-earth and earth-space horizon crossings for each sensor; and Separation Angles, the angle between the meridian sun pulse and the centre of the two earth pulses for each earth sensor. All angles are rotation angles measured around the spin axis.

The QUARTZ attitude determination software uses a Gauss-Newton weighted least squares algorithm to minimise the residuals between real and simulated observations and produce an updated attitude estimate. To reduce the errors on the simulated observations, both the earth sensor field of view and electronic delays caused by the components of the earth sensor are modelled. This procedure is fully described in reference 1.

The software identifies the attitude state by an inertial representation of the attitude along with biases for each of the observations and the earth sensor cant angles. Any number of these parameters can be optionally solved for, or fixed to a-priori values. (Chord width biases are not actually contained within the solution state - estimation of these values is made based upon minimising residual values still further once the WLS iteartion has been completed).

Results Summary

ORION has a small misalignment (about 1°) between the spacecraft spin-axis and the body Z axis, which also varies with propellant fill fraction. Measuring this dynamic imbalance angle (wobble angle) was particularly significant for attitude determination since the angle between the sensor and the spin axis contains a component of the wobble. With the help of apogee and perigee attitude data passes, this value was able to be refined in-flight, removing a significant error source from the Attitude Determination.

The solution method adopted was to fix only the sun phase bias and solve for all other biases (cant angle, separation angles and chord width). This method proved to be both consistent and robust.

The attitude determination results are shown in Table 5. All values are in degrees. From the table it is apparent that the attitude determination solutions are consistent with the orbit match solutions. The largest discrepancy between determined solutions and their subsequent orbit match is 0.32 degrees (AD A3 & P4 Orbit Match) for any of the data passes. It is interesting that the spin-axis right ascension seems almost as difficult to solve for as the spin-axis declination - this could be attributed to a residual wobble angle error.

Table 5 Comparison of Attitude Solutions from Attitude Determination and Orbit Matching

Solution	Attitude		Biases					
	RA	Dec.	Cant Angles		Separation Angles		Chord Widths	
			ES1	ES2	ES1	ES2	ES1	ES2
GD Injection	136.793	-15.901						
A1 AD	137.169	-15.925	0.074	0.079	0.013	0.093	-0.026	-0.014
A1 Orbit Match	137.275	-15.991						
P2 AD	137.117	-15.998	-0.149	-0.168	0.223	0.309	0.039	0.034
A2 AD	137.102	-16.083	0.191	0.185	0.043	0.115	-0.034	-0.012
A2 Orbit Match	137.230	-15.958						
Dog Leg Slew + Trim								
P3 AD	137.510	7.675	-0.100	-0.115	0.317	0.394	0.081	0.077
A3 AD	137.354	7.145	0.127	0.109	-0.103	-0.041	-0.031	-0.044
P4 AD*	137.570	7.578	0.094	0.063	0.399	0.491	0.116	0.090
P4 Orbit Match	137.491	7.436						
A4 AD	137.58	7.459	-0.012	-0.012	0.206	0.308	-0.169	-0.181
P5 AD	137.568	7.508	0.128	0.112	0.428	0.526	0.068	0.064
Trim (2 pulses)								
A5 AD	137.594	7.898	0.011	0.000	0.302	0.417	-0.090	-0.098
P6 AD*	137.516	7.769	0.297	0.289	0.505	0.627	0.061	0.059
P6 Orbit Match	137.445	7.634						

* Data incomplete due to LAE firing during data pass.

The perigee passes generated solutions which were closer (0.12-0.24 degrees) to the respective orbit match solution than the solutions generated from the apogee passes (0.24-0.32 degrees). This is consistent with pre-launch covariance analysis that predicted that the apogee pass accuracy would not be as good. This is described more fully in Reference 1.

A selection of residuals from the mission are shown in Figures 6 to 11. Only the earth sensor residuals are shown as the sun phase residuals are similar to typical GTO missions and show very well the observation quantisation (in this case 0.011 deg).

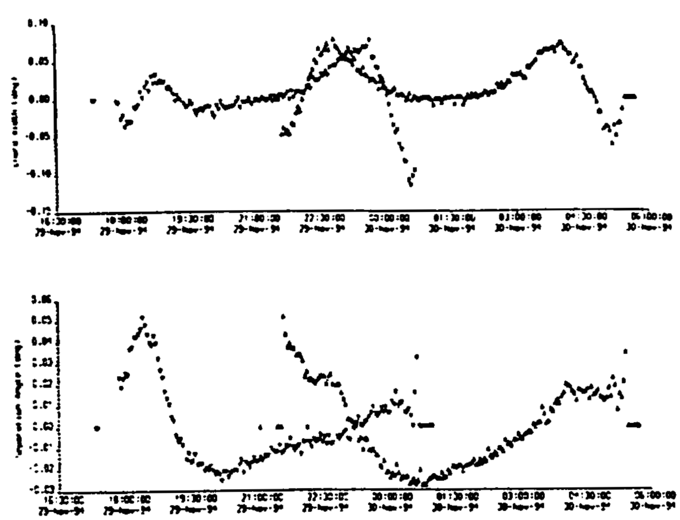

Figure 6 A1 Earth Chord & Separation Residuals

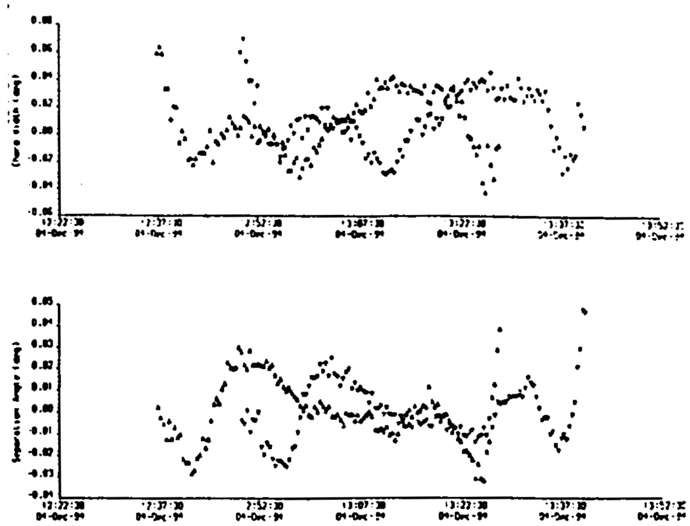

Figure 7 P3 Earth Chord & Separation Residuals

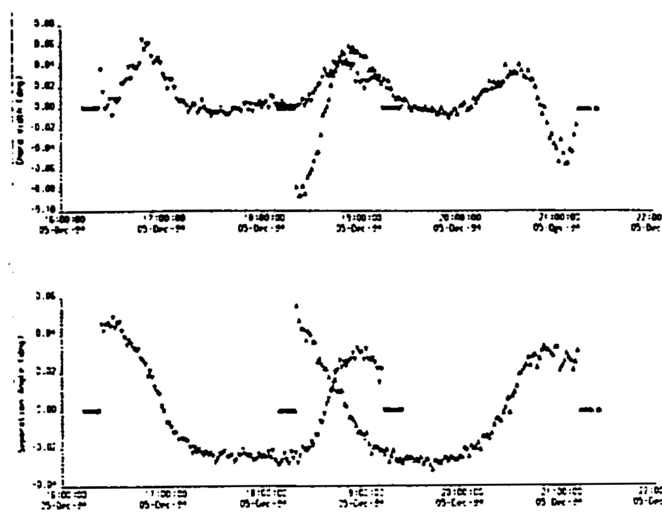

Figure 8 A3 Earth Chord & Separation Residuals

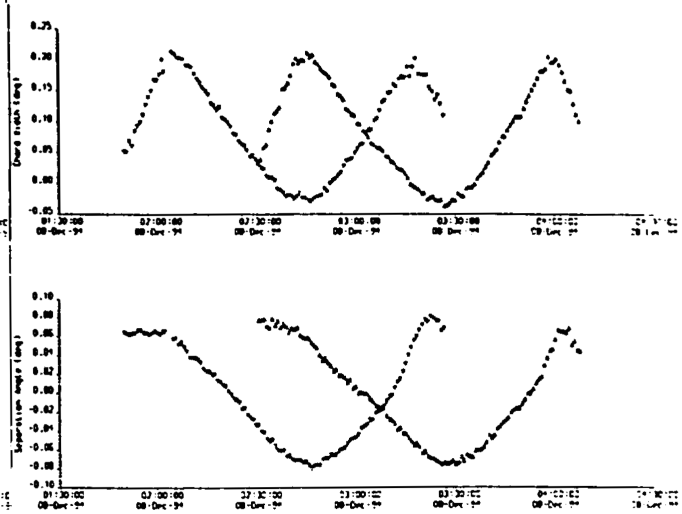

Figure 9 A4 Earth Chord & Separation Residuals

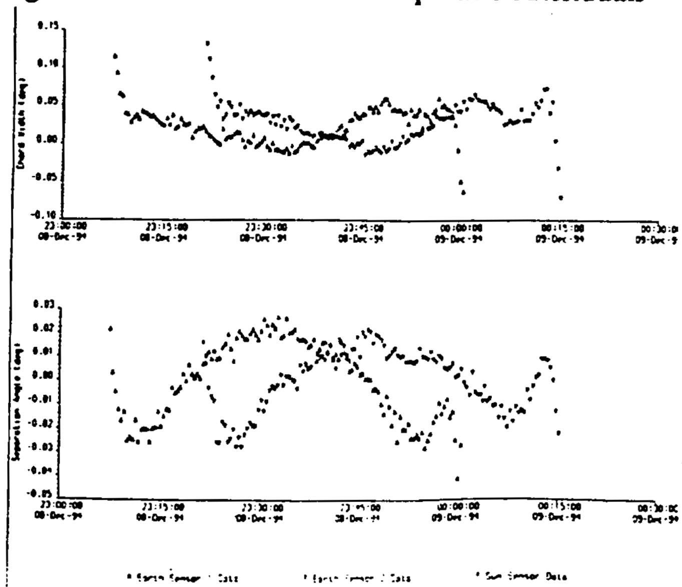

Figure 10 P5 Earth Chord & Separation Residuals

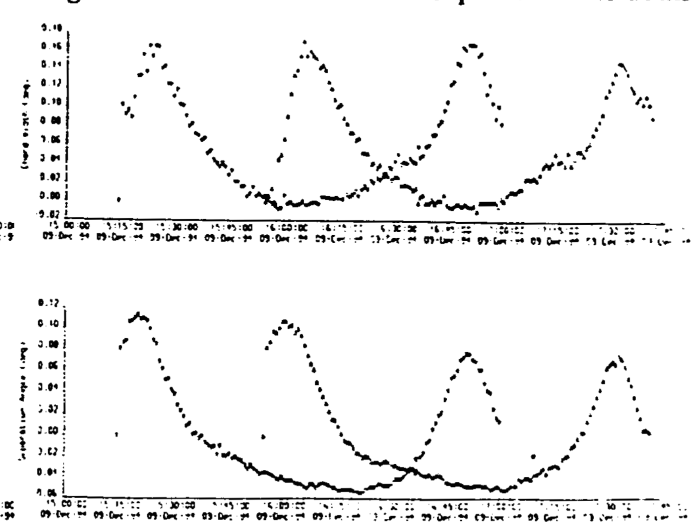

Figure 11 A5 Earth Chord & Separation Residuals

The passes shown are:

First apogee	(A1)	altitude 123,000 km
Third perigee	(P3)	altitude 25,000 km
Third apogee	(A3)	altitude 123,000 km
Fourth apogee	(A4)	altitude 65,000 km
Fifth perigee	(P5)	altitude 36,000 km
Fifth apogee	(A5)	altitude 65,000 km.

Looking at the residuals from the mission, the overwhelming first impression is of the clearly defined symmetric patterns in the SSTO, ITO apogee residuals. Their magnitudes are at least as small as GTO without the usual increase at the ends. This non-divergence at the start and end of coverage allowed the processing of all data from the apogee passes without having to cull any data. The perigee passes have less structure and are more similar to GTO but have extremely small separation angle residuals, barely exceeding 0.03 deg. The level of noise is similar for both apogee and perigee and is of the order of 0.01° and with at most 14 data points smoothed into one value the 'cleanliness' of the data has not been artificially introduced.

The patterns in the apogee data residuals are very definite and there are clear similarities between residuals patterns for data produced at similar altitudes. The patterns generated at 120,000 km (A1 & A3) have a shape very similar to that produced by the field of view and electronic delay models and therefore these patterns could be attributed to mis-modelling. It is quite significant if at super-synchronous altitudes, these modelling errors are the only significant source of error.

Discussion of Attitude Determination in SSTO

The geometry of the transfer orbit was favourable being closer to the winter solstice and avoiding sun-orbit-earth coplanarity. The large declination of the sun allowed good definition of the attitude to offset the slowly varying separation angle partial derivatives due to the low inclination.

It was thought prior to launch that diurnal earth luminance variations might produce unstable and unusable observations during the apogee passes. In fact the residual patterns are more symmetrically structured at super-synchronous altitudes which implies that the diurnal luminance variations (not simulated in QUARTZ) decrease with altitude above geosynchronous. Above geosynchronous altitudes, the relative size of the sensor field of view is larger with respect to the apparent size of the earth and the variations in the infra-red profile across the earth's surface are thus 'smoothed out'. At the highest altitude of 120,000 km, the residual patterns show virtually no random features apart from the noise at all.

Concerns were also raised that for apogee passes, very short chord data, at the beginning and end of each pass, may not be usable, and because the earth would appear smaller, it may be necessary to cull a larger percentage of data than in GTO. Pre-flight analysis had shown us that our sensor modelling ought to allow us to use all the data from each of the apogee passes, from the first chord to the last and the mission proved that this was indeed the case.

In fact, it turned out that the edge chords were extremely well behaved (compared to the simulated values) and no edge chord data had to be routinely culled, suggesting that the modelling is adequate. Modelling the electronic delays in the earth sensor reduces the errors in modelling small chords where the two horizon crossing pulses may merge, but even this is unlikely to occur at 13 rpm since the sensor transfer function is optimised up to spin rates of 90 rpm.

The chord width biases in GTO missions have tended to show a similarity between solved for values on odd revs. and even revs. It has been postulated that this is because the scanned earth longitudes are similar on alternate revs. In SSTO, two patterns emerge; there is a similarity between apogee passes (negative biases) and perigee passes (positive biases); and the biases are smaller in absolute value, when the data is produced further away from the earth. The explanation for the first pattern is not clear. The second pattern is explained as a seasonal earth luminance variation where this apparent change in earth size varies as an angular measure with distance from the earth.

GTO residuals often exhibit random patterns within themselves which vary from one pass to the next. They are thought to be due to diurnal variations in the earth's infra-red profile. The general 'W' or 'U' shapes seen in the residuals are most likely caused by modelling errors increasing when the scan of the sensor crosses close to the earth limb. The SSTO apogee residuals show very pronounced patterns which are extremely symmetrical and seem to depend on altitude, the GTO-like patterns re-emerging at lower altitudes. Apart from white noise, there is virtually no random fluctuation at all. It is therefore assumed that most of the residual patterns in SSTO are due to modelling errors.

We can conclude that the methods and modelling employed for GTO can be transported wholesale to the SSTO case. The enormous amount of data produced in SSTO can be reduced to manageable proportions with an efficient smoothing process, and the representation of the attitude state and sensor modelling are still adequate for the purpose.

The mission design of not relying on any passes of apogee data to produce an attitude solution can now be seen to be over-cautious because the solutions generated from the apogee data passes were very consistent with the solutions from the perigee passes and with the orbit matches.

MANOEUVRE OPTIMISATION

Optimisation of the four firings of the Liquid Apogee Engine required a more sophisticated approach than adopted for GTO. For each firing, the attitude, burn duration and start epoch were to be optimised, giving a maximum of sixteen optimisation variables.

The algorithm selected was the Multiple Shooting Algorithm. This divides up the Transfer Orbit into a four segments (the four coast arcs between firings) and matches the interface between segments as a set of extra internal constraints. At each interface, the six keplerian elements and the spacecraft mass must match. This leads to a total of 28 internal constraints.

Although the method is first order and progress towards the optimum slows down as the optimum is reached, it is nevertheless robust and does not rely on computing second order differentials. Progress may be slow, but it is guaranteed (given a suitable selection of the tuning constants). This turned out to be cost effective, since we were able to speed up the convergence after development by upgrading to a VAX Workstation 4000/90. The software now performs a typical convergence of 100 iterations in four minutes.

Options available were to optimise all four parameters per burn, to fix those parameters to a given value or in the case of burns 2 and 4, to set the attitude parameters to those on the previous burn. This latter option allows the optimiser to keep the attitude the same between the two apogee and the two perigee burns, thus minimising the need for an attitude manoeuvre between these firings. This strategy resulted in no trim being required between the two apogee burns, and only a 0.3° trim between the perigee burns, which in turn allowed a greater reliance to be placed on the attitude solutions achieved from the ESS and the orbit matches, since the attitude had not been altered significantly.

The manoeuvre optimisation became progressively more simple with each firing since their were less manoeuvres to optimise and less constraints to meet. Optimising the first firing was the most intensive, and many cases were studied in the twenty four hours leading to the first firing.

The nominal cases for LAE1 were referred to as the OPOP, OOOP, FOOP and FPOP solutions. These nomenclatures refer to whether the manoeuvres were optimised, fixed or set to previous attitude, with one letter for each burn. Note that in all these cases the third firing is always optimised (since it must be slewed to anyway) but the fourth firing is assumed to be the same attitude as the third. The penalty in assuming no trim between LAE3 and LAE4 during the LAE1 optimisation was shown before launch to be negligible (< 1 gram).

FPOP represents the worst case by not trimming the attitude before the first or second firing. OOOP represents the best case by trimming before both. OPOP trims before the first firing but keeps this attitude for the second firing, whilst FOOP fixes the attitude for the first firing but trims before the second firing.

No difference was found between the OOOP and OPOP cases showing there would be no penalty in adopting a no trim between apogee burns strategy. However, there was significant uncertainty in the attitude prior to LAE1, so that trimming to an optimised attitude was not considered viable.

Given an uncertainty in attitude, the best strategy was to defer any attitude trim to before LAE2. This allows the AD solution to be compared to the orbit matched solution once the first burn has taken place.

As well as optimising the nominal case, studies around the optimum were performed by varying the attitude in Right Ascension and Declination. Grids of 7 by 7 points were performed, each point being a full optimisation of 50 iterations, so that a contour plot could be produced of the attitude sensitivity. These studies took about an hour to run on a VAX 4000/90 Work Station.

For the first firing, the following results were obtained for propellant penalties with respect to the optimum OOOP strategy:

OOOP	0 grams
OPOP	0 grams
FOOP	29 grams
FPOP	54 grams

As the maximum penalty was only 54 grams even if no trim before LAE2 was performed, then it was clear that staying in the separation attitude for both apogee burns was an acceptable strategy (54 grams of propellant is less than one day of life on-station).

This situation did not change between the first and second firing and in fact no attitude manoeuvre was performed until after the second firing, when a slew was necessary to achieve the perigee firings.

THE DOG LEG SLEW

Contrary to normal multi-burn GTO missions, a 22° slew in declination between the second and third firings is mandatory. In order to collect attitude data through perigee 3 close to the optimum LAE3 attitude, the slew was performed as two segments separated in phase by 70°. This strategy is referred to as a Dog-Leg Slew and allows the calibration of both phase angle and precession using only solar aspect angle data (Reference 4). Once both legs have been performed and the calibrations made, the final attitude can be computed and a trim back to the target attitude can be performed.

An additional tactic employed to minimise the possibility of any subsequent trim was to calibrate the phase angle only after the first segment.

The first segment of the slew was performed five hours after LAE2 and the second segment two hours later. The results of the calibration for the two segments were 0.97 for the precession calibration and 14.8 msec for the phase calibration. The required trim was 3.3° in declination to the target LAE3 attitude - this manoeuvre was performed two hours before perigee 3 and allowed the attitude determination to be performed in the LAE3 attitude.

STATION ACQUISITION

The spacecraft was eventually placed in a Drift Orbit 4° West of station with a 234 km apogee bias and a -162 km perigee bias, resulting in a drift rate of 0.3°W per day. A drift rate away from station was chosen to avoid any possibility of passing in front of TDRSS4 located 3.5°W of station, since the intense and time critical operations associated with acquiring Earth pointing may have caused RF interference with that spacecraft.

Two West manoeuvres were performed to remove the apogee bias on the 16th and 19th of December. This resulted in a drift back toward station of 1.1°E per day. Three East manoeuvres were then performed to remove the perigee bias and slow the spacecraft, on the 22nd and 23rd December. The spacecraft was finally placed in the centre of the station-keeping box with a drift rate of 0.01°E per day. This resulted in a full station-keeping cycle, with no further East-West manoeuvres required until 6th January 1995. The eccentricity vector was correctly initialised for the sun-pointing strategy, with an eccentricity of 2.5×10^{-4} and perigee towards the sun.

CONCLUSION

The ORION SuperSynchronous Transfer Orbit mission was a complete success. The strategy of employing two apogee and two perigee burns resulted in maximum spacecraft life. A multi-burn strategy also allowed the final burn to take place at the On-Station longitude with a near zero drift rate after the final firing. This is desirable to avoid any RF interference whilst drifting past existing spacecraft in this crowded GEO sector.

The main conclusions for attitude determination are that the Galileo ESS can cope with the smaller energy throughput and smaller earth size typical in super-synchronous transfer orbits and produce sensible data. Both apogee and perigee data can be processed on the ground with no upgrade from modelling suitable for GTO to produce accurate solutions, and no data needs to be routinely culled as all the data in apogee passes is accurate enough to be used. Finally, apogee and perigee chords together allow the determination of the wobble angle, removing a significant error source from the attitude determination.

On the down side, the Transfer Orbit phase is long compared to GTO. Five and a half revolutions took just over eleven days, compared to a GTO mission of between two and five days. Although in many ways the mission was more complex than a standard GTO, the longer orbit periods did allow the intensive Flight Dynamics activities to be performed in the greater time available.

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